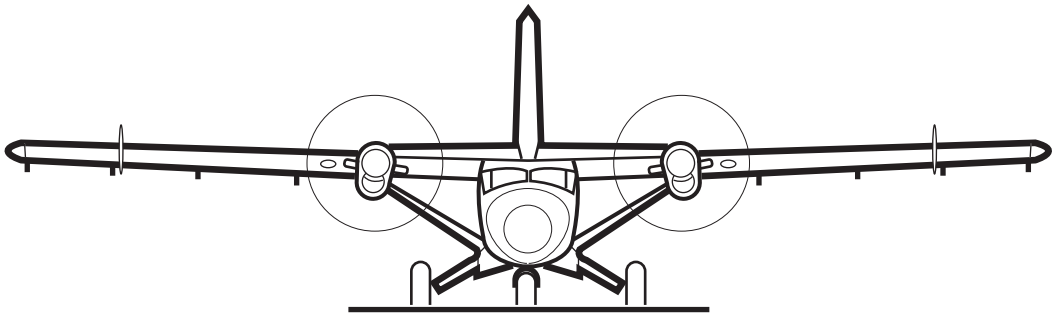
PSM 1-64-2



DHC-6 TWIN OTTER
SERIES 400

MAINTENANCE MANUAL
VOLUME 1

VIKING

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Revision 8 - Sep. 28/18

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January 31, 2006

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February 4th, 2020

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revisions 31-37 and 31-38 – dated February 4, 2020

1 Insert the attached Temporary Revisions (TRs) 31-37 and 31-38 into PSM 1-64-2, as directed in the placement instructions on the cover page of each TR.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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June 27th, 2019

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revisions 20-13, 27-35 and 91-5 – dated June 27, 2019

1 Insert the attached Temporary Revisions (TRs) 20-13, 27-35 and 91-5 into PSM 1-64-2, as directed in the placement instructions on the cover page of each TR.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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May 29th, 2019

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revision 28-16 – dated May 29, 2019

1 Insert the attached Temporary Revision (TR) 28-16 into PSM 1-64-2, as directed in the placement instructions on the cover page.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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May 21st, 2019

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revisions 07-2 and 27-34 – dated May 21, 2019

1 Insert the attached Temporary Revisions (TRs) 07-2 and 27-34 into PSM 1-64-2, as directed in the placement instructions on the cover page of each TR.

Please Note: TR 07-2 Supersedes TR 07-1.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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April 2nd, 2019

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revision 76-2 – dated April 2, 2019

1 Insert the attached Temporary Revision (TR) 76-2 into PSM 1-64-2, as directed in the placement instructions on the cover page.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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March 27th, 2019

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revision 12-18 – dated March 26, 2019

1 Insert the attached Temporary Revision (TR) 12-18 into PSM 1-64-2, as directed in the placement instructions on the cover page.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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November 19th, 2018

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revision 76-1 – dated November 19, 2018

1 Insert the attached Temporary Revision (TR) 76-1into PSM 1-64-2, as directed in the placement instructions on the cover page.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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November 14th, 2018

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revisions 27-33 – dated November 14, 2018

1 Insert the attached Temporary Revision (TR) 27-33 into PSM 1-64-2, as directed in the placement instructions on the cover page.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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October 31st, 2018

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revisions 11-9, 24-17 & 24-18 – dated October 30, 2018

1 Insert the attached Temporary Revisions (TRs) 11-9, 24-17 and 24-18 into PSM 1-64-2, as directed in the placement instructions on the cover page of each Temporary Revision.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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October 16th, 2018

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revision 07-1 – dated October 16, 2018

1 Insert the attached Temporary Revision (TR) 07-1 into PSM 1-64-2, as directed in the placement instructions on the cover page.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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October 10th, 2018

To: Distribution

Subject: PSM 1-64-2, DHC-6 Series 400 – Maintenance Manual

Incorporate Temporary Revisions 25-19 and 33-3 – dated October 10, 2018

1 Insert the attached Temporary Revisions (TRs) 25-19 and 33-3 into PSM 1-64-2, as directed in the placement instructions on the cover page of each Temporary Revision.

2 Update the Temporary Revisions Index in the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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September 28th, 2018

To: Distribution

Subject: PSM 1-64-2, DHC-6 Twin Otter Series 400 – Maintenance Manual

Incorporate Revision 8, dated September 28, 2018

This package contains Revision 8 of PSM1-64-2, DHC-6 Twin Otter, Series 400 – Maintenance Manual which is a complete rewrite of the manual. Change bars are applied only to changes incorporated in this manual.

Please note that the following Temporary Revisions have been incorporated into this revision:

Introduction	–	TR-6
Chapter 11	–	TRs 11-7 & 11-8
Chapter 12	–	TRs 12-14 to 12-17
Chapter 20	–	TRs 20-11 & 20-12
Chapter 23	–	TRs 23-16 to 23-18
Chapter 24	–	TRs 24-12 to 24-16
Chapter 25	–	TRs 25-15 to 25-18
Chapter 27	–	TRs 27-23 to 27-32
Chapter 28	–	TRs 28-13 to 28-15
Chapter 31	–	TRs 31-32 to 31-36
Chapter 32	–	TR 32-18
Chapter 33	–	TRs 33-1 & 33-2
Chapter 34	–	TRs 34-34 to 34-46
Chapter 52	–	TR 52-8
Chapter 53	–	TR 53-4
Chapter 71	–	TR 71-5

Please note that the following new sections have been added at Revision 8 and will not have change bars:

20-50-04 – Environmental Sealing (Self Leveling Green Sealant)

25-50-01 – STA 332 Modified Bulkhead and Cargo Net Installation

28-10-31 – Fuel Gallery Access Panel Seals (Mod 6/2408)

31-60-00 – Remote Image Bus (RIB) (S.O.O. 6212)

Additional highlights

Change bars will indicate new Mods, S.O.O.s and other technical updates included in the following sections:

09-00-00; 11-00-00; 12-10-15; 12-20-05; 20-60-01; 20-75-00; 26-10-00; 26-20-00; 26-20-11; 27-00-00; 27-10-11; 27-20-11; 27-30-11; 27-36-11; 27-38-00; 27-50-21; 27-50-31; 28-10-11; 30-10-00; 31-10-00; 31-15-00; 31-31-00; 31-32-00; 33-00-00; 61-10-00; 61-22-02; 71-00-00; 76-10-00

1. Remove and replace all pages except for the Record of Revisions and Record of Temporary Revisions
2. Update the Temporary Revisions Index with all the TRs incorporated in this Revision.
3. Record the insertion of Revision 8 into the Record of Revisions at the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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June 28th, 2017

To: Distribution

Subject: PSM 1-64-2, DHC-6 Twin Otter Series 400 – Maintenance Manual

Incorporate Revision 7, dated June 28, 2017

1. This package contains Revision 7 of PSM1-64-2, DHC-6 Twin Otter, Series 400 – Maintenance Manual.

2. Using the List of Effective Pages for each chapter, remove and replace the revised pages, dated June 28, 2017.

3. Please note that the following Temporary Revisions have been incorporated into this revision:

Chapter 8	–	TR 08-1
Chapter 11	–	TR 11-6
Chapter 12	–	TR 12-11
Chapter 20	–	TRs 20-9 & 20-10
Chapter 21	–	TRs 21-3 & 21-4
Chapter 22	–	TRs 22-6 to 22-8
Chapter 24	–	TRs 24-6 to 24-10
Chapter 25	–	TRs 25-12 to 25-14
Chapter 28	–	TRs 28-8 to 28-12
Chapter 31	–	TRs 31-22 to 31-31

Chapter 34 – TRs 34-29 to 34-33

Chapter 45 – TR 45-12

Chapter 71 – TR 71-4

Chapter 80 – TR 80-2

Chapter 91 – TR 91-4

4. Please remove and destroy the TRs listed.

5. Update the Temporary Revisions Index.

6. Record the insertion of Revision 7 into the Record of Revisions at the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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May 20th, 2016

To: Distribution

Subject: PSM 1-64-2, DHC-6 Twin Otter Series 400 – Maintenance Manual

Incorporate Revision 6, dated May 20, 2016.

1. This package contains Revision 6 of PSM1-64-2, DHC-6 Twin Otter, Series 400 – Maintenance Manual.

2. Using the List of Effective Pages for each chapter, remove and replace the revised pages, dated May 20, 2016.

3. Please note that the following Temporary Revisions have been incorporated into this revision:

Introduction	–	TR-4
Chapter 12	–	TR 12-10
Chapter 22	–	TRs 22-1 to 22-5
Chapter 23	–	TR 23-15
Chapter 24	–	TR 24-4
Chapter 28	–	TRs 28-6 & 28-7
Chapter 31	–	TRs 31-17 to 31-21
Chapter 32	–	TR 32-17
Chapter 34	–	TRs 34-23 to 34-28
Chapter 45	–	TRs 45-7 to 45-11

Chapter 52 – TRs 52-6 & 52-7

Chapter 91 – TRs 91-1 to 91-3

Appendix A – TR-5

4. Please remove and destroy the TRs listed.

5. Update the Temporary Revisions Index.

6. Record the insertion of Revision 6 into the Record of Revisions at the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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August 28th, 2015

To: Distribution

Subject: PSM 1-64-2, DHC-6 Twin Otter Series 400 – Maintenance Manual

Incorporate Revision 5, dated August 28, 2015.

1 This package contains Revision 5 of PSM1-64-2, DHC-6 Twin Otter, Series 400 – Maintenance Manual.

2 Using the List of Effective Pages for each chapter, remove and replace revised pages, dated August 28, 2015.

3 Please note that the following Temporary Revisions have been incorporated into this revision:

Chapter 11 – TRs 11-4

Chapter 25 – TRs 25-10, 25-11

Chapter 29 – TRs 29-1 to 29-3

Please remove and destroy the TRs listed above.

4 Update the Temporary Revisions Index.

5 Record the insertion of Revision 5 into the Record of Revisions at the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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April 30th, 2014

To: Distribution

Subject: PSM 1-64-2, DHC-6 Twin Otter – Maintenance Manual

Incorporate Revision 3, dated April 30, 2014.

1 This package contains Revision 3 of PSM1-64-2, DHC-6 Twin Otter, Series 400 – Maintenance Manual.

2 Using the List of Effective Pages for each chapter, remove and replace revised pages, dated April 30, 2014.

3 Please note that the following Temporary Revisions have been incorporated into this revision:

Chapter 11	–	TR 11-3
Chapter 12	–	TRs 12-6 & 12-7
Chapter 20	–	TR 20-8
Chapter 21	–	TR 21-1
Chapter 23	–	TRs 23-8 to 23-12
Chapter 24	–	TR 24-1
Chapter 25	–	TR 25-4

Chapter 28 – TRs 28-3 to 28-5

Chapter 46 – TRs 46-1 & 46-2

Also included in this revision are the incorporation of new S.O.O.s and Mods (highlighted by change bars) that have been developed since the last revision of the Series 400 AMM.

4 Update the Temporary Revisions Index.

5 Record the insertion of Revision 3 into the Record of Revisions at the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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July 12, 2013

To: Distribution

Subject: PSM 1-64-2, DHC-6 Twin Otter – Maintenance Manual

Incorporate Revision 2, dated July 10, 2013

1 This package contains Revision 2 of PSM 1-64-2, DHC-6 Twin Otter, Series 400 – Maintenance Manual.

2 Using the List of Effective Pages for each chapter, remove and replace revised pages dated July 10, 2013.

3 Please note that the following Temporary Revisions have been incorporated in this revision:

Introduction	–	TR 3
Chapter 11	–	TR 11-2
Chapter 12	–	TRs 12-2 to 12-5
Chapter 23	–	TR 23-7
Chapter 24	–	TRs 24-2 & 24-3
Chapter 25	–	TRs 25-2 & 25-3
Chapter 28	–	TR 28-2

Chapter 31 – TRs 31-10, 31-11 & 31-13

Chapter 32 – TRs 32-2 to 32-8, 32-10 to 32-16

Please Note: TR 32-9 was cancelled (not issued)

Chapter 34 – TRs 34-20 to 34-22

Chapter 53 – TR 53-3

Chapter 55 – TR 55-1

Also included in this revision are the incorporation of new S.O.O.s and Mods (highlighted by change bars) that have been developed since the last revision of the Series 400 AMM.

4 Update the Temporary Revisions Index.

5 Record the insertion of this revision in the Record of Revisions.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

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November 30, 2011

To: Distribution

Subject: PSM 1-64-2, DHC-6 Twin Otter – Maintenance Manual

Incorporate Revision 1 dated November 30, 2011

1 This package contains a complete rewrite of PSM 1-64-2, DHC-6 Twin Otter, Series 400, Maintenance Manual.

2 Remove and replace all pages except for the Record of Temporary Revisions.

3 Please note that the following Temporary Revisions have been incorporated in this revision:

Introduction – TR-1

Appendix A – TR-2

Chapter 20 – TRs 20-1 to 20-7

Chapter 23 – TRs 23-1 to 23-6

Chapter 25 – TR 25-1

Chapter 27 – TR 27-1 to 27-20

Chapter 28 – TR 28-1

Chapter 31 – TR 31-1 to 31-2, TR 31-6 and TR 31-9

Please Note: TR 31-3 to 31-5 and 31-7 to 31-8 were not issued.

Chapter 34 – TRs 34-1 to 34-12 and 34-14 to 34-19

Please Note: TR 34-13 was not issued.

Chapter 45 – TRs 45-1 to 45-6

Chapter 52 – TRs 52-1 to 52-5

Chapter 53 – TRs 53-1 and 53-2

Chapter 54 – TR 54-1

Chapter 56 – TR 56-1

Chapter 71 – TRs 71-1 and 71-3

Please Note: TR 71-2 was not issued.

Chapter 80 – TR 80-1

4 Record the insertion of this revision in the Record of Revisions, supplied in this package, and place at the front of the manual.

Note: We recommend that this transmittal letter be kept for record purposes and inserted at the front of the manual.

PSM 1-64-2

DHC-6 TWIN OTTER SERIES 400

MAINTENANCE MANUAL

Retain this record in the front of the manual. On receipt of revisions, insert revised pages in the manual and enter date inserted and initials against the appropriate Revision Number and Date.

Rev. No.	Revision Date	Insertion Date	By	Rev. No.	Revision Date	Insertion Date	By
1	30-NOV-11	1/13/2012	ATP/RLL				
2	7/10/2013	8/2/2013	ATP/SS				
3	4/30/2014	5/15/2014	ATP/VP				
4	MAR 31, 2015	APR 8, 2015	ATP/RD				
5	AUG 28, 2015	SEP 2, 2015	ATP/RLL				
6	MAY 20, 2016	MAY 24, 2016	ATP/RLL				
7	JUN 28, 2017	JUL 3, 2017	ATP/RLL				
8	SEP 28, 2018	OCT 8, 2018	ATP/RD				

RECORD OF REVISIONS

TEMPORARY REVISION INDEX

Retain this index in the front of the manual. When Temporary Revisions are issued or cancelled, enter the relevant information in the columns below.

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27-2	15-OCT-10	30-NOV-11	27-33-212 face pg.3	34-1	06-DEC-10	30-NOV-11	34-30-00 after pg.2
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53-1	15-OCT-10	30-NOV-11	53-10-01 after 53-10-00	34-7	06-DEC-10	30-NOV-11	34-53-00 after pg.2
53-2	15-OCT-10	30-NOV-11	53-10-02 after TR53-1	34-8	06-DEC-10	30-NOV-11	34-54-00 after pg.2
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25-1	06-DEC-10	30-NOV-11	25-60-11 after pg.2	2	06-DEC-10	30-NOV-11	Appendix A after Chap-91

TEMPORARY REVISION INDEX

PSM 1-64-2

DHC-6 SERIES 400 MAINTENANCE MANUAL

TEMPORARY REVISION INDEX

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1. Organization

This manual has been prepared generally in accordance with A.T.A. Specification No. 100. This manual can be consulted in electronic form or as a paper copy. Although this manual is contained in two volumes, future revisions may necessitate it being divided into more volumes.

Note At initial certification of the Viking DHC-6 Series 400, the public commercial operations bundle (S.O.O 6210) which includes the IFR (Instrument Flight Rules) operations bundle (S.O.O. 6209) is the Phase 1 configuration. Further configurations can be found in PSM 1-6-12, with associated limitations in PSM 1-6-13.

2. Content

This manual contains descriptive and maintenance information on systems and items of equipment to the level required to service the aircraft, to understand, test, and trouble shoot the systems, and to remove and install the units when this is not self-evident. Work of a more complex nature which requires a unit to be off the aircraft, is covered in the relevant Maintenance, Repair, or Overhaul Manuals. Listing of these manuals are available in Appendix A.

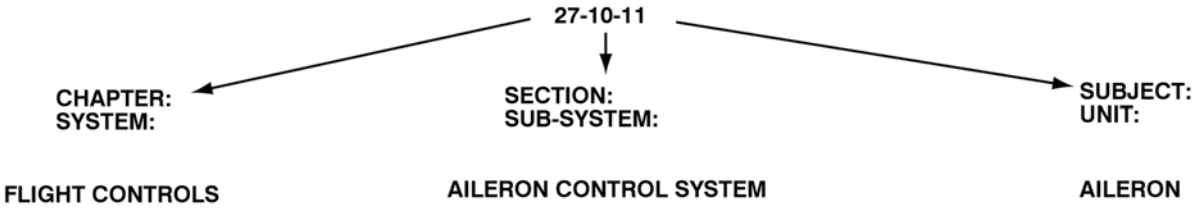
3. Using the Manual

A. Chapterization

Chapters are allocated in accordance with A.T.A. Specification No. 100, which provides a functional breakdown of the entire aircraft. Information on all units comprising a system will be found in the chapter identified by the name of that system. For example, all units relating to the generation and distribution of electrical power are covered in Chapter 24 – ELECTRICAL POWER, but the electrically operated aileron trim tab actuator, is covered in Chapter 27 – FLIGHT CONTROLS.

B. Chapter Breakdown

Each chapter is divided into sections and, where necessary, each section into subjects. These three elements (Chapter – Section – Subject) are allocated numbers which are located on the bottom outer corner of each page. The following is an example of a three element number:



Pages bearing the first element number only (27–00–00), contain descriptive and maintenance information relating to the complete system, but where necessary, the system is broken down into sub-systems which are covered on pages bearing the first and second element numbers (27–10–00). The sub-systems are broken down further into units, and will be found on pages bearing all three element numbers (27–10–11). Each system, sub-system and unit, is allocated a block of numbers to allow for coverage of various modification states; this also allows an operator to remove any information, which does not relate to his aircraft, from the manual. For example, if a modified aileron is introduced, both the original and modified ailerons would be covered under a separate subject number: Original aileron 27–10–11, Modified aileron 27–10–12.

C. Page Numbers

Page number blocks are allocated for the different subject topics (modules), the pages in each module being numbered from page 1 through 999. Since no one block of page numbers is likely to be fully used, and other blocks may not be used at all, it is inevitable that page numbering for a subject will have unused numbers in the sequence. The Lists of Effective Pages (refer to Para 3.D), can however, be used to check the completeness of the manual.

D. List of Effective Pages

In order that operators can establish that a Chapter is complete, and whether it contains the latest issue of all pages, a List of Effective Pages is located at the front of each chapter, immediately following the chapter divider card. On each list, the date against each page number should correspond with the date on the relevant page.

E. Airworthiness Limitations

The Airworthiness Limitations are covered under two separate documents; the Structural Components Service Life Limits Manual, PSM 1-6-11, as mandated by TCCA Airworthiness Directive CF-2000-14, (FAA Airworthiness Directive AD-2008-03-01) and the Airworthiness Limitation-Avionics Manual, PSM 1-6-13.

F. Inspection Requirements Manual

Chapter 5 – Time Limits/Maintenance Checks, is covered under a separate document, titled Inspection Requirements Manual PSM 1-6-7.

G. Corrosion Prevention and Control Manual

The Corrosion Prevention and Control Manual, PSM 1-6-5, provides mandatory supplementary inspections that are required to maintain the integrity of an aging aircraft.

The relationship between the Inspections Requirements Manual, PSM 1-6-7 and the Corrosion Prevention and Control Manual, PSM 1-6-5, is that PSM 1-6-7 provides overall coverage of the inspections programs considered necessary to maintain the aircraft in a continuous state of airworthiness. PSM 1-6-5 provides supplementary structural inspections that may be required to maintain the integrity of structures of aircraft approaching their originally intended economic design life.

H. Associated Manuals

Throughout this manual, cross-reference is made to other publications; a complete list of which, appears below:

Flight Manual	PSM 1-64-1A
Pilot Operating Handbook	PSM 1-64-POH
Ground Support Manual	PSM 1-6-2T
Wiring Diagram Manual	PSM 1-64-2W
Structural Repair Manual	PSM 1-6-3
Illustrated Parts Catalog	PSM 1-64-4
Corrosion Prevention and Control Manual	PSM 1-6-5
Corrosion Prevention and Control Manual	PSM 1-GEN-5
Component Maintenance Manuals	PSM 1-6-6A
*See Appendix A for a detailed list of CMM documents.	

Inspection Requirements Manual	PSM 1-6-7
Structural Components Service Life Limits Manual	PSM 1-6-11
Avionics Airworthiness Manual	PSM 1-6-13
Hartzell Propeller Owners Manual	139
Hartzell Propeller Overhaul Instructions	118
United Aircraft –34	Engine Manuals
Specific Operating Instructions	3021241
Maintenance Manual	3021242
Parts Catalogue	3021244
Honeywell Manuals	
Installation Engineering Bulletin of the APEX Avionics System in the Viking DHC-6 Series 400 Aircraft	EB60000477
Flight Line Maintenance Manual Primus Apex® For the Viking Twin Otter Series 400 Systems	D201202000012
Electrical Equipment Chasis MAU–12U Cabinet	CMM A09–6710–012
Power Supply Module	CMM A09–6710–015
Single NIC Module Assembly	CMM D200704000034
AGM–200 Advanced Graphics Module	CMM D200703000019
CIO–AP2 Custom I/O Card	CMM D200705000015
Dual Generic I/O Module Assembly	CMM A09–6710–018
AIO–F01 Actuator I/O Module	CMM A09–6710–032
IM–950 MAU Configuration Module	CMM A09–6710–007
KMC 9200 Flight Controller	006–10674–0000
KDU–1080 10" Flat Panel MFD/EICAS	CMM K5920078–00
KMC 2210 Display Controller	006–10675–0000
KMC 2210 Display Controller	CMM 071–01596–0201
KMC 2220 Multi-Function Display Controller	006–10676–0000
KMC 2220 Multi-Function Display Controller	071–01598–0301
Power Supply Module	CMM A09–6710–015
KSG 7200 ADAHRS, KMG 7010 Magnetometer	006–10662–0007
KSG 7200 ADAHRS	CMM 006–15662–0000
KMG 7010 Magnetometer	CMM 006–15663–0000

KGS 200 Global Navigation System Sensor Unit	006–10679–0001
KGS 200 Global Navigation System Receiver Unit	CMM 006–015706–0000
KTR 2280 Multi-Mode Digital Radio (MMDR)	CMM 006–15654–0000
KXP 2290/2290A Mode S Transponder	006–10689–0001
KXP 2290/2290A Mode S Transponder	CMM 006–15689–0000
KMA 29 Audio Amplifier /Intercom/Marker Beacon Receiver	0006–50678–0000
KA 44B ADF Antenna	CMM 006–05535–0003
KN 63 Distance Measuring Equipment	006–05176–0004
KN 63 Distance Measuring Equipment	CMM 006–05176–0001
Installation Manual for the Advanced Series Solid State Recorder System (AR–CVR, AR–FRD, AR–COMB)	012–0708
KHF 1050 Communications System	006–10640–0006
KHF 1050 Communications System	CMM 006–15642–0001
KHM 920/KTA970 Multi-Hazard Awareness/Traffic Alert & Collision Avoidance TCAS I	006–15633–0001
CAS 67A TCAS II System	006–05340–0008
KRA 405B Radar Altimeter	006–10536–0010
KRA 405B Radar Altimeter	CMM 006–15536–0010
Mark V Warning Computer	CMM 012–0534–001
MK VI MK VII Enhanced Ground Proximity Warning System (Class A TAWS) Installation Design Guide	060–4314–150
KRX 1053 Receiver/Exciter	CMM 006–15642–0001
RDR 2000 Colour Weather Radar System	006–00643–0007
RDR 2000 Colour Weather Antenna/Receiver/Transmitter	CMM 006–05332–0003
XM Weather Data Receiver System	A15–7162–001
XM Weather Data Receiver System	CMM XMD157–01–8
KSA 272A Trim Servo	MM 006–05290–0002

L3 Manuals	
GH–3100 Electronic Standby Instrument System Installation Manual	TP–559
GH–3900.2 Electronic Standby Instrument System Installation Manual	0040–34404–01
Magnetometer	0040–21200–01
Emergency Power Supply PS–850/PS–855	TP–470
WX–500 Installation Manual	009–11500–001

I. Test Definitions

The definitions of the tests detailed in the maintenance procedures are as follows:

1) Operational Test

A test used to ascertain that a system or component is in operable conditions. It can be done with the facilities installed in the aircraft.

2) Function Test

A test used to ascertain that a system or unit is functioning properly in all aspects in conformance with minimum acceptable design specifications. The test may require supplemental ground support or test equipment.

3) Bench Test

A test which requires removal of the item from the aircraft; a visual inspection for general condition, impending failure, need for lubrication, repair or replacement of parts and adjustment or calibration to proper specifications.

4. Revisions

A. Normal Revisions

Revisions will be made to this manual periodically, and active subscription holders will receive copies without request.

B. Temporary Revisions

Temporary Revisions will be issued at the earliest opportunity, to notify the operator of important changes affecting the maintenance of the aircraft. These revisions will be printed on yellow paper for instant identification, when incorporated in the manual.

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CHAPTER 6

DIMENSIONS AND AREAS

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CHAPTER 6 – DIMENSIONS AND AREAS

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

REFERENCE	PAGE	DATE	REFERENCE	PAGE	DATE
06	* 1	Sep. 28/18	06-00-00	* 1	Sep. 28/18
	* 2	Blank Page		* 2	Sep. 28/18
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CHAPTER 6 – DIMENSIONS AND AREAS

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SUBJECT	CHAPTER SECTION SUBJECT	PAGE
DIMENSIONS AND AREAS	**06-00**	
GENERAL INFORMATION	06-00-00	
Aircraft Dimensions and Ground Clearances		1
Aircraft Stations Diagram		3
Symmetry and Alignment Checks		5

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Dimensions and Areas – General Data

1. Aircraft Dimensions and Ground Clearances

With the aircraft supported on its landing gear, refer to Figure 1, the principal dimensions and ground clearances are as follows:

Note Dimensions to Ground Line are approximate only and vary depending on aircraft configuration and loading conditions.

(1) Overall Dimensions

Span	65 ft – 0 in.
Length	51 ft – 9 in.
Height (tires normal)	19 ft – 9 in.
Propeller ground clearance (static)	5 ft – 0 in.
Design gross weight	12,500 pounds

(2) Wings

Type	High
Chord	6 ft – 6 in.
Incidence	2 – 1/2°
Dihedral	3°
Aspect Ratio	10.0
Airfoil Section	High Lift Section

(3) Horizontal stabilizer and elevators

Span	20 ft – 8 in.
Chord	4 ft – 9 in.
Incidence	0°
Dihedral	0°
Aspect ratio	4.35
Airfoil Section	63_A213 Inverted and Modified

(4) Vertical stabilizer

Height	13 ft – 6 in.
Root Chord	10 ft – 0 in. (Basic)
Tip Chord	5 ft – 0.6 in.
Aspect ratio	1.84

(5) Fuselage

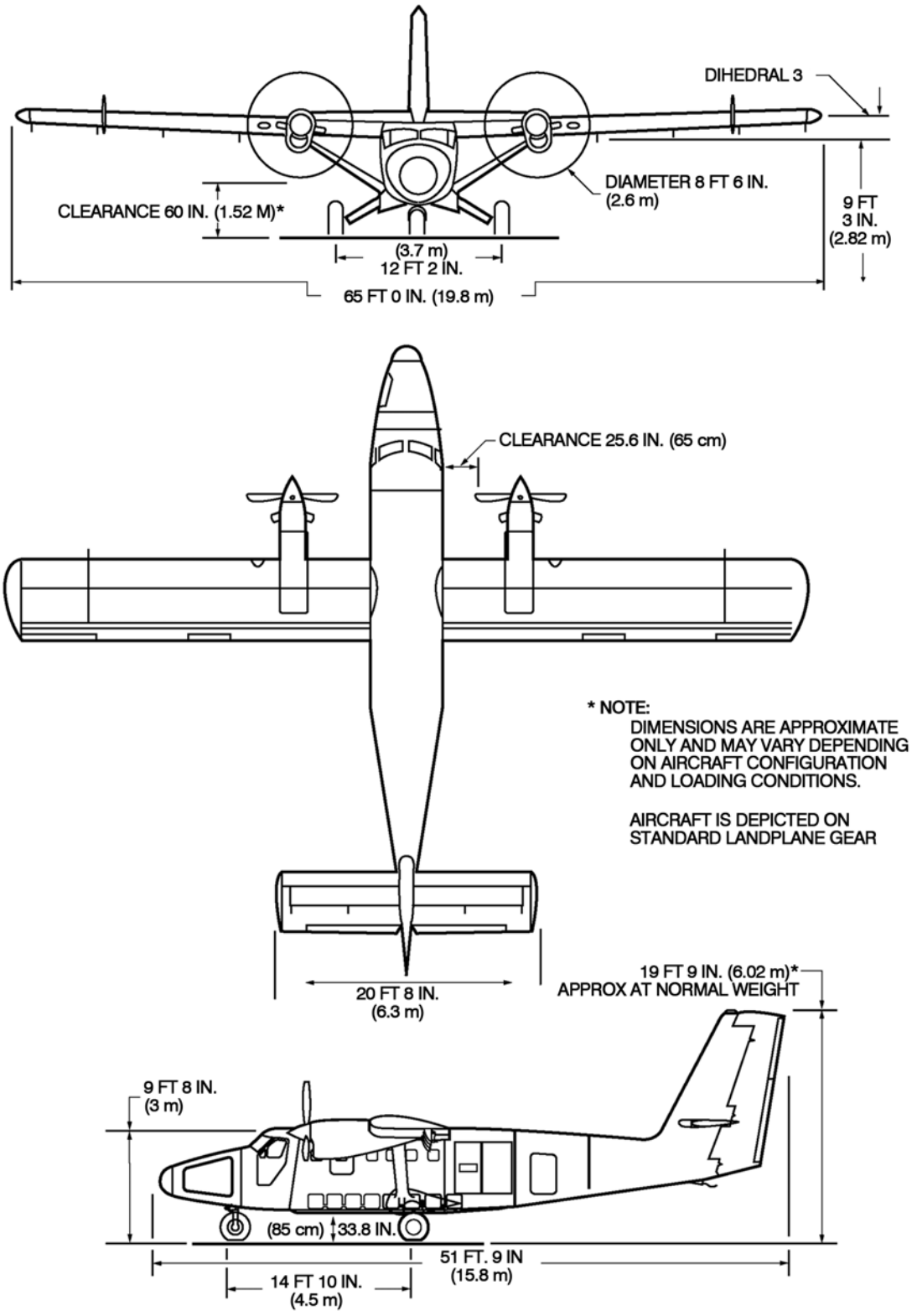
Width (maximum)	5 ft – 9.2 in.
Height (top of center fuselage)	9 ft – 8 in.

(6) Areas

Wing	420 sq ft
Horizontal stabilizer and elevators	98 sq ft

(7) Landing gear

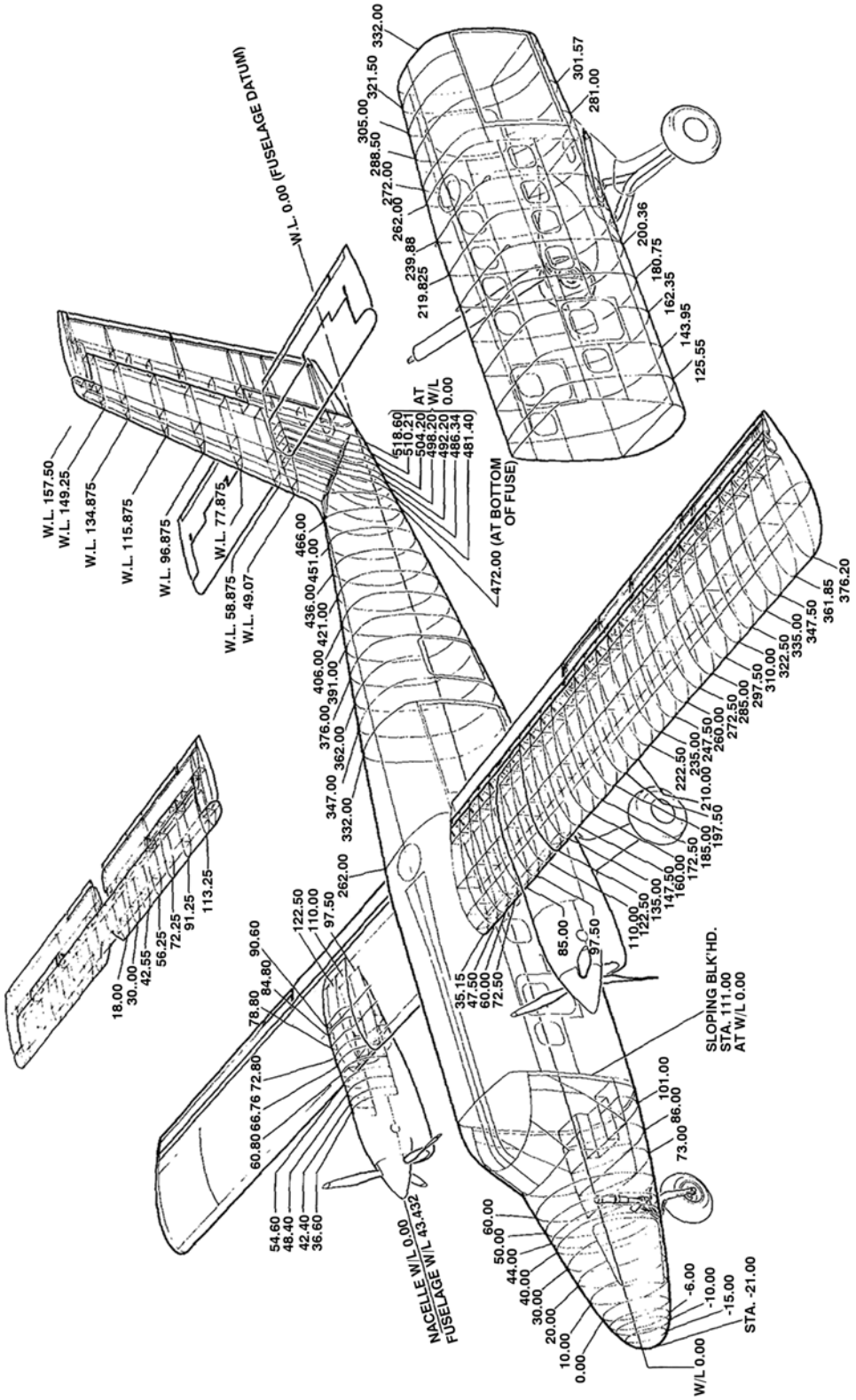
Main Wheels track	12 ft – 2 in.



Aircraft Dimensions
Figure 1

2. Aircraft Stations Diagram

In Figure 2, the stations diagram shows the location of the structural members of the aircraft in inches from three reference sources, each designated as station 0. Station 0 of the fuselage is 60.00 inches forward of the flight compartment front bulkhead, station 0 of the wing, horizontal stabilizer and elevator is the centerline of the aircraft; and station 0 of the vertical stabilizer and rudder is the aircraft datum line.



Aircraft Stations Diagram
Figure 2

3. Symmetry and Alignment Checks

Symmetry and alignment checks are required if the aircraft has suffered a heavy landing, sustained damage or any of the major airframe components (wings, horizontal tail, vertical tail, fuselage) have been replaced. Complete the following symmetry and alignment checks. See Figure 3 for definition of points A, B, C, D, E, F and G. A difference of ± 0.5 inches from the production symmetry and alignment values should be taken as a warning of possible damage. Inspection of the relevant parts should be carried out to determine the extent of the damage.

Note Measurement techniques, measurement equipment and temperature may affect measurement accuracy; consult with Viking Air Limited Engineering if the above tolerances are exceeded.

A. Verify alignment by measuring the following distances per Figure 3:

1) Point A to B (37' – 2.5" ± 1.0") (_____)

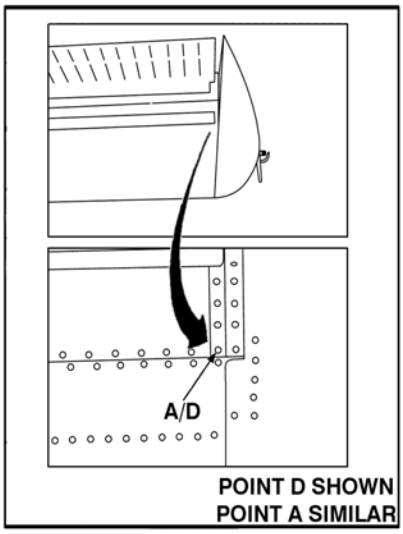
2) Point A to C (32' – 3.0" ± 1.0") (_____)

3) Point A to G (32' – 6.0" ± 1.0") (_____)

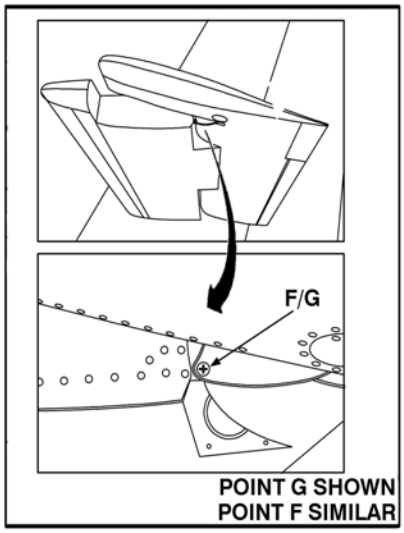
4) Point D to B (37' – 2.5" ± 1.0") (_____)

5) Point D to E (32' – 3.0" ± 1.0") (_____)

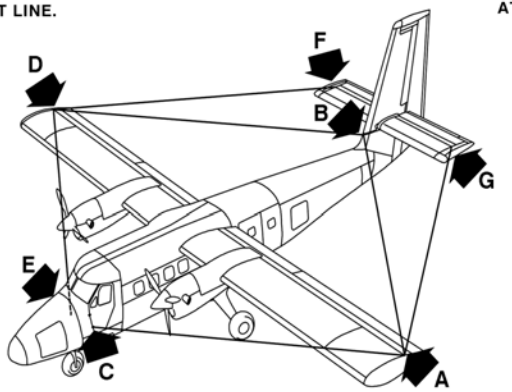
6) Point D to F (32' – 6.0" ± 1.0") (_____)

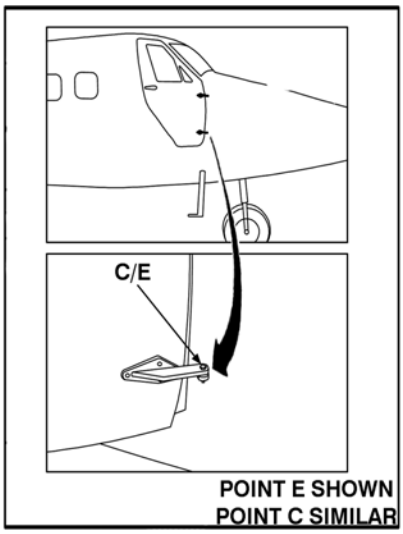


NOTE: VIEW LOOKING FORWARD, BOTTOM OF WING TIP, REAR SPAR RIVET LINES. POINT A AND D ARE THE FARTHEST OUTBOARD RIVETS ON THE REAR SPAR AFT RIVET LINE.

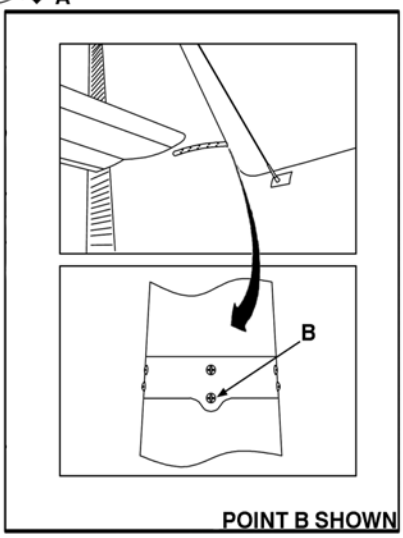


NOTE: VIEW LOOKING INBOARD ON HORIZONTAL TAIL (LEFT SIDE). POINT F AND G ARE THE OUTBOARD ELEVATOR HINGE LINE ATTACHMENT BOLTS.





NOTE: VIEW LOOKING AT RIGHT SIDE FORWARD COCKPIT DOOR. POINT C AND E (LEFT SIDE) ARE LOWER DOOR HINGE ATTACHMENT BOLTS.



NOTE: VIEW LOOKING AFT ON THE VERITCAL TAIL. POINT B IS THE BOTTOM SCREW ON THE VERTICAL TAIL/FUSELAGE JOINT FAIRING.

Symmetry and Alignment Checks
Figure 3

CHAPTER 7

LIFTING AND SHORING

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CHAPTER 7 – LIFTING AND SHORING

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

REFERENCE	PAGE	DATE	REFERENCE	PAGE	DATE
07	* 1	Sep. 28/18	07-00-00	* 1	Sep. 28/18
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CHAPTER 7 – LIFTING AND SHORING

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Hoisting the Aircraft		1
Hoisting the Fuselage		1
Hoisting a Wing		1
Hoisting the Vertical Stabilizer		1
Hoisting the Horizontal Stabilizer		1
Hoisting an Engine		1
Hoisting a Propeller		2
Jacking the Aircraft		2

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TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR07-2**

Insert in 07-00-00 before Page 1

LIFTING AND SHORING

This Temporary Revision Supersedes Temporary Revision 07-1

REASON FOR ISSUE:

To introduce instructions for hoisting the Wing, Vertical Stabilizer, Horizontal Stabilizer and Rudder.

ACTION:

In Chapter 7, Section 07-00-00, remove Temporary Revision 07-1 and replace with Temporary Revision 07-2.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

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Lifting and Shoring – General Data

1. Information and Warnings

   A. Refer to PSM 1-6-2T for storage, maintenance and inspection requirements.

   B. The slings and lifting devices are designed in accordance with ASME B30.20 “Safety Standard for Below-the Hook Lifting Devices”. It is the responsibility of the owner/user to inspect, maintain and use the lifting device in accordance with ASME B30.20 “Safety Standard for Below-the-Hook Lifting Devices”, PSM 1-6-2T and PSM 1-64-2. It is the responsibility of the owner/user to ensure compliance with all standards if the sling or lifting device is used in conjunction with other lifting systems, such as, but not limited to ASME B30.16 “Safety Standards for Overhead Hoists (Underhung)”.

   C. Modifications to slings and lifting devices are not permitted unless approved by Viking.

   D. Viking slings and lifting devices are recommended for indoor use, or outdoor use when the following conditions are met:

      - Adequate sunlight or supplementary light is present
      - Winds do not exceed 3 kts (5.6 km/h)
      - Precipitation is not present

   E. Viking slings and lifting devices must be used to lift loads vertically only.

   F. Slings and lifting devices are not recommended for transporting a suspended load.

2. Approximate Weight of Components

Table 1 Approximate Weight of Components

ITEM	APPROXIMATE WEIGHT (POUNDS)
Engine	324
Propeller	134
Wing	910
Wing Strut	90
Aileron	79
Flaps	133
Vertical Stabilizer	53
Rudder	67
Horizontal Stabilizer	94
Elevator	70

3. Hoisting the Aircraft

CAUTION **SLING AND LIFTING DEVICE OPERATORS ARE RESPONSIBLE FOR COMPLIANCE WITH ALL APPLICABLE INDUSTRY STANDARDS, AND WITH ALL APPLICABLE LOCAL REGULATIONS.**

   A. Prior to carrying out a lift, the sling or lifting device must be inspected in accordance with inspection requirements given in PSM 1-6-2T, Chapter 7.

CAUTION **A REGULAR SCHEDULE OF INSPECTION OF SLINGS AND LIFTING DEVICES IN ACCORDANCE WITH THE REQUIREMENTS OF ASME B30.20 SHALL BE ESTABLISHED AND RECORDS MAINTAINED.**

B. The entire aircraft may be hoisted by using an approved aircraft sling or lifting device (refer to PSM 1-6-2T).

C. Read and follow all information in Para 1.

WARNING **THE SLINGS AND LIFTING DEVICES ARE NOT DESIGNED FOR, AND MUST NOT BE USED FOR LIFTING, SUPPORTING, OR TRANSPORTATION OF PERSONNEL.**

D. Remove any cargo, loose items, passengers and personnel from aircraft prior to hoisting. The aircraft may be lifted with interior fixtures.

E. It is recommended that the aircraft be defueled.

F. Confirm aircraft weight is within limitation of the sling or lifting device.

WARNING **THE SLINGS AND LIFTING DEVICES MUST ONLY BE USED TO LIFT THE DHC-6 COMPONENT THAT THEY ARE INTENDED FOR AND MUST NOT BE USED TO LIFT ANY PARTS, ASSEMBLIES OR AIRCRAFT OTHER THAN THOSE DESCRIBED WITHIN THIS MANUAL. IF AN INCORRECT SLING OR LIFTING DEVICE IS USED, FAILURE OF THE SLING OR LIFTING DEVICE MAY OCCUR AND INJURE PERSONNEL OR CAUSE DAMAGE TO AIRCRAFT OR EQUIPMENT.**

G. Attach front sling attachment brackets in accordance with Figure 1, Detail B. Insert the pin through the bracket and the forward wing attachment bolt. Confirm pin is locked in place.

Note The bushings are not required when the aircraft is lifted with wings attached.

H. Attach rear sling attachment brackets in accordance with Figure 1, Detail A. Remove the shackle, to allow the shackle to open in a scissor fashion to fit over the aft wing attachment bolt. Close the bracket and re-insert the bracket to lock the bracket from opening.

Note The bushings are not required when the aircraft is lifted with wings attached.

I. Attach master link to a lifting crane with the applicable lifting capacity.

J. Connect front and rear sling attachment brackets, installed in Para 3.G and Para 3.H, to lifting device lower chains.

K. Confirm attachments are secure and personnel are clear.

CAUTION **SLING AND LIFTING DEVICE OPERATORS MUST BE TRAINED IN OPERATION OF ANY EQUIPMENT HE OR SHE IS OPERATING.**

L. Carry out lift in accordance with all applicable local regulations.

4. Hoisting the Fuselage

A. Read and follow all information in Para 1.

B. The fuselage cannot be lifted without the wings if the empennage is installed (refer to Figure 1).

C. Remove any cargo, loose items, passengers and personnel from aircraft prior to hoisting. The fuselage may be lifted with interior fixtures.

D. Carry out lift as described in Para 3.F through Para 3.L.

5. Hoisting a Wing

Note Sling and lifting device operators are responsible for compliance with all applicable industry standards, and with all applicable local regulations.

Note Sling and lifting device operators must be trained in operation of any equipment they are operating.

A. Prior to carrying out a lift, the sling or lifting device must be inspected in accordance with inspection requirements given in PSM 1-6-2T, Chapter 7.

CAUTION A REGULAR SCHEDULE OF INSPECTION OF SLINGS AND LIFTING DEVICES IN ACCORDANCE WITH THE REQUIREMENTS OF ASME B30.20 SHALL BE ESTABLISHED AND RECORDS MAINTAINED.

B. The rudder may be hoisted by using an approved wing sling or lifting device (refer to PSM 1-6-2T).

C. Read and follow all information in Para 1.

WARNING THE SLINGS AND LIFTING DEVICES ARE NOT DESIGNED FOR LIFTING, SUPPORTING OR TRANSPORTATION OF PERSONNEL, AND MUST NOT BE USED FOR THOSE PURPOSES.

D. Confirm that the total weight of the vertical stabilizer (refer to Table 1 — Approximate Weight of Components), the lifting device, any rigging hardware, and any other additional weight being lifted is within limitations of the sling or lifting device, and the crane being used to carry out the lift.

WARNING THE SLINGS AND LIFTING DEVICES MUST ONLY BE USED TO LIFT THE DHC-6 COMPONENT THAT THEY ARE INTENDED FOR. THEY MUST NOT BE USED TO LIFT ANY PARTS, ASSEMBLIES, OR AIRCRAFT OTHER THAN THOSE DESCRIBED WITHIN THIS MANUAL.

E. Remove Wing

Refer toFigure 1 and 57-00-00, Wings – General Data for details.

1) Disconnect the wing sling cradles from the spreader.

2) Attach the spreader lifting eye to the crane and suspend over the wing.

3) Place one wing cradle below the wing at approximately wing station 137.

4) Place the remaining wing below the wing cradle at approximately wing station 209.

5) Attach the cables to the cradles using the quick-release ball-lock pins. Select the appropriate attachment, depending on whether the wing is being lifted with or without flaps and ailerons.

6) Lift crane until slight tension is applied to all cables.

7) Confirm all attachments are secure.

8) Lift crane until full weight of wing is supported by the lifting sling.

9) Remove wing. Refer to 57-00-00, Wings – General Data.

10) Check the balance of the wing. If the wing is not balanced, a 30-40 lbs ballast weight may be placed on top of the wing near the wing tip. Balance can be adjusted by moving the ballast weight inboard or outboard. Ensure the weight does not damage the wing skin.

11) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations.

12) Lower wing on to a suitable support surface.

13) When lift is complete and wing is securely supported, remove the sling from the wing.

F. Install Wing

Refer to Figure 1 and 57-00-00, Wings – General Data for details.

1) Disconnect the wing sling cradles from the spreader.

2) Attach the spreader lifting eye to the crane and suspend over the wing.

3) Place one wing cradle below the wing at approximately wing station 137.

4) Place the remaining wing below the wing cradle at approximately wing station 209.

5) Attach the cables to the cradles using the quick-release ball-lock pins. Select the appropriate attachment, depending on whether the wing is being lifted with or without flaps and ailerons.

6) Lift crane until slight tension is applied to all cables.

7) Confirm all attachments are secure.

8) Lift crane until full weight of wing is supported by the lifting sling.

9) Check the balance of the wing. If the wing is not balanced, a 30-40 lbs ballast weight may be placed on top of the wing near the wing tip. Balance can be adjusted by moving the ballast weight inboard or outboard. Ensure the weight does not damage the wing skin.

10) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations.

11) Move wing to the aircraft and position in place.

12) Install wing. Refer to 57-00-00, Wings – General Data.

13) When lift is complete and wing is secured to the aircraft, remove the sling from the wing.

6. Hoisting the Vertical Stabilizer

Note Sling and lifting device operators are responsible for compliance with all applicable industry standards, and with all applicable local regulations.

Note Sling and lifting device operators must be trained in operation of any equipment they are operating.

A. Prior to carrying out a lift, the sling or lifting device must be inspected in accordance with inspection requirements given in PSM 1-6-2T, Chapter 7.

CAUTION **A REGULAR SCHEDULE OF INSPECTION OF SLINGS AND LIFTING DEVICES IN ACCORDANCE WITH THE REQUIREMENTS OF ASME B30.20 SHALL BE ESTABLISHED AND RECORDS MAINTAINED.**

B. The rudder may be hoisted by using an approved wing sling or lifting device (refer to PSM 1-6-2T).

C. Read and follow all information in Para 1.

WARNING **THE SLINGS AND LIFTING DEVICES ARE NOT DESIGNED FOR LIFTING, SUPPORTING OR TRANSPORTATION OF PERSONNEL, AND MUST NOT BE USED FOR THOSE PURPOSES.**

D. Confirm that the total weight of the vertical stabilizer (refer to Table 1 — Approximate Weight of Components), the lifting device, any rigging hardware, and any other additional weight being lifted is within limitations of the sling or lifting device, and the crane being used to carry out the lift.

WARNING **THE SLINGS AND LIFTING DEVICES MUST ONLY BE USED TO LIFT THE DHC-6 COMPONENT THAT THEY ARE INTENDED FOR. THEY MUST NOT BE USED TO LIFT ANY PARTS, ASSEMBLIES, OR AIRCRAFT OTHER THAN THOSE DESCRIBED WITHIN THIS MANUAL.**

E. Remove Vertical Stabilizer

Refer to Figure 1 and 55-30-00, Vertical Stabilizer – General Data for details.

1) Remove and retain AN525-10R10 screws and AN315-3 nuts supplied in thimble of the sling.

2) Attach the cables to the vertical stabilizer using the supplied AN525-10R10 screws. Do not use the AN315-3 nuts while lifting.

3) Confirm the sling can rotate freely about the attachments.

4) Suspend crane above the vertical stabilizer and attach the sling to the crane.

5) Lift crane until slight tension is applied to all cables.

6) Confirm all shackles, ball-lock pins, and other attachments are secure.

7) Remove vertical stabilizer. Refer to 55-30-00, Vertical Stabilizer – General Data.

8) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations.

9) Lower vertical stabilizer on to a suitable surface.

10) When lift is complete and vertical stabilizer is securely supported, remove the sling from the vertical stabilizer.

11) Re-install the AN525-10R10 screws into the thimbles of the sling and retain screws with AN315-3 nuts.

F. Install Vertical Stabilizer

Refer to Figure 1 and 55-30-00, Vertical Stabilizer – General Data for details.

1) Remove and retain AN525-10R10 screws and AN315-3 nuts supplied in thimble of the sling.

2) Attach the cables to the vertical stabilizer using the supplied AN525-10R10 screws. Do not use the AN315-3 nuts while lifting.

3) Confirm the sling can rotate freely about the attachments. (Refer to Figure 1).

4) Suspend crane above the vertical stabilizer and attach the sling to the crane.

5) Lift crane until slight tension is applied to all cables.

6) Confirm all shackles, ball-lock pins, and other attachments are secure.

7) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations, allowing vertical stabilizer to rotate on sling attachments.

8) Move vertical stabilizer to the aircraft and position in place.

9) Install vertical stabilizer. Refer to 55-30-00, Vertical Stabilizer – General Data.

10) When lift is complete and vertical stabilizer is secured to the aircraft, remove the sling from the vertical stabilizer.

11) Re-install the AN525-10R10 screws into the thimbles of the sling and retain screws with AN315-3 nuts.

7. Hoisting the Horizontal Stabilizer

Note Sling and lifting device operators are responsible for compliance with all applicable industry standards, and with all applicable local regulations.

Note Sling and lifting device operators must be trained in operation of any equipment they are operating.

A. Prior to carrying out a lift, the sling or lifting device must be inspected in accordance with inspection requirements given in PSM 1-6-2T, Chapter 7.

CAUTION **A REGULAR SCHEDULE OF INSPECTION OF SLINGS AND LIFTING DEVICES IN ACCORDANCE WITH THE REQUIREMENTS OF ASME B30.20 SHALL BE ESTABLISHED AND RECORDS MAINTAINED.**

B. The rudder may be hoisted by using an approved wing sling or lifting device (refer to PSM 1-6-2T).

C. Read and follow all information in Para 1.

WARNING **THE SLINGS AND LIFTING DEVICES ARE NOT DESIGNED FOR LIFTING, SUPPORTING OR TRANSPORTATION OF PERSONNEL, AND MUST NOT BE USED FOR THOSE PURPOSES.**

D. Confirm that the total weight of the vertical stabilizer (refer to Table 1 — Approximate Weight of Components), the lifting device, any rigging hardware, and any other additional weight being lifted is within limitations of the sling or lifting device, and the crane being used to carry out the lift.

WARNING **THE SLINGS AND LIFTING DEVICES MUST ONLY BE USED TO LIFT THE DHC-6 COMPONENT THAT THEY ARE INTENDED FOR. THEY MUST NOT BE USED TO LIFT ANY PARTS, ASSEMBLIES, OR AIRCRAFT OTHER THAN THOSE DESCRIBED WITHIN THIS MANUAL.**

E. Remove Horizontal Stabilizer

Refer to Figure 1 and 55-10-00, Horizontal Stabilizer – General Data for details.

1) Locate each of the forward lugs of the sling on the forward upper surface of the horizontal stabilizer and install the supplied bolts as illustrated in Figure 1.

2) Select the appropriate cable for attachment to the aft lifting point of the horizontal stabilizer. Use the short cable to lift the horizontal stabilizer with the elevator attached, or use the long cable to lift the horizontal stabilizer without the elevator attached.

3) Place the shackle of the aft cable over the lug located above the aft spar of the horizontal stabilizer and install the quick-release ball-lock pin attached to the sling.

4) Suspend crane above the horizontal stabilizer and attache the lifting ring to the crane.

5) Lift crane until slight tension is applied to all cables.

6) Confirm all attachments are secure.

7) Remove horizontal stabilizer. Refer to 55-10-00, Horizontal Stabilizer – General Data.

8) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations.

9) Lower horizontal stabilizer on to a suitable surface.

10) When lift is complete and horizontal stabilizer is securely supported, remove the sling from the horizontal stabilizer.

F. Install Horizontal Stabilizer

Refer to Figure 1 and 55-10-00, Horizontal Stabilizer – General Data for details.

1) Locate each of the forward lugs of the sling on the forward upper surface of the horizontal stabilizer and install the supplied bolts as illustrated in Figure 1.

2) Select the appropriate cable for attachment to the aft lifting point of the horizontal stabilizer. Use the short cable to lift the horizontal stabilizer with the elevator attached, or use the long cable to lift the horizontal stabilizer without the elevator attached.

3) Place the shackle of the aft cable over the lug located above the aft spar of the horizontal stabilizer and install the quick-release ball-lock pin attached to the sling.

4) Suspend crane above the horizontal stabilizer and attache the lifting ring to the crane.

5) Lift crane until slight tension is applied to all cables.

6) Confirm all attachments are secure.

7) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations.

8) Move horizontal stabilizer to the aircraft and position in place.

9) Install horizontal stabilizer. Refer to 55-10-00, Horizontal Stabilizer – General Data.

10) When lift is complete and horizontal stabilizer is secured to the aircraft, remove the sling from the horizontal stabilizer.

8. Hoisting the Rudder

Note Sling and lifting device operators are responsible for compliance with all applicable industry standards, and with all applicable local regulations.

Note Sling and lifting device operators must be trained in operation of any equipment they are operating.

A. Prior to carrying out a lift, the sling or lifting device must be inspected in accordance with inspection requirements given in PSM 1-6-2T, Chapter 7.

CAUTION **A REGULAR SCHEDULE OF INSPECTION OF SLINGS AND LIFTING DEVICES IN ACCORDANCE WITH THE REQUIREMENTS OF ASME B30.20 SHALL BE ESTABLISHED AND RECORDS MAINTAINED.**

B. The rudder may be hoisted by using an approved wing sling or lifting device (refer to PSM 1-6-2T).

C. Read and follow all information in Para 1.

WARNING **THE SLINGS AND LIFTING DEVICES ARE NOT DESIGNED FOR LIFTING, SUPPORTING OR TRANSPORTATION OF PERSONNEL, AND MUST NOT BE USED FOR THOSE PURPOSES.**

D. Confirm that the total weight of the rudder (refer to Table 1 — Approximate Weight of Components), the lifting device, any rigging hardware, and any other additional weight being lifted is within limitations of the sling or lifting device, and the crane being used to carry out the lift.

WARNING **THE SLINGS AND LIFTING DEVICES MUST ONLY BE USED TO LIFT THE DHC-6 COMPONENT THAT THEY ARE INTENDED FOR. THEY MUST NOT BE USED TO LIFT ANY PARTS, ASSEMBLIES, OR AIRCRAFT OTHER THAN THOSE DESCRIBED WITHIN THIS MANUAL.**

E. Remove Rudder

Refer to Figure 1 and 27-20-11, Rudder – Maintenance Practices for details.

1) Attach the links of sling to the top of the rudder using the supplied hardware. Ensure that the Teflon rubbing plate faces the rudder skin.

2) Confirm the sling can rotate freely about the attachments.

3) Suspend crane above the rudder and attach the sling to the crane.

4) Lift crane until slight tension is applied to all cables.

5) Confirm all attachments are secure.

6) Remove rudder. (Refer to 27-20-11, Rudder – Maintenance Practices).

7) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations.

8) Lower rudder on to a suitable surface and rotate manually, as required, to prevent damage, ensuring personnel do not position themselves under the suspended load.

9) When lift is complete and rudder is securely supported, remove the sling from the rudder.

F. Install Rudder

Refer to Figure 1 and 27-20-11, Rudder – Maintenance Practices for details.

1) Attach the links of sling to the top of the rudder using the supplied hardware. Ensure that the Teflon rubbing plate faces the rudder skin.

2) Confirm the sling can rotate freely about the attachments.

3) Suspend crane above the rudder and attach the sling to the crane.

4) Lift crane until slight tension is applied to all cables.

5) Confirm all attachments are secure.

6) Confirm personnel are clear and carry out lift in accordance with all applicable local regulations, allowing rudder to rotate on sling attachments.

7) Move rudder to the aircraft and position in place.

8) Install rudder. (Refer to 27-20-11, Rudder – Maintenance Practices).

9) When lift is complete and rudder is secured to the aircraft, remove the sling from the rudder.

9. Hoisting an Engine

The engine may be hoisted using the engine sling (SD12541) attached to the lifting points of the engine. If desired, the engine may be removed or installed with propeller attached (refer to Figure 2).

10. Hoisting a Propeller

A. From a vertical position:

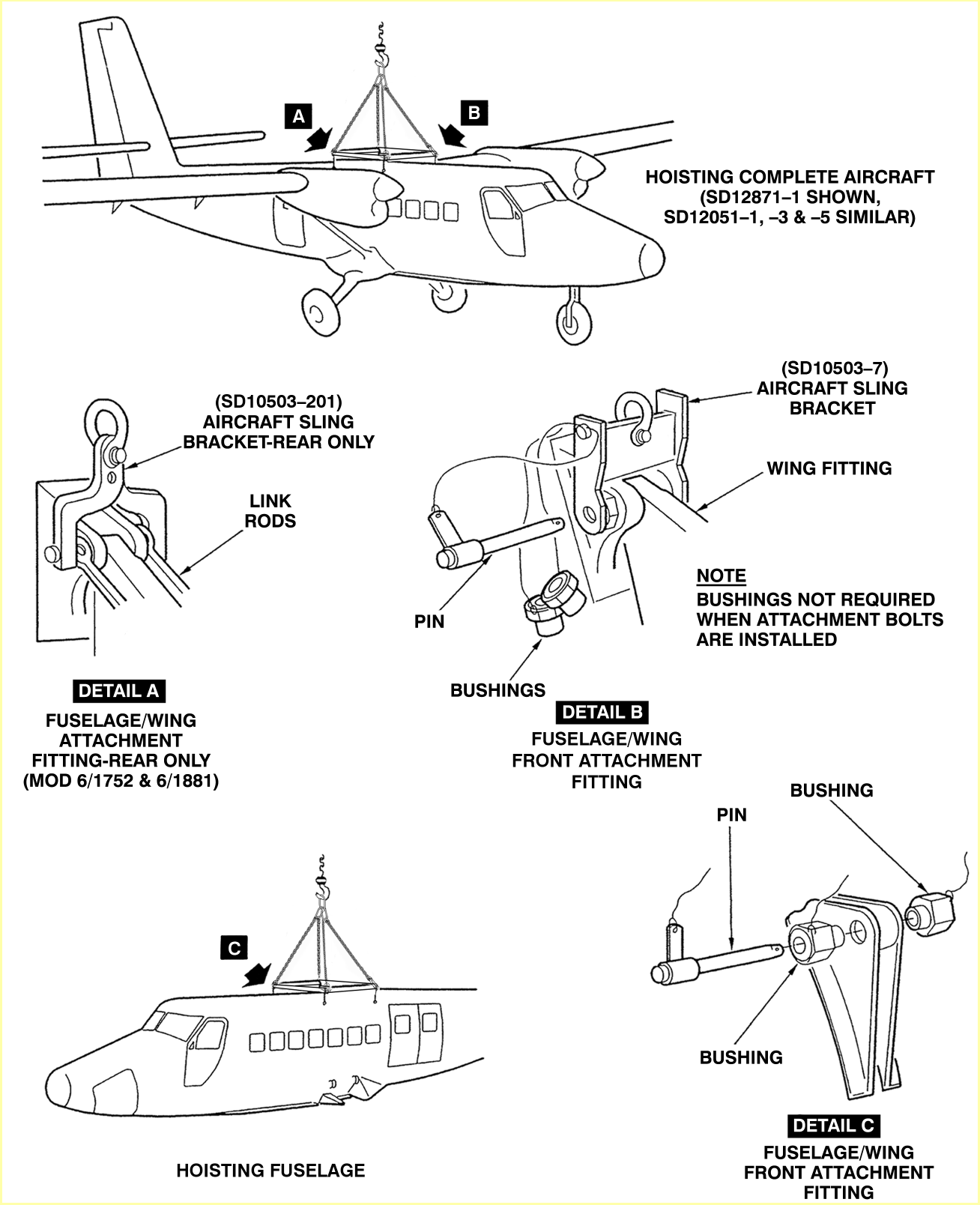
Position the propeller with one blade pointing vertically down, loop sling (SD5534) around roots of upper blades and hoist using a crane of sufficient capacity (refer to Figure 2).

B. From a horizontal position:

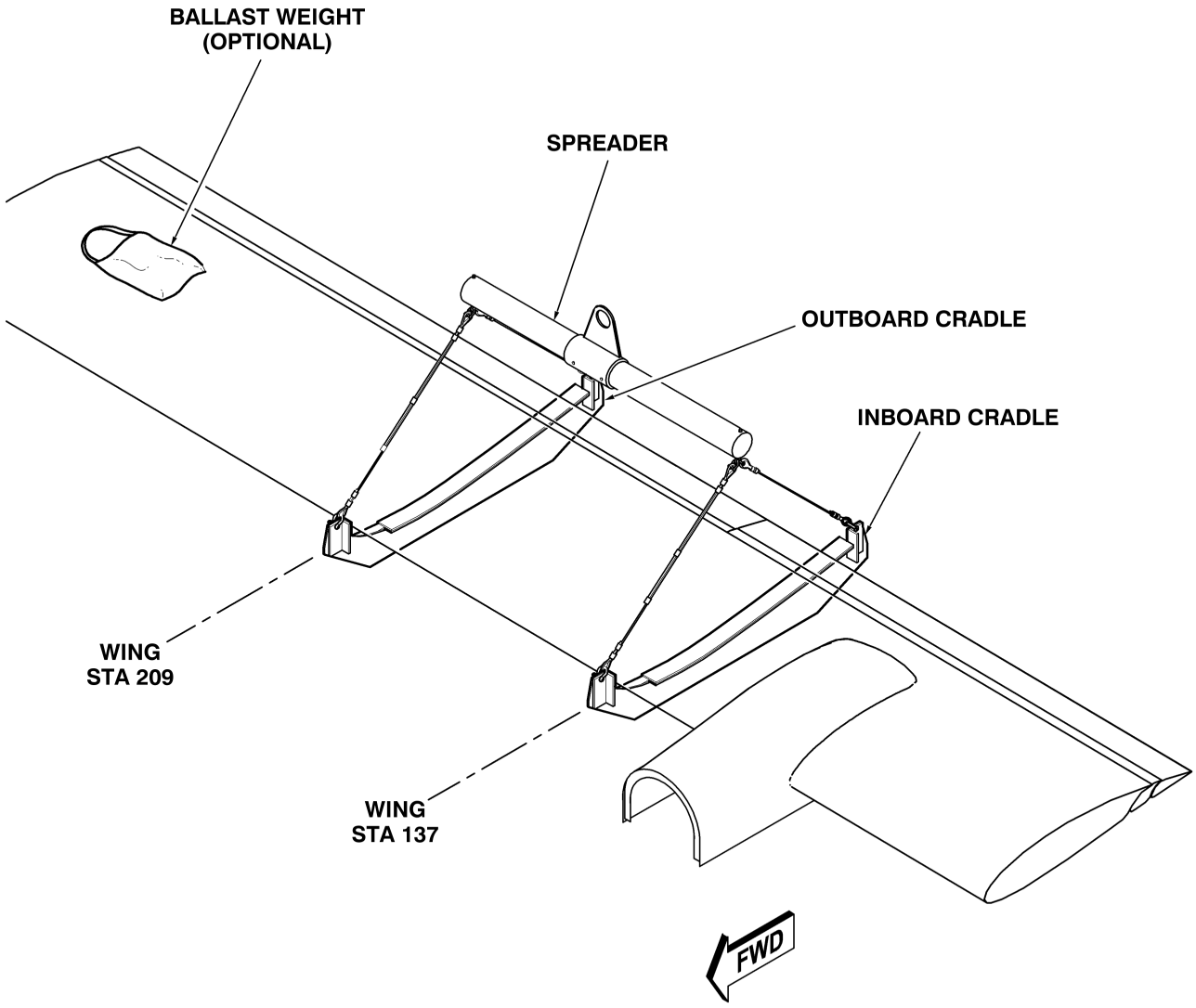
Loop sling (SD5534) around roots of two blades, support unslung blade to prevent damage to tip, and hoist using a crane of sufficient capacity.

11. Jacking the Aircraft

Five jacking points are provided on the aircraft and are located on the underside of the fuselage and main gear. Each jacking pad is coded with a letter, the related insert shows the jacking application. Tail lifting or slinging is accomplished by a bar inserted in the rear fuselage lifting tube and supported by two jacks, one at each end (refer to Figure 3).

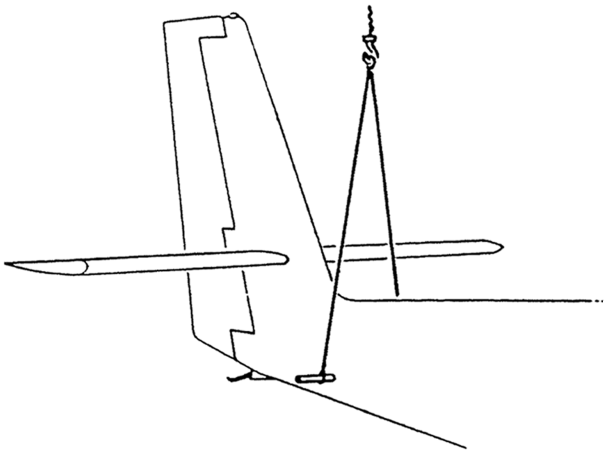


Aircraft Hoisting
Figure 1 (Sheet 1 of 4)

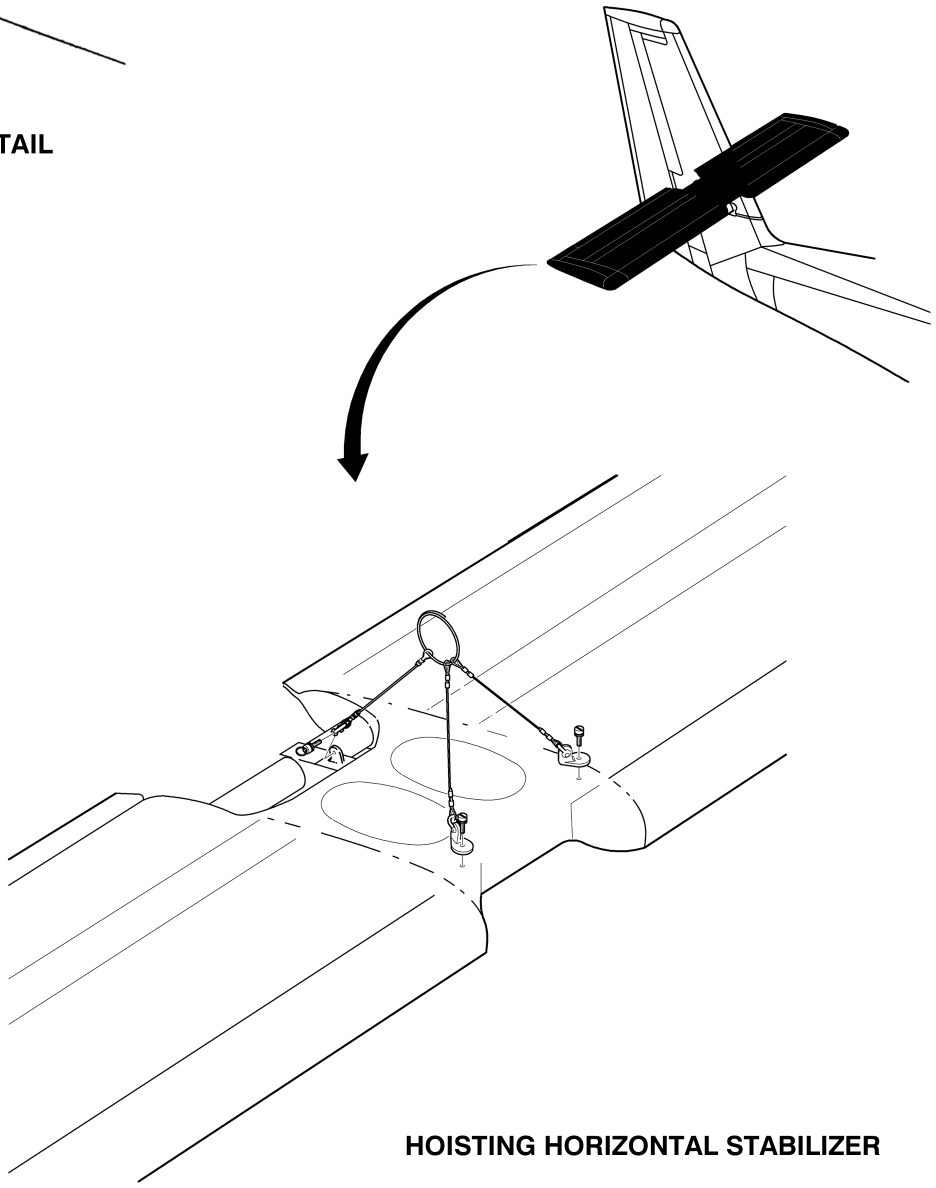


HOISTING WING

Aircraft Hoisting
Figure 1 (Sheet 2 of 4)

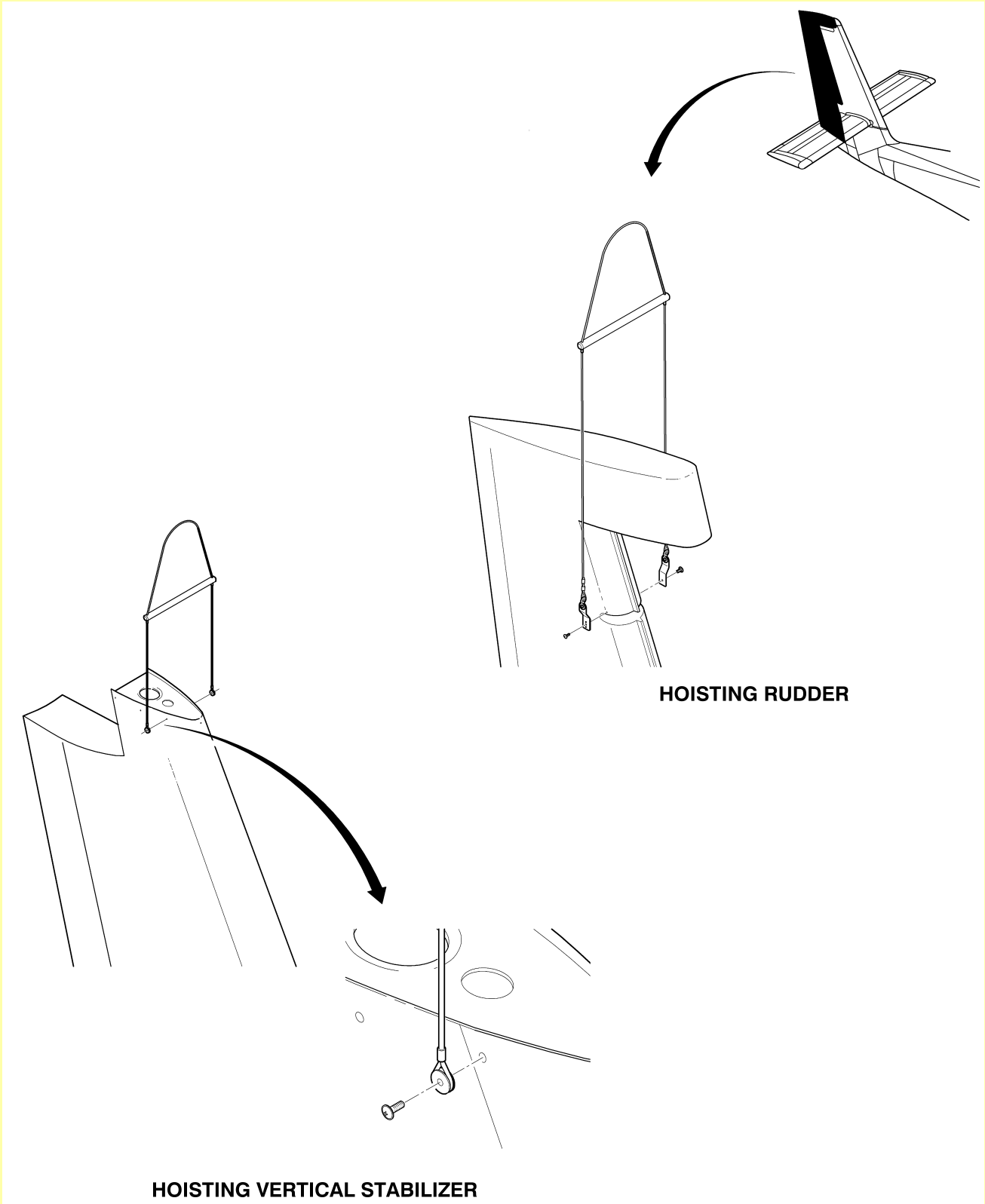


HOISTING TAIL



HOISTING HORIZONTAL STABILIZER

Aircraft Hoisting
Figure 1 (Sheet 3 of 4)



HOISTING RUDDER

HOISTING VERTICAL STABILIZER

Aircraft Hoisting
Figure 1 (Sheet 4 of 4)

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NOTE: Please see the TEMPORARY REVISION that revises this page.

Lifting and Shoring – General Data

1. Approximate Weight of Components

Table 1 Approximate Weight of Components

ITEM	APPROXIMATE WEIGHT (POUNDS)
Engine	324
Propeller	134
Wing	910
Wing Strut	90
Aileron	79
Flaps	133
Vertical Stabilizer	53
Rudder	67
Horizontal Stabilizer	94
Elevator	70

2. Hoisting the Aircraft

The entire Aircraft (Mod 6/1752 and 6/1881) may be hoisted by an aircraft hoisting sling (SD12501–5) attached to the special pins which are inserted through four hollow wing-to-fuselage bolts. The hardware securing the two rear wing-to-fuselage attachment differs from the forward attachment points. The sling is designed to enable the aircraft to be hoisted complete with furnishings and full fuel load (refer to Figure 1).

3. Hoisting the Fuselage

The fuselage, with empennage removed, is hoisted using the same sling and attachment points as for the aircraft hoisting. The aircraft sling cannot be used if empennage is installed to fuselage (refer to Figure 1).

4. Hoisting a Wing

A wing may be hoisted by means of the wing sling (SD12505), two lifting cradles and a crane with a sufficient lifting capacity. The cradles should be placed about ten feet apart and positioned slightly toward the wing root from center (refer to Figure 1).

5. Hoisting the Vertical Stabilizer

The vertical stabilizer is hoisted by means of a vertical stabilizer hoisting sling (SD12507), which is secured to the top rib of the vertical stabilizer, after first removing the stabilizer cap (refer to Figure 1).

6. Hoisting the Horizontal Stabilizer

The horizontal stabilizer is hoisted by means of a horizontal stabilizer and elevator hoisting sling (SD 12508), which is secured at two forward points and one rear point about the center of the stabilizer (refer to Figure 1).

7. Hoisting an Engine

The engine may be hoisted using the engine sling (SD12541) attached to the lifting points of the engine. If desired, the engine may be removed or installed with propeller attached (refer to Figure 2).

NOTE: Please see the TEMPORARY REVISION that revises this page.

8. Hoisting a Propeller

   A. From a vertical position:

      Position the propeller with one blade pointing vertically down, loop sling (SD5534) around roots of upper blades and hoist using a crane of sufficient capacity (refer to Figure 2).

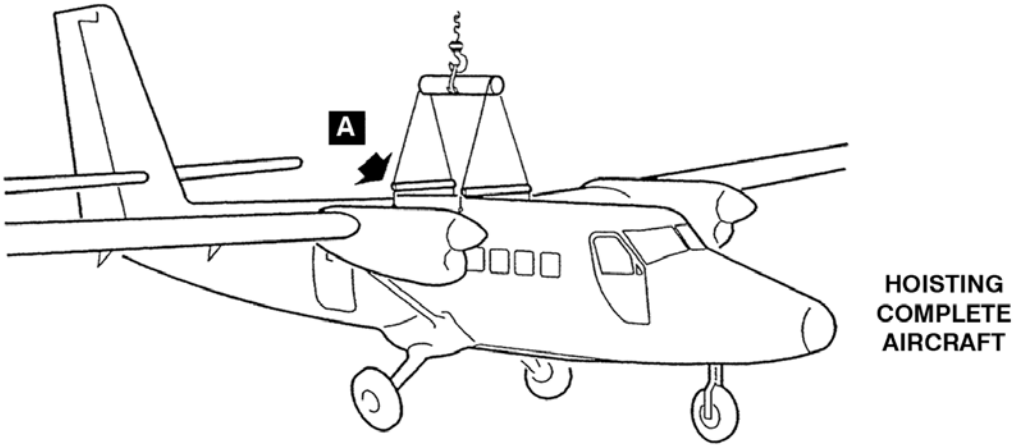
   B. From a horizontal position:

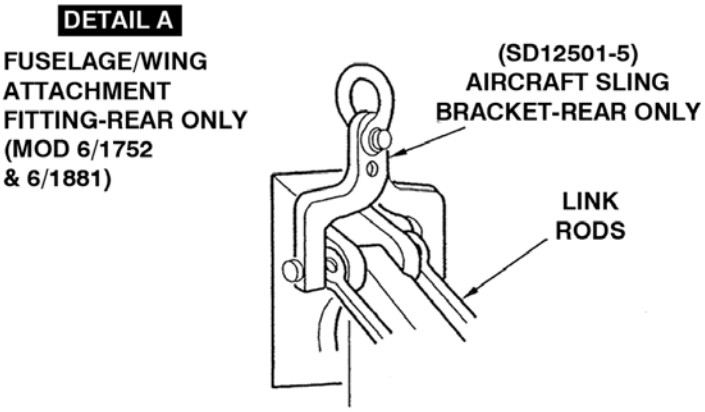
      Loop sling (SD5534) around roots of two blades, support unslung blade to prevent damage to tip, and hoist using a crane of sufficient capacity.

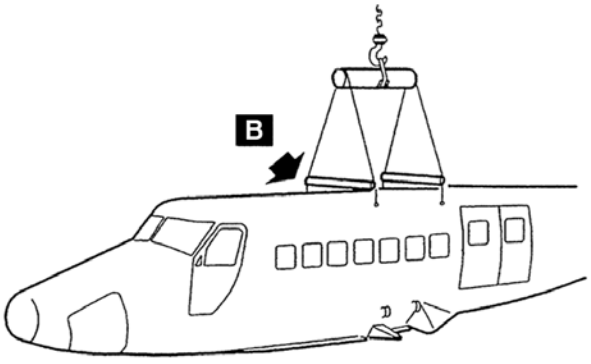
9. Jacking the Aircraft

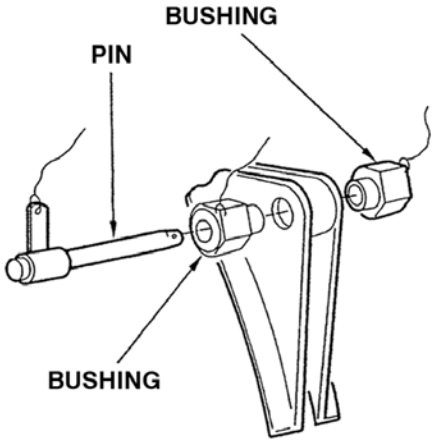
   Five jacking points are provided on the aircraft and are located on the underside of the fuselage and main gear. Each jacking pad is coded with a letter, the related insert shows the jacking application. Tail lifting or slinging is accomplished by a bar inserted in the rear fuselage lifting tube and supported by two jacks, one at each end (refer to Figure 3).

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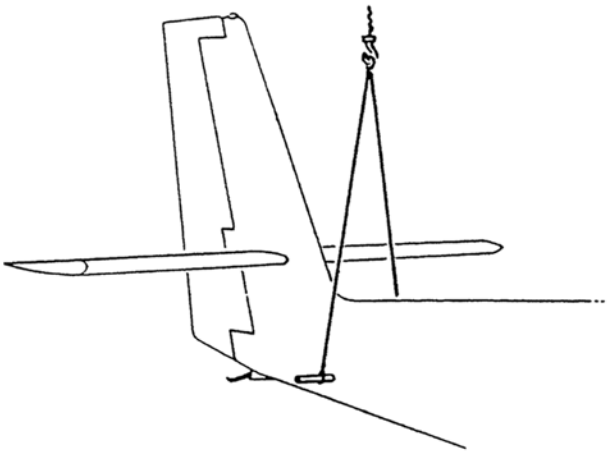




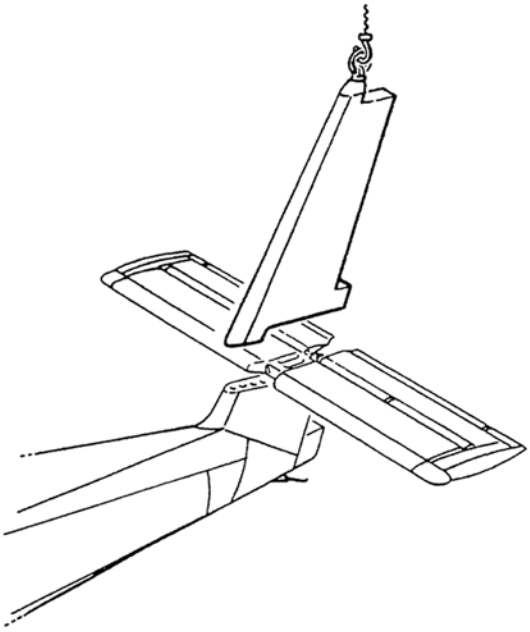




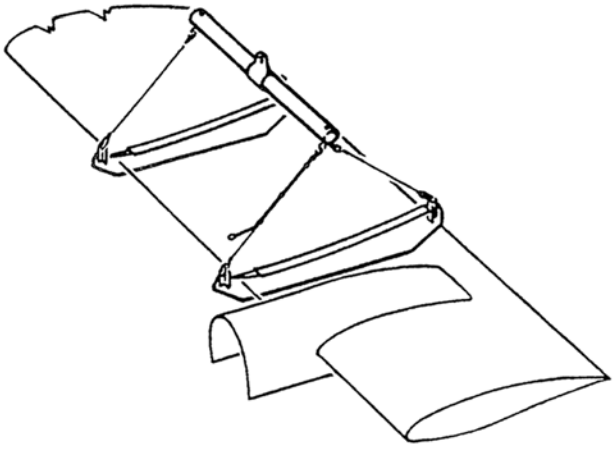
Hoisting Aircraft (Complete) and Fuselage
Figure 1 (Sheet 1 of 2)



HOISTING TAIL

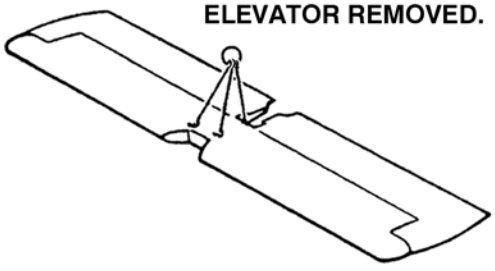


HOISTING VERTICAL STABILIZER



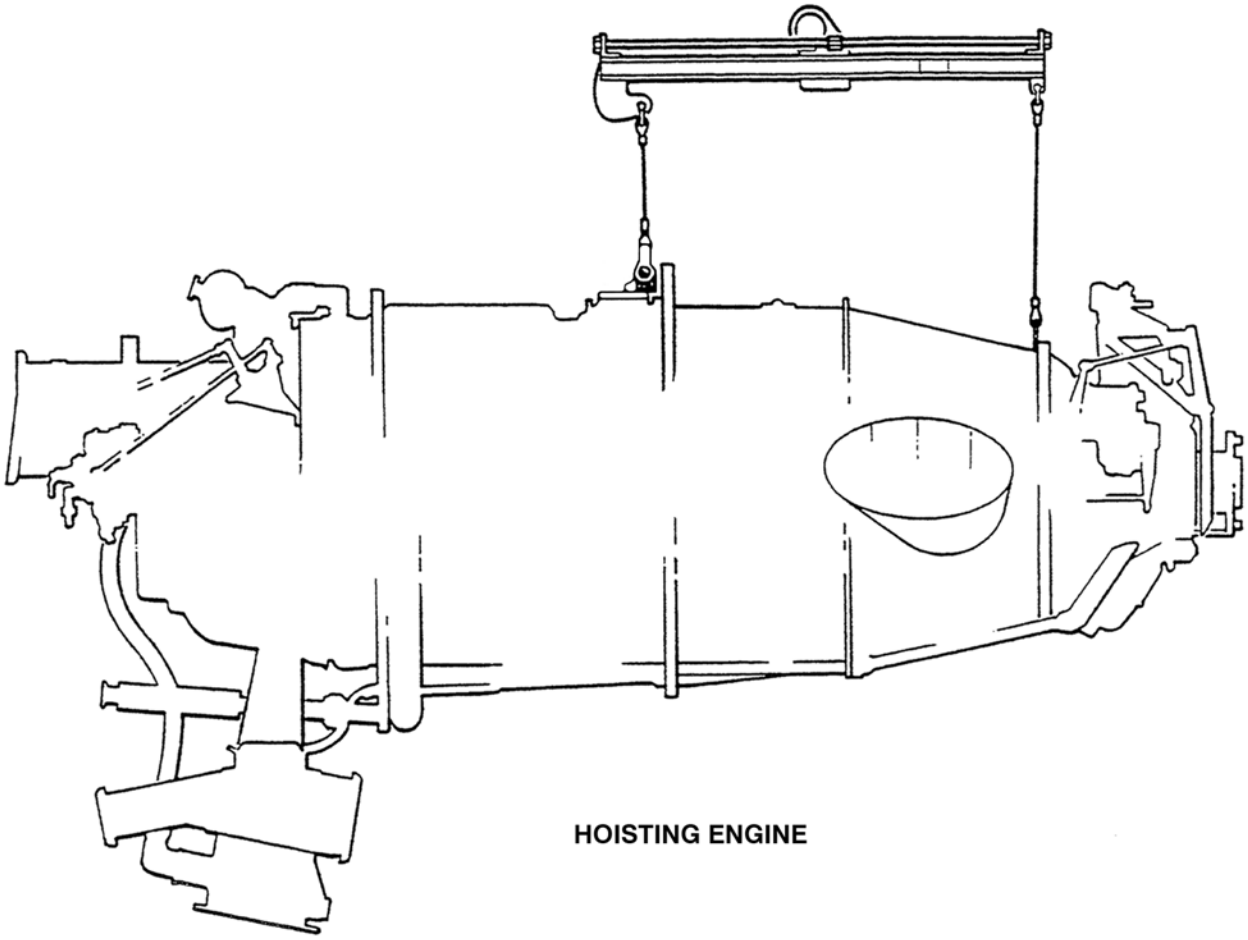
HOISTING WING

NOTE:
USE SHORT REAR CABLE TO HOIST STABILIZER WITH ELEVATOR ATTACHED. LONG CABLE WHEN ELEVATOR REMOVED.

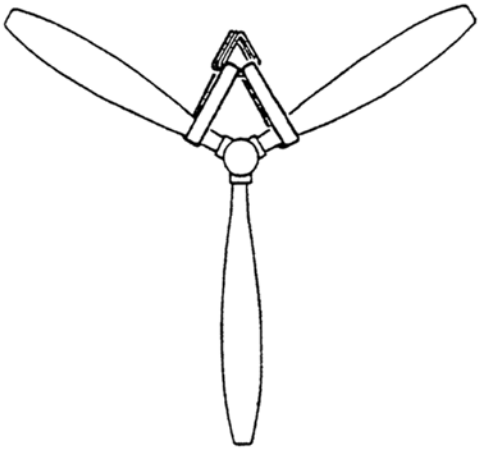


HOISTING HORIZONTAL STABILIZER

Hoisting Aircraft (Complete) and Fuselage
Figure 1 (Sheet 2 of 2)

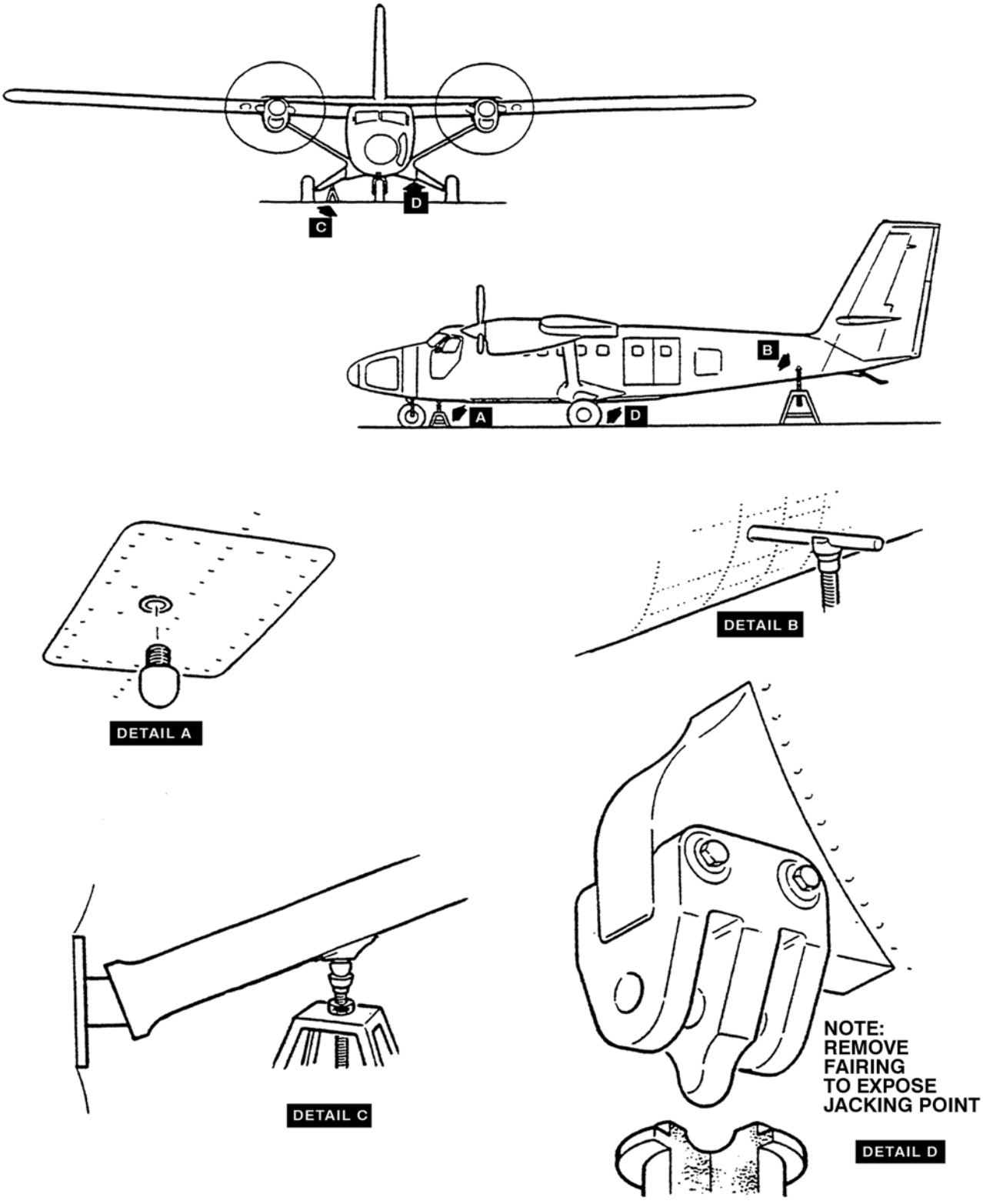


HOISTING ENGINE



HOISTING PROPELLER

Engine and Propeller Hoisting
Figure 2



Aircraft Jacking
Figure 3

CHAPTER 8

LEVELING AND WEIGHING

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CHAPTER 8 – LEVELING AND WEIGHING

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

REFERENCE	PAGE	DATE	REFERENCE	PAGE	DATE
08	* 1	Sep. 28/18		* 5	Sep. 28/18
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	* 3	Sep. 28/18	08-00-00	* 1	Sep. 28/18
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CHAPTER 8 – LEVELING AND WEIGHING

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LEVELING AND WEIGHING	**08-00**	
GENERAL INFORMATION	08-00-00	
Aircraft Leveling		1

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Levelling and Weighing – General data

1. Aircraft Leveling

   Leveling the aircraft laterally is accomplished by placing a straight-edge and spirit level or digital protractor (for higher degree of accuracy) across the seat rail, or rail and track assembly, and longitudinally by placing the spirit level or digital protractor along the rail, or rail and track assembly, then adjusting the supporting jacks until the aircraft is level.

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CHAPTER 9

TOWING AND TAXIING

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CHAPTER 9 – TOWING AND TAXIING

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	* 3	Sep. 28/18	09-00-00	* 1	Sep. 28/18
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CHAPTER 9 – TOWING AND TAXIING

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GENERAL INFORMATION	09-00-00	
General		1

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Towing and Taxiing – General Data

1. General

CAUTION **ENSURE THAT BRAKE ACCUMULATOR IS FULLY CHARGED TO 1500 PSI BEFORE COMMENCING TO TOW AIRCRAFT.**

Towing the aircraft is accomplished by attaching a towbar, approved by Viking for the aircraft landing gear configuration (refer to PSM 1-6-2T), to the nose wheel axle and disconnecting the nose leg torque links. To connect the tow bar to the nose wheel axle, insert pawl pins through ends of the tow bar into the axle spools, then insert L-shaped locking pins vertically through ends of the tow bar and the groove in pawl pins.

Station a crew member in flight compartment to operate brakes when aircraft is being towed.

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CHAPTER 10

PARKING AND MOORING

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CHAPTER 10 – PARKING AND MOORING

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	* 5	Sep. 28/18		* 4	Blank Page

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CHAPTER 10 – PARKING AND MOORING

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Parking Procedure		1
Mooring Procedure		1

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Parking and Mooring – General Data

1. Parking Procedure

   A. General

   Park the aircraft by heading it into the wind, applying the parking brakes, locking the control surfaces, fitting protective covers and propeller restrainers as necessary, and closing all doors and access provisions. Before leaving the flight compartment ensure that controls are set as follows:

   - Propeller levers to full INCREASE
   - Power levers to IDLE (If propeller zero thrust latches are fitted and engaged, leave power levers at the ZERO THRUST STOP/START line)
   - Fuel levers to OFF
   - POWER SOURCE switch to OFF

   B. Parking Brake

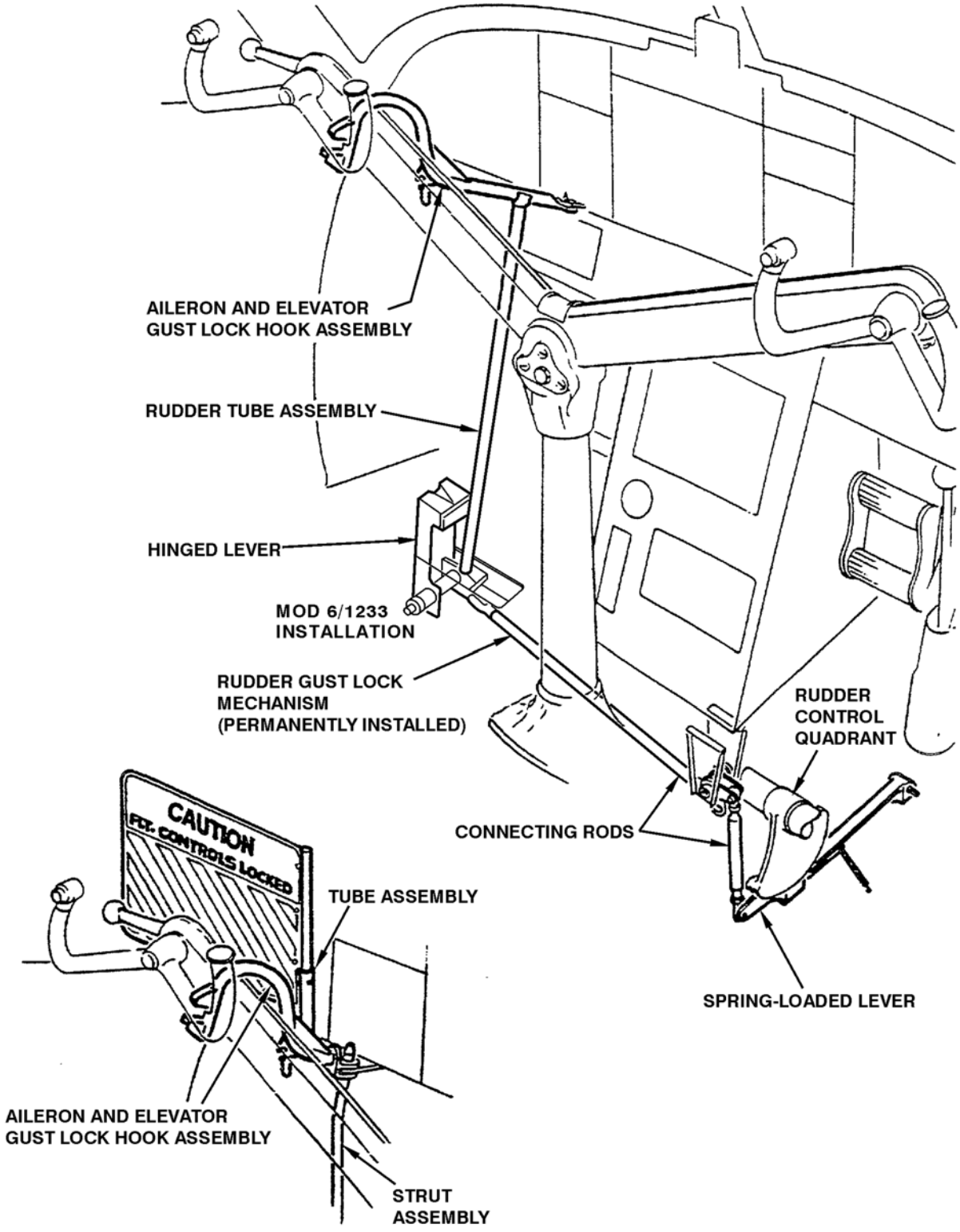
   A push-pull type parking brake handle is located on a pedestal between the pilot's rudder pedals. To apply the parking brake, depress both brake pedals fully, pull out the parking brake handle and release pedals. To release parking brake, apply pressure to both brake pedals, push brake handle fully in and release pedals.

   C. Gust Lock Installation – Mods 6/1676 and 6/1726

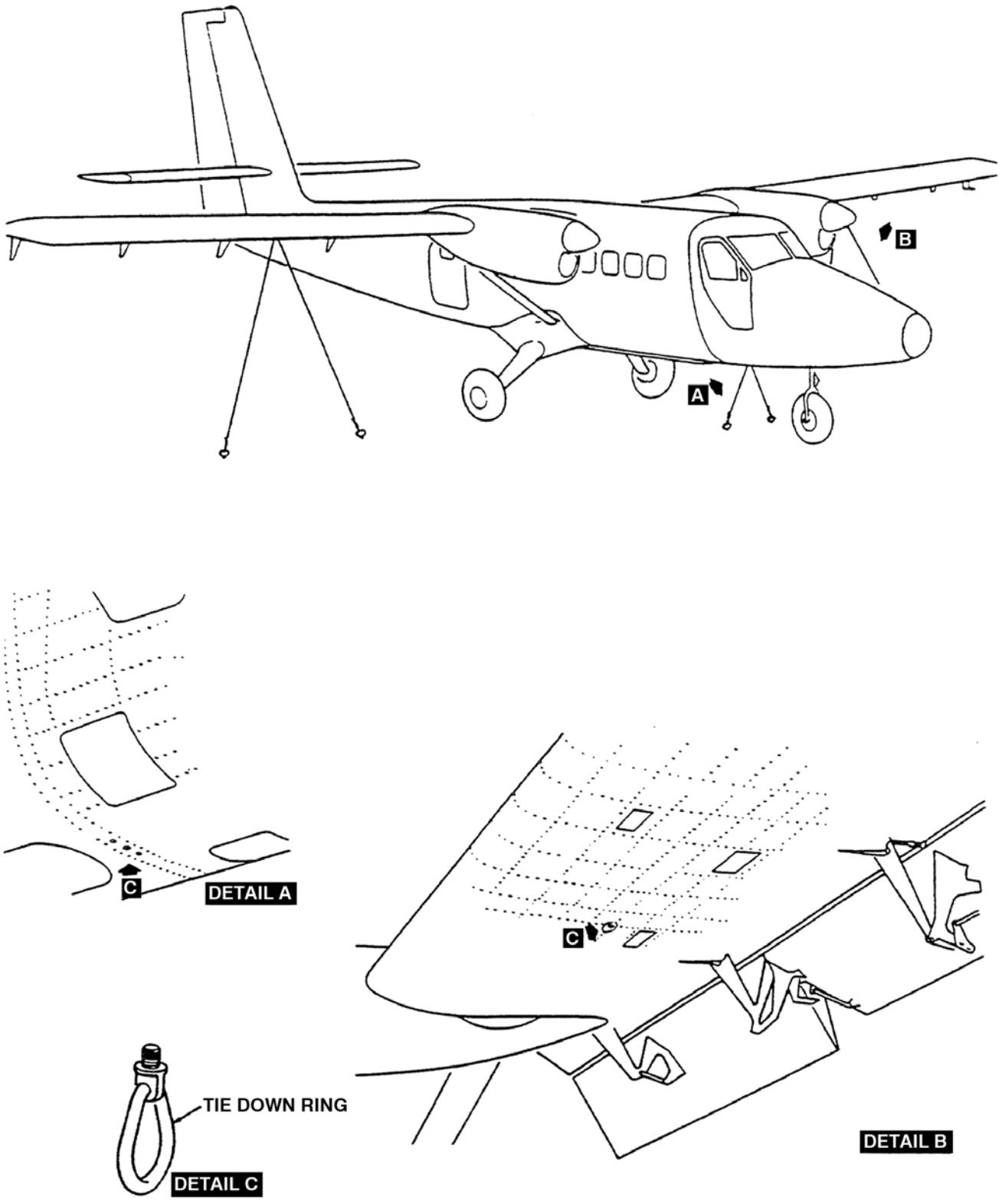
   Position aileron and elevator gust lock hook assembly over left arm of control column and engage hand wheel inner spoke in padded support. Align hook assembly lug with eyebolt on control column and install pip pin. Move rudder pedals to neutral position, lift rudder gust lock lever and insert spring loaded spigot of strut assembly into socket below floor. Position the caution flag (Mod 6/1726) in the tube assembly so that the flag obscures the instrument panel. Move control column forward until end fitting of hook assembly is aligned with gust lock bracket on instrument panel. Push down on strut assembly and engage strut upper spigot pin through hook assembly end fitting and gust lock bracket (refer to Figure 1).

2. Mooring Procedure

   Mooring is accomplished by parking the aircraft as in Para 1., place chocks on the forward and aft side of each nose and main landing gear wheel, and tying aircraft down with mooring rope (SD12519) using aircraft tie-down points (refer to Figure 2).



Gust Lock Installation
Figure 1



Aircraft Mooring Points
Figure 2

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CHAPTER 11

REQUIRED PLACARDS

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CHAPTER 11 – REQUIRED PLACARDS

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

REFERENCE	PAGE	DATE	REFERENCE	PAGE	DATE
11	* 1	Sep. 28/18		* 12	Sep. 28/18
	* 2	Blank Page		* 13	Sep. 28/18
	* 3	Sep. 28/18		* 14	Sep. 28/18
	* 4	Blank Page		* 15	Sep. 28/18
	* 5	Sep. 28/18		* 16	Sep. 28/18
	* 6	Blank Page		* 17	Sep. 28/18
11-00-00	* 1	Sep. 28/18		* 18	Sep. 28/18
	* 2	Sep. 28/18		* 19	Sep. 28/18
	* 3	Sep. 28/18		* 20	Sep. 28/18
	* 4	Sep. 28/18		* 21	Sep. 28/18
	* 5	Sep. 28/18		* 22	Sep. 28/18
	* 6	Sep. 28/18		* 23	Sep. 28/18
	* 7	Sep. 28/18		* 24	Sep. 28/18
	* 8	Sep. 28/18		* 25	Sep. 28/18
	* 9	Sep. 28/18		* 26	Sep. 28/18
	* 10	Sep. 28/18		* 27	Sep. 28/18
	* 11	Sep. 28/18		* 28	Blank Page

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CHAPTER 11 – REQUIRED PLACARDS

TABLE OF CONTENTS

SUBJECT	CHAPTER SECTION SUBJECT	PAGE
REQUIRED PLACARDS	**11-00**	
GENERAL INFORMATION	11-00-00	
General		1

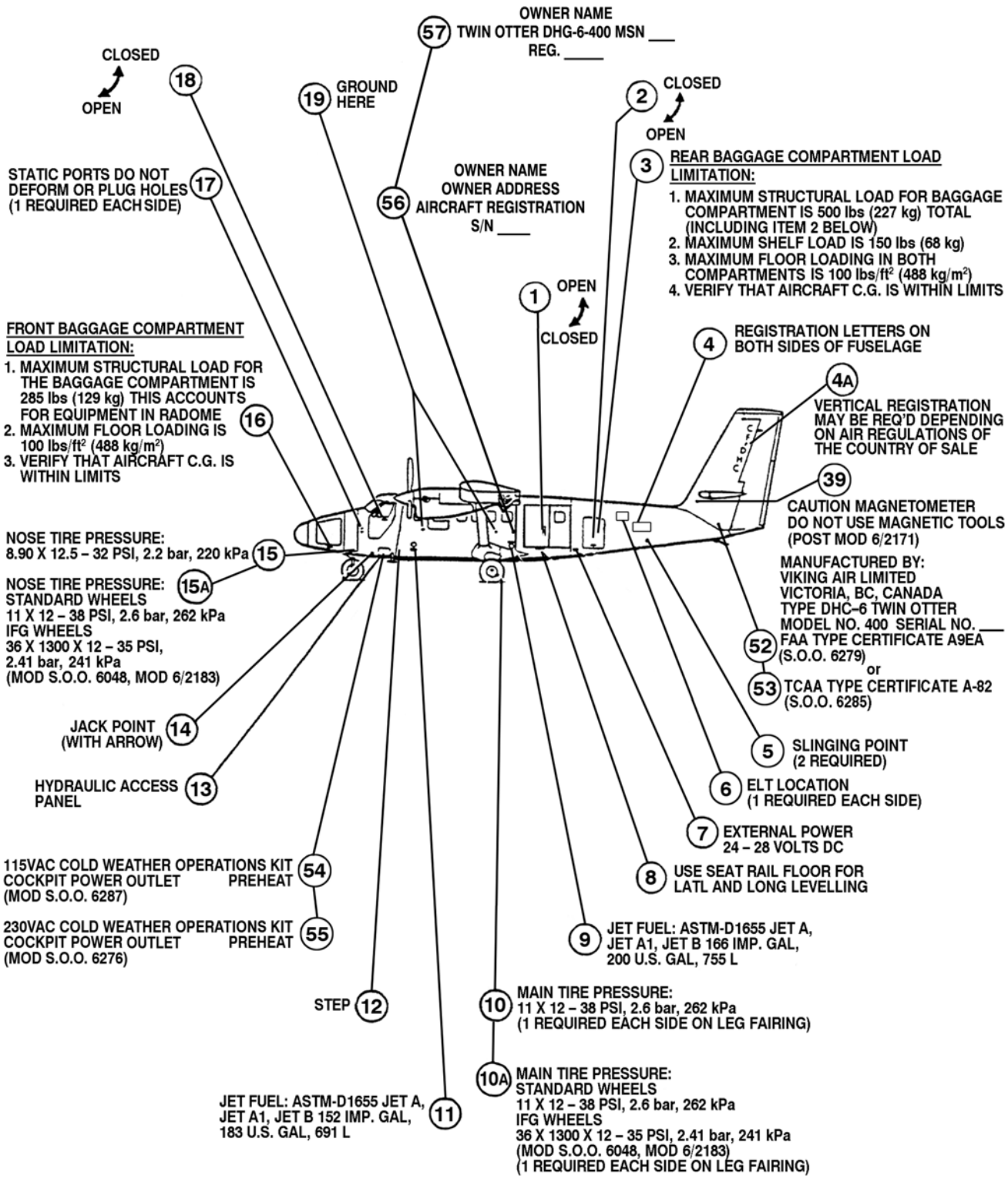
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Required Placards – General Data

1. General

All internal and external placards and stencils are shown in Figure 1 and Figure 2 respectively. Details of these placards and stencils are shown in Table 1 — Details of External Placards and Stencils and Table 2 — Details of Internal Placards . Always refer to the requirements of the country of registration. Those typically required by government regulations are defined as follows:

- Baggage
- Fuel and oil
- Emergency
- Safety
- Airspeed
- Operating limitations
- Aircraft registration



External Placards and Stencils
Figure 1 (Sheet 1 of 5)

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR11-9**

Insert in 11-00-00 after Page 2
REQUIRED PLACARDS

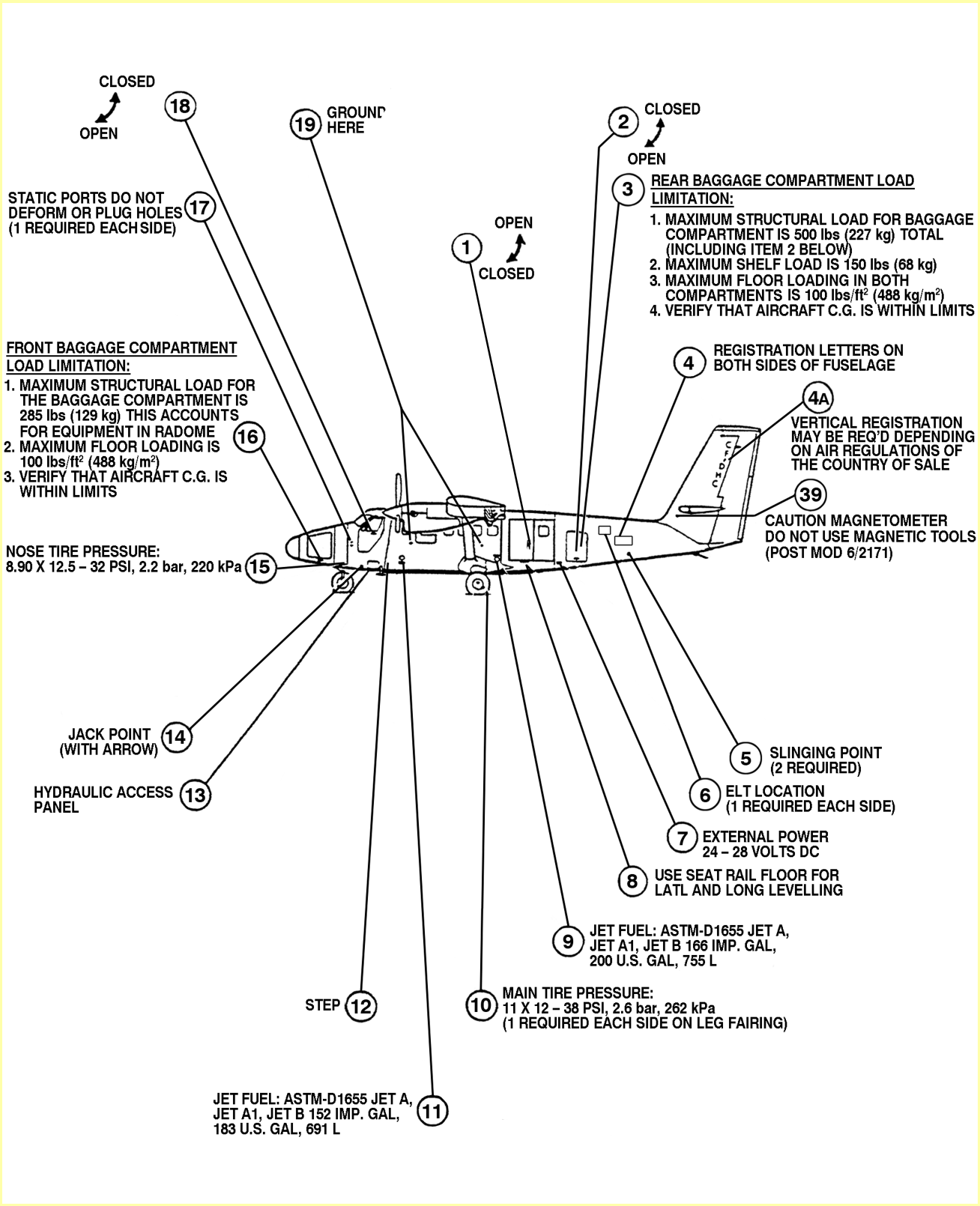
REASON FOR ISSUE:
To incorporate S.O.O. 6371, Relocation of the Main Battery and the Ground Power Unit Receptacle and S.O.O. 6431, Introduction of Station 15.45 Bulkhead – Nose Luggage Compartment.

ACTION:
In Chapter 11, Section 11-00-00, ignore existing Pages 2 to 9 and insert revised pages, as shown on Pages 2 to 12 of this Temporary Revision.

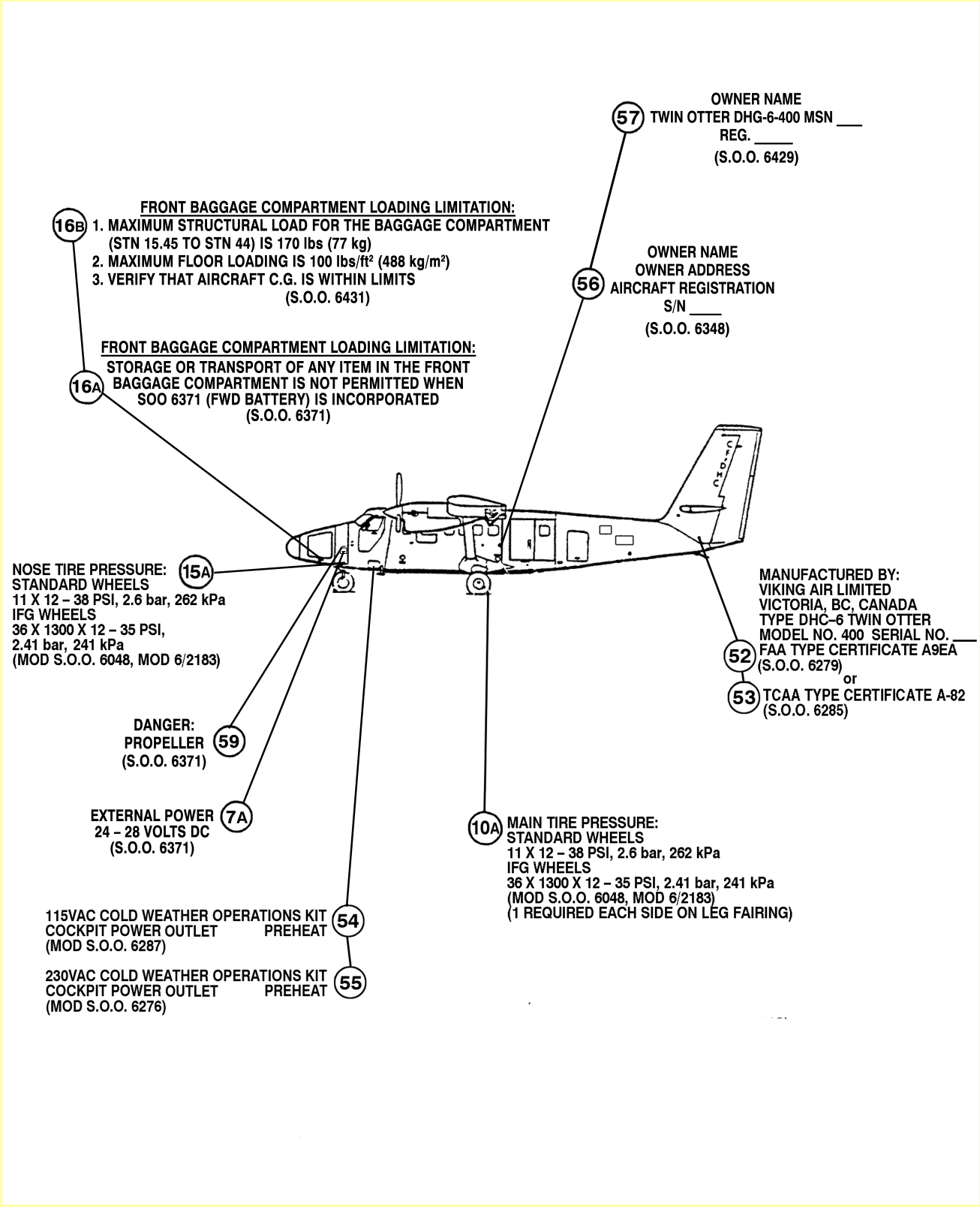
-Continued overleaf-

RECORDING INSTRUCTIONS:

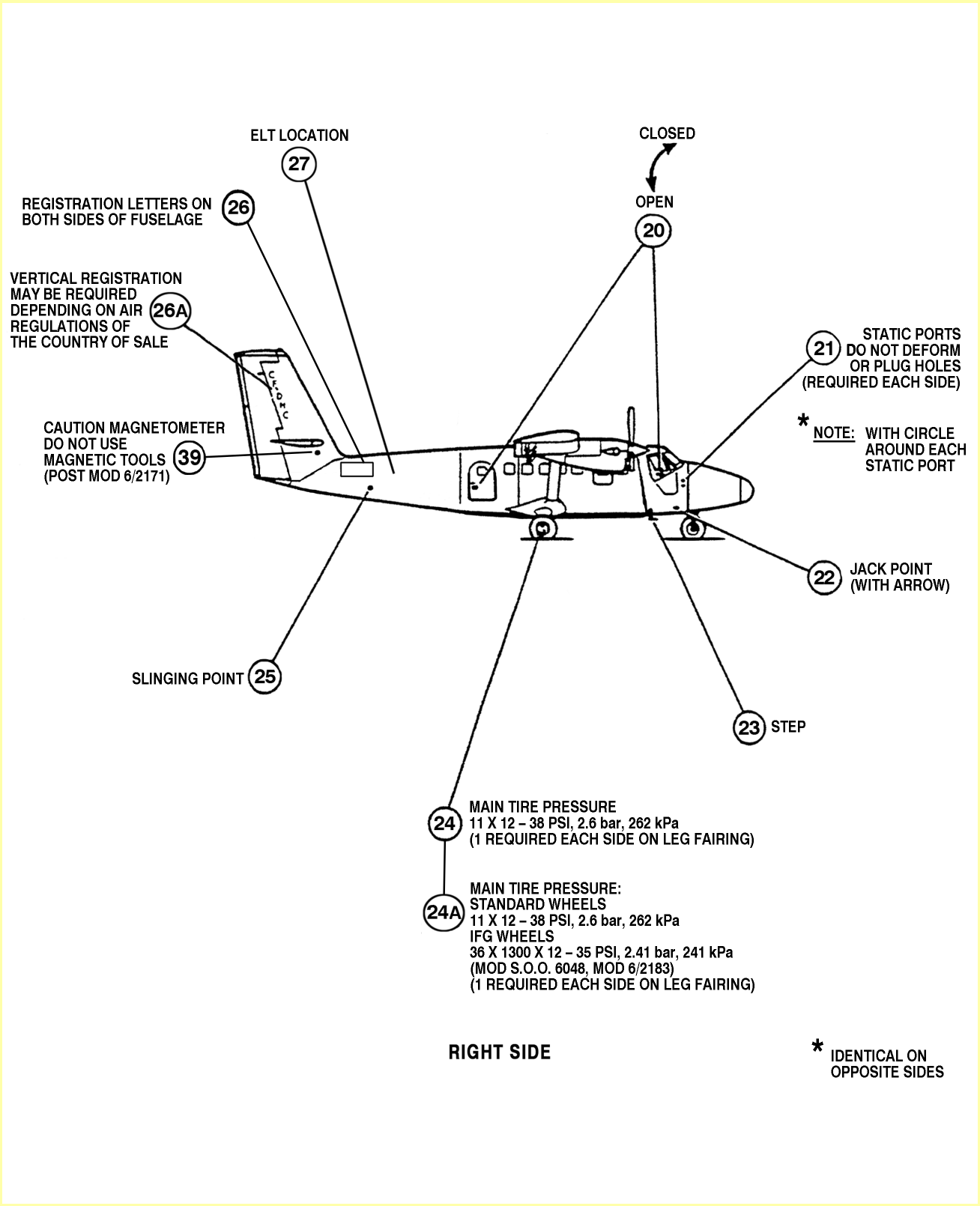
Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.



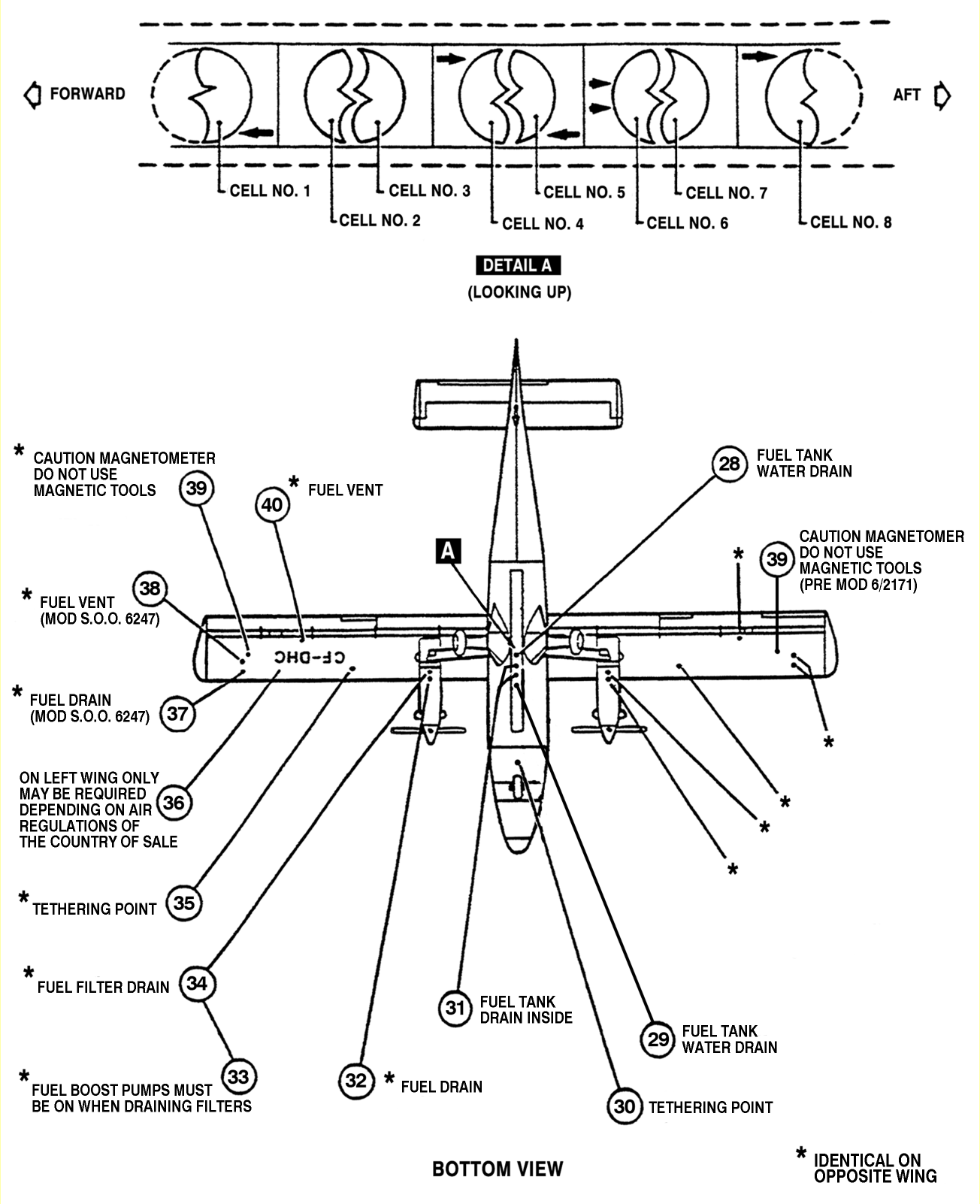
External Placards and Stencils
Figure 1 (Sheet 1 of 6)



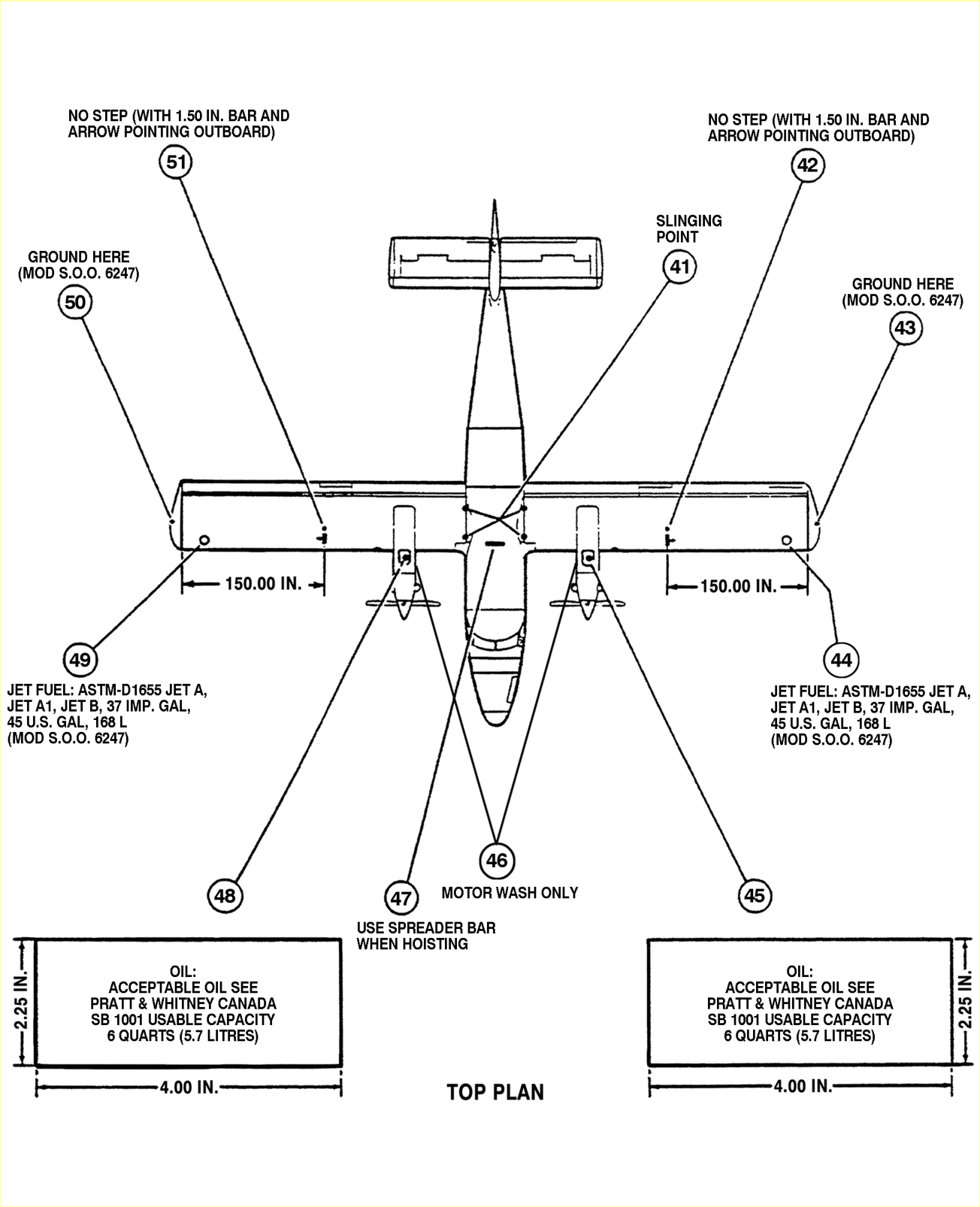
External Placards and Stencils
Figure 1 (Sheet 2 of 6)



External Placards and Stencils
Figure 1 (Sheet 3 of 6)

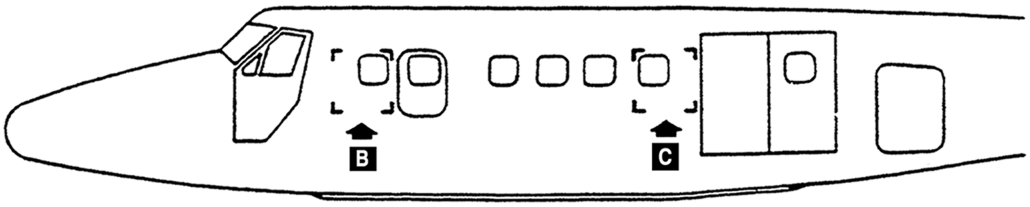


External Placards and Stencils
Figure 1 (Sheet 4 of 6)



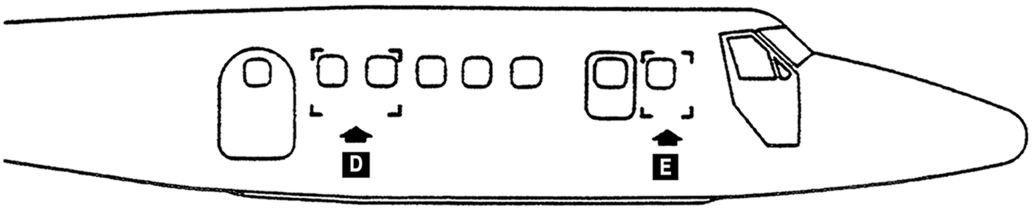
External Placards and Stencils
Figure 1 (Sheet 5 of 6)

EXTERNAL BREAK-IN MARKINGS
(MOD S.O.O. 6150)





FUSELAGE LEFT SIDE





FUSELAGE RIGHT SIDE

NOTE: LOCATE CORNER MARKINGS AT STATIONS AND WATER LINES AS INDICATED. (CABIN FLOOR IS AT W/L - 15.0 IN.)

External Placards and Stencils
Figure 1 (Sheet 6 of 6)

Table 1 Details of External Placards and Stencils

FIGURE	NUMBER	MARKING		
		TYPE	LETTER SIZE (INS)	COLOUR/DESCRIPTION
1	1	Label	0.25	))
	2	Label	0.25	)
	3	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	4	Lettering/ Stencil	12.00	Black (Green A/C)
	4A	Lettering/ Stencil	6.00	To suit (see sales order for Country of sale)
	5	Label	0.38	Black letters on light surface Yellow letters on dark surface
	6	Label	0.50	) Red letters
	7	Label	0.25	)
	7A	Label	0.25	(S.O.O. 6371)
	8	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	9	Label	0.25	)
	10	Label	0.25	) Black letters on light surface) Yellow letters on dark surfaces
	10A	Label	0.25	) Black letters on light surface) Yellow letters on dark surfaces
	11	Label	0.25	Red Letters
	12	Stencil	0.38	) Black letters on light surface) Yellow letters on dark surface
	13	Stencil	0.38	) Black letters on light surface) Yellow letters on dark surface
	14	Label	0.38	))
	15	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	15A	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	16	Label	0.19	))
	16A	Label	0.19	(S.O.O. 6371)
	16B	Label	0.19	(S.O.O. 6431)
	17	Stencil	0.38	))

Table 1 Details of External Placards and Stencils (continued)

FIGURE	NUMBER	MARKING		
		TYPE	LETTER SIZE (INS)	COLOUR/DESCRIPTION
	18	Label	0.25	))
	19	Label	0.38	))
	20	Stencil	0.25	))
	21	Stencil	0.38	) Black letters on light surface) Yellow letters on dark surface
	22	Label	0.38	))
	23	Stencil	0.38	))
	24	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	24A	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	25	Label	0.38	)
	26	Lettering/ Stencil	12.00	Black (Green A/C)
	26A	Lettering/ Stencil	6.00	To suit (see sales order for Country of sale)
	27	Label	0.50	) Red letters
	28	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	29	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	30	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	31	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	32	Label	0.38	) Red letters
	33	Label	0.38	) Red letters
	34	Label	0.38	Black letters on light surface Yellow letters on dark surface
	35	Label	0.38	Black letters on light surface Yellow letters on dark surface
	36	Stencil	20.00	Black letters unless otherwise specified (see sales order for Country of sale)
	37	Stencil	0.50	))

Table 1 Details of External Placards and Stencils (continued)

FIGURE	NUMBER	MARKING		
		TYPE	LETTER SIZE (INS)	COLOUR/DESCRIPTION
	38	Stencil	0.50	) Red letters)
	39	Label	0.38	) Red letters
	40	Label	0.50	)
	41	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	42	Label	1.00	) Black letters on light surface) Yellow letters on dark surface
	43	Stencil	0.38	)
	44	Label	0.50	)
	45	Label	0.25	)
	46	Label	0.38	Black letters on light surface
	47	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	48	Label	0.50	)
	49	Label	0.50	) Black letters on light surface) Yellow letters on dark surface
	50	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	51	Label	1.00	)
	52	Plate		Dimensions 2.48 x 4.00 (Ref C6G1504–27) (S.O.O. 6279)
	53	Plate		Dimensions 2.48 x 4.00 (Ref C6G1513–27) (S.O.O. 6285)
	54	Placard	0.15	) Black letters on light surface) Yellow letters on dark surface
	55	Placard	0.15	) Black letters on light surface) Yellow letters on dark surface
	56	Plate	0.30	Stainless Steel Engraved (S.O.O. 6348)
	57	Plate	0.30	Stainless Steel Engraved (S.O.O. 6429)
	58	Label	0.50	(S.O.O. 6371)
	DETAIL A	Stencil		Red arrows

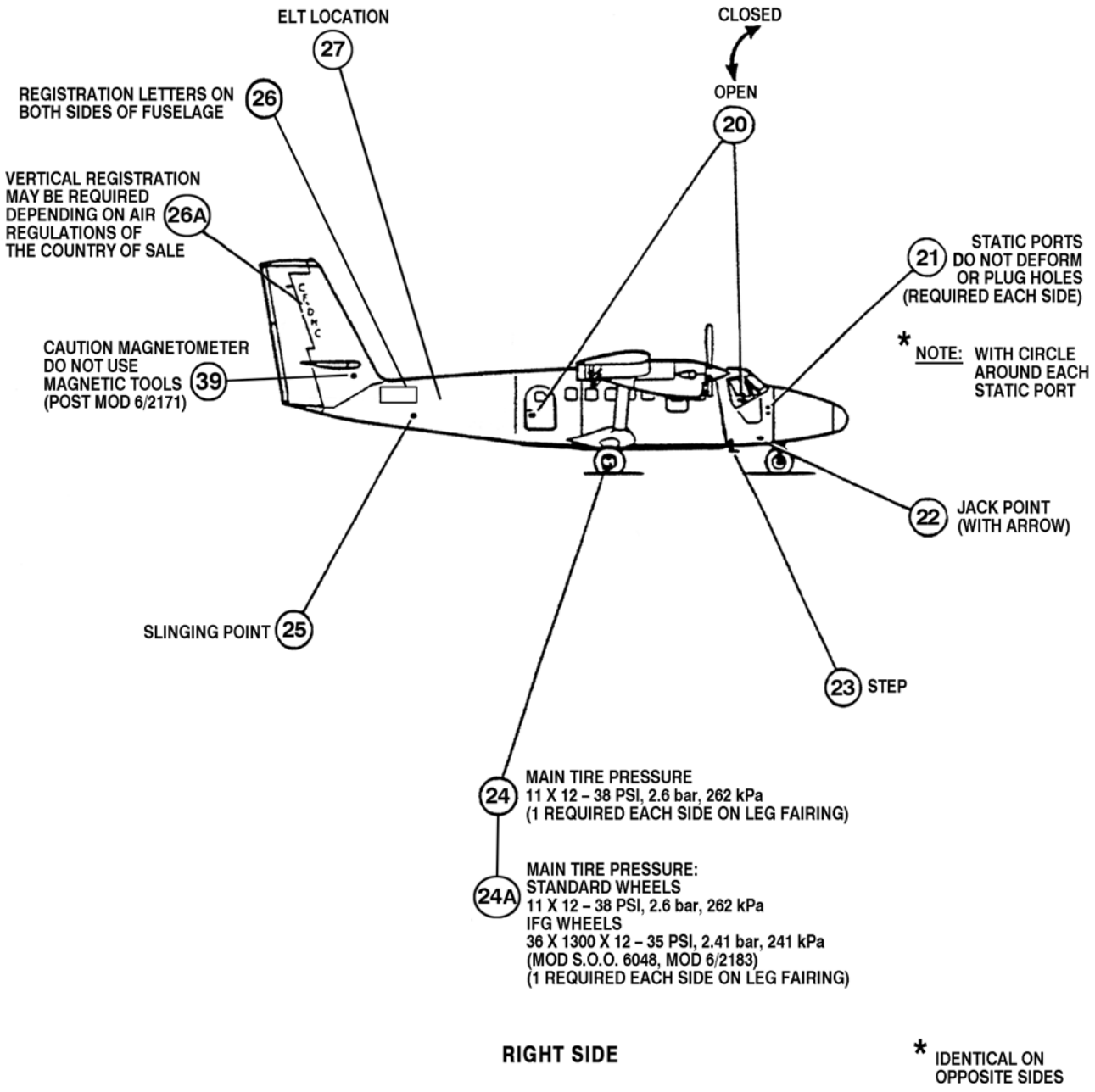
Table 1 Details of External Placards and Stencils (continued)

FIGURE	NUMBER	MARKING		
		TYPE	LETTER SIZE (INS)	COLOUR/DESCRIPTION
	DETAIL B, C, D, AND E	Stencil (lettering)	1.00	) Red if background is not) contrasting, outline with) white or other contrasting colour
		Stencil (Corner markings)	4.00 x 4.00 x 1.00	))

Note Silk-screening may be used in lieu of stencils.

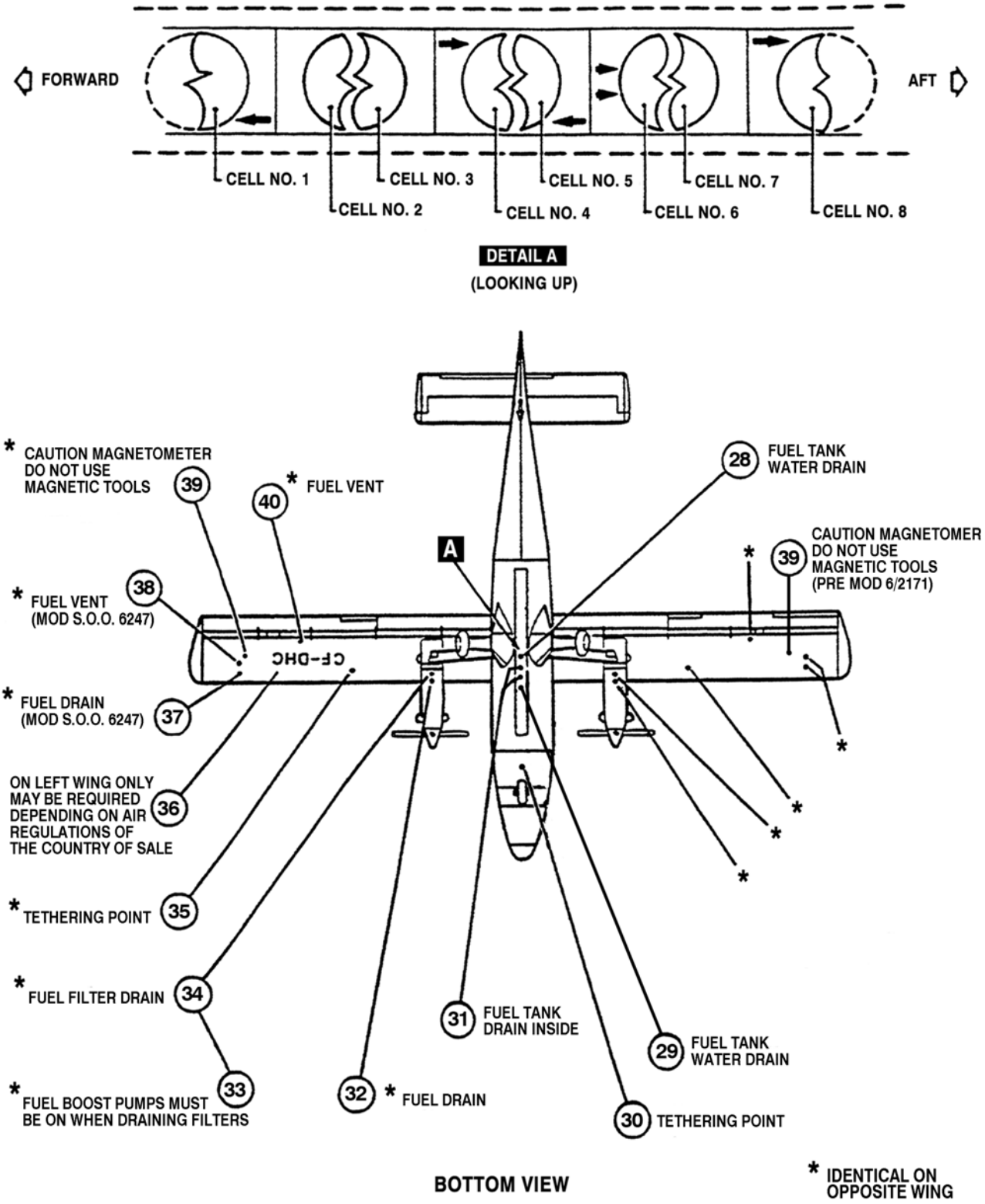
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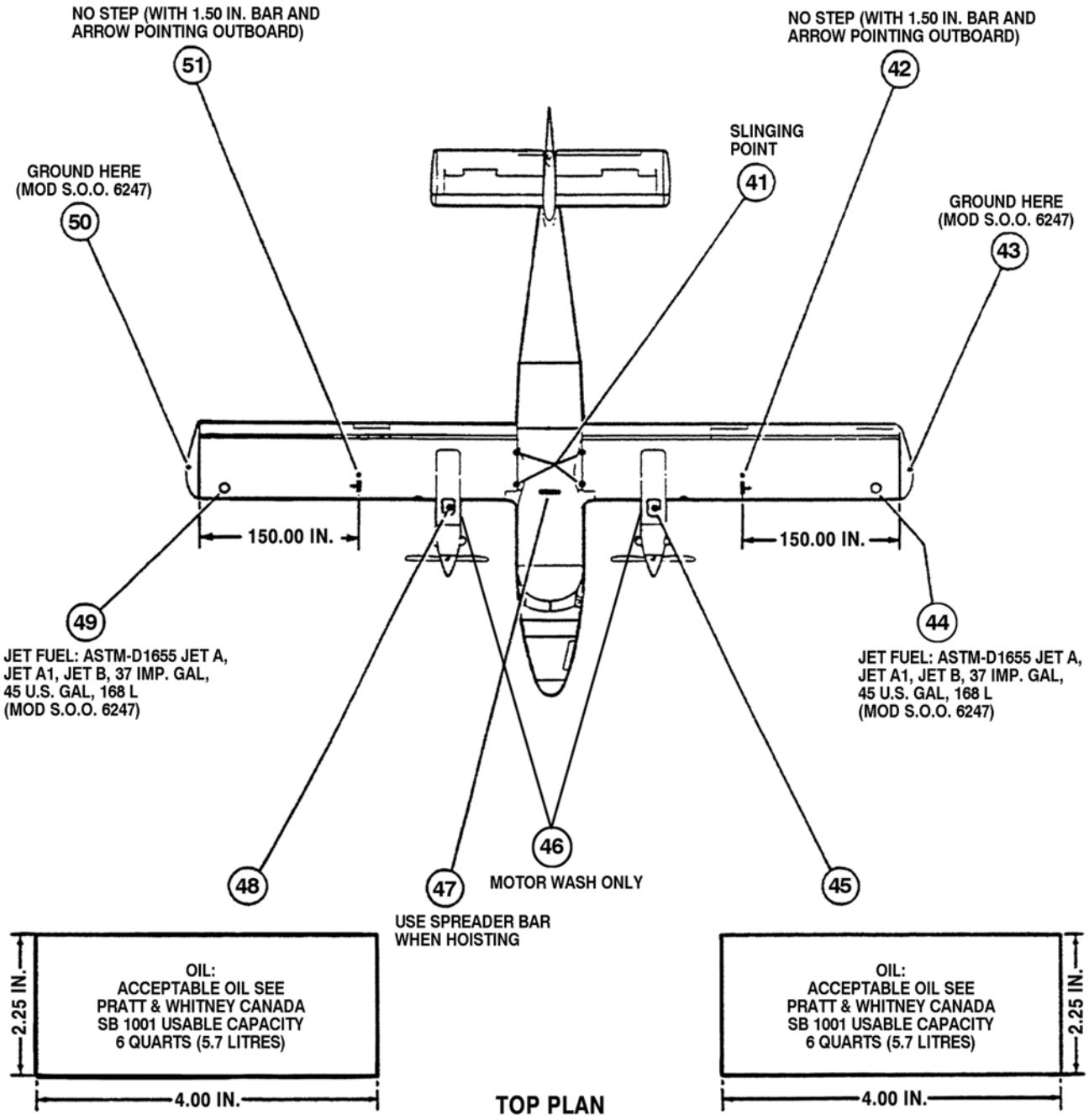


External Placards and Stencils
Figure 1 (Sheet 2 of 5)

NOTE: Please see the TEMPORARY REVISION that revises this page.

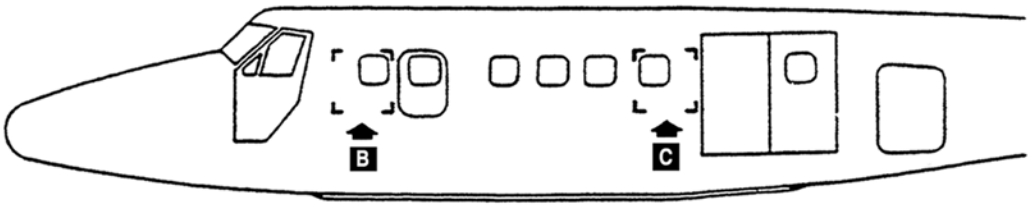


External Placards and Stencils
Figure 1 (Sheet 3 of 5)



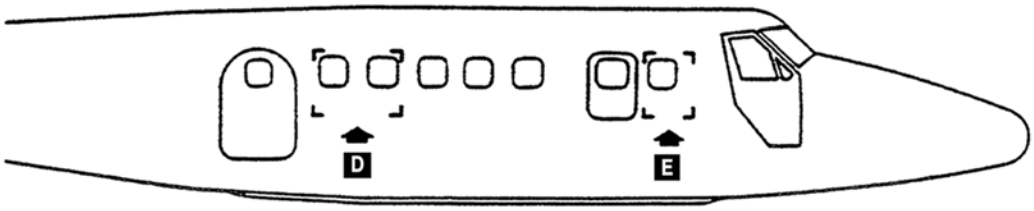
External Placards and Stencils
Figure 1 (Sheet 4 of 5)

EXTERNAL BREAK-IN MARKINGS
(MOD S.O.O. 6150)





FUSELAGE LEFT SIDE





FUSELAGE RIGHT SIDE

NOTE: LOCATE CORNER MARKINGS AT STATIONS AND WATER LINES AS INDICATED. (CABIN FLOOR IS AT W/L - 15.0 IN.)

External Placards and Stencils
Figure 1 (Sheet 5 of 5)

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 1 Details of External Placards and Stencils

FIGURE	NUMBER	MARKING		
		TYPE	LETTER SIZE (INS)	COLOUR/DESCRIPTION
1	1	Label	0.25	))
	2	Label	0.25	)
	3	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	4	Lettering/ Stencil	12.00	Black (Green A/C)
	4A	Lettering/ Stencil	6.00	To suit (see sales order for Country of sale)
	5	Label	0.38	Black letters on light surface Yellow letters on dark surface
	6	Label	0.50	) Red letters
	7	Label	0.25	)
	8	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	9	Label	0.25	)
	10	Label	0.25	) Black letters on light surface) Yellow letters on dark surfaces
	10A	Label	0.25	) Black letters on light surface) Yellow letters on dark surfaces
	11	Label	0.25	Red Letters
	12	Stencil	0.38	) Black letters on light surface) Yellow letters on dark surface
	13	Stencil	0.38	) Black letters on light surface) Yellow letters on dark surface
	14	Label	0.38	))
	15	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	15A	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	16	Label	0.19	))
	17	Stencil	0.38	))
	18	Label	0.25	))
	19	Label	0.38	))

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 1 Details of External Placards and Stencils (continued)

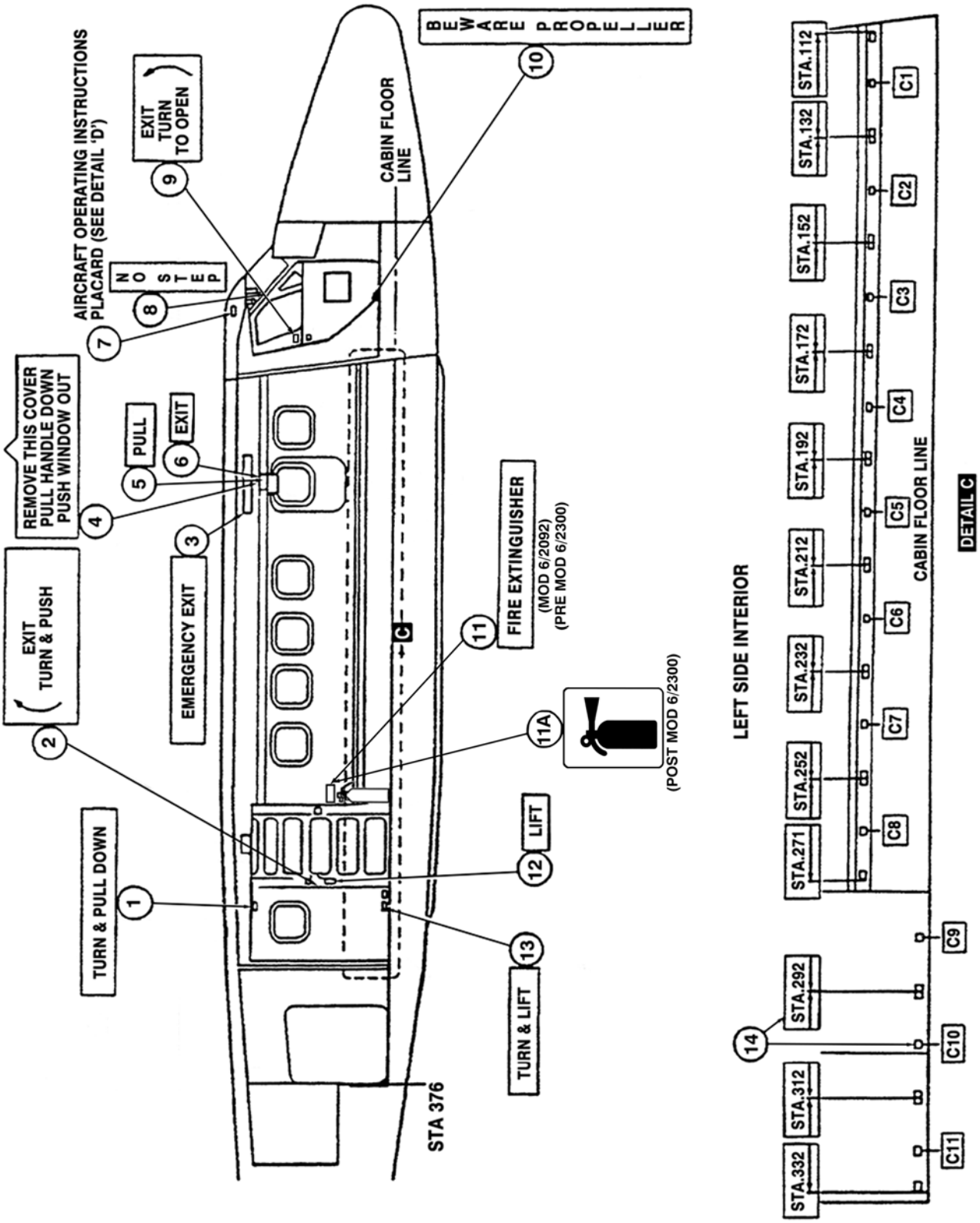
FIGURE	NUMBER	MARKING		
		TYPE	LETTER SIZE (INS)	COLOUR/DESCRIPTION
	20	Stencil	0.25	))
	21	Stencil	0.38	) Black letters on light surface) Yellow letters on dark surface
	22	Label	0.38	))
	23	Stencil	0.38	))
	24	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	24A	Label	0.25	) Black letters on light surface) Yellow letters on dark surface
	25	Label	0.38	)
	26	Lettering/ Stencil	12.00	Black (Green A/C)
	26A	Lettering/ Stencil	6.00	To suit (see sales order for Country of sale)
	27	Label	0.50	) Red letters
	28	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	29	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	30	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	31	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	32	Label	0.38	) Red letters
	33	Label	0.38	) Red letters
	34	Label	0.38	Black letters on light surface Yellow letters on dark surface
	35	Label	0.38	Black letters on light surface Yellow letters on dark surface
	36	Stencil	20.00	Black letters unless otherwise specified (see sales order for Country of sale)
	37	Stencil	0.50	))
	38	Stencil	0.50	) Red letters)
	39	Label	0.38	) Red letters

NOTE: Please see the TEMPORARY REVISION that revises this page.

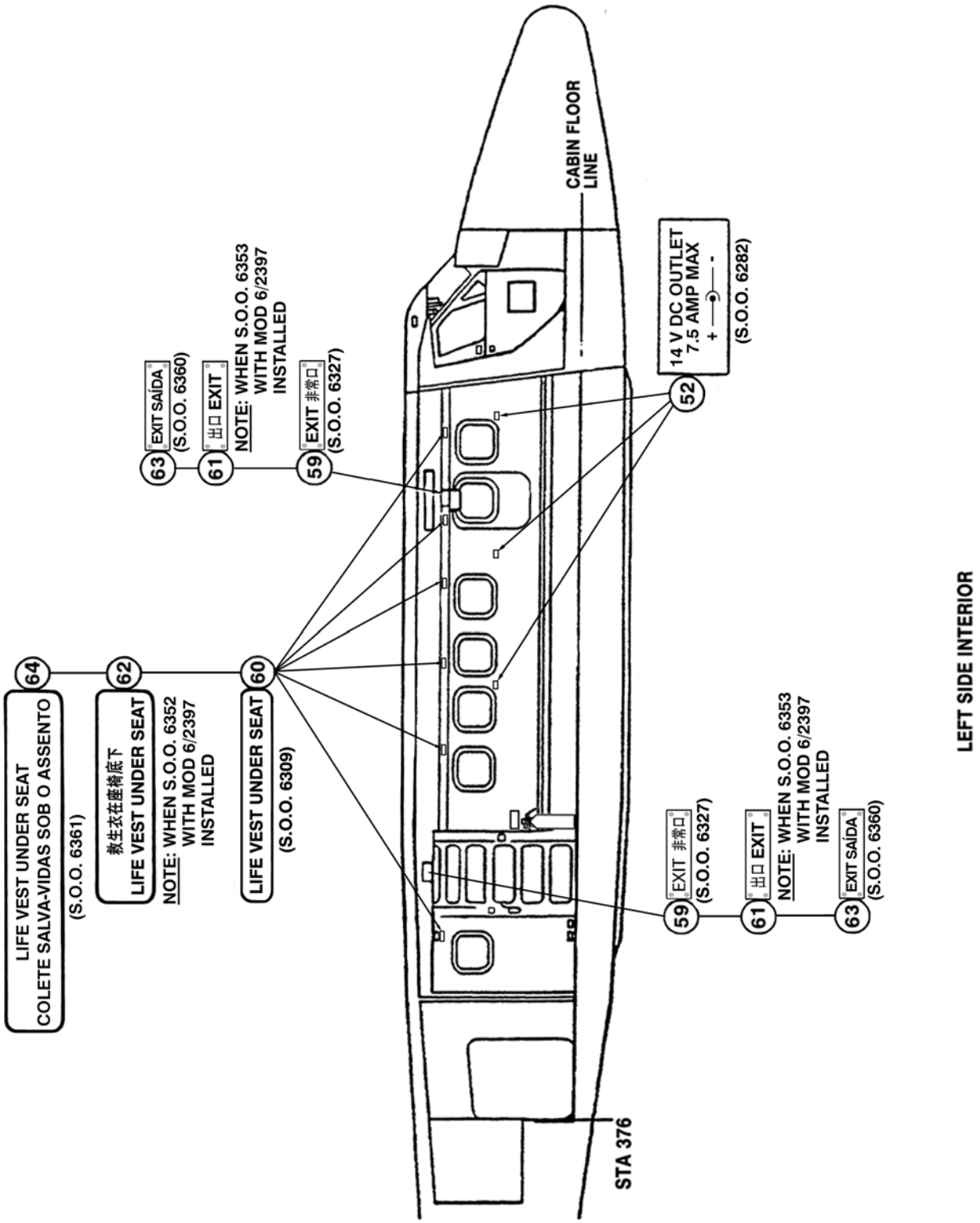
Table 1 Details of External Placards and Stencils (continued)

FIGURE	NUMBER	MARKING		
		TYPE	LETTER SIZE (INS)	COLOUR/DESCRIPTION
	40	Label	0.50	)
	41	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	42	Label	1.00	) Black letters on light surface) Yellow letters on dark surface
	43	Stencil	0.38	)
	44	Label	0.50	)
	45	Label	0.25	)
	46	Label	0.38	Black letters on light surface
	47	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	48	Label	0.50	)
	49	Label	0.50	) Black letters on light surface) Yellow letters on dark surface
	50	Label	0.38	) Black letters on light surface) Yellow letters on dark surface
	51	Label	1.00	)
	52	Plate		Dimensions 2.48 x 4.00 (Ref C6G1504–27) (S.O.O. 6279)
	53	Plate		Dimensions 2.48 x 4.00 (Ref C6G1513–27) (S.O.O. 6285)
	54	Placard	0.15	) Black letters on light surface) Yellow letters on dark surface
	55	Placard	0.15	) Black letters on light surface) Yellow letters on dark surface
	56	Plate	0.30	Stainless Steel Engraved (S.O.O. 6348)
	57	Plate	0.30	Stainless Steel Engraved (S.O.O. 6429)
	DETAIL A	Stencil		Red arrows
	DETAIL B, C, D, AND E	Stencil (lettering)	1.00	) Red if background is not) contrasting, outline with) white or other contrasting colour
		Stencil (Corner markings)	4.00 x 4.00 x 1.00	))

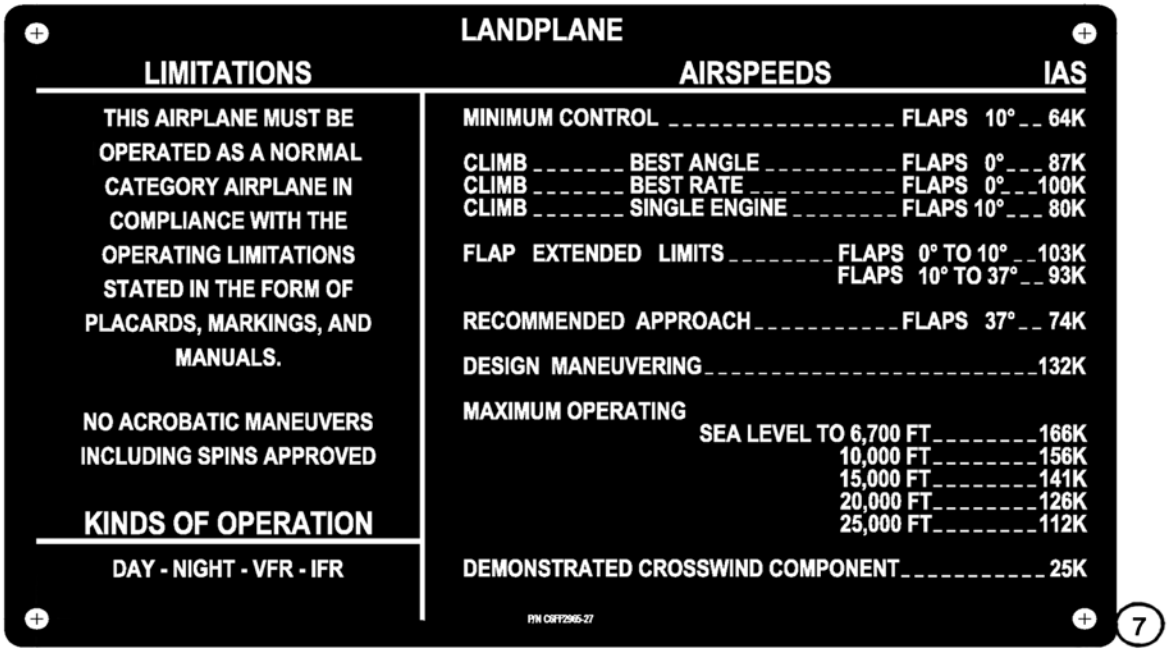
Note Silk-screening may be used in lieu of stencils.



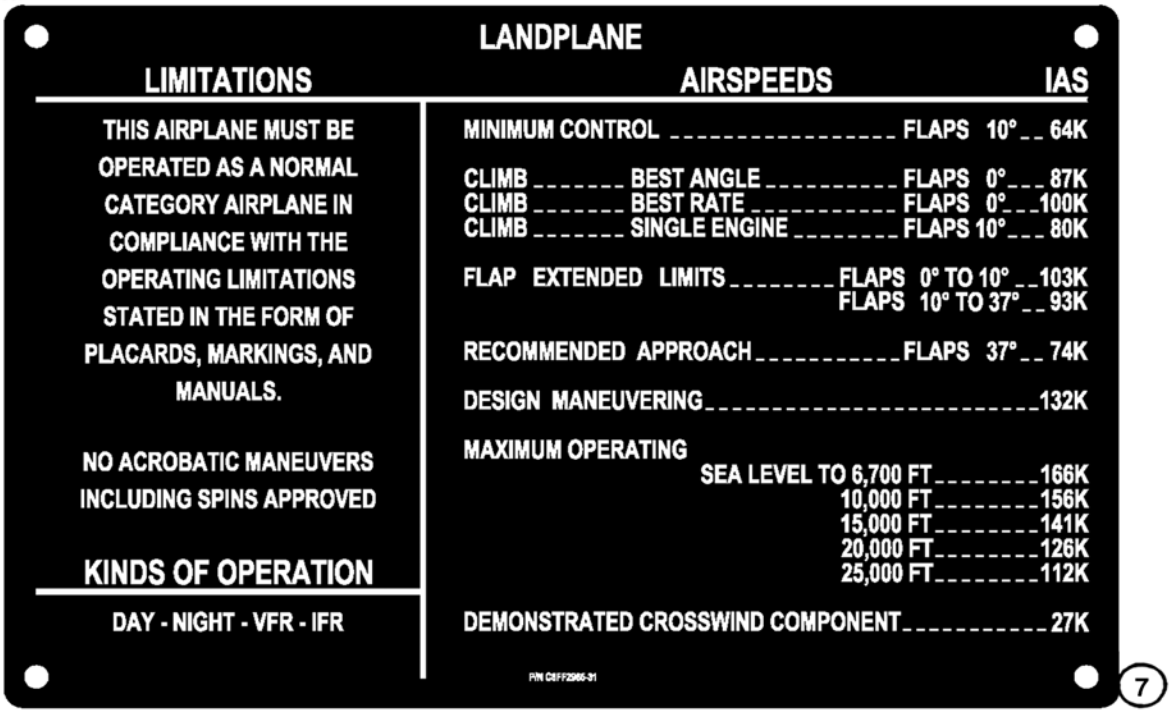
Internal Placards
Figure 2 (Sheet 1 of 14)



Internal Placards
Figure 2 (Sheet 2 of 14)



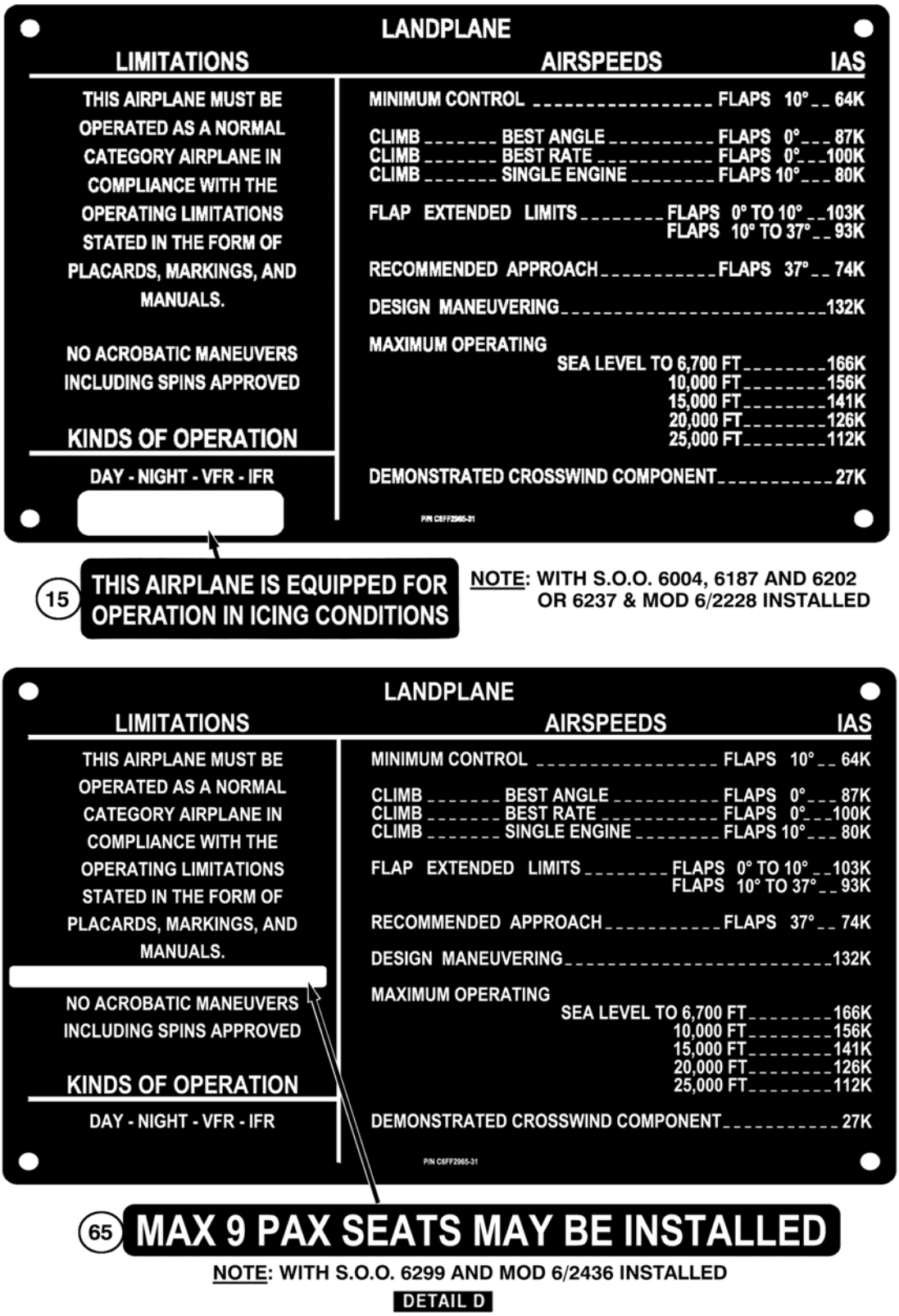
MOD 6/2059
PRE MOD 6/2434



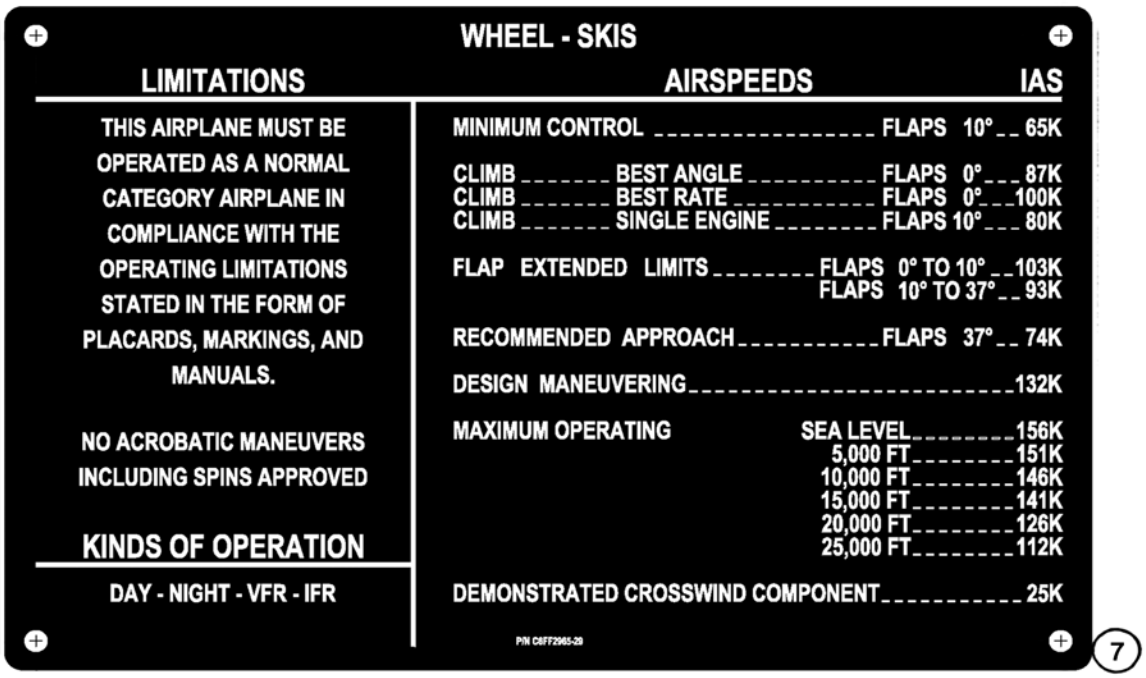
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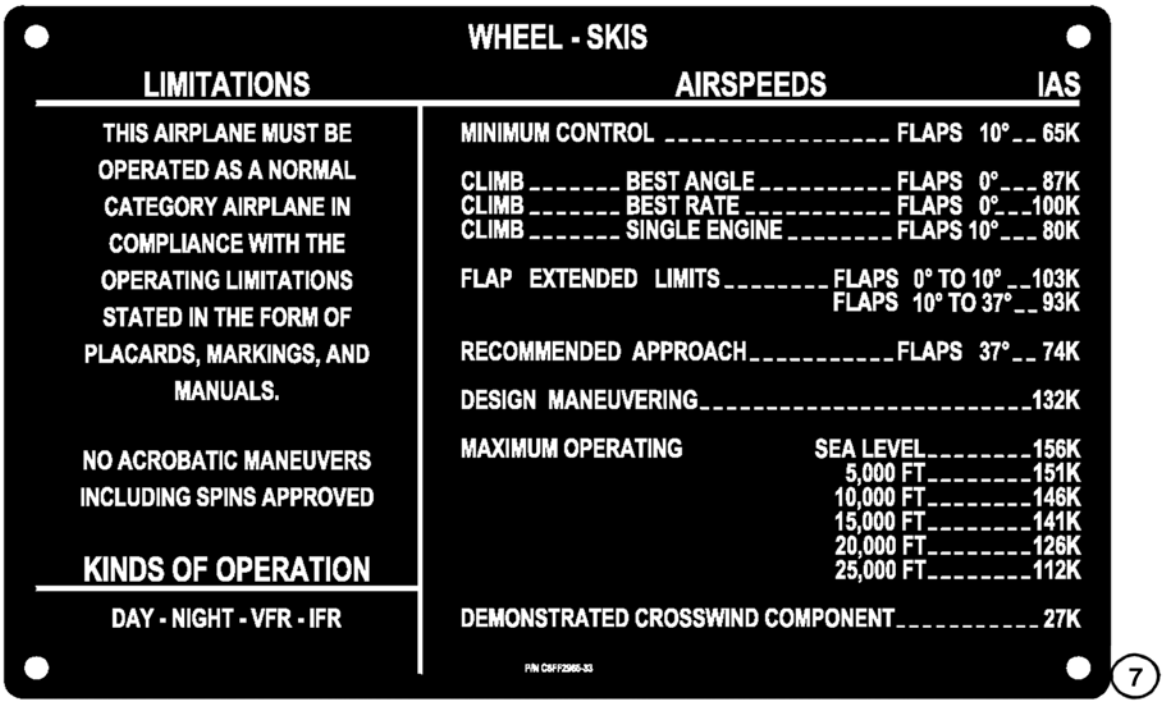
Internal Placards
Figure 2 (Sheet 3 of 14)



Internal Placards
Figure 2 (Sheet 4 of 14)



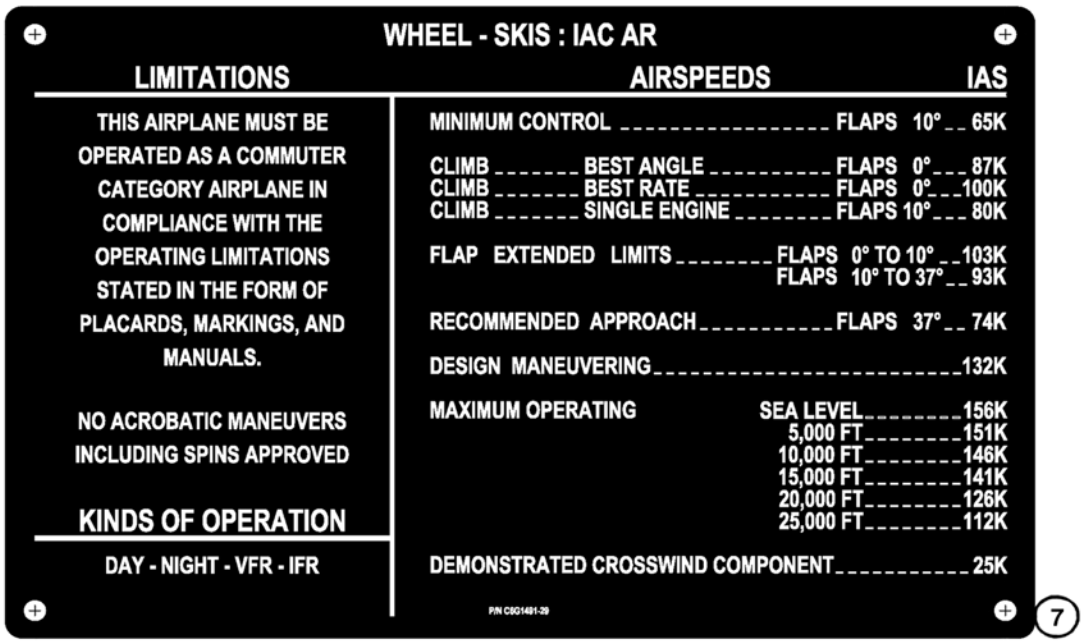
MOD S.O.O. 6001 WITH MOD 6/2182
PRE MOD 6/2434



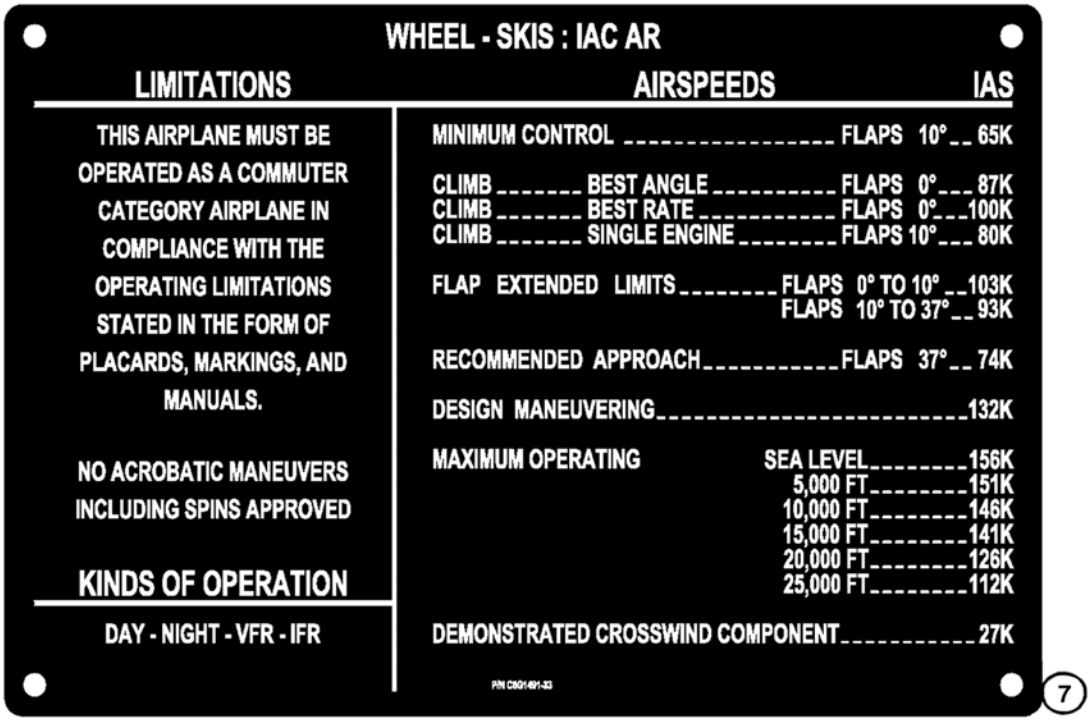
MOD S.O.O. 6001 WITH MOD 6/2182
POST MOD 6/2434

DETAIL D

Internal Placards
Figure 2 (Sheet 5 of 14)



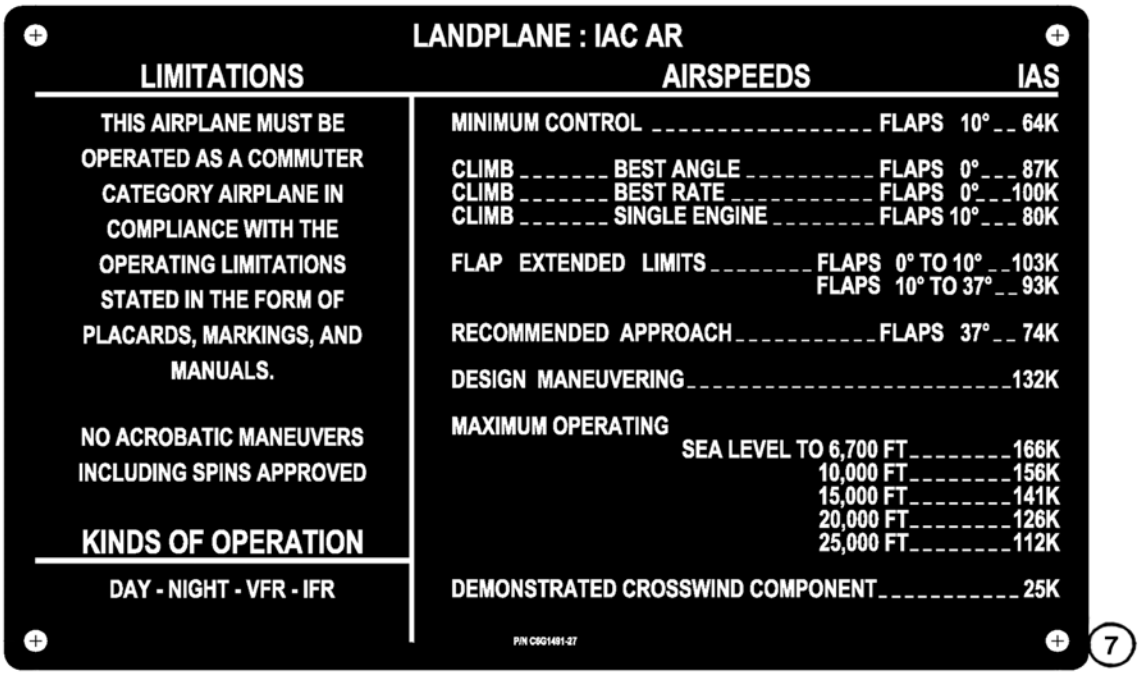
MOD S.O.O. 6269 AND S.O.O. 6001 WITH MOD 6/2182
(IAC AR REQUIREMENTS)
PRE MOD 6/2434



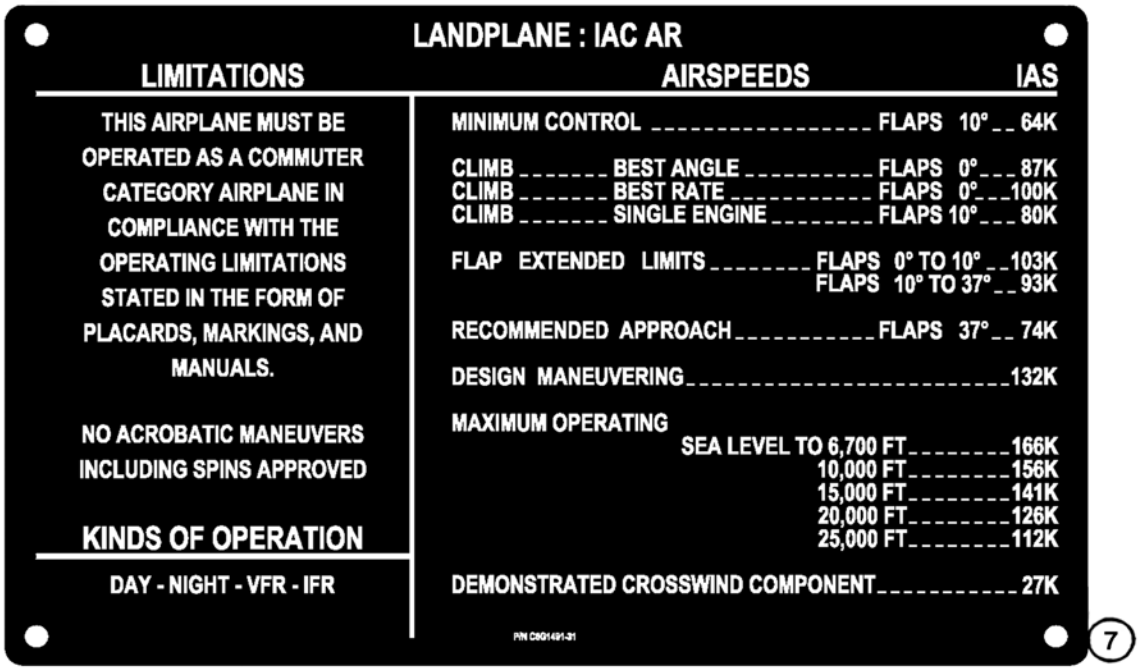
MOD S.O.O. 6269 AND S.O.O. 6001 WITH MOD 6/2182
(IAC AR REQUIREMENTS)
POST MOD 6/2434

DETAIL D

Internal Placards
Figure 2 (Sheet 6 of 14)



MOD S.O.O. 6269
(IAC AR REQUIREMENTS)
PRE MOD 6/2434



MOD S.O.O. 6269
(IAC AR REQUIREMENTS)
PRE MOD 6/2434

DETAIL D

Internal Placards
Figure 2 (Sheet 7 of 14)



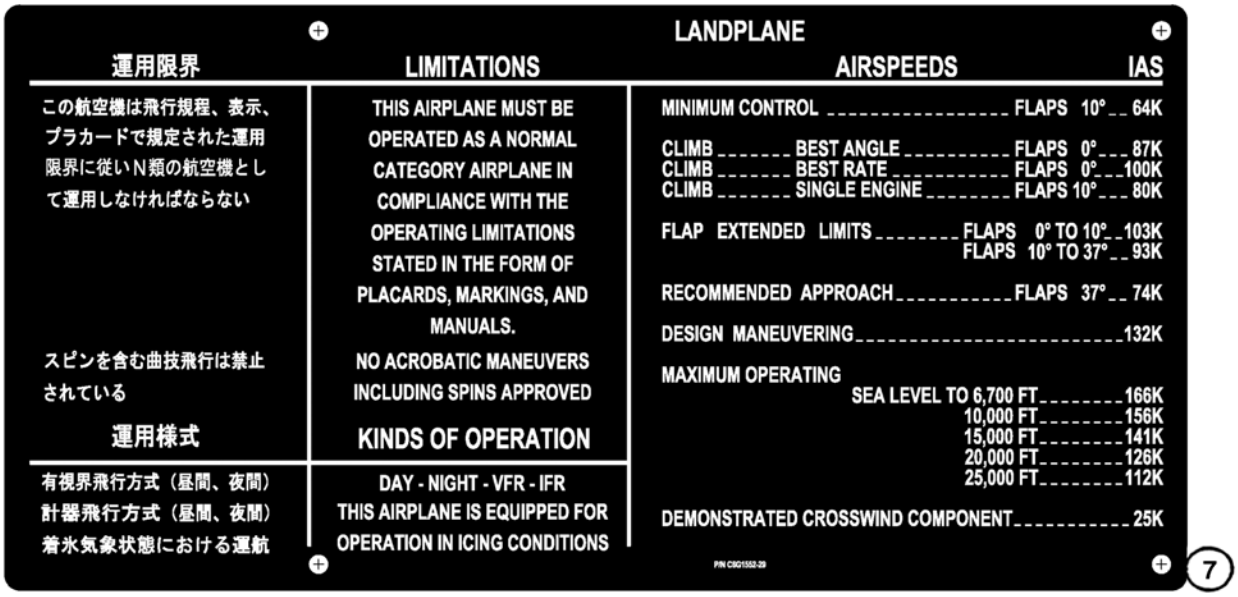
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(JCAB REQUIREMENTS)
PRE MOD 6/2434



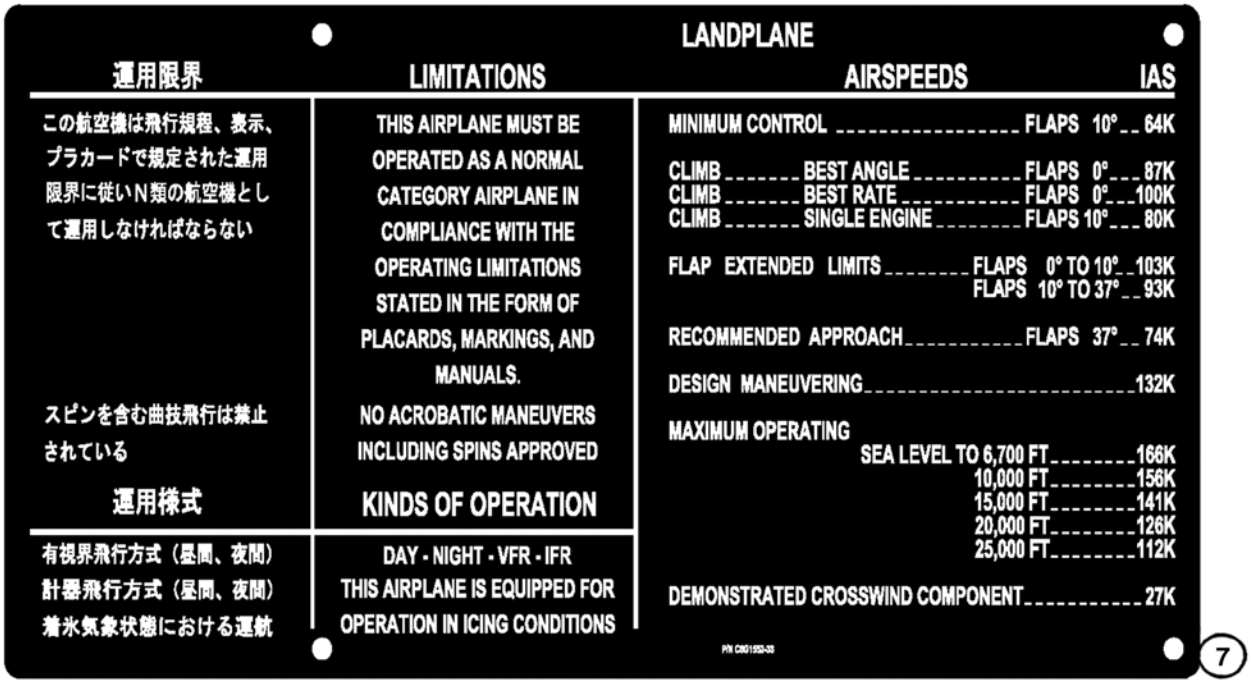
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(JCAB REQUIREMENTS)
POST MOD 6/2434

DETAIL D

Internal Placards
Figure 2 (Sheet 8 of 14)



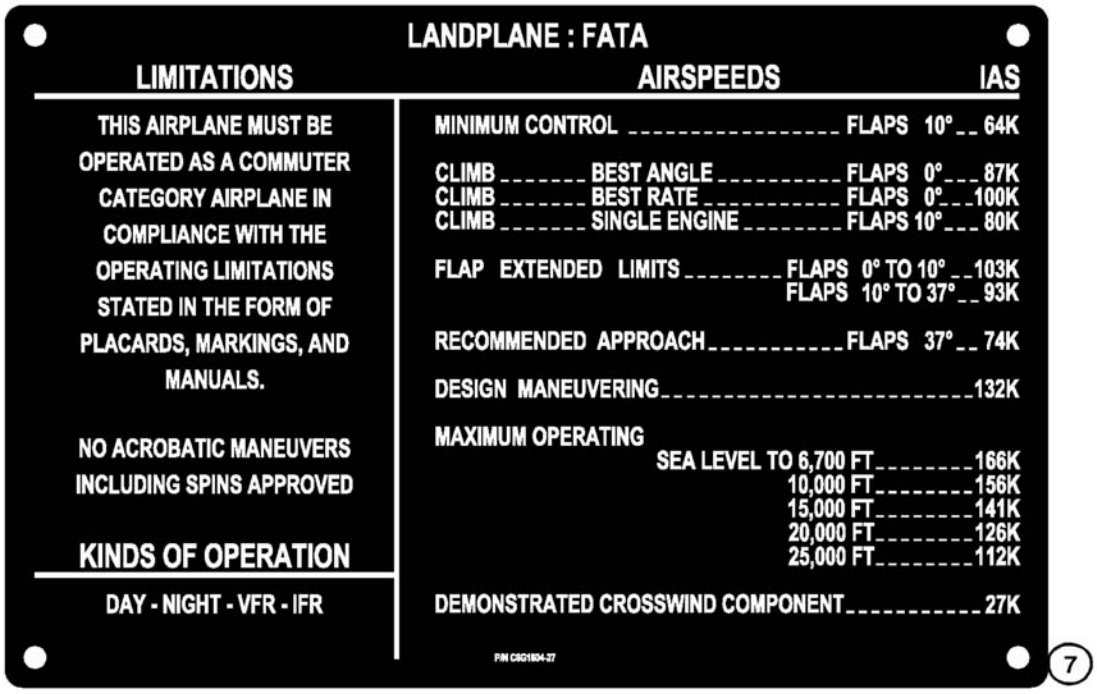
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(JCAB REQUIREMENTS)
PRE MOD 6/2434



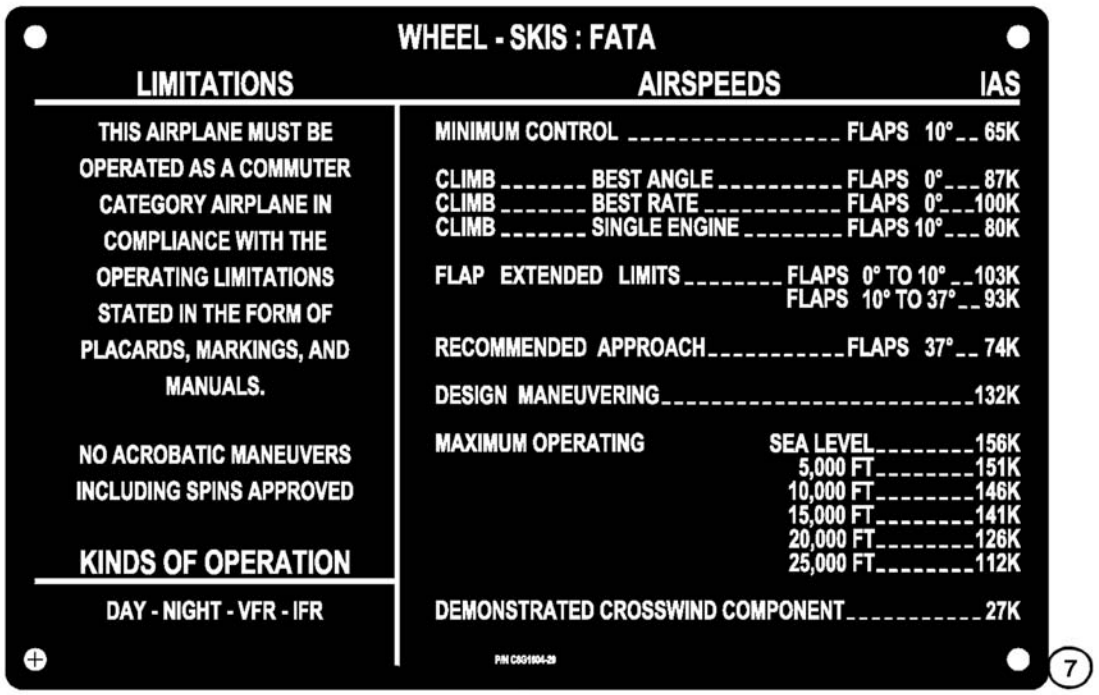
MOD S.O.O. 6328 WITH S.O.O. 6004, 6187 AND 6202 OR 6237
(JCAB REQUIREMENTS)
POST MOD 6/2434

DETAIL D

Internal Placards
Figure 2 (Sheet 9 of 14)



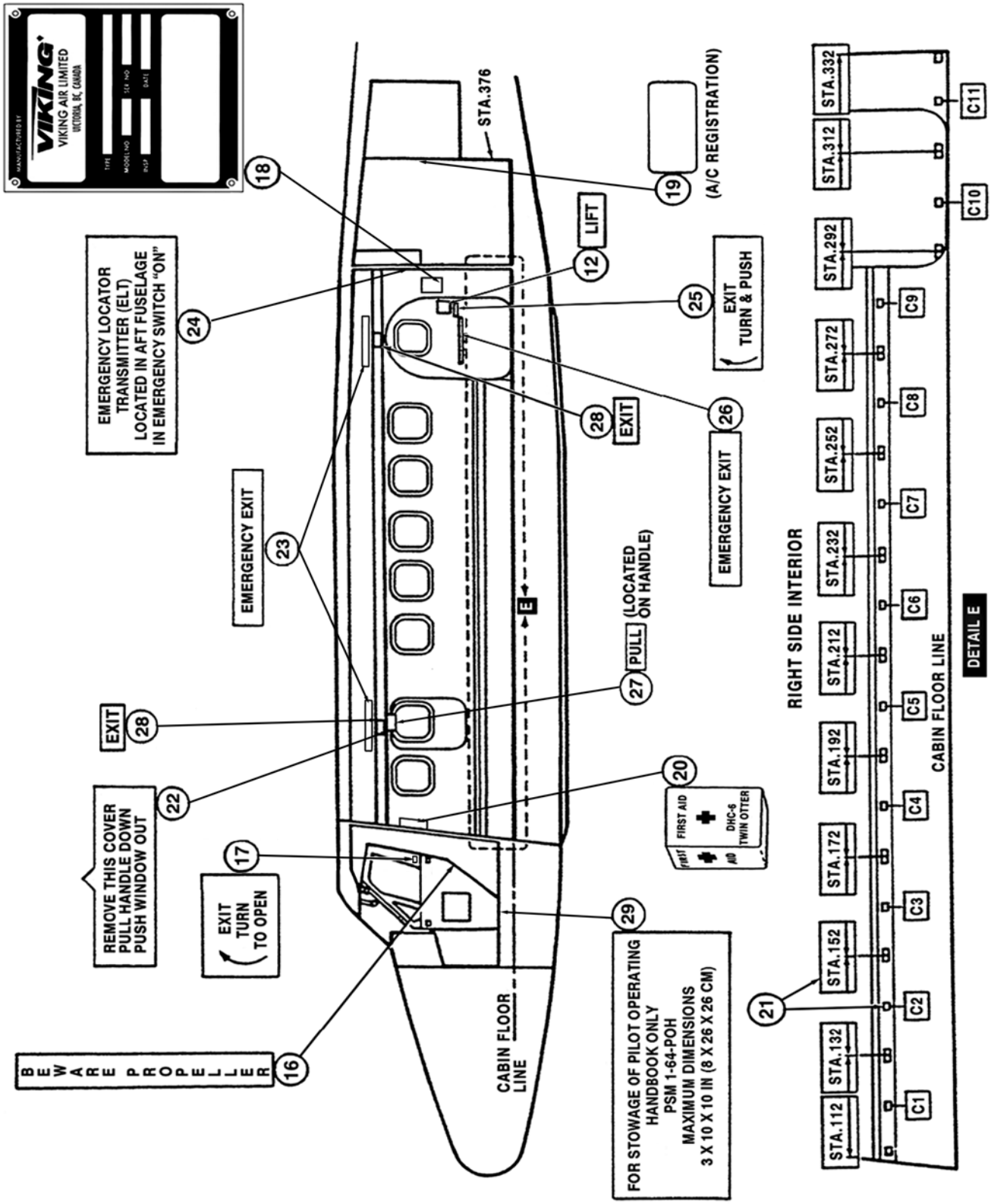
MOD S.O.O. 6411
(FATA REQUIREMENTS)



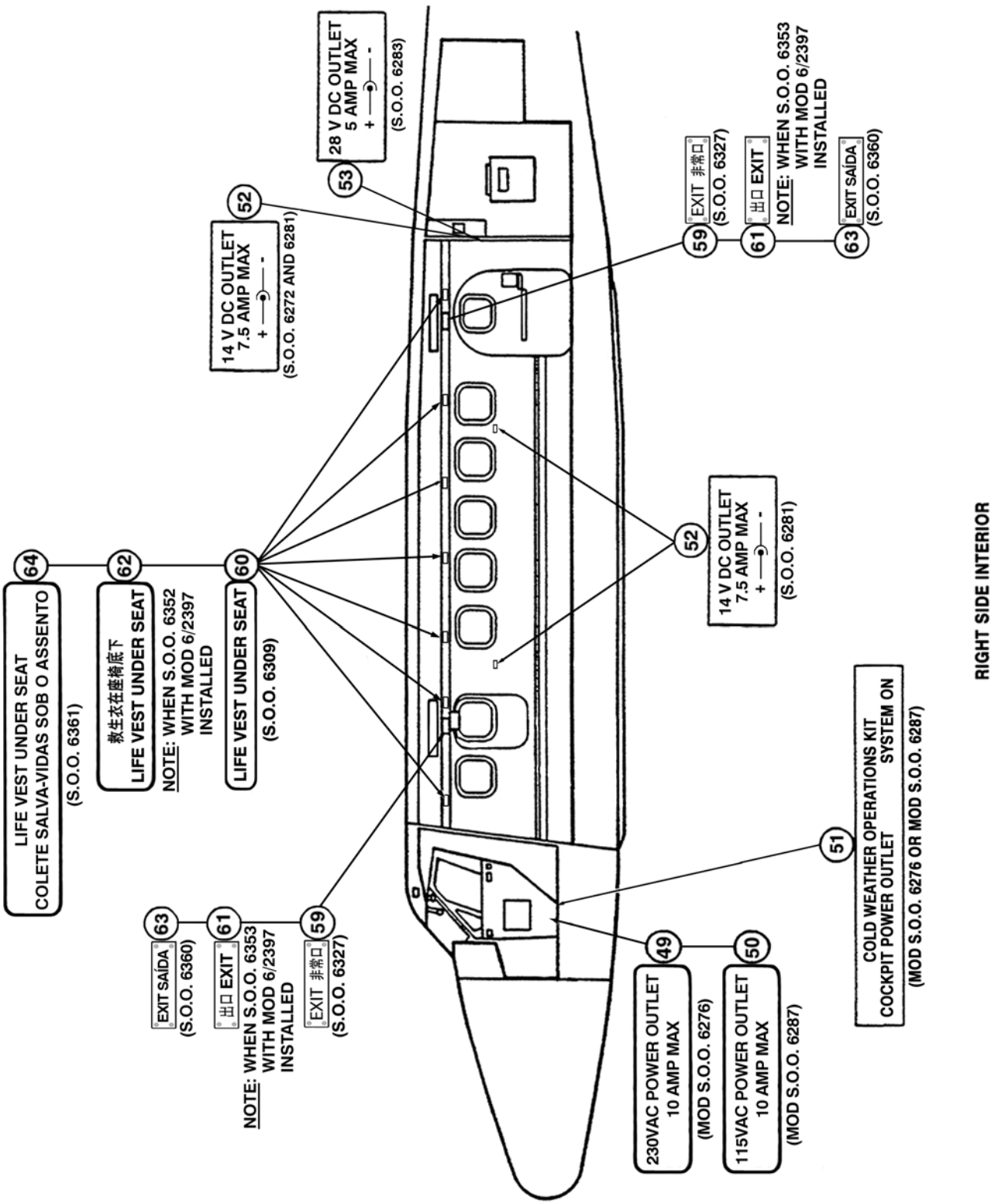
MOD S.O.O. 6411 AND S.O.O. 6001 WITH MOD 6/2182
(FATA REQUIREMENTS)

DETAIL D

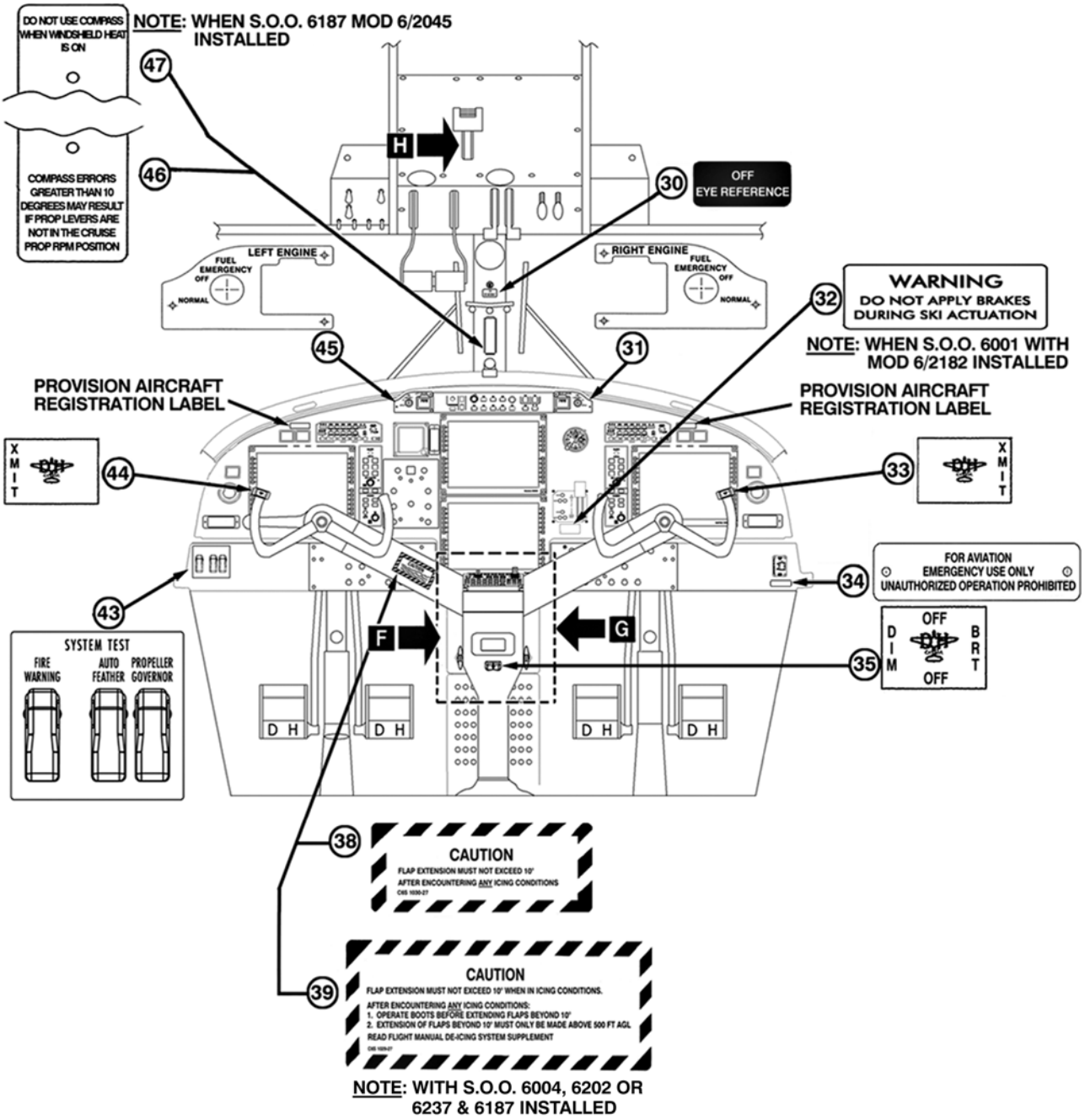
Internal Placards
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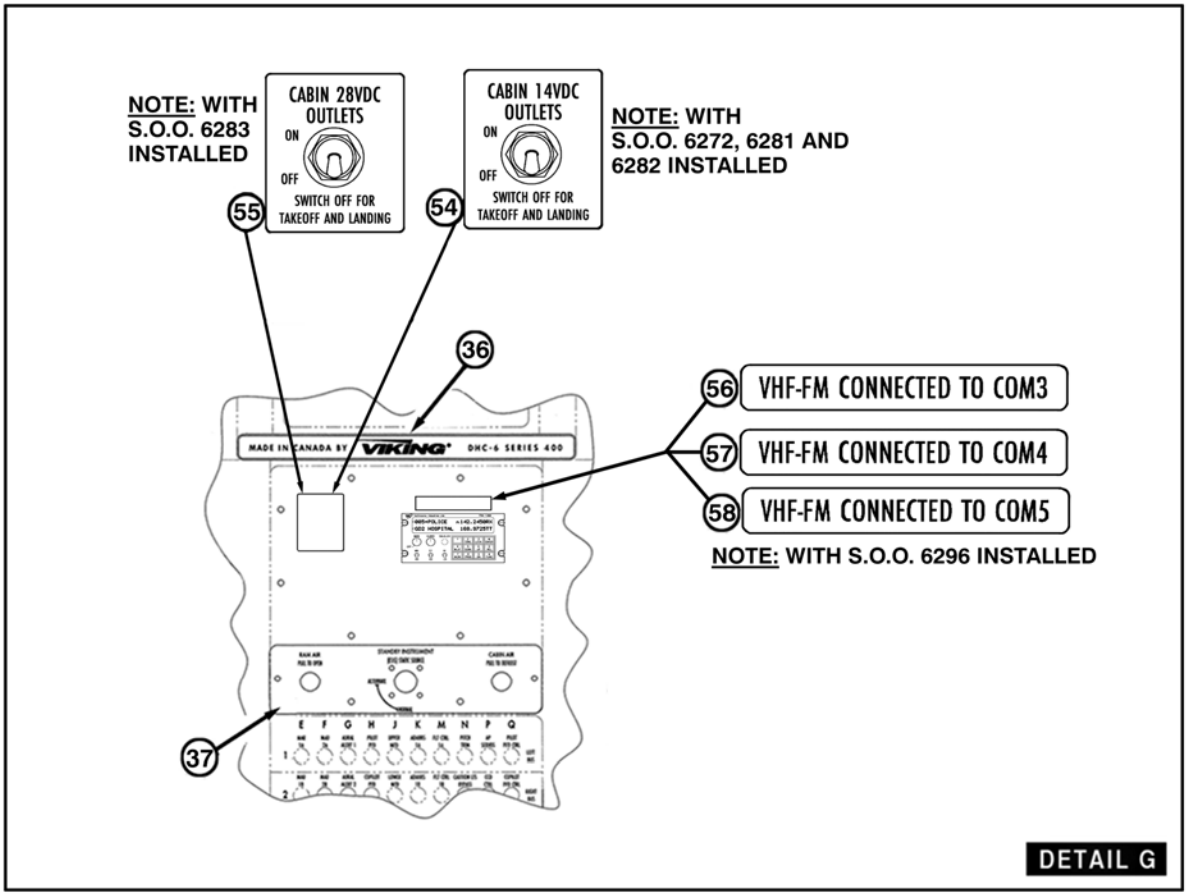
Internal Placards
Figure 2 (Sheet 11 of 14)

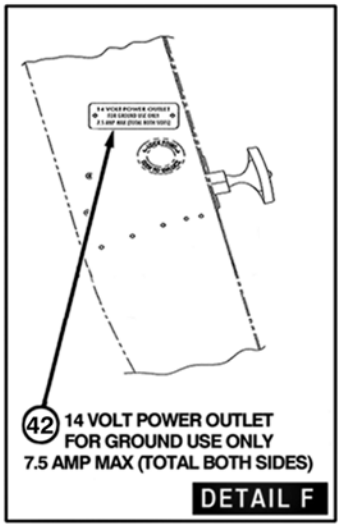


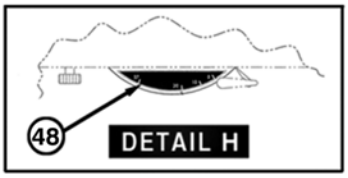
Internal Placards
Figure 2 (Sheet 12 of 14)

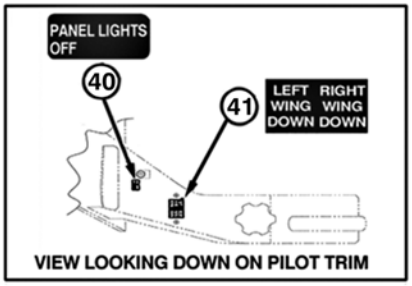


Internal Placards
Figure 2 (Sheet 13 of 14)









Internal Placards
Figure 2 (Sheet 14 of 14)

Table 2 Details of Internal Placards

FIGURE	NUMBER	TYPE	PART NUMBER
2	1	Label	C6–2020–35 (Pre Mod 6/2198)
		Label	C6–2020–49 (Post Mod 6/2198)
	2	Label	C6–2020–37 (Pre Mod 6/2198)
		Label	C6–2020–51 (Post Mod 6/2198)
	3	Label	C6–2020–53 (Large) (Post Mod 6/2198)
	4	Label	C6FF2201–35 (Pre Mod 6/2198)
		Label	C6FF2201–37 (Post Mod 6/2198)
	5	Label	C6–2020–29 (Pre Mod 6/2198)
		Label	C6–2020–43 (Post Mod 6/2198)
	6	Sign	CANAC–12A (Pre Mod 6/2308)
		Sign	C6SC1161–1 (Post Mod 6/2308)
	7	Placard	C6FF2965–27 (Pre Mod 6/2434)
		Placard	C6FF2965–31 (Post Mod 6/2434)
		Placard	C6FF2965–29 (Mod S.O.O. 6001 with Mod 6/2182) (Pre Mod 6/2434)
		Placard	C6FF2965–33 (Mod S.O.O. 6001 with Mod 6/2182) (Post Mod 6/2434)
		Placard	C6G1491–29 (Mod S.O.O. 6269 and S.O.O. 6001 with Mod 6/2182) (Pre Mod 6/2434)
		Placard	C6G1491–33 (Mod S.O.O. 6269 and S.O.O. 6001 with Mod 6/2182) (Post Mod 6/2434)
		Placard	C6G1491–27 (Mod S.O.O. 6269) (Pre Mod 6/2434)
		Placard	C6G1491–31 (Mod S.O.O. 6269) (Post Mod 6/2434)
		Placard	C6G1552–27 (Mod S.O.O. 6328) (Pre Mod 6/2434)
		Placard	C6G1552–31 (Mod S.O.O. 6328) (Post Mod 6/2434)
		Placard	C6FF1552–29 (Mod S.O.O. 6328 with S.O.O. 6004, 6187 and 6202 or 6237) (Pre Mod 6/2434)
		Placard	C6FF1552–33 (Mod S.O.O. 6328 with S.O.O. 6004, 6187 and 6202 or 6237) (Post Mod 6/2434)
		Placard	C6G1604–27 (Mod S.O.O. 6411)
		Placard	C6G1604–29 (Mod S.O.O. 6411 and S.O.O. 6001 with Mod 6/2182)

Table 2 Details of Internal Placards (continued)

FIGURE	NUMBER	TYPE	PART NUMBER
	8	Placard	C6–2021–49
	9	Label	C6JF1432–27 (Pre Mod 6/2198)
		Label	C6JF1432–31 (Post Mod 6/2198)
	10	Label	C6–2023–67
	11	Label	C6FF2961–29 (Mod 6/2092) (Pre Mod 6/2300)
	11A	Label	C6FF2976–27 (Post Mod 6/2300)
	12	Label	C6–2020–33 (Pre Mod 6/2198)
		Label	C6–2020–47 (Post Mod 6/2198)
	13	Label	C6–2020–31 (Pre Mod 6/2198)
		Label	C6–2020–45 (Post Mod 6/2198)
	14	Label	C6–1044–1 (For hardwall)
	15	Label	C6S1051–27 (S.O.O. 6004, 6187 and 6202 or 6237) (Mod 6/2228)
	16	Label	C6–2023–67
	17	Label	C6JF1432–29 (Pre Mod 6/2198)
		Label	C6JF1432–33 (Post Mod 6/2198)
	18	Label	VSP11–1 (Pre Mod 6/2345)
		Label	VSP11–3 (Post Mod 6/2345)
	19	Plate	CSP274–1
	20	Silkscreen	C6FF1697–37 (Front)
		Silkscreen	C6FF1697–39 (Side)
	21	Label	C6–1044–1 (For hardwall)
	22	Label	C6FF2201–35 (Pre Mod 6/2198)
		Label	C6FF2201–37 (Post Mod 6/2198)
	23	Label	C6–2020–53 (Large) (Post Mod 6/2198)
	24	Placard	C6–2021–57
	25	Label	C6–2020–37
	26	Label	C6–1004–31 (Small)
	27	Label	C6–2022–29
	28	Sign	CANAC–12A (Pre Mod 6/2308)
		Sign	C6SC1161–1 (Post Mod 6/2308)
	29	Placard	C6–2021–51
	30	Label	C6–2021–45
	31	Placard	C6JF1431–28

Table 2 Details of Internal Placards (continued)

FIGURE	NUMBER	TYPE	PART NUMBER
	32	Label	C6HF1055–31 (S.O.O. 6001 with Mod 6/2182)
	33	Label	C6–2021–36
	34	Placard	C6–2021–55
	35	Label	C6–2021–31
	36	Placard	C6JF1433–27
	37	Placard	C6–2021–27
	38	Label	C6S1030–27
	39	Label	C6S1029–27 (S.O.O. 6004, 6202 & 6187)
	40	Placard	C6–2021–43
	41	Label	C6–2021–39
	42	Placard	C6–2021–33
	43	Placard	C6–2021–37
	44	Label	C6–2021–35
	45	Placard	C6JF1431–27
	46	Placard	C6–2021–53
	47	Placard	C6–2021–29 (S.O.O. 6187, Mod 6/2045)
	48	Placard	C6–2021–47 (Left Hand)
		Placard	C6–2021–48 (Right Hand)
	49	Placard	C6G1512–29 (Mod S.O.O. 6287)
	50	Placard	C6G1512–31 (Mod S.O.O. 6276)
	51	Label	C6G1512–33 (Mod S.O.O. 6276 or Mod S.O.O. 6287)
	52	Placard	C6G1515–37 (S.O.O. 6272, 6281 or 6282)
	53	Placard	C6G1515–39 (S.O.O. 6283)
	54	Placard	C6G1515–41 (S.O.O. 6272, 6281 or 6282)
	55	Placard	C6G1515–43 (S.O.O. 6283)
	56	Placard	C6G1528–27 (S.O.O. 6296)
	57	Placard	C6G1528–29 (S.O.O. 6296)
	58	Placard	C6G1528–31 (S.O.O. 6296)
	59	Sign	C6SC1161–3 (S.O.O. 6327)
	60	Placard	C6G1546–27 (S.O.O. 6309)
	61	Sign	C6SC1161–11 (S.O.O. 6353 with Mod 6/2397)
	62	Placard	C6G1558–29 (S.O.O. 6352 with Mod 6/2397)

Table 2 Details of Internal Placards (continued)

FIGURE	NUMBER	TYPE	PART NUMBER
	63	Sign	C6SC1161–7 (S.O.O. 6360)
	64	Placard	C6G1563–27 (S.O.O. 6361)
	65	Placard	C6G1612–27 (S.O.O. 6299 with Mod 6/2436)

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CHAPTER 12

SERVICING

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CHAPTER 12 – SERVICING

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

REFERENCE	PAGE	DATE	REFERENCE	PAGE	DATE
12	* 1	Sep. 28/18		* 2	Sep. 28/18
	* 2	Blank Page	12-10-25	* 1	Sep. 28/18
	* 3	Sep. 28/18		* 2	Sep. 28/18
	* 4	Blank Page	12-20-05	* 1	Sep. 28/18
	* 5	Sep. 28/18		* 2	Sep. 28/18
	* 6	Sep. 28/18		* 3	Sep. 28/18
12-00-00	* 1	Sep. 28/18		* 4	Sep. 28/18
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	* 3	Sep. 28/18		* 6	Sep. 28/18
	* 4	Sep. 28/18		* 7	Sep. 28/18
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Servicing – General Data

1. General

Safety precautions to be observed by personnel working on the aircraft are given, if required, as part of the individual maintenance procedures. Danger areas are identified on the aircraft by red stencil markings, e.g. BEWARE PROPELLERS, marked on forward face of pilot's and co-pilot's door jamb.

2. Application of DC Power for Ground Servicing

CAUTION DO NOT COMMENCE WORK ON AIRCRAFT ELECTRICAL EQUIPMENT WITH ELECTRICAL POWER CONNECTED, UNLESS WORK CONSISTS OF TESTS REQUIRING USE OF ELECTRICAL POWER.

The following descriptions of external power, ground power and GND PWR all refer to the application of 28 Volt DC power to the aircraft using the external power receptacle. The external power receptacle is located on the left side of the fuselage aft of the cargo door. A spring loaded flap is fitted over the power receptacle to protect it from dirt and weather. For aircraft incorporating S.O.O. 6371, the external ground power receptacle is located forward with the main battery in the outer metallic nose structure, at approximately STA 55. A cover is fitted and is secured with quarter turn fasteners over the power receptacle to protect it from dirt, weather and water ingress.

A. Application of DC Power (Pre Mod 6/2314)

DC Power may be provided to the aircraft using either external power or battery power. If battery power is used, amount of time in use should be minimized to prevent excessive battery discharge.

1) To apply external DC Power proceed as follows:

a) Ensure that POWER SOURCE switch is set to OFF.

b) Connect 28 Volt DC power source to aircraft external DC power receptacle.

c) Ensure circuit breakers are engaged.

d) Set POWER SOURCE switch to EXTERNAL.

e) Set MASTER switch to ON; set BUS TIE switch to NORMAL.

f) After the cockpit display screens power on, check the ELECTRICAL window. GND PWR should be ON (green) and ground power voltage is displayed in the lower left.

g) When ground servicing is complete, disconnect DC Power:

1 Set POWER SOURCE switch to OFF.

2 Set MASTER switch to OFF.

2) To apply DC Power from the battery proceed as follows:

a) Ensure circuit breakers are engaged.

b) Set POWER SOURCE switch to BATTERY.

c) Set MASTER switch to ON; set BUS TIE switch to NORMAL.

d) After the cockpit display screens power on, check the ELECTRICAL window. Battery VOLTS and AMPS are displayed in the lower area of the window. Negative AMPS readings indicate battery discharge (e.g. 34 A).

e) When ground servicing is complete, disconnect DC Power:

1 Set POWER SOURCE switch to OFF.

2 Set MASTER switch to OFF.

B. Application of DC Power (Post Mod 6/2314)

DC Power may be provided to the aircraft using external power, battery power or both. If battery power is used without external power, amount of time in use should be minimized to prevent excessive battery discharge.

1) To apply external DC Power proceed as follows:

a) Ensure that EXTERNAL switch is set to OFF.

b) Ensure that AVIONICS START BATTERY switch is set to OFF (Post Mod 6/2315).

c) Connect 28 Volt DC power source to aircraft external DC power receptacle.

d) Ensure circuit breakers are engaged.

e) Set EXTERNAL switch to ON and confirm that the BUS TIE switch is set to NORMAL.

f) After the cockpit display screens power on, check the ELECTRICAL window. GND PWR should be ON (green) and ground power voltage is displayed in the lower left.

g) When ground servicing is complete, disconnect DC Power:

1 Set EXTERNAL switch to OFF.

2) To apply DC Power from the battery proceed as follows:

a) Ensure circuit breakers are engaged.

b) Ensure that AVIONICS START BATTERY switch is set to OFF (Post Mod 6/2315).

c) Set BATTERY switch to ON and confirm that the BUS TIE switch is set to NORMAL.

d) After the cockpit display screens power on, check the ELECTRICAL window. Battery VOLTS and AMPS are displayed in the lower area of the window. Negative AMPS readings indicate battery discharge (e.g. 34 A).

e) When ground servicing is complete, disconnect DC Power:

1 Set BATTERY switch to OFF.

3) To apply DC Power from the battery and external proceed as follows:

a) Ensure that BATTERY and EXTERNAL switches are both set to OFF.

b) Ensure that AVIONICS START BATTERY switch is set to OFF (Post Mod 6/2315).

c) Connect 28 Volt DC power source to aircraft external DC power receptacle.

d) Ensure circuit breakers are engaged.

e) Set BATTERY switch to ON and confirm that the BUS TIE switch is set to NORMAL.

f) After the cockpit display screens power on, check the ELECTRICAL window. Battery VOLTS and AMPS are displayed in the lower area of the window. If external voltage is present at the DC power receptacle, the bottom of the screen will display GROUND POWER AVAILABLE and the ground power voltage is displayed.

g) If the ground power voltage is higher than the battery voltage (between approximately one and five volys higher), then the EXTERNAL may be switched ON.

h) Review the ELECTRICAL display screen. GND PWR should be green, BATTERY VOLTS and GND PWR VOLTS should be approximately equal (within 0.2 volts) and BATTERY AMPS should be a positive number, indicating that the battery is being charged by the GND PWR.

Note Avoid excessive battery charge currents using external power. Typically charging currents should be below 50 AMPS within one or two minutes of BATTERY and EXTERNAL both being switched ON.

i) When ground servicing is complete, disconnect DC Power:

1 Set BATTERY switch to OFF.

2 Set EXTERNAL switch to OFF.

3. Access and Inspection Provisions

CAUTION BEFORE REFITTING ACCESS PANELS, CHECK FOR INTERNAL CLEANLINESS TO ENSURE THAT NO TOOLS, LOCKWIRE, ETC. HAVE BEEN LEFT INSIDE THE STRUCTURE. ALSO CHECK THAT FUEL DRAIN VALVES ARE LOCKWIRED IN THE CLOSED POSITION.

Removable panels necessary for access to equipment for inspection and maintenance purposes are shown in Figure 1 and Table 1 — Key to Figure 1. Access and Inspection Provisions. Interior upholstery is removable in all places where access is necessary for maintenance.

4. Servicing Provisions

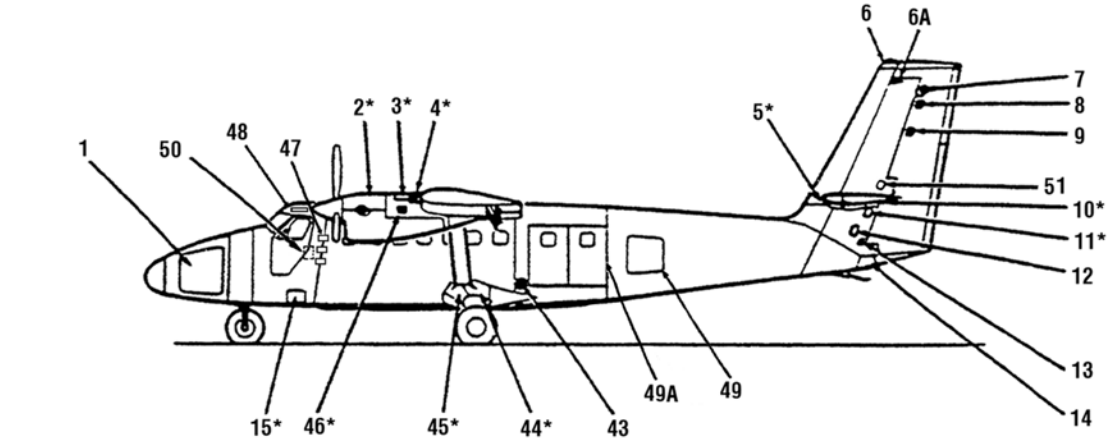
For servicing provisions refer to Figure 2.

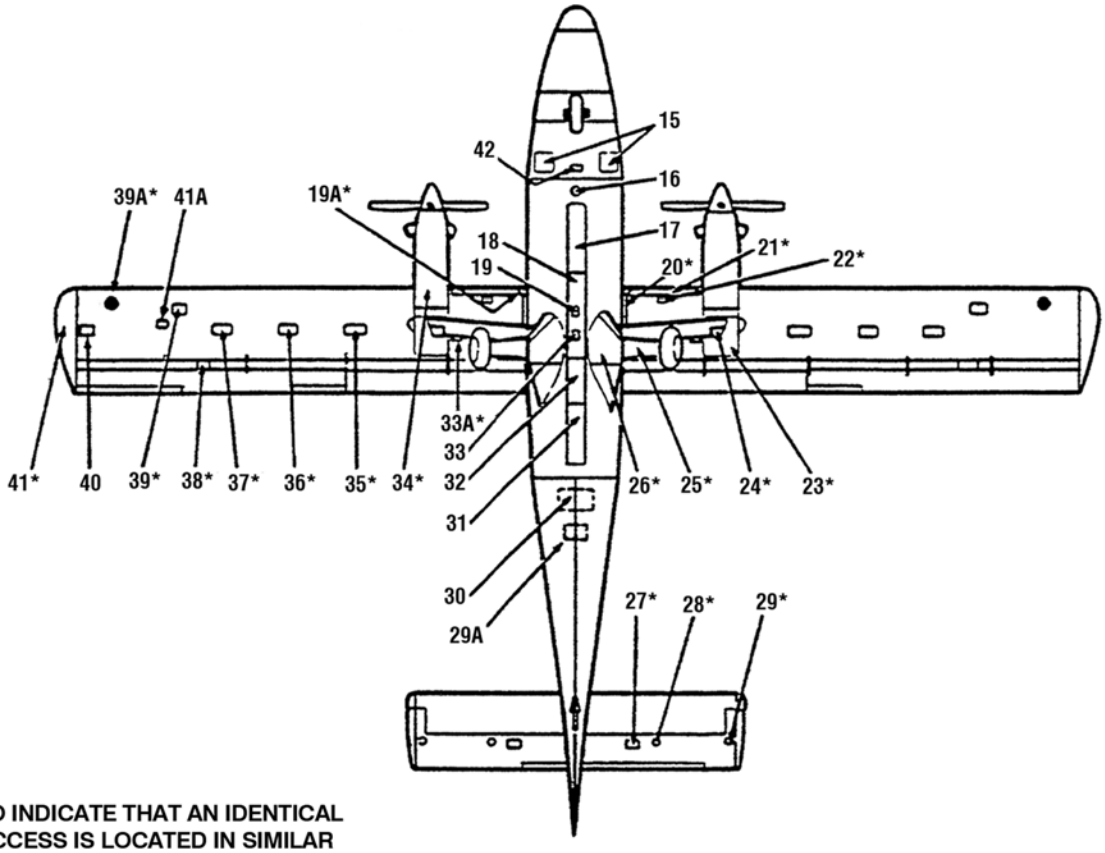
5. Application of Roof Access Steps

The roof access steps are located on the left side of the fuselage aft of the cockpit door. A handle hold is installed on the roof panel above the cockpit door to aid the usage of the roof access steps.

The function of the roof access steps is to provide easy access to the upper fuselage panels.

WARNING PROPER FALL ARREST SAFETY EQUIPMENT SHOULD BE USED WHEN ACCESS IS REQUIRED TO THE UPPER FUSELAGE.





* TO INDICATE THAT AN IDENTICAL ACCESS IS LOCATED IN SIMILAR POSITION ON OPPOSITE SIDE.

ONE OR BOTH SIDES

OPPOSITE SIDE ONLY

INTERNAL

Access and Inspection Provisions
Figure 1

Table 1 Key to Figure 1. Access and Inspection Provisions

1.	Nose Gear Shock Strut Upper Mounting Bolts Air Inflation Valve Nose Wheel Steering Actuator, Cable and Pulleys Hydraulic Lines Static Manifold, Pitot and Static Lines Caution Light Control Panel Nose Compartment Utility Light Electrical Wiring Radio Equipment Fwd Avionics Shelves and Maintenance Panel
2.	Propeller Governor, Propeller Overspeed Governor, Breather Propeller Control Propeller Tachometer Generator Engine Front Lifting Eye Vibration Isolators Spark Ignitors Torque Pressure Transmitter
3.	Engine Control Cables and Pulleys Engine Rear Lifting Eye Starter-Generator and Wiring Fuel Pump Oil Scavenge Return Line Oil Tank Filler Cap and Dipstick
4.	Engine Controls
5.	Vertical Stabilizer Forward Attachment Horizontal Stabilizer Forward Attachment Electrical Wiring Apex Magnetometer x2 (Post Mod 6/2171)
6.	Anti-Collision Light
6A.	VOR/LOC Antenna Cable Harness (Mod 6/1473)
7.	Rudder Top Hinge
8.	Rudder Trim Tab Screw Jack
9.	Rudder Trim Tab Turnbuckles

Table 1 Key to Figure 1. Access and Inspection Provisions (continued)

10. Horizontal Stabilizer Rear Attachment
Elevator and Flap/Elevator Interconnect Trim Cables
Elevator Torque Tube Bellcrank and Support Bracket

11. Elevator Quadrant
Rudder Trim Tab Control Cables

12. Rudder Cables and Pulleys
Elevator Trim Tab Control Cables and Pulleys

13. Rudder Control Cable Attachment Bracket
Rudder Lower Hinge
Rudder Stops
Rudder Geared Tab Gearbox

14. Tail Bumper Shock Strut Attachment
Tail Light Electrical Wiring

15. Hydraulic Power Package
Flying Control Cables and Pulleys
Rudder Gust Lock Mechanism
Electrical Wiring
Heating System:

- Hot Air Valve
- Expansion Chamber
- Recirculated Air Duct
- Silencer
- Ram Air Valve
- Cabin Air Control Valve
- Ejector
- Main Duct Fan

16. Electrical Wiring

17. No. 1, 2, and 3 Fuel Cells, Sump Plates and Tank Units
Forward Tank Interconnector Manifold
Forward Tank Fuel Ejector

18. No. 4, 5, and 6 Fuel Cells, Sump Plates and Tank Units
Forward and Aft Tank Interconnect Drain Valves
Forward and Aft Tank Booster Pumps
Fuel Pressure Switches
Crossfeed Valve

Table 1 Key to Figure 1. Access and Inspection Provisions (continued)

19.	Forward Tank Drain Valve Forward Tank Booster Pumps Drain Valves
19A.	Wing-to-Fuselage Electrical Connectors
20.	Wing-to-Fuselage Front and Rear Attachments Aileron Cables Bleed Air Line Flap Push-Pull Rod Electrical Wiring De-icing Line
21.	Engine Control Cables and Fairleads Electrical Wiring
22.	Bleed Air Line De-icing Line
23.	Wing Strut Upper Attachment Fuel Main Feed Line
24.	Wing Strut
25.	Brake Hydraulic Line Wheel-Ski Hydraulic Lines and Electrical Wiring (if fitted)
26.	Main Landing Gear Attachment Shock Absorber Brake Hydraulic Line Jacking Point
27.	Flap/Elevator Interconnect Jack (Right Side) Elevator Trim Jack (Left Side)
28.	Elevator Mid Hinge Attachment
29.	Elevator Outboard Hinge Attachment
29A.	Rudder and Elevator Cables
30.	Main Battery

Table 1 Key to Figure 1. Access and Inspection Provisions (continued)

31.	No. 8 Fuel Cell. Sump Plate, and Tank Unit Aft Tank Fuel Ejector Aft Tank Interconnector Manifold
32.	No. 6 and 7 Fuel Cells, Sump Plate, and Tank Unit Aft Tank Interconnect Manifold
33.	Aft Tank Drain Valve Aft Tank Booster Pumps Drain Valves
33A.	Aileron Control Cables and Fairleads Flap Inboard Operating Bellcrank
34.	Fuel Strainer and Fuel Lines Fuel Purge Connections Firewall Shut-off Valve Fire Extinguisher Bottle Ignition Current Regulator Box Gas Generator Tachometer Generator Oil Cooler and Oil Lines Vibration Isolators Electrical Wiring
35.	Aileron Control Cables and Fairleads Flap Idler Lever and Push-Pull Rod
36.	Aileron Control Cables and Fairleads Flap Idler Lever and Push-Pull Rod
37.	Aileron Control Cables and Pulleys Flap Outboard Operating Bellcrank
38.	Aileron Linkage and Flap Outboard Attachment
39.	Wing Electrical Wiring Conduit De-icing Distributor Valve Wing De-icing Water Separator (Mod 6/1440)
39A.	Wing Tank Components (Mod S.O.O. 6247)
40.	Wing Tank Capacitance Unit Level Control (Mod S.O.O. 6247)

Table 1 Key to Figure 1. Access and Inspection Provisions (continued)

41. Navigation Light
Wing Tank Components (Mod S.O.O. 6247)

- Wing Tank Low Pressure Switch
- Fuel Transfer Valve
- Check Valve
- Pressure Pump
- Refuel Shut-off Valve
- Radio Noise Filter (Mod 6/1374)
- Wing Tank Fuel Filter (Mod 6/2006)

41A. Standby Magnetometer One Side (Mod 6/2049) Apex Magnetometer x2 other side (Mod 6/2049) (Pre Mod 6/2171)

42. Hydraulic System Filter

43. Rudder and Elevator Control Cables and Pulleys

44. Fuel Main Feed Line
Wheel Brake Hydraulic Line
Shock Absorber

45. Wing Strut Lower Attachment

46. Engine Oil Filter

47. Roof Access Steps

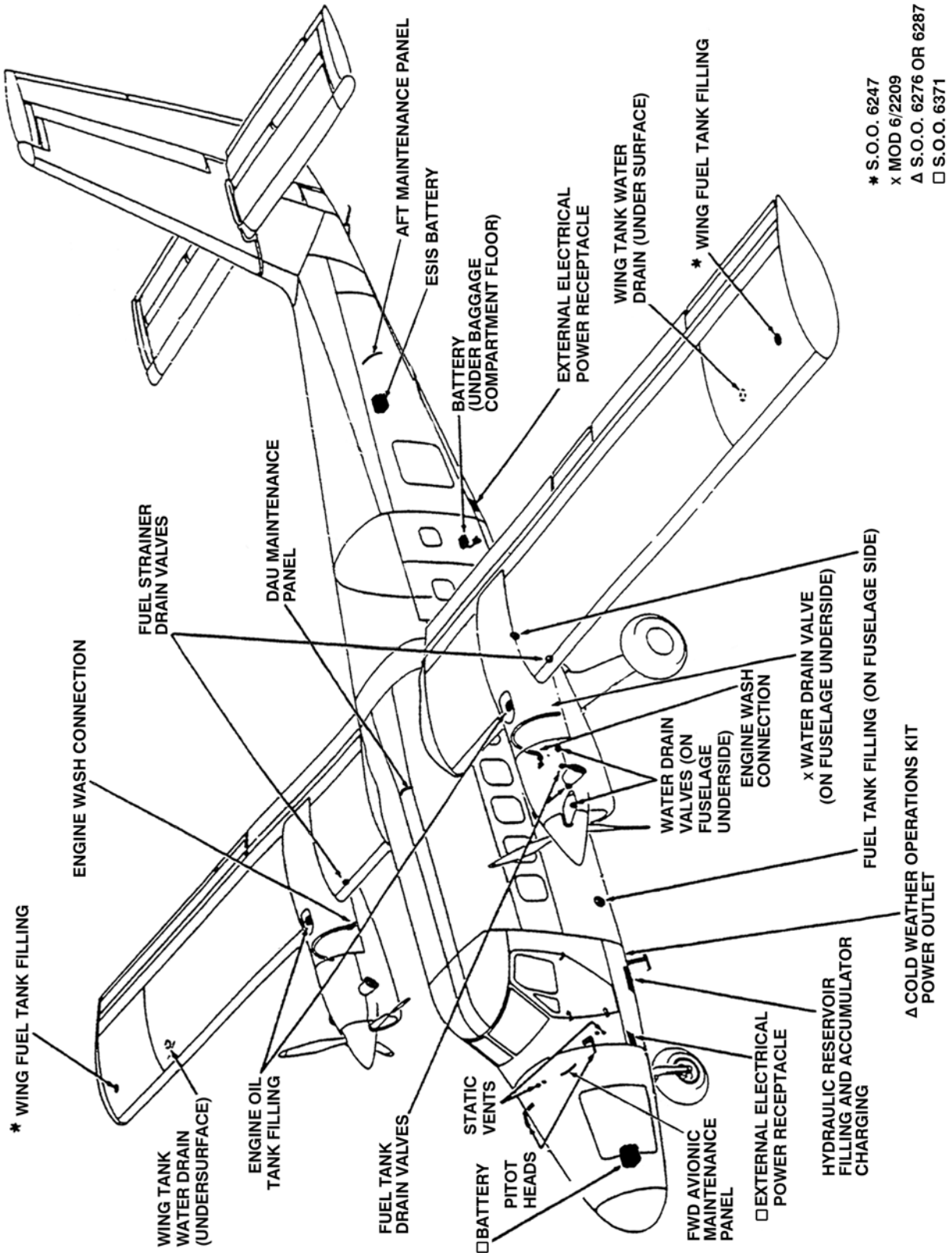
48. Handle Hold

49. Aft Avionics Shelves and Maintenance Panel at Station 421.00
Rudder and Elevator Servo's (S.O.O. 6221)
Rudder Trim Servo (S.O.O. 6221)

49A. For Aircraft with S.O.O. 6393:
Aft Avionics Shelves and Maintenance Panel at Station 421.00
Rudder and Elevator Servo's (S.O.O. 6221)
Rudder Trim Servo (S.O.O. 6221)

50. Aileron Servo (Co-pilot side) (S.O.O. 6221)

51. Elevator Position Transmitter (if fitted with Mod 6/2417)



Servicing Diagram
Figure 2

Fuel System – General Data

1. General

Fuel specifications and tank capacities are given in Table 1 — Fuel Specifications and Grade and Table 2 — Fuel Tank Capacities and Fuel Weights JP–4 Fuel. The fuselage tank fuel filler caps are located on the left side of the fuselage, one forward of the wing strut and the other aft. When wing tanks are fitted (Mod S.O.O. 6247) a filler cap is located on each outer wing upper surface.

Table 1 Fuel Specifications and Grade

Canadian Specification	U.S. Specification	British Specification	NATO Symbol	Civil Specification
3–GP–22b	MIL–T–5624G	D Eng RD2486	F–40	Turbine Fuel JP–4
Note Aviation Kerosene or JP–5 may also be used. MIL–G–5572C AVGAS (all grades) is allowed for emergency use only and is limited to a total of 150 hours in any one overhaul cycle.				

Table 2 Fuel Tank Capacities and Fuel Weights JP–4 Fuel

Location	Approximate Weight (7.8 Lb./Imp Gal)	Capacity (Gallons) U.S.	Imperial
Forward Tank	1185	183	152
Aft Tank	1295	200	166
Wing Tanks (if fitted)	577	89	74
Total With Wing Tanks	3057	472	392
Total Without Wing Tanks	2480	383	318
Note Weight (approximately) of alternate fuels are as follows:			
Aviation Kerosene	8.2 lb/Imperial Gallon		
JP–5	8.3 lb/Imperial Gallon		
AVGAS (Emergency)	7.2 lb/Imperial Gallon		

2. Refueling

A. Refueling Precautions

WARNING A CHAMOIS OR FELT FILTER MUST NOT BE USED AS THESE CONTRIBUTE TO CONTAMINATION AND IN ADDITION, ELECTROSTATICALLY CHARGE THE FUEL.

Before attempting to refuel aircraft, observe the following precautions:

1) No smoking in the vicinity of the aircraft.

2) Position a suitable fire extinguisher in the immediate area of the aircraft, and ensure personnel involved are acquainted with its use.

3) Ground both aircraft and fuel tender.

4) Ensure aircraft electrical power is OFF.

5) Aircraft with Cold Weather Operations Kit installed (S.O.O. 6276 or 6287) ensure system is disconnected.

6) Before placing nozzle in the filler neck, ground nozzle.

7) Avoid carrying out other work on aircraft during refueling operations.

8) Avoid spilling fuel.

Note It is important that only clean fuel be used. If the presence of contaminates is suspected, tanks should be filled through a funnel with a 100 – 200 mesh metal screen.

B. Filler Neck Refueling

1) Observe refueling precautions. Refer to Para 2.A.

2) Ensure external power supply to aircraft is disconnected and aircraft supply is OFF.

3) Ground supply nozzle to ground socket, remove filler cap, insert nozzle and fill tank. When tank is full, remove nozzle, replace and secure filler cap and disconnect ground plug from ground socket.

4) Repeat Para 2.B.3) for other tank (and wing tanks if fitted) as necessary.

C. Alternate Method of Wing Tank Refueling

If over wing refueling of wing tanks cannot be carried out, the wing tanks may be refueled from the fuselage tanks as follows:

1) Apply DC power. (Refer to 12-00-00, Servicing – General Data, Para 2.).

2) Position fuel levers in the overhead console to OFF and select fuselage tank BOOST PUMP 1 switches to ON.

3) Wing fuel tank switches – REFUEL (left and/or right).

Note Using above procedure, the forward fuselage tank will refill the right wing tank and the rear fuselage tank will refill the left wing tank.

4) Observe that the wing tank fuel indications register gradual increase in contents, and that AFT and FWD (fuselage tank) fuel quantity indicators register a corresponding decrease. When tank content indicators show no further change, select wing fuel tank switch(es) and boost pump switch(es) to OFF.

Note As above operations require 15 to 20 minutes to completely fill a wing tank, an external power source should be used to power the electrical system.

3. Draining the Fuel System

CAUTION OBSERVE PRECAUTIONS DETAILED IN Para 2.A.

To drain fuselage fuel tanks, use the drains on the sump plates of fuel cells No. 4 and No. 5. No. 4 cell drains the forward tank and No. 5 cell drains the aft tank. To drain a wing tank (if fitted) switch on battery supply or connect an external electrical power source to aircraft, disconnect flexible hose at nacelle fuel filter tee (cap flexible hose), attach suitable length of hose to filter tee and run open end into container. Switch wing fuel tank switch to L or R ENGINE as applicable. When draining is complete select switch OFF, disconnect electrical supply, connect and secure flexible hose to filter tee.

4. Draining of Water and Contaminates from Fuel System

A. General

Drain valves are provided at various locations in the fuel system to allow removal of accumulated water and/or contaminates.

B. Locations of Water Drain Valves

Drain valves are located on the two main fuel filters and under No. 4 and No. 5 cells on the forward and aft fuselage tank interconnect manifolds, respectively, for all aircraft models.

On aircraft with Mod S.O.O. 6247 incorporated, an additional drain valve is provided on lower surface of each integral wing tank.

On aircraft with Mod S.O.O. 6048, an additional drain valve (Mod 6/1498) is provided under No. 8 cell on the aft fuselage tank interconnector manifold.

On aircraft with Mod 6/2209 incorporated, an additional drain valve is provided under No. 8 cell, on the forward and aft fuselage interconnector manifolds.

C. Draining Procedure

CAUTION OBSERVE PRECAUTIONS DETAILED IN Para 2.A.

Water and/or contaminants must be drained from integral wing tanks, fuel strainers, and fuselage tank interconnector manifolds at regular intervals as prescribed in the Inspection Schedule. To accomplish this, open water drain valves provided, keep valves open until clean fuel only is ejected and close valves.

Note Due to the higher locations of fuel filters in engine nacelles, water and/or contaminants from filters may be drawn into fuel tanks when associated drain valves are opened. To prevent contamination of fuel tanks, booster pumps must be operating before fuel filter drain valves are opened.

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Hydraulic System – General Data

1. General

   The hydraulic system is supplied by one reservoir mounted on the hydraulic power package which is installed beneath the flight compartment floor. The reservoir is accessible through an access panel located below the pilot's door (refer to Figure 1). The fluid level is recorded on a dipstick which is attached to the reservoir filler cap.

2. Checking Fluid Level

   A. Ensure that brakes are off and flaps up.

   B. Ensure discharge of brake and system damping accumulators by operating the brakes and wing flaps until accumulators are completely discharged (brake and hydraulic pressure indicators read zero).

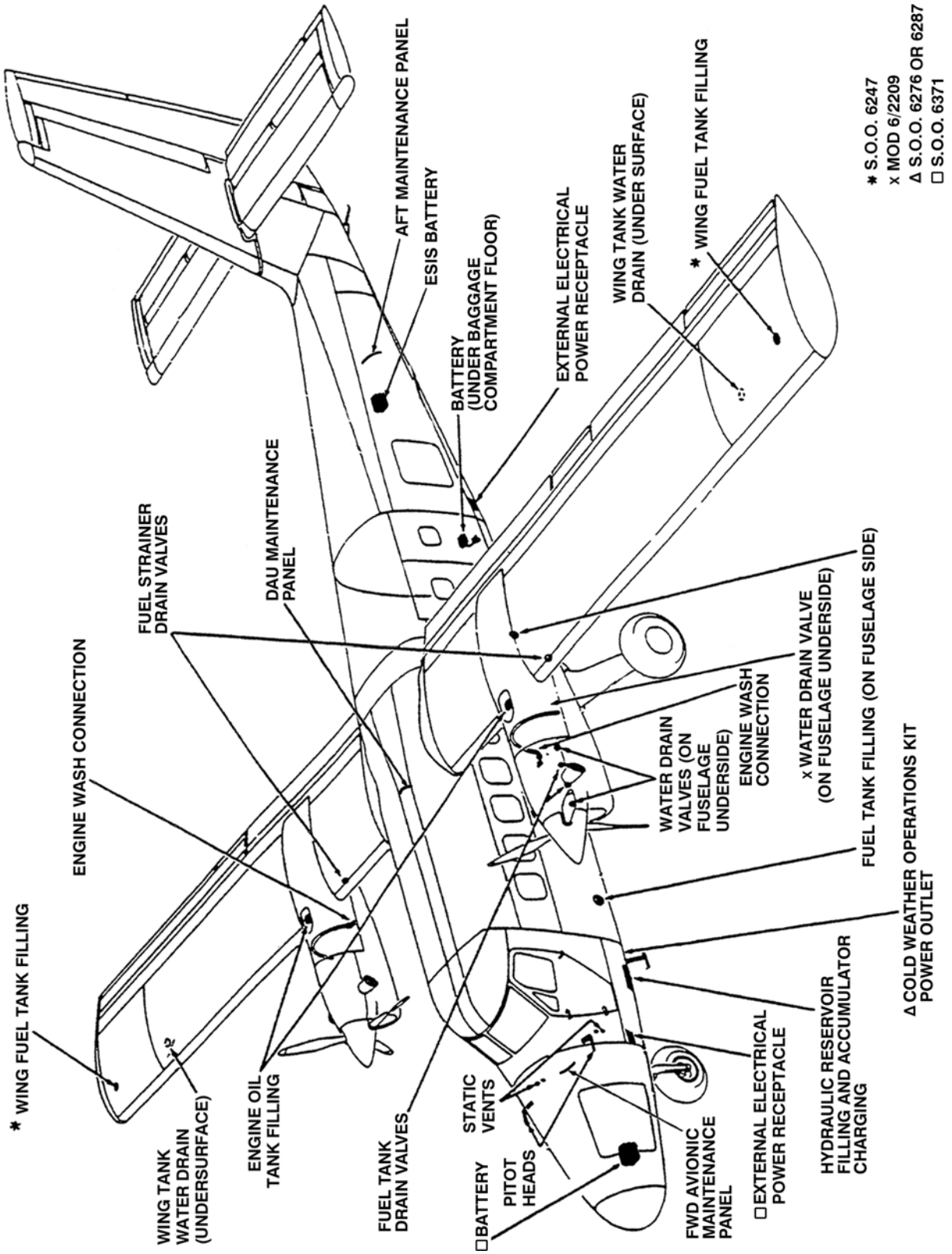
   C. Unscrew hydraulic reservoir filler cap and check that fluid level is between REFILL and FULL marks on dipstick. Add hydraulic fluid. Specification MIL–H–5606, as necessary.

3. Charging the Wheel Brakes Accumulator

   The wheel brakes accumulator is mounted on the hydraulic power package located beneath the flight compartment floor. A remote air charging valve and pressure indicator are located at the left of the accumulator. To charge the accumulator, release accumulator hydraulic pressure by operating the brake pedals, then charge the accumulator with nitrogen or dry air to 700 to 800 psi.

4. Charging the Damping Accumulator

   The damping accumulator is mounted on the hydraulic power package alongside the wheel brakes accumulator, and is accessible through the same access door. To charge the accumulator, release accumulator hydraulic pressure by operating the wing flaps, then charge the accumulator with nitrogen or dry air to 700 to 800 psi.



Servicing Diagram
Figure 1

Landing Gear – General Data

1. General

   Replenishing the landing gear is confined to inflating the tires and servicing the nose gear shock strut. The main gear shock strut, because of its design, does not require servicing.

2. Servicing Nose Gear Shock Strut

   CAUTION **DO NOT FILL SHOCK STRUT WITH OIL UNDER PRESSURE AS THIS CAN RESULT IN ITS FAILURE TO TELESCOPE. OIL FILLING BY GRAVITY IS THE ONLY ACCEPTED PROCEDURE.**

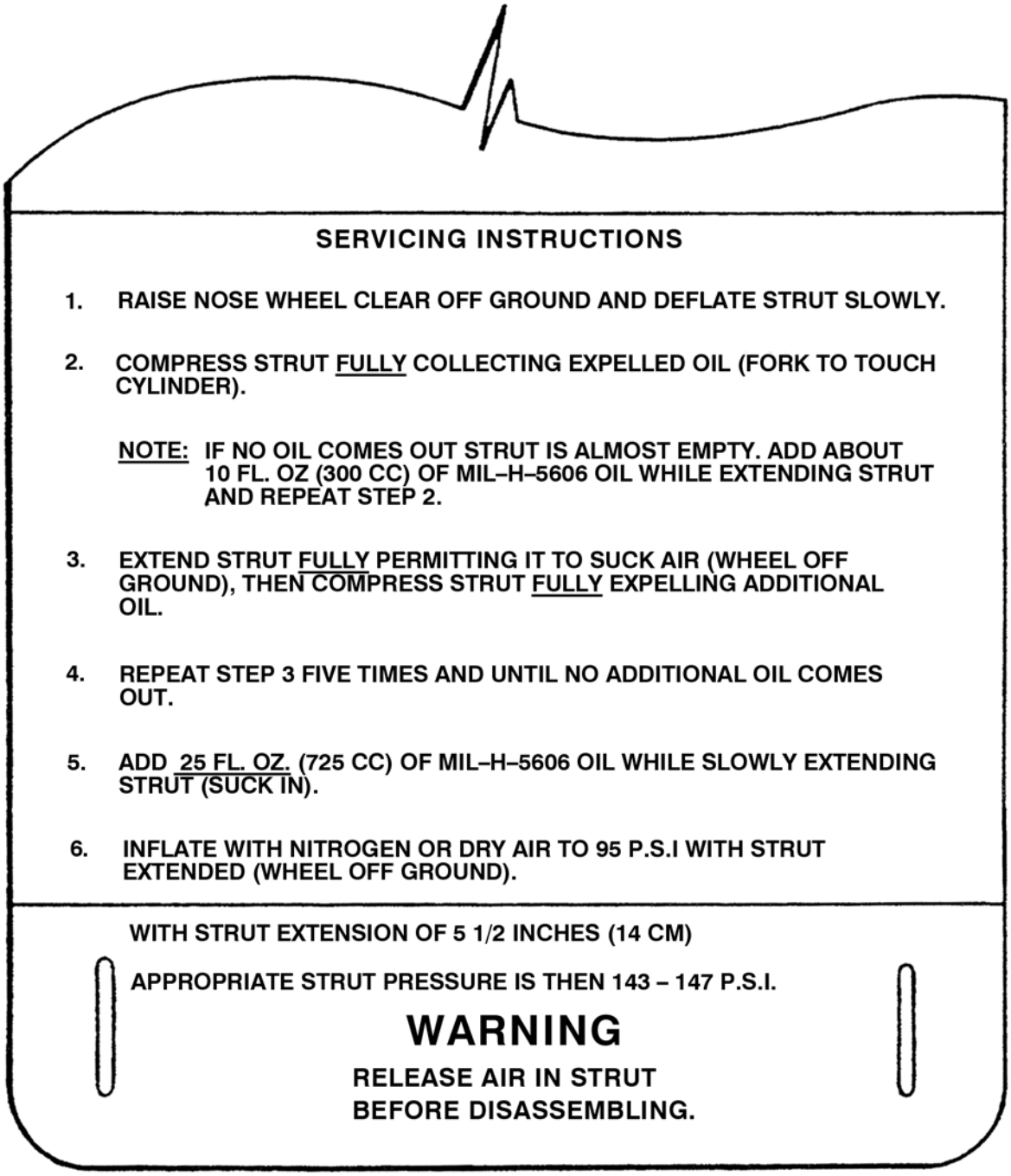
   The nose gear shock strut must be serviced in accordance with instructions on the nose gear identification plate. (Refer to Figure 1).

   Note Dry compressed nitrogen is preferred to compressed air for charging the shock strut.

3. Tires

   Inflation pressures for normal operating conditions, using standard size tires, are 32 psi for nose wheel tire and 38 psi for main wheel tires. During operation at airports where the temperature is below –20°F (–29°C), inflation pressure for main wheel tires must be reduced to 34 psi to meet landing energy absorption requirements.

   Inflation pressures for soft field operating conditions, using large size tires (Mod S.O.O. 6048), are 24 psi for nose wheel tire and 35 psi for main wheel tires (Mod 6/1574).



Nose Gear Identification Plate (71–101–11) – Servicing Instruction
Figure 1

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR12-18**

Insert in 12-10-25 before Page 1

OIL SYSTEM

REASON FOR ISSUE:

To update the instructions for checking the engine oil level.

ACTION:

In Chapter 12, Section 12-10-25, ignore existing Para 2. and insert revised Para, as shown on Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

2. Checking Oil Quantity

A quantity dipstick is contained in the filler cap and indicates the quantity of oil required to top up tank to the full mark. To check oil quantity, open panel to gain access to oil filler cap, remove cap and check oil level in normal manner. Install and secure cap on completion of check.

Note Take care to avoid overfilling the oil tank, which may cause discharge through breather and high oil consumption.

If level indication is low when checking oil quantity after engine has been shutdown for more than 30 minutes, or after an oil change, then start the engine and run at ground-idle for minimum five minutes, feather the propeller, shutdown the engine, and recheck the oil quantity.

NOTE: Please see the TEMPORARY REVISION that revises this page.

Oil System – General Data

1. General

CAUTION **DO NOT MIX DIFFERENT BRANDS OF OIL WHEN CHANGING OR ADDING OIL BETWEEN OIL CHANGES, SINCE THE CHEMICAL STRUCTURE OF DIFFERENT BRANDS OF SYNTHETIC OIL MAY DIFFER SUFFICIENTLY TO MAKE THEM INCOMPATIBLE.**

Each engine is equipped with an integral oil tank. The filler cap is accessible through an access panel in the top of the nacelle structure. The total oil capacity for each tank is 1.9 Imp (2.3 U.S.) gallons and the usable oil capacity is 5 Imp (6 U.S.) quarts. The approved synthetic lubricating oils are listed in Pratt and Whitney Canada SB No. 1001.

2. Checking Oil Quantity

A quantity dipstick is contained in the filler cap and indicates the quantity of oil required to top up tank to the full mark. To check oil quantity, open panel to gain access to oil filler cap, remove cap and check oil level in normal manner. Install and secure cap on completion of check.

Note Normal oil level is at the one quart mark on the dipstick. Overfilling may cause discharge through breather.

If oil change has been done or engine has remained stationary for a period of more than 12 hours, start engine, run at idle for 2 minutes, feather propeller, shutdown engine, check oil and replenish if required with same brand of oil.

3. Draining Oil Systems

To drain the oil system, open the lower engine cowling and remove the drain plugs from the bottom of the accessory gear case, oil cooler, reduction gearbox and oil tank.

4. Oil System Flushing

If engine is to be operated on a different oil than the oil on which it was previously operated, proceed as follows:

A. Remove oil drain plugs from engine oil tank, reduction gear box, accessories gearbox and oil cooler.

CAUTION **LIMIT ENGINE ROTATION TO SHORTEST TIME NECESSARY TO ACCOMPLISH COMPLETE DRAINING.**

B. With drains open, place power lever to IDLE and fuel lever to OFF, motor engine with starter only (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data) to allow scavenge pumps to clear engine.

C. Install drain plugs in engine and oil cooler.

D. Fill engine oil tank with the new oil.

E. Start engine and run at low IDLE for a minimum of two minutes. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

F. Feather propeller.

G. Shutdown Engine. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

H. Repeat Para 4.A and Para 4.B.

I. Remove engine oil filter. Install new element.

J. Reinstall all drain plugs in engine and cooler.

K. Replenish engine oil tank with applicable oil.

L. Start engine. Run at low IDLE for a minimum of two minutes and check for leaks.

M. Feather propeller.

N. Shutdown engine and refill oil tank to proper level.

Cleaning – General Data

1. General

Directions are outlined for engine compressor wash, cleaning windshield, cabin windows, floor panels, upholstery, display screens and fuel gallery.

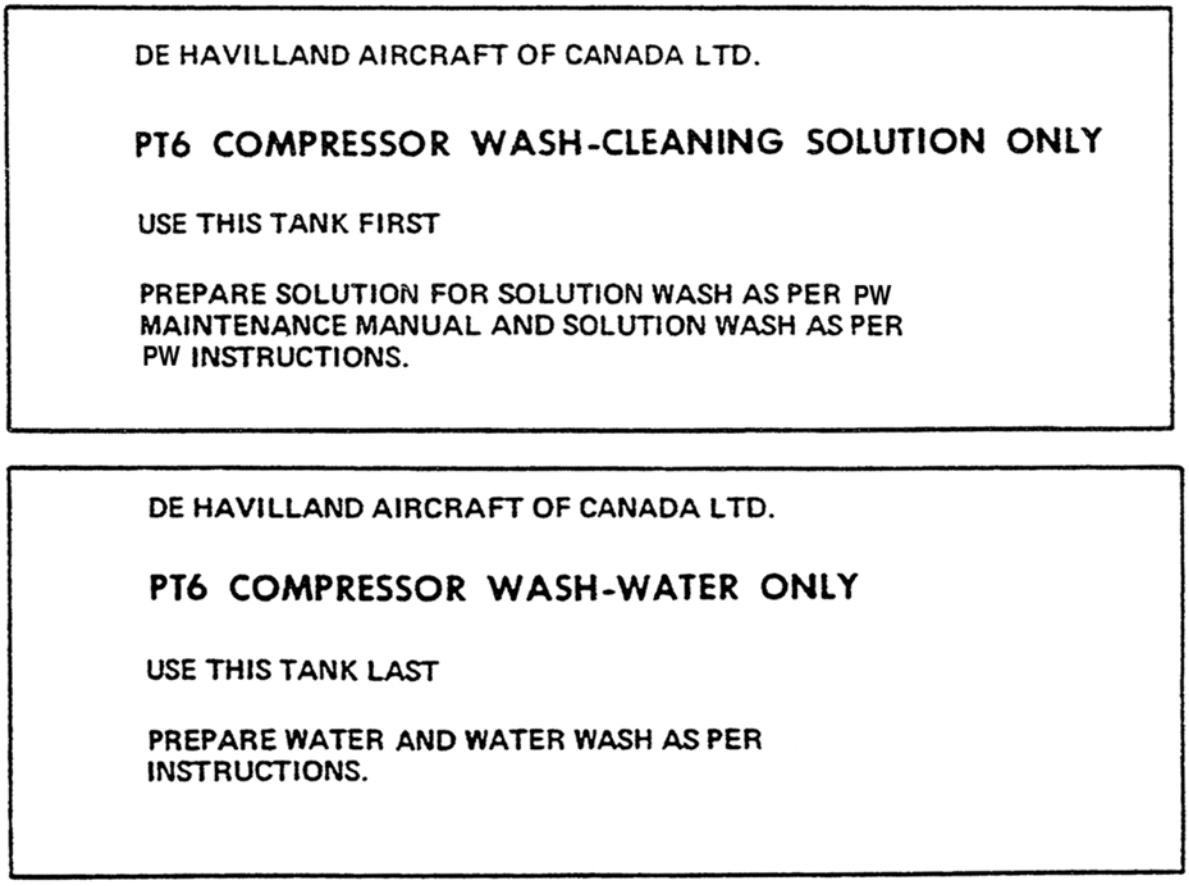
2. Engine Compressor Wash

WARNING ALL PERSONNEL AND EQUIPMENT MUST BE KEPT CLEAR OF PROPELLER WHEN CARRYING OUT ENGINE COMPRESSOR WASH.

A. General

An engine compressor wash may be carried out with the engine motoring, depending on requirements. Motoring washes are done using a wash wand (refer to Pratt and Whitney Engine Maintenance Manual) at the engine air inlet screen.

All engine compressor washes must be carried out strictly in accordance with instructions given in the Pratt and Whitney Engine Maintenance Manual.



Instruction Labels – Engine Wash System Compression Tank Rig
Figure 1

B. Wash Procedures

1) Ensure engine is shutdown.

2) Fill compression tanks of engine wash rig (Mod 6/1507) as required for type of wash, in accordance with instructions on tank labels. (Refer to Figure 1).

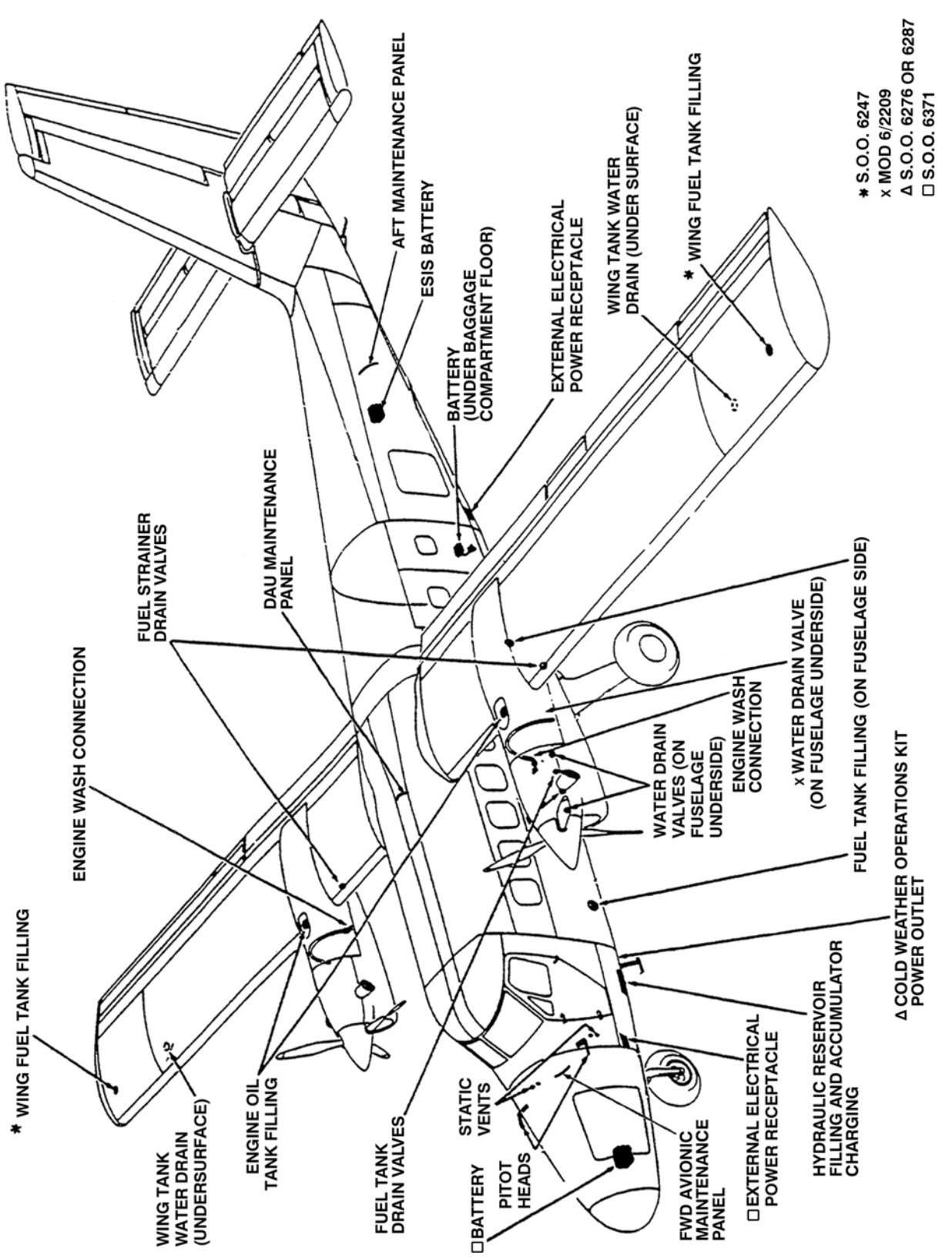
3) Connect engine wash rig as follows:

a) Aircraft fitted spray ring connection is located at inboard side of engine nacelle (refer to Figure 2 for details).

b) For compressor wash with engine motoring, connect rig to wash wand and position wand at engine air inlet screen.

4) Start up or motor engine and proceed with appropriate engine compressor wash in accordance with instructions given in Pratt and Whitney Engine Maintenance Manual.

5) On completion of satisfactory compressor wash, shutdown engine or stop motoring, disconnect wash rig from spray ring or wash wand, and cap line connection in aircraft fitted spray ring. Remove engine compressor wash rig from vicinity of aircraft.



Servicing Diagram
Figure 2

3. Cleaning Windshield and Cabin Windows

To clean windshield and cabin windows, use copious quantities of soap and water applied by sponge or rag. Any rubbing necessary to remove obstinate spots of dirt should be done with bare hands so that static electricity (which will attract dust) is not generated. After cleaning, polish with Anti-static Plastic Cleaner and finish with a clean, dry, soft cloth to a high polish.

4. Cleaning Floor Panels (Post Mod 6/2166)

CAUTION **NEVER USE PAINT THINNERS, ACETONE, MEK OR OTHER SOLVENT BASED CHEMICALS. IF YOU ARE UNSURE TRY IT ON A SMALL SECTION OF THE PLASTIC BEFORE APPLYING IT TO MAIN PANEL.**

Remove stubborn marks and oily marks with jet A1 fuel (kero) or Mineral Turpentine followed by a detergent cleaner to remove any traces of the fuel/turps and smell from the fuel/turps.

Note Methylated spirits are also useful for some marks and stains.

Note General cleaning should be carried out using normal detergent sprayed or wiped onto the surface and wiped off with a clean cloth.

5. Cleaning Upholstery

A. Cleaning Polyvinyl Upholstery (Pre Mod 6/2281)

1) Easy Maintenance

a) All commercial cleaning agents and solvents can be used, including bleach.

2) Stain Removal

Almost all stains can be removed or improved if blotted up immediately, while still wet or fresh.

a) Use a white or light coloured paper towel, tissues, or any absorbent fabric and blot away at the stain.

b) Keep blotting and changing the absorbent material until the spill or coloured matter is no longer being removed.

c) If possible, leave an absorbent towel pressed against the spot for 5 to 10 minutes to slowly absorb any residue.

d) Repeat as needed and be patient.

3) Removal Methods

a) Pretest all solutions in an inconspicuous area by applying a few drops.

b) Hold a white towel against the wet area and count to ten.

c) Examine the towel to determine if there is any colour transfer or change in fabric colour.

d) Apply small amounts of the cleaning solution by pressing or padding the area with the absorbent towelling.

e) Wipe the area down and allow it to dry for at least six hours.

4) Special Tips

a) Scrape, vacuum or brush up any dried powdery residue (i.e. mud, talc, powders, etc.).

b) When using water, detergent or solvent, work from the outer edge of the stain towards the centre to avoid ring marks.

c) If using wet extraction or wet/dry vacuum, use overlapping strokes to avoid cleaning marks.

d) Do not over wet or over use cleaning solutions.

e) Contact professional cleaner to remove stains.

B. Cleaning Leather Upholstery (Post Mod 6/2281)

CAUTION DO NOT USE DETERGENTS, CITRUS CLEANERS OR SOLVENTS FOR CLEANING LEATHER.

1) Clean leather covers with commercial saddle soap.

C. Cleaning Fabric Upholstery (Post Mod 6/2281)

1) External surfaces and upholstery should be periodically checked and cleaned. Carefully follow instructions of cleaning product. The normal cleaning method for seat covers is to remove them from the seat and commercially dry clean them with perchlorethane (refer to PSM 1-6-6A – Component Maintenance Manual).

Note Prior to dry cleaning, mate hook attachment with opposing piece of pile.

Note Under normal conditions and controlled dry cleaning processes, cover assemblies hold their flammability resistance up to five cleaning cycles. This is not a definite limit since many factors can shorten or extend the flammability life of the cover assembly.

Note Use only nonionic detergents and turn covers inside out if laundering upholstery.

D. Cleaning Sheepskin Covers (S.O.O. 6268)

The sheepskin seat covers are constructed of both sheepskin and Douglass Leather. Each area must be cleaned in accordance with the following manufacturers recommendations.

CAUTION DO NOT MACHINE WASH COVERS UNDER ANY CIRCUMSTANCES.

1) Sheepskin

Basic cleaning should require only brushing and vacuuming. Severe heat can damage sheepskin, resulting in shrinking and hardening of pelts.

Note Do not use alkaline products to clean sheepskin.

a) For mild cleaning:

1 Sponge clean with water and a mild detergent.

2 Comb the wool while wet and air dry skins away from artificial or excessive heat.

3 Skins may crimp after washing. If a straight hair appearance is preferred, brush fleece with a pet brush.

b) For deeper cleaning:

1 Hand scrub with warm water not exceeding 110°F (90°C).

2 Use a small amount of water for each area.

Note Do not soak the covers.

3 Use only pure, mild soap.

Note Do not use Woolite.

4 Comb the wool while wet and air dry skins away from artificial or excessive heat.

5 Skins may crimp after washing. If a straight hair appearance is preferred, brush fleece with a pet brush.

2) Douglass Leather

Leather is resistant to soil and moisture. Basic care should involve no more than regular dusting. For topical spot cleaning, Leather Cleaner Wipes are available from the manufacturer.

Douglass Leather Cleaning Products:

- Leather Cleaner Wipes – A simple, fast, effective way to remove topical soil, these are excellent for spot cleaning. Available from the manufacturer.
- D.I.P. Stick – A powerful and concentrated cleaner specifically for ink or lipstick stain removal. The D.I.P. Stick is compact and easy to use.

Note Do not use detergents, saddle soap, oils or waxes.

a) Basic cleaning:

1 Use a solution of lukewarm water and a pure, mild soap.

2 Dampen (don't wet) a soft cloth or sponge with solution and apply in circular motions.

3 Wipe dry with a soft, clean cloth.

4 For ink or lipstick stain removal, use Douglass D.I.P. Stick.

E. Cleaning Stretchers (S.O.O. 6208)

1) Support Assemblies and Stretcher Frame

a) Remove dirt with a dry, non-abrasive hard-bristled brush.

b) Wipe with a moist towel.

Note Do not immerse in water.

c) Dry thoroughly before storage or use.

2) Stretcher Fabric

a) Clean fabric with warm water and mild commercial soap.

CAUTION DO NOT USE BLEACH OR HARSH CHEMICALS.

b) Dry thoroughly before storage or use.

6. Cleaning the Display Screens and Bezel Buttons

CAUTION **DO NOT USE PAPER MATERIALS TO CLEAN THE SCREEN, THIS MAY CAUSE DAMAGE TO THE SCREEN.**

CARE MUST BE TAKEN WHEN CLEANING SCREEN AND BEZEL BUTTONS, DO NOT PUSH ON SCREEN OR BEZEL BUTTONS, THIS MAY CAUSE DAMAGE.

Note Remove all jewelry before cleaning screen, this may cause scratches to display screen.

A. Cleaning the Display Screens

Required Materials and Equipment:

- Silk cloth or brush
- Precision wipe
- Substrate cleaner
- Isopropyl alcohol

1) Ensure the aircraft electrical power is OFF (refer to 24-00-00, Electrical Power – General Data).

2) To remove finger prints and non grease type marks, breath on marked area and gently wipe area with clean cloth.

3) To remove oil or grease marks:

a) Dampen a precision wipe with substrate cleaner.

b) Gently wipe screen until marked area is clean.

c) Repeat if required.

B. Cleaning the Bezel Buttons

1) Ensure the aircraft electrical power is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Use a clean non-abrasive lint-free cotton cloth dampened with a multi-purpose detergent **containing no alcohol**.

3) Gently wipe bezel button surface until clean.

4) Repeat if required.

7. Cleaning Fuel Gallery Electrical Connections

A. Ensure the aircraft electrical power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

B. Clean the electrical connectors and surrounding area with Isopropyl alcohol, using a brush H–B–681 or toothbrush A–A–59.

C. Remove extra solvent and residue with cleaning cloth CCC–C–46 (Type I, Class 4).

8. Cleaning STA 332 Bulkhead Cargo Net

Required Materials and Equipment:

- Mild Detergent or Dish Soap
- Soft Scrub Brush

A. Cleaning Procedure

CAUTION **DO NOT USE BLEACH OR HARSH CHEMICALS.**

1) Remove cargo netting from aircraft.

2) Wash cargo net with mild detergent and warm water.

3) Gently scrub the cargo net as required to remove dirt/stains.

4) Rinse well with fresh warm water.

5) Allow to air dry thoroughly before storage or use.

Lubrication – General Data

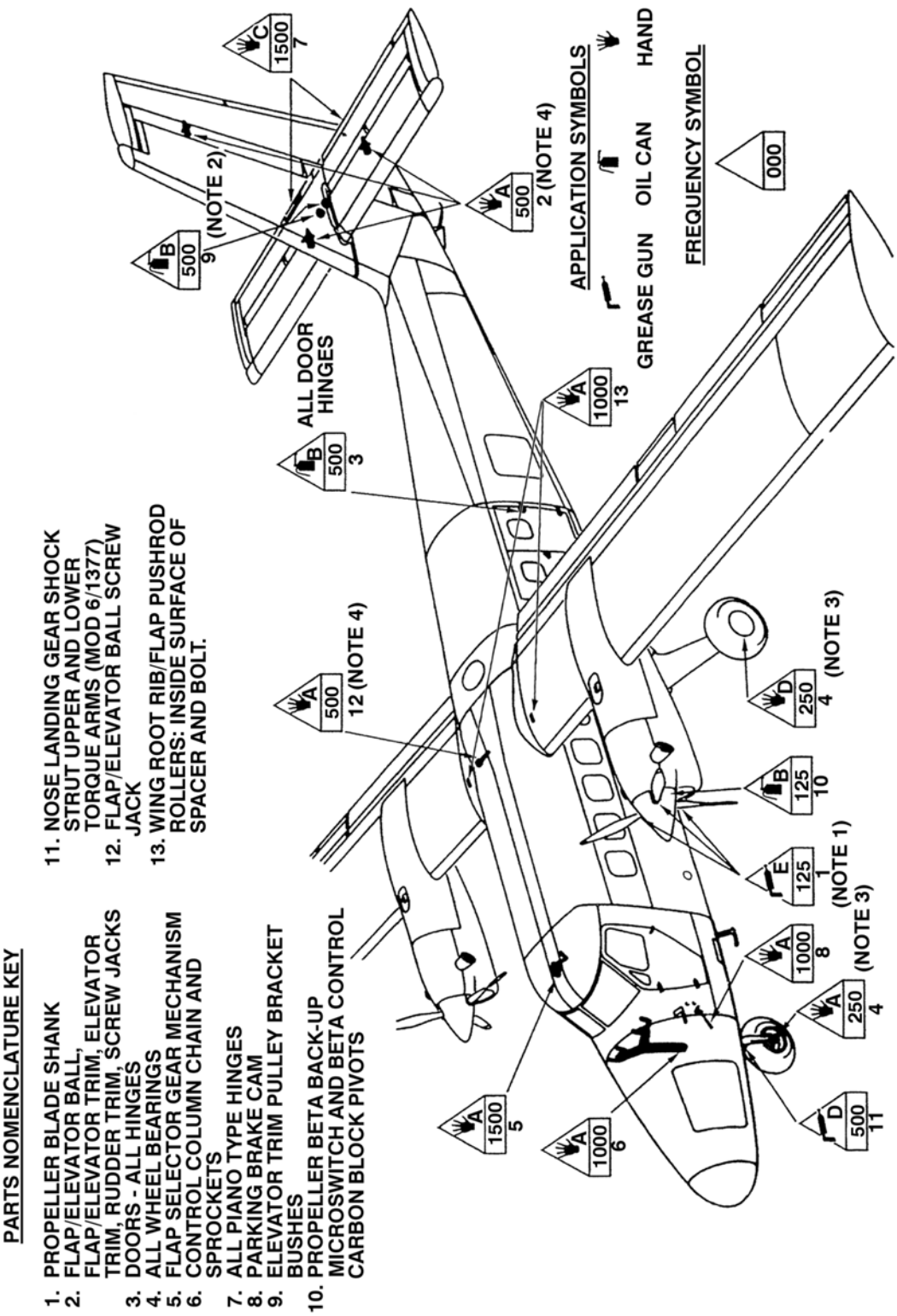
1. General

Lubrication points, methods, and frequency of application are given in Lubrication Diagram, Figure 1. Extensive use of sealed bearings and dry-film lubricant, applied during manufacture, keeps the need for external lubrication to a minimum. Consequently, daily and intermediate lubrication is unnecessary, and periodic application is indicated by a triangle, the frequency of which is denoted by the figure on the block at the base of the triangle. The specification and equivalent types of lubrication recommended for use at each point is given in Table 1 — Lubricants, and is used in conjunction with the frequency figure in Figure 1. Before applying lubricant, wipe off any dirt and moisture from the point with a clean, lint-free cloth, lightly moistened with dry cleaning solvent. Avoid excessive use of lubricant when applying by hand to prevent spillage. Carefully clean up spillage immediately.

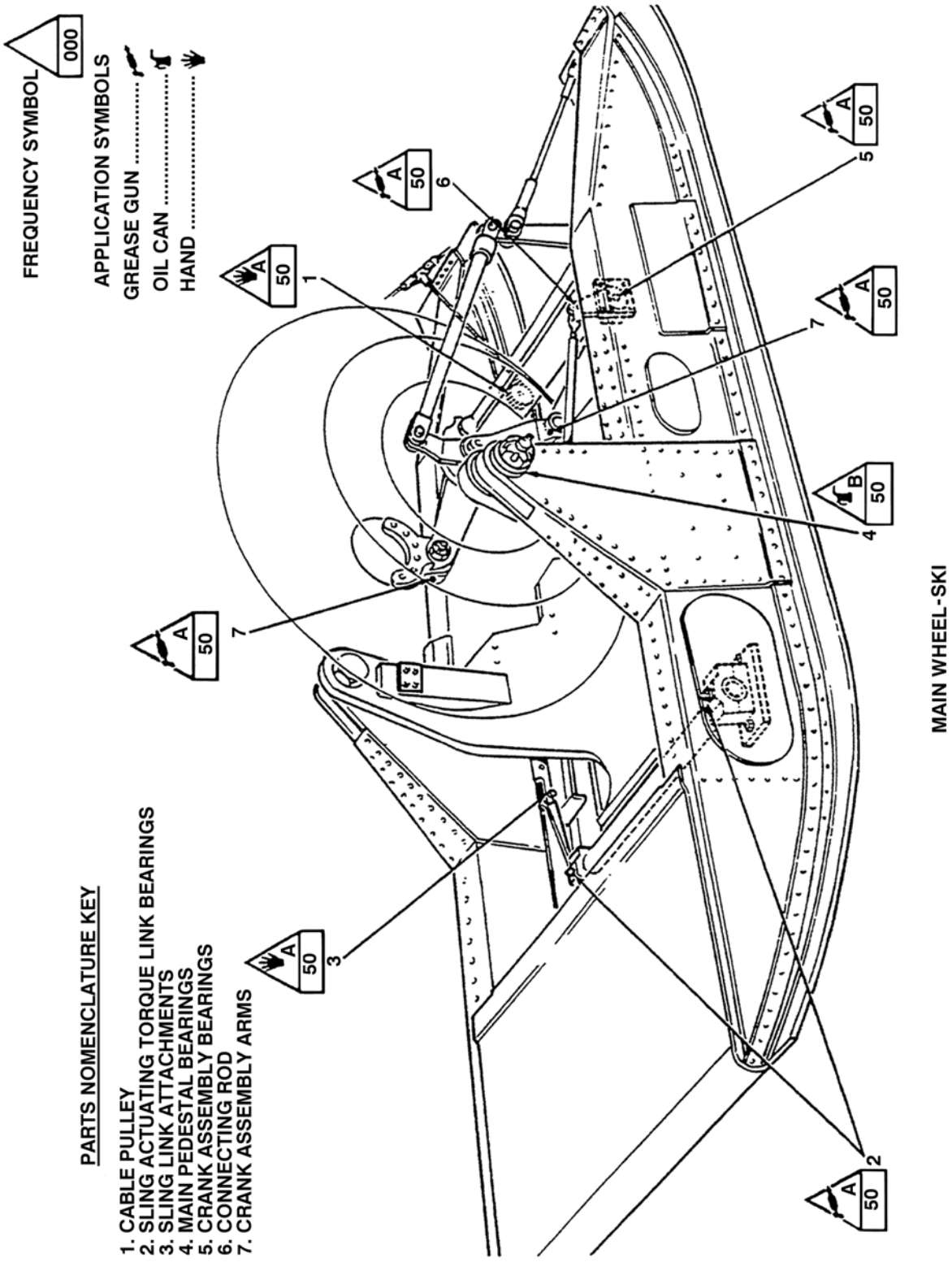
Table 1 Lubricants

SYMBOL	A	B	C	D	E
Type of Lubricant	Grease – Low and High Temperature	Oil – General Purpose	Silicone	Grease – General Purpose Wide Temperature Range	Grease – General Purpose Limited Operating Temperature Range
Canadian Specifications	3–GP–606A 3–GP–683A 3–GP–689A MIL–PRF–23827 MIL–PRF–81322	3–GP–335A	MIL–S–8660	MIL–PRF–81322	MIL–PRF–24139
U.S. Specifications	MIL–PRF–23827 MIL–PRF–81322	MIL–L–7870A		MIL–PRF–81322	MIL–PRF–24139
SHELL Aviation Products	Aeroshell Grease 7 Aeroshell Grease 22	Aeroshell Fluid 3		Aeroshell Grease 22	Aeroshell Grease 6
ESSO Aviation Products	Beacon 325	ESSO Instrument Oil			
TEXACO	Low Temperature Grease EP	1692 Low Temperature Oil			
MOBIL Aviation	SHC 100				

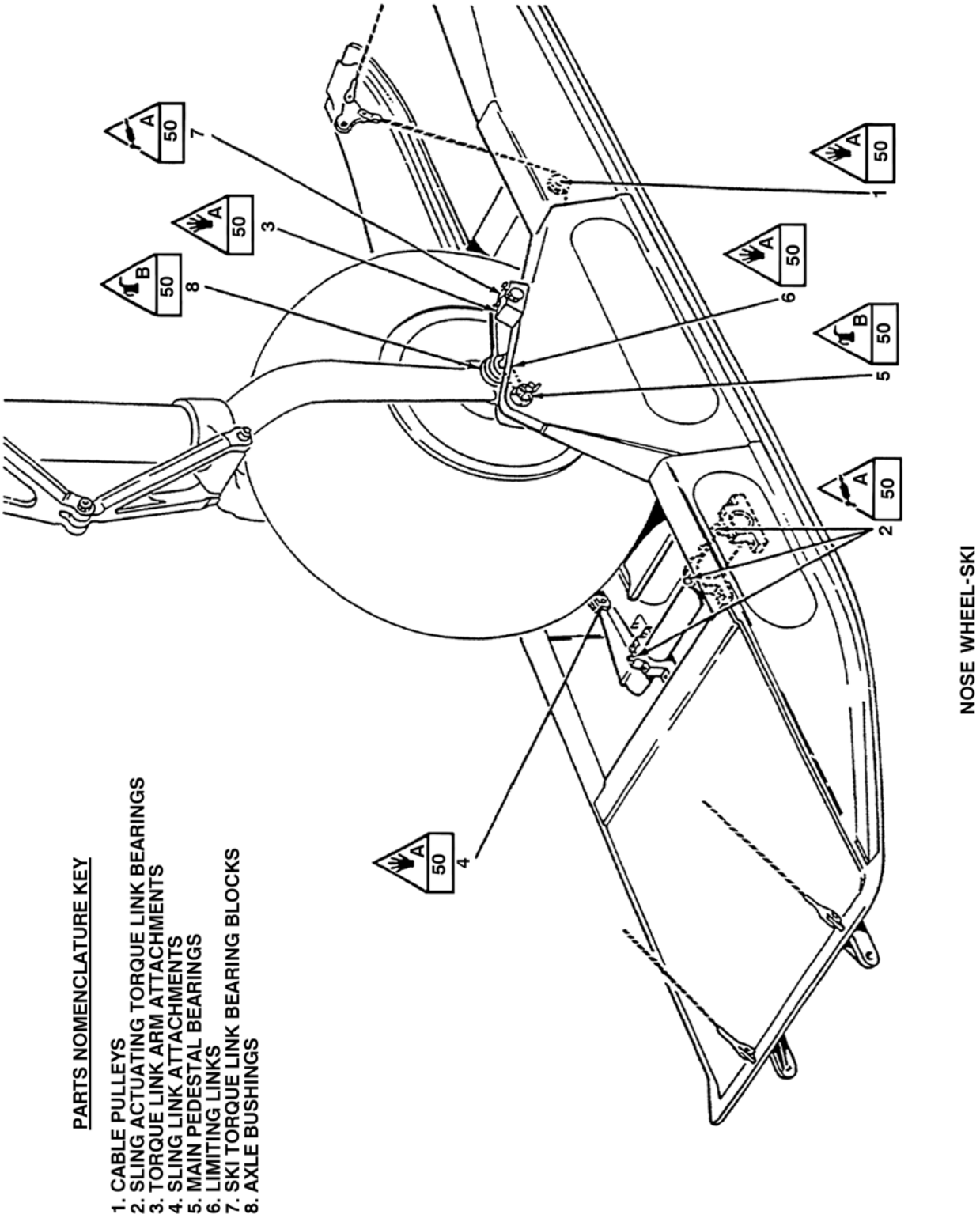
Note Greases shown under Symbol A have a synthetic base and as such are not compatible with grease having a mineral base.



Lubrication Diagram
Figure 1



Lubrication Diagram - Wheel-Skis
Figure 2 (Sheet 1 of 2)



Lubrication Diagram - Wheel-Skis
Figure 2 (Sheet 2 of 2)

Unscheduled Servicing – General Data

1. General

This section contains information necessary to carry out servicing that is normally unscheduled. It includes instructions for cleaning and cold weather protection.

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Aircraft External Cleaning – General Data

1. External Cleaning

A. Washing Aircraft Exterior

Equipment and Materials:

- Cleaner MIL–C–25769
- Solvent MIL–C–25179
- Solvent P–D–680 (AMS 3160)
- Isopropanol
- Aliphatic Naphtha TT–N–95, Type II
- Brush, Mop or Sponge
- Cheesecloth

1) General

a) Frequent washing of the aircraft should be carried out to maintain its appearance and to minimize corrosion. The painted area of the aircraft should be polished at periodic intervals to remove chalking of paint and to restore its gloss.

b) Flammable cleaners and solvents should not be used in areas where combustion may occur.

c) To prevent corrosion, do not use high pressure spraying of cleaning detergents on units having holes which drain into internal cavities.

d) Lubricate aircraft components, e.g. pivot points and accessible bearings whenever necessary after washing.

2) Preparation for Cleaning

a) Mask all areas where contact with, or penetration of, cleaning concentrate is to be avoided. To include windshields, windows, wheels, pitot heads, static vents, all top surface openings, doors, nacelle and nose assembly fairings.

b) Use an alkaline water base cleaner Specification MIL–C–25769 diluted in the ratio of one part cleaner to ten parts fresh water for lightly soiled conditions. For moderately soiled conditions, dilute one part cleaner to five to eight parts water.

3) Application of Cleaner

a) Cleaner can be applied by spraying, mop, sponge or brush. Allow cleaner to remain on surface for approximately five minutes and agitate with a brush or mop, rinse thoroughly with copious amounts of clean (potable) water.

Note Do not allow cleaner to dry before rinsing, as less effective cleaning will result. Do not apply the cleaner to excessively large areas which cannot be brushed and thoroughly rinsed before the cleaner dries on the surface.

b) Where electrical equipment is present, limit cleaning wherever possible to wiping clean using cloths moistened with solvent. Specification P–D–680 (AMS3160). Wheels or related brake parts should be hand cleaned using alkaline cleaners or solvents as appropriate.

c) When alkaline cleaning concentrate is not available, solvent emulsion compound Specification MIL–C–25179 may be used, in dilution of one part compound to nine parts cleaning solvent Specification P–D–680 (AMS3160). Procedures are the same as with alkaline water based cleaner but extreme care must be exercised to prevent solution coming into contact with rubber components such as hoses, tires, seals, etc.

d) On completion of cleaning operations, ensure that all masking applied in Para 1.A.2).a), is removed.

B. Cleaning Composite Components

1) Doors, Nose Baggage Compartment, Lower Aft Nacelle Fairing and Fwd/Aft Gallery Fairings. Refer to manufacture's instructions for cleaning procedures for the radome.

2) Apply small amount of solvent to a clean wiping cloth.

Note Apply the minimum amount of solvent, Isopropyl Alcohol, IPA or Plastic Prep/Cleaner eg. Plas-Stick 2319s by Dupont when solvent cleaning.

3) Wipe the composite components with the solvent dampened wiping cloth.

4) Wipe dry with another clean wiping cloth **before** the solvent evaporates.

Note It is important to dry the surface immediately after cleaning because if the solvent evaporates before it is wiped up, oil and grease residue will be left on the clean surface.

5) If more cleaning is required, repeat Para 1.B.2) and Para 1.B.4).

C. Cleaning Aircraft Windows and Windshields

CAUTION DO NOT USE ANY ABRASIVE MATERIALS, STRONG ACIDS, METHANOL OR METHYL ETHYL KETONE.

1) Remove excessive amounts of dirt and other substances from the glass surfaces with clean water applied with a sponge or soft cloth.

2) Wash gently with mild soap and water, or a 50/50 solution of isopropanol and water or aliphatic naphtha TT–N–95, Type II.

3) Rinse thoroughly using clean water and dry using clean, soft cloth or sponge; use a straight rubbing motion.

D. Cleaning Aircraft Propellers

CAUTION DO NOT ALLOW SOLVENT TO CONTACT RUBBER, PLASTIC OR FIBERGLASS PARTS.

1) Clean external metallic surfaces with solvent P–D–680 (AMS3160D) or equivalent. Apply solvent sparingly with a soft cloth or soft brush.

2) Remove solvent by thoroughly rinsing with clean water and wipe dry with a soft cloth.

3) Clean rubber, plastic and fiberglass surfaces with mild soap and water applied with a soft cloth.

4) Rinse with clean water and dry with soft cloth.

Cold Weather Protection – General Data

1. General

CAUTION SAE AEROSPACE MATERIALS SPECIFICATION (AMS) 1428; "AIRCRAFT DE-ICING/ANTI-ICING FLUID SAE TYPE II AND IV" ARE NOT ACCEPTABLE.

Operating the aircraft in cold weather conditions presents several problems due to the effects of low temperature, such as ice and snow buildup on wings and control surfaces. Corrective maintenance and lost time can be kept to a minimum by an organized program of preventative maintenance. Ensure aircraft parking is observed in accordance with procedures detailed in 10-00-00, Para 1.

Note Procedures given may be varied according to conditions and available equipment. In extremely cold weather, covers can be prevented from freezing to the aircraft by applying a light brush coat of freezing point depressant fluid.

Equipment and Materials:

- Long handled broom
- Rig, De-icer/Washer FMC 239–5500, Model SA–600, or equivalent
- SAE Aerospace Materials Specification (AMS) 1424; "Aircraft De-icing/Anti-icing Fluid SAE Type I"
- SAE Aerospace Materials Specification (AMS) 1428; "Aircraft De-icing/Anti-icing Fluid SAE Type III"

2. Aircraft De-icing

Aircraft de-icing is defined as the process of removing accumulation of snow or ice from the aircraft. Where facilities exist for heating fluids contained in the de-icer rig, note that heated fluids are more effective in the de-icing procedure than unheated fluids. Remove snow and ice from aircraft as follows:

A. Remove snow from fuselage, wings and empennage as follows:

1) Immediately after a snow fall, clean the aircraft of all snow to prevent any melting snow from freezing to the surface and interior areas of the aircraft into which water may flow.

CAUTION ALL EXTERIOR SURFACES, PARTICULARLY AIRFOIL SURFACES, MUST BE COMPLETELY CLEAR OF SNOW, SLUSH OR ICE IMMEDIATELY PRIOR TO TAKE-OFF.

2) Wet snow can best be removed from the fuselage, wings and empennage by means of a long handled broom if depth of snow warrants it, and by application of freezing point depressant (FPD) fluids. When sweeping snow off aircraft surfaces, work from support stands placed close to the aircraft. If stands are not available, exercise extreme caution when walking on aircraft surfaces during snow clearing operations to avoid slipping or sliding off aircraft. Wear rubber or fabric footwear while on aircraft surfaces.

Note FPD fluids must be applied at the correct mixture strength and temperature as determined by manufacturer.

3) Check for snow accumulation between aileron leading edges and flaps, elevator leading edges and shrouds, wing flap leading and trailing edges and all tabs and shrouds.

4) If snow accumulation is dry and light, it may be blown off the aircraft using compressed air.

Note Do not use hot air to remove dry snow.

5) Clear aircraft parking area of snow, slush and ice. If this is impractical, clear the area around the main wheels and nose wheels to reduce the possibility of tires being frozen to the ground.

B. Remove ice accumulation from aircraft as follows:

1) Plug engine and air intake ducts when de-icing is done in their vicinity.

CAUTION **USE FREEZING POINT DEPRESSANT (FPD) FLUIDS WITH CAUTION NEAR ENGINES AND EXHAUST GAS AREAS TO AVOID ACCUMULATIONS THAT COULD BE INGESTED OR IGNITED ON STARTING ENGINES. REMOVE ANY ACCUMULATION BEFORE STARTING ENGINES.**

2) Drain holes, static ports, pitot heads, area between inboard fore flap trailing edge and trailing flap leading edge, outboard flap trailing edge and aileron leading edge, and control surfaces must be free of ice accumulation.

CAUTION **DO NOT USE HARD OR SHARP TOOLS FOR REMOVING ICE. USE FPD FLUIDS SPARINGLY AROUND HINGE POINTS AND BEARINGS. DO NOT SPRAY DIRECTLY ONTO THESE AREAS.**

3) Remove ice from the aircraft surface by spray application of FPD fluids; as the ice loosens, sweep it off.

Note Application of hot air can also be used to melt ice from surfaces but do not direct hot air on window surfaces. Melting ice must be mopped dry to prevent subsequent refreezing on aircraft.

4) When operating from snow or slush covered runways, monitor the nose gear area externally for ice buildup, and if necessary remove any ice buildup by spray application of FPD fluids.

3. Aircraft Anti-icing

A. General

Anti-icing is defined as the procedure used to prevent or delay the formation of ice accumulations on the aircraft and flight control surfaces. Application of unheated freezing point depressant (FPD) fluids is more effective in the anti-icing procedures than heated fluids. Carry out anti-icing as follows:

1) Apply FPD fluids at the correct mixture as determined by the manufacturer.

2) Ensure spray application of FPD fluid covers all surfaces with a fine film of fluid.

Note Complete access must be gained to upper surfaces of horizontal stabilizer and elevators.

B. Elevator Assembly Anti-icing

1) Prevent ice accumulation at the elevator trim tabs by spraying FPD fluid on to the tab leading edge and around hinge locations.

2) Additional anti-icing protection of the elevator can be achieved by the periodic application of silicone grease to the tab leading edges.

Note Two operators are required to carry out this task. Repeat silicone grease application as frequently as weather conditions demand.

If the aircraft is washed (particularly the horizontal stabilizer area) or de-iced while operating in climatic conditions where freezing or ice buildup could occur, repeat the silicone grease application to the tabs.

3) De-ice horizontal stabilizer and elevator surfaces if required.

4) Engage elevator gust lock.

5) Ensure all tabs are free of snow and ice accumulation.

6) Operate trim tabs to fully up position. Using a clean cotton cloth dampened with isopropyl alcohol, clean the full length of the leading edge tabs; clean as much of the leading edge area as possible, dry immediately with a clean dry cloth.

7) Operate trim tabs to fully down position; repeat cleaning as at Para 3.B.6).

CAUTION EXERCISE CARE WHEN APPLYING SILICONE GREASE.

8) Using a soft bristle brush, apply a thin film of silicone grease to the full length of the leading edge of the tabs; cover as much of the leading edge as possible.

9) Operate trim tabs to full up positions; repeat application of silicone grease as in Para 3.B.8).

10) Disengage elevator gust lock.

11) Operate trim tabs to take-off position.

C. Propeller Anti-icing

1) To help prevent ice accumulation on propeller blades, Icex is applied to blades.

2) Lightly scrub blades and heaters with a mild soap and water solution and dry thoroughly.

3) Apply Icex with a soft cloth or aerosol can sparingly over blades.

4) Buff the entire surface with a soft dry cloth until a smooth gloss is obtained.

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CHAPTER 20

STANDARD AIRFRAME PRACTICES

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CHAPTER 20 – STANDARD AIRFRAME PRACTICES

LIST OF EFFECTIVE PAGES

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General – General data

1. General

The standard practices continued in this chapter apply to adhesives, fasteners, sealants and protective coatings as used and recommended by Viking Air Limited.

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Fasteners – Torquing

1. General

This standard specifies the procedure and requirements for torquing bolts, nuts and screws (excluding fluid line fittings), which have torque values given in Maintenance Practices.

2. Materials and Equipment

A. Materials:

- MIL–PRF–83483 anti-seize thread compound
- Tamper proof sealant (putty), white, pink or red – Organic Products Co. F-900 or #3M EC1252. Refer to appropriate MSDS for the storage life of tamper proof sealant

B. Equipment:

- Torque wrenches
- Sockets and adapters as required

3. Requirements

A. Bolts, nuts and screws must only be torqued when specified in Maintenance Practices.

B. Lubricating and anti-seize compounds must not be used on threaded fasteners unless such preparation is specifically called up.

C. All paint, dirt and corrosion must be removed from threads prior to mating up and torquing. Mating parts must be smooth and clean.

D. Threaded fasteners must engage freely.

E. The installed bolt must not engage imperfect threads of tapped blind holes, and the threads of tapped holes must not engage the run-out of the bolt or screw threads.

F. Bolts, nuts and washers must seat squarely with the mating surfaces.

4. Torquing

A. Installed fasteners must be torqued to values specified in Maintenance Practices.

B. Wherever possible, in a bolt-nut or screw-nut combination, the bolt or screw must be held stationary and the nut turned.

C. When it is necessary to tighten a fastener from the bolt or screw side, the torque value must approach the high side of the specified range.

D. Whenever torqued fasteners are to be secured by means of cotter pins or lockwire, the low side of the torque range must be used for tightening. If necessary, tighten the fastener so that the next slot aligns with the hole, provided the maximum torque value is not exceeded.

E. When the maximum torque value is reached and the slot in the nut does not line up with the hole in the bolt, the nut and washer must be changed.

F. When installation of a part involves tightening a group of more than three fasteners, pre-torque all fasteners to 75% of final torque value in the following sequence, as applicable:

1) For relatively solid parts where length to width ratio of fastener pattern is less than two, pre-torque in a criss-cross order.

2) When length to width ratio of fastener pattern is two or more, or when the flatnose of thin parts or gaskets may be affected by the torquing sequence, pre-torquing must be carried out starting from the center of the pattern and working outwards to the ends.

G. After pre-torquing, torque all fasteners to final torque value in same sequence as used in pre-torquing.

5. Overtorqued Fasteners

A. Fasteners which have been tightened beyond the maximum specified torque value must be removed, rendered unserviceable, and scrapped.

B. Overtorqued fasteners must not be backed off and retorqued to the correct value.

C. In the case of overtorqued bolt-nut or screw-nut fasteners, both bolt or screw and nut must be discarded.

6. Retorquing

A. Where it is necessary to retorque a fastener assembly, the nut must be backed off one full turn and retightened to the specified value.

B. The bolt or screw must not be allowed to turn throughout the operation, unless the installation requires the torque to be applied to the bolt or screw in the first instance.

7. Use of Torque Wrenches and Adapters

A. Before use, pre-set click type torque wrenches to the specified value as follows.

1) Release the locking mechanism located on the wrench handle.

2) Adjust the wrench by rotating the handle until the desired setting is reached, as indicated by the graduated scale.

3) Engage the locking mechanism and visually re-check the torque setting.

B. When torquing fasteners, the torque wrench drive must be held squarely to the mating surface receiving the fastener.

C. A slow steady motion must be applied to torque wrenches when tightening a fastener.

D. Torque wrenches must be checked for accuracy before use.

E. When using offset adapters, the adapter must be in line with the torque wrench arm.

F. Allowance for the offset adapter must be made to calculate torque value in accordance with the following formula:

Dial Reading = (Specified Torque x L) ÷ (L + L1)

L = Distance in inches from mid point of grip to centerline of wrench drive.

L1 = Distance in inches from centerline of wrench drive to centerline of offset adapter drive.

8. Identification of Torqued Fasteners

A. All torqued fasteners should be identified after torquing.

B. Identification should consist of a stripe or blot of red lacquer.

C. Identification must be placed on the nut of any bolt-nut or screw-nut combination, whenever practicable, and mating surface, otherwise head of bolt or screw must be marked.

D. Where external appearance of aircraft may be affected, identification may be omitted.

E. If at any time misalignment or displacement of the identification stripe or bolt indicates relative movement of the fastener, the fastener must be retorqued in accordance with Para 6., and identification reapplied.

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Fluid Lines and Fittings – Torquing

1. General

Unless otherwise detailed in individual maintenance practices this standard specifies procedures and requirements for torquing fluid line tapered thread fittings, AN818 and NAS591 fluid line coupling nuts.

2. Materials and Equipment

A. Materials:

- Castrol anti-seize compound with zinc oxide base
- SAE AMS 2518
- ROYCO 1MS MIL–PRF–83483
- SAE AMS–G–4343 (e.g., ROYCO 43)
- VV–P–236 (e.g., PENRECO Amber Petrolatum)
- Titeseal #2 or #3
- Monsanto MCS 352
- Allube HI–LO MS #1

B. Equipment:

- Torque wrenches, sockets and adapters as required

3. Lubricate Threads

A. Refer to Table 1 — Fluid Lines and Fittings Thread Lubricants for required lubricant.

B. Apply lubricant sparingly to male threads only (to prevent clogging of system or contamination of fluid).

C. On straight threads, apply lubricant to all but the first two threads.

D. On tapered threads (except oxygen and pitot-static systems), apply lubricant to the first three threads and ensure that no lubricant enters the bore of the fitting.

WARNING DO NOT LUBRICATE OXYGEN SYSTEM FITTINGS. OIL OR GREASE EXPOSED TO OXYGEN UNDER PRESSURE COULD CAUSE AN EXPLOSION RESULTING IN SERIOUS INJURY TO PERSONNEL AND/OR DAMAGE TO EQUIPMENT.

E. On oxygen and pitot-static system tapered threads, wrap the thread circumference once with teflon tape, cut tape length to obtain a minimum overlap, and work into threads with fingers.

F. When the system fluid is specified in Table 1 — Fluid Lines and Fittings Thread Lubricants, as the lubricant, the fluid may be applied generously.

Table 1 Fluid Lines and Fittings Thread Lubricants

Line Content	Type of Thread	Thread Lubricant
Fuel or Oil	Straight Thread	Castrol anti-seize compound or PENRECO amber petrolatum
	Tapered Thread	Titeseal #2 or #3
Hydraulic fluid (petroleum base)	Straight Thread	PENRECO amber petrolatum or Aeroshell fluid #4 or Univis J43
	Tapered Thread	PENRECO amber petrolatum

Table 1 Fluid Lines and Fittings Thread Lubricants (continued)

Line Content	Type of Thread	Thread Lubricant
Hydraulic fluid (synthetic phosphate ester)	Straight Thread	Monsanto MCS 352 or Allube HI–LO MS #1 or Hydraulic fluid (synthetic phosphate ester) or Phosphocent 246
	Tapered Thread	MCS 352 (Monsanto) or HI–LO MS #1 (allube) or Phosphocent 246
Water/Methanol	All	PENRECO amber petrolatum
Oxygen	Straight Thread	None
	Tapered Thread	Teflon tape
Pitot – Static	Straight Thread	AMS–G–4343
	Tapered Thread	Teflon tape
Compressed air (not oxygen)	All	AMS–G–4343
Hot gases (up to 800°F)	All	MIL–PRF–83483
Other fluids (up to 300°F)	All	Castrol anti-seize

4. Torquing

A. Torque Oxygen and Pitot System Tapered Thread Fittings

1) Screw fitting in not less than two and one half turns then torque between 22 and 30 pounds–inches.

2) Replace fittings that leak when torqued to 30 pound–inches.

3) Remove and discard any fitting torqued to more than 30 pounds–inches. Do not loosen and retorque.

B. Torque Fluid Lines and Fittings

1) Refer to Table 2 — Torque Coupling Nuts on Flared Tubes and Fittings for Torque Values.

2) Correct size and proper fitting wrenches must be used.

3) Torque fittings to a minimum value indicated in Table 2 — Torque Coupling Nuts on Flared Tubes and Fittings.

4) Torque fittings to maximum value indicated in Table 2 — Torque Coupling Nuts on Flared Tubes and Fittings only to adjust a position type fitting or to correct a leaking connection.

5) Remove and discard fitting or line on connections that continue to leak after being torqued to maximum value indicated in Table 2 — Torque Coupling Nuts on Flared Tubes and Fittings.

6) Remove and discard overtorqued fittings. Do not loosen and retorque.

7) When torquing the nut on a union, restrain union with a wrench while torquing nut.

8) Ensure that flexible hoses and rigid lines are not twisted or under preload during torquing.

Table 2 Torque Coupling Nuts on Flared Tubes and Fittings

SIZE (Note 2)	TORQUE VALUE (in–lbs) (Note 1)		
	STRAIGHT THREADS ON ALUMINUM ALLOY TUBE OR FITTING	**STRAIGHT THREADS ON STEEL ALLOY, CRES OR TITANIUM TUBE OR FITTINGS**	**TAPERED PIPE THREADS (except oxygen & pitot systems)**
1/8”	20 – 30	75 – 85	40 – 300
3/16”	25 – 35 (15 – 35 for 5052–0 alu-minum tubing with a wall thick-ness of 0.022”)	95 – 105	—
1/4”	50 – 65 (20 – 65 for 5052–0 alu-minum tubing with a wall thick-ness of 0.022”)	135 – 150	60 – 600
5/16”	70 – 90 (30 – 90 for 5052–0 alu-minum tubing with a wall thick-ness of 0.022”)	170 – 200	—
3/8”	110 – 130 (35 – 130 for 5052–0 alu-minum tubing with a wall thickness of 0.022”)	270 – 300	75 – 700
1/2”	230 – 260	450 – 500	100 – 900
5/8”	330 – 360	650 – 700	—
3/4”	460 – 500	900 – 1000	300 – 1600
1”	500 – 700	1200 – 1400	400 – 2200
1 1/4”	800 – 900	1520 – 1680	500 – 2500
1 1/2”	800 – 900	1900 – 2100	—
2”	1800 – 2000	2660 – 2940	—

Note 1 – If the hose fitting material and the tube material are different, use the lower torque value.

Note 2 – For the purpose of this table, size may be considered the tube size, nut size or fitting size, as applicable.

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Protective Coatings Hard and Soft Film Corrosion Preventive Compound – General Data

1. General

WARNING **THIS PROCEDURE USES MATERIALS THAT MAY BE DANGEROUS TO THE PERSONNEL'S HEALTH AND/OR THE ENVIRONMENT. SUFFICIENT PRECAUTIONS MUST BE TAKEN AT ANY STEP OF USE. IF NOT, THIS CAN CAUSE DAMAGE TO THE ENVIRONMENT AND DANGEROUS EFFECTS ON THE PERSONNEL'S HEALTH AND SAFETY.**

This standard specifies the procedures and requirements for the application of hard and soft film corrosion preventive compound to aircraft parts and assemblies.

A. Materials

- Hard Film Corrosion Preventive Compound to MIL–C–16173 Grade 1 (TECTYL 890, TECTYL 891, PETROTECT 1–X, NOX–RUST 501, ESGARD PL–1)
- Soft Film Corrosion Preventive Compound to MIL–C–16173 Grade 2 (TECTYL 502C, NOX–RUST 502, ESGARD PL–2, PETROTECT 2 OR LPS 3)
- Soft Film Corrosion Preventive Compound, Transparent, Water Displacing, Type I (colourless Boeshield T9 and LPS No. 3) and Type II (coloured Boeshield T9, ZC 023, ZCD–5023NS, COR–BAN35, LPS HARD COAT, PROTECTOR 100 and AV08)
- Solvent – DS108
- Solvent – NAPHTHA
- Solvent – Diestone DLS or HFP
- Solvent – Varsol
- Solvent – Toluol

B. Safety Precautions

1) Smoking, open flames or naked lights are prohibited in areas where spray application of corrosion preventive compound is being carried out.

2) The aircraft and aircraft major assemblies shall be electrically grounded during spraying operations.

3) Operators shall wear suitable protective respiratory equipment when applying corrosion preventive compound.

C. Preparation of Corrosion Preventive Compounds

WARNING **KEEP SOLVENT AWAY FROM FIRE AND OTHER SOURCES OF IGNITION, PROVIDE SUFFICIENT VENTILATION WHEN WORKING IN CONFINED SPACES, AND AVOID SKIN CONTACT WITH SOLVENT.**

1) Hard Film Compound

For brush, spray, dip, oil can or pour application, use as received.

Note Do not thin hard film, Grade 1 compounds, with solvents.

2) Soft Film Compound

For dip or brush application, dilute soft film compound with varsol to the consistency of thin syrup.

3) Soft Film Compound (Boeshield and LPS No. 3).

For brush or spray application, use as received from manufacturer.

2. Preparation of Parts

   A. Prepare Parts

      1) All operations, including welding, heat treatment, sand blasting, plating and painting, must be completed prior to applying corrosion preventive compound, except that open end tubes may be treated before painting, provided that sufficient time is allowed for the film to dry, and the outside of the tube is adequately cleaned prior to painting.

      2) Parts must be free from rust, oxides, etc., before applying corrosion preventive compound.

      3) Except for large parts or assemblies, degrease parts contaminated with shop oil or grease before applying corrosion preventative compound solvent clean large assemblies.

3. Application of Hard Film Compound (Grade 1)

   A. Open End Tubes

      1) Apply F13 Grade 1 hard film to the inside of open ended tubes as follows:

         a) Mask off any open holes or cut-outs. If the tube is open at both ends, mask off one end.

         b) Fill the tube with F13 Grade 1 hard film slowly to prevent any air bubble formation.

         c) Completely drain excess film from the inside of the tube.

         d) Position the part vertically and allow to dry for at least 4 hours.

         e) Ensure that the inside of the tube is completely covered with the film (i.e., the film should be continuous and uniformly black). Re-coat if necessary.

   B. Sealed Tubes

      1) Sealed or closed tubes or parts must be treated by pressure oiling through the drilled holes provided for this purpose. When sufficient compound is inside, seal the holes temporarily with tape. Rotate and upend the part to ensure that the film contacts all interior surfaces.

      2) Remove tape and allow excess compound to drain completely. Multi-sectioned parts must be turned frequently to ensure drainage from all low points and pockets.

      3) When the parts are thoroughly drained, seal the drill holes with self-plugging cherry rivets.

         WARNING KEEP SOLVENT AWAY FROM FIRE AND OTHER SOURCES OF IGNITION, PROVIDE SUFFICIENT VENTILATION WHEN WORKING IN CONFINED SPACES, AND AVOID SKIN CONTACT WITH SOLVENT.

      4) Clean all excess compound from exterior surfaces by wiping with a clean cloth dampened with NAPHTHA. Wipe dry with a clean cloth before solvent evaporates.

   C. Small Parts

      1) Small parts and assemblies on which the surfaces to be treated are readily accessible may be treated by dipping or brush coating with a thin uniform film of compound.

4. Application of Soft Film Compound (Grade 2)

   A. Small Parts

      Small parts and assemblies may be treated by dipping or brush coating with a thin uniform film of compound.

5. Application of Soft Film Compound (Grade 3 Type I and Type II)

A. Small Parts

Small parts and assemblies may be treated by brush or spray coating using mist action hand trigger spray bottles or standard paint spraying equipment to achieve a thin uniform film of compound.

B. Large Parts

Large parts or areas shall be treated by spraying using mist action hand trigger spray bottles, standard paint spraying equipment or extension spray equipment to achieve a thin film of compound.

Note When spraying compound on to the interior of an aircraft, the following items shall be suitably masked or shielded from direct spray or over spray contamination:

- Black boxes with cooling holes
- Windows
- Oxygen tubes and fittings
- All other system tubing open ends
- Control cables and associated hardware
- Systems components

Note A light over spray on the following items is not detrimental, but direct spray shall be avoided:

- Bearings
- Pulleys
- Rubber items
- Non-oxygen system tubes

Note Subsequent to spray application, enclosed areas shall be allowed to dry in accordance with one of the following schedules before resuming work on the aircraft or assembly:

- Drying Time:
  - 1 hour with forced ventilation
  - 6-8 hours with natural ventilation

6. Removal of Film

Hard and soft film may be removed by washing the part with DS108, Diestone DLS or Diestone HFP.

7. Touch-Up and Repair

A. If corrosion preventive compound has been removed from treated surfaces, touch-up these areas with the same type of compound as originally used on the parts. It is acceptable to use aerosol cans, if available, containing the same compound for touch-up.

B. If parts have been reworked after application of corrosion preventive compound, ensure that the surfaces are free of swarf or chips before re-application of compound.

8. Requirements

A. All areas must be completely covered with a continuous film of compound.

B. Insides of open tubes must be examined for continuity of coating by examining the bore against a light. Uncovered areas will show up as dark patches.

C. The film thickness shall be such that the applied coating is continuous, but running of the compound is kept to a minimum.

Chemical Conversion Coatings (Alodine) – Manual Application

1. General

WARNING **THIS PROCEDURE USES MATERIALS THAT MAY BE DANGEROUS TO THE PERSONNEL'S HEALTH AND/OR THE ENVIRONMENT. SUFFICIENT PRECAUTIONS MUST BE TAKEN AT ANY STEP OF USE. IF NOT, THIS CAN CAUSE DAMAGE TO THE ENVIRONMENT AND DANGEROUS EFFECTS ON THE PERSONNEL'S HEALTH AND SAFETY.**

Manually apply chemical conversion coating solution only if application by immersion is not permitted due to the part's size or the assemblies' potential for solution entrapment. For touch-up of small surfaces (i.e. spotfacing, countersinks, repair areas etc.), it is recommended that Touch-N-Prep Alodine 1132 markers be used.

If assemblies contain materials other than aluminum or titanium, take extreme care when processing these assemblies because the chemical conversion coating solution may cause corrosion. It is recommended that material other than aluminum or titanium be masked off when processing using aluminum foil tape.

Manual application meets all requirements of MIL–DTL–5541 and covers two classes of chemical conversion coatings formed by the reaction of the chemical materials and the surfaces of aluminum and aluminum alloys:

- Class 1A coatings are corrosion protective films for parts, repair and touch-up applications and will improve paint adhesion for aluminum and aluminium alloys.
- Class 3 coatings are corrosion preventative films for electrical and electronic applications where low resistance contacts are required.

2. Materials and Equipment

- Alodine 1200S
- De-Ionized Water
- Scotch-Brite Pads
- Fine Abrasive Paper
- Suitable Brushes
- Polyethylene Container
- Clean Cotton Wipers
- Approved Safety Glasses
- Protective Rubber Gloves
- Deoxidine 624 (or Alumi Prep 33 or Turco W.O. 1)
- Approved Respirator
- Cellulose Sponge
- Touch-N-Prep Alodine 1132 Marker
- Aluminum Foil Tape, adhesive backed (3M #425)

3. Safety Precautions

A. Wear neoprene rubber gloves (refer to Para 2.), aprons, boots and approved chemical splash goggles while carrying out chemical conversion coating operations.

B. Ensure adequate ventilation is supplied to the areas where the chemical conversion coating process is being carried out. Avoid breathing the fumes or vapours during application.

C. Take care to prevent the powder from being distributed into the air. Airborne powder will cause a serious health hazard. Wear protective respiratory equipment when handling conversion coating materials.

D. Avoid skin contact with solutions. If skin contact occurs, wash the affected area immediately with large quantities of water. If irritation of the skin occurs, contact a First Aid attendant immediately.

E. Avoid eye contact with chemicals or solutions. If eye contact occurs, immediately flush eyes in a directed stream of water for at least 15 minutes while forcibly holding eyelids apart to ensure completed irrigation of all eye and lid tissue. Contact a First Aid attendant and a physician.

F. Avoid ingestion of chemicals or solutions. If ingestion occurs, do not induce vomiting. Drink large amounts of water or milk of magnesia, but never give anything by mouth to an unconscious person. Immediately contact a First Aid attendant and a physician.

4. Cleaning before Manual Application

WARNING KEEP SOLVENT AWAY FROM FIRE AND OTHER SOURCES OF IGNITION. PROVIDE SUFFICIENT VENTILATION WHEN WORKING IN CONFINED SPACES. AVOID SKIN CONTACT WITH SOLVENT.

Before manual chemical conversion coating as specified in Para 6. (except Type II IVD aluminum coated parts):

- Degrease as per Para 4.A
- If heavy oxides are present, mechanically clean according to Para 4.B
- Manually deoxidize according to Para 4.C only if using Alodine 1200S
- If Touch-up refer to Para 6.B, do not deoxidize

A. Degreasing and Manual Solvent Cleaning

1) Apply a small amount of solvent to a clean wiping cloth. Refer to 20-40-01, Silicone Adhesives – General Data Table 1, for the solvent appropriate for the surface to be cleaned. For solvents which evaporate quickly, apply more solvent than for solvents which evaporate slowly.

2) Wipe the area with the solvent dampened wiping cloth.

a) When cleaning bond surfaces with solvent saturate a clean lint-free cloth, and thoroughly scrub the bond surfaces. Wipe dry with a clean lint-free cloth before the solvent evaporates.

Note Wiping cloth (CID A–A–59323) shall be lint free, 100 percent cotton cloth conforming to A–A–2522, Grade A, colour 1: cotton gauze/cheese cloth conforming to CCC–C–440, Type II or III, which are unbleached or white cleaning cloths conforming to AMS 3819A, Class 2, Grade A, and have not been exposed to any other chemical solutions.

b) Keep solvent cloth saturated and turn frequently to avoid reusing a dirty area.

3) Wipe dry with another clean wiping cloth before the solvent evaporates. It is important to dry the surface immediately after cleaning because if the solvent evaporates before it is wiped up, oil and grease residue will be left on the cleaned surface. Although DS108 solvent has a slow evaporation rate, wipe dry as soon as possible as delay will result in a longer lingering odour.

4) Check for a water break free surface.

5) If more cleaning is needed, apply fresh solvent to a clean portion of the cloth and repeat Para 4.A.2), Para 4.A.3) and Para 4.A.4).

B. Mechanical Cleaning

1) Remove surface oxides or heavy deposits of contamination using aluminum wool or scotch-brite pad.

2) Dust off loose particles with a clean cloth.

C. Manual Deoxidizing (Deoxidine 624, Alumi Prep 33 or Turco W.O.1)

1) Apply manual deoxidizer using a cellulose sponge or sheepskin roller.

Note Close tolerance holes and surfaces must be masked before deoxidizing. Close tolerance holes do not require masking if final reaming of close tolerance holes will be done at the assembly stage.

Note Clean only small areas at a time. Do not allow the solution to dry on the surface. If necessary, apply the deoxidizer to a section of the part at a time.

2) Initial water rinse (must occur within 5 minutes of applying deoxidizer).

3) Wipe surface clean with water soaked cloths.

Note If the surfaces have been chemically conversion coated, wipe lightly to prevent damage to the coating.

Note Rinse cloths with clean water frequently.

4) Apply water moistened litmus paper while the surface is wet. If the litmus paper changes colour repeat Para 4.C.

5) If the component has laps or seams then apply water moistened litmus paper to the laps or seams. If litmus paper changes colour, blow out seams and laps with clean filtered compressed air. Repeat step until paper does not change colour.

If the component does not have laps or seams, if specified, manually apply chemical conversion coating in accordance to Para 6.

5. Preparation of Alodine 1200S Solution

WARNING WEAR RESPIRATOR TO PREVENT INHALATION OF ALODINE POWDER.

A. Prepare Alodine 1200S solution as follows:

1) Fill an acid resistant mixing container with clean luke warm water.

2) Carefully add the Alodine 1200S powder 1.0 to 3.0 wt oz per US gallon.

3) Thoroughly stir the solution until the powder is dissolved. Disregard the small amount of insoluble material which may settle out of solution.

B. Use Touch-N-Prep Coating Alodine 1132 marker as supplied.

6. Manual Application of Alodine 1200S Solution

A. Application of Alodine 1200S Chemical Conversion Coating

CAUTION TAKE EXTREME CARE WHEN PROCESSING ASSEMBLIES CONTAINING METALS OTHER THAN ALUMINUM. THE SOLUTIONS USED MAY BE CORROSIVE TO THESE METALS.

1) Apply chemical conversion coating to the surfaces of cleaned parts as follows:

a) For aluminum parts, apply conversion coating solution to the surface using a brush. Maintain a continuous wet surface.

b) Leave the solution on the part for 1 to 5 minutes. Do not allow the solution to dry on the surface. If drying does occur, re-wet the surfaces with solution before water rinsing.

c) Gently wipe all surfaces with a clean cloth soaked in the conversion coating solution just before rinsing.

d) Rinse parts thoroughly with water not exceeding a conductivity limit of 600 µmhos and pH of 5.0 to 8.0. Chemical salts trapped under a paint film will eventually result in blistering or corrosion problems.

e) Wipe all part surfaces dry using a clean lint-free cloth. Use clean oil free compressed air to blow dry treated surfaces such as lap joints, seams, pockets, etc.

f) Carefully inspect all treated surfaces and if necessary, re-apply the solution to bare or starved areas according to Para 6.A.1).a) through Para 6.A.1).e) in order to get the desired coating thickness. The coating colour ranges from light gold to tan depending on the metal type, temperature, solution concentration and number of applications. Ensure that the coating is not powdery.

g) For aluminum parts, prime parts as soon as possible after drying.

B. Touch Up of Chemical Conversion Coating Using Touch-N-Prep Alodine 1132 or Alodine 1200S

1) If using a Touch-N-Prep Coatings Alodine 1132 marker, touch-up parts as follows:

a) Remove cap.

b) If necessary, the applicator tip can be cut to conform to any shape with a single edge razor blade.

c) Hold unit with applicator tip down.

d) To activate, press tip against a firm surface. This will open the valve allowing Alodine 1132 to reach the applicator tip. A new unit should charge in 30 – 45 seconds. When the Alodine wets the tip, release pressure. Unit is ready to use.

e) Check to ensure that the surface to be coated is dry.

f) Apply Touch-N-Prep Coatings Alodine 1132 to the repaired surface in smooth, even strokes. Be sure to cover edges. For best results, the initial coating should be applied in one direction using a 50% overlap on each pass. After allowing the first coat to dry, apply the second coat at right angles to the first coat with 50% overlap. Do not allow the solution to puddle (puddling may result in improper drying leading to paint adhesion failures, flaking, etc.). A pre-warmed surface will result in a smoother coating. To recharge the applicator tip, repeat Para 6.B.1).d). Replace cap when not in use.

g) Allow the chemical conversion coating to air dry, force dry with warm air (eg. hair dryer) or using a heat lamp (130°F maximum). Do not rinse.

2) Except as specified in Para 6.B.1), touch-up parts with small surface areas with chemical conversion coating solution as follows:

a) Apply chemical conversion coating solution to prepared surface using a pencil brush, small Scotch-Brite pad or foam tip applicator to fully cover the applicable area. Prevent excess solution from dripping onto the surrounding structure.

b) Allow the solution to remain on the part surface for 1 to 5 minutes before washing off. Do not allow the solution to dry on the surface. If drying does occur, re-wet the surface with solution before water rinsing.

c) Remove all traces of solution by rinsing with clean water or wiping with water-damp cloths followed by wiping dry with clean lint-free cloth or blowing dry with clean oil-free compressed air.

C. Equipment Notes

1) Containers to hold the Alodine 1200S solution must be constructed from stainless steel, plastic, rubber or other acid proof material. Lead, glass, tin or galvanized iron must not be used.

2) Storing the solution in mild steel containers will result in a slow decomposition of the fluid.

7. Requirements

A. The treated surfaces must be free from bare or starved areas.

B. The coating must be continuous and free from powder, imperfections and excessive blotches which may result in poor paint adhesion.

C. The coating must be light gold to tan in colour.

8. Additional Information

A. Handling of Processed Parts

1) Do not touch conversion coated surfaces with bare hands. Wear clean cotton gloves when handling such parts.

2) For parts to be transported for further processing or to be held for more than 24 hours, wrap the parts completely or interlace with Kraft paper to ensure that the conversion coated surfaces are not exposed to contamination. Coated parts that are to be primed within 24 hours do not require protective wrapping but must be kept free from contamination and be handled only while wearing clean cotton gloves.

B. Defective Coatings

1) One or more of the following conditions may cause the conversion coating to be too light:

- The dwell time of alodine on the part was too short.
- The chemical concentration of the alodine solution is too low.
- The pH of the alodine solution is not within the specified limits.

2) One or more of the following conditions may be the cause of abnormal amounts of powder being encountered:

- The parts were improperly cleaned.
- The chemical concentration of the alodine solution is too high.
- The dwell time of the alodine on the part was too long.
- The pH of the alodine solution is not within the specified limits.

9. Storage

A. Store Alodine 1200S in a cool dry area away from organic compounds or easily oxidizable materials. Alodine 1200S is an acidic oxidizing agent and is not affected by freezing.

B. Store Touch-N-Prep Coatings Alodine 1132 markers away from incompatible materials. Alodine 1132 contains chromate. Protect from freezing. Do not tamper with packaging and do not try to refill used markers.

10. Disposal of Solutions

A. Dispose of chemical conversion coating solutions according to MSDS and all applicable regional regulations.

B. Dispose of used Touch-N-Prep Coatings Alodine 1132 markers by re-packaging used markers and mail back to supplier for disposal (mailing label is supplied in original packaging).

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Silicone Adhesives – General Data

1. General

WARNING **THIS PROCEDURE USES MATERIALS THAT MAY BE DANGEROUS TO THE PERSONNEL'S HEALTH AND/OR THE ENVIRONMENT. SUFFICIENT PRECAUTIONS MUST BE TAKEN AT ANY STEP OF USE. IF NOT, THIS CAN CAUSE DAMAGE TO THE ENVIRONMENT AND DANGEROUS EFFECTS ON THE PERSONNEL'S HEALTH AND SAFETY.**

This standard specifies the procedure and requirements for bonding aircraft parts with PSA 529/SRC–18 two part adhesive.

A. Materials and Equipment

- NAPHTHA
- Suitable grooved work holder
- Disposable wax-free paperboard containers
- Eye Dropper
- Clean cotton gloves
- Masking tape
- Weighing scale; capable of weighing to within + or – 0.1 grams
- PSA 529/SRC–18
- Self-adhesive labels (refer to Figure 1)
- Disposable wooden tongue depressor

B. Surface Treatment

1) Aluminum alloy parts must have bonding surfaces primed with fluid resistant epoxy primer.

2) Bonding surfaces of cadmium plated parts must be primed with fluid resistant epoxy primer.

3) Precautions must be taken and masking employed where necessary to ensure that plexiglass parts have only their bonding surfaces scuffed.

C. Safety Precautions

1) Adhesive, primer and solvents must be kept away from fire and other sources of ignition.

2) Sufficient ventilation must be provided when using solvent, primer or adhesive in confined areas.

3) Skin contact with solvent, primer and adhesive must be avoided; wear protective gloves.

4) Protective hand cream must not be used as it may cause contamination of cleaned or adhesive coated surfaces.

D. Cleaning the Bonding Surfaces

1) Surfaces to be bonded must be clean and completely free of contamination such as dirt, mould release agent, grease or finger marks.

2) When cleaning bonding surfaces with solvent saturate a clean lint-free cloth with solvent, as specified, and thoroughly scrub the bond surfaces. Wipe dry with a clean lint-free cloth before the solvent evaporates.

3) Keep solvent cloth saturated and turn frequently to avoid reusing a dirty area.

4) When cleaning or applying adhesive to long lengths of silicone rubber extrusion, a suitable grooved work holder should be used to hold and support part.

5) Cleaned, primed or adhesive coated surfaces must not be touched with bare hands or otherwise subjected to contamination. Wear clean cotton gloves when handling prepared surfaces.

6) Refer to for cleaning and preparation of various materials.

E. Preparation of Adhesives

Note PSA 529/SRC–18 adhesive is received in kits from the supplier. Each kit includes resin in a 4 fl. oz. glass bottle and a 3.3 ml glass bottle of catalyst.

1) Remove cap from the bottle of adhesive resin.

2) Add the contents of the bottle of catalyst directly into the bottle of resin. The kit supplies the resin and catalyst in the correct mixing ratio.

3) Using the disposable wooden tongue depressor, stir the resin/catalyst mixture thoroughly to obtain a homogeneous air-free mix. Immediately cap and close the bottle tightly.

4) Label the bottle according to Figure 1 and stamp with the expiry date (mixing date + 3 months pot life).



Shelf Life Adhesive Label
Figure 1

Table 1 Solvent Selection For General Materials

MATERIAL TO BE CLEANED	SOLVENT (Note 1)
Electrical connections	IPA
Bare metal	DS108, Diestone DLS, Diestone HFP or MEK
Primed or painted metal parts	DS108, Diestone DLS, Diestone HFP or MEK (Note 1)
Unprimed plastic parts (Note 2)	NAPHTHA
Unprimed fiberglass or phenolic (Note 3)	DS108 or MEK
Thermoplastic (Note 4)	Methanol or Naphtha
Thermoset plastic (Note 5)	MEK
Rulon A	DS108
Rubber (Note 6) except silicone	Naphtha

Table 1 Solvent Selection For General Materials (continued)

MATERIAL TO BE CLEANED	SOLVENT (Note 1)
Silicone rubber	IPA
Porous materials (Note 7)	Do not clean porous materials in any way. If the surfaces are contaminated, do not use the materials for bonding.

Note 1 – Use DS108 for cleaning aircraft exterior before painting.

Note 2 – Plastic parts include Formica, Arborite, etc. Plastic parts do not include Kevlar and fiberglass.

Note 3 – Fiberglass includes Kevlar laminates and composites. Phenolic laminates do not include Formica, Arborite, etc.

Note 4 – Thermoplastic: acrylic (Plexiglas), Acrylic/PVC (kydex), nylon (Zytel), polycarbonate (Lexan), PTFE (teflon), PVC/PVF (Tedlar).

Note 5 – Thermoset plastics: fibre-reinforced epoxy, polyester or phenolic composites.

Note 6 – Rubber: neoprene, Buna-N (nitrile, NBR).

Note 7 – Porous materials: Velcro, fabrics, balsa, cork, etc.

2. Bonding

A. Application

1) Apply masking tape adjacent to area to be bonded to prevent adhesive contamination of the adjacent surfaces.

2) Ensure areas to be bonded are clean (refer to Para 1.D and Table 1 — Solvent Selection For General Materials).

3) Pour required amount of catalyzed adhesive into suitable wax-free container. Immediately re-cap bottle.

4) Apply a thin uniform coat (0.010 to 0.015 inch) to both joining surfaces using a bristle brush of suitable size. Ensure that both surfaces are completely covered, including butt ends of silicone foam extrusions.

Note A grooved holder may be used to support the work (if applicable).

5) Allow adhesive to dry until a firm tack develops.

6) Join parts together and roll down with a rubber roller or stitch roller, or press firmly with fingers to ensure complete contact throughout.

Note Parts must be carefully positioned initially as bond must not be broken and repositioned.

7) Where the silicone extrusion tends to lift due to distortion, as around corners, apply moderate pressure by means of masking tape or other suitable means, until bond has cured sufficiently.

8) The bond must be allowed to cure for a minimum of 24 hours at 72°F (22°C) (room temperature) before further working the assembly or installing it in the aircraft.

Note Discard unused adhesive. Do not pour back into catalyzed bottle.

a) Bonded parts must have complete contact over the full bonding area.

b) Visual indication of poor adhesion is cause for rejection.

3. Storage

A. Store adhesive components at 60°F to 80°F (15°C to 26°C) with the storage precautions necessary for flammable materials.

B. Ensure that the containers are clearly stamped with the storage life expiry date. Refer to the MSDS and/or manufacturer's recommendations for the storage life of the adhesive components.

C. Keep containers of adhesives tightly closed when not in use.

Weather Sealing – General Data

1. General

WARNING **THIS PROCEDURE USES MATERIALS THAT MAY BE DANGEROUS TO THE PERSONNEL'S HEALTH AND/OR THE ENVIRONMENT. SUFFICIENT PRECAUTIONS MUST BE TAKEN AT ANY STEP OF USE. IF NOT, THIS CAN CAUSE DAMAGE TO THE ENVIRONMENT AND DANGEROUS EFFECTS ON THE PERSONNEL'S HEALTH AND SAFETY.**

For fay surface sealing use sealant to MIL–PRF–81733 Type IV, Class 1, Grade A (P/S 870 C). For fillet or butt joint sealing use sealant PR–1776MB.

A. Location of Sealant

The sealant is used:

1) To fill butt joints around door sills, hatches, and external fittings which penetrate skin panels.

2) Between faying surfaces.

3) Fillet sealing along edges of skin panels.

4) To dome seal fasteners.

B. Safety Precautions

1) Keep solvents and sealant away from fire and other sources of ignition.

2) Provide sufficient ventilation when working in confined spaces; avoid breathing of solvent or sealant fumes.

3) Wear protective respiratory equipment when spraying sealant or working in spray sealing areas.

4) Avoid skin contact with solvents or sealants. Where skin contact has occurred, the affected area should be washed thoroughly with soap and water.

5) In case of sealant or solvent ingestion, immediate medical attention should be obtained.

6) Splash goggles should be worn when mixing and handling sealant components.

7) When eye contact with sealing components has occurred, thoroughly flush eye with water and obtain medical attention.

2. Preparation of Sealant

A. Prepare Sealant as follows:

1) Thoroughly stir base compound and accelerator in their original separate container until a uniform consistency is achieved.

2) When supplied in kit form, add all accelerator supplied to the base compound and mix slowly and thoroughly. Scrape sides and bottom of container and mixing paddle periodically, to ensure thorough mixing. Mix for five or six minutes. If required, transfer mixed sealant to polyethylene cartridge and use with applicator gun.

3) When packaged in bulk, mix according to manufacturer's instructions. Mix thoroughly as in Para 2.A.2).

B. Pot Life of Mixed Adhesive

1) Pot life is the time and condition during which mixed sealant remains suitable for application.

2) Refer to manufacturer's data for sealant pot life. High humidity and temperature conditions will shorten the pot life.

3. Preparation of Parts

A. Prepare parts as follows:

WARNING KEEP SOLVENT AWAY FROM FIRE AND OTHER SOURCES OF IGNITION. PROVIDE SUFFICIENT VENTILATION WHEN WORKING IN CONFIRMED SPACES. AVOID SKIN CONTACT WITH SOLVENT.

1) Apply small amount of solvent to a clean wiping cloth. Refer to 20-40-01, Silicone Adhesives – General Data, Table 1 for the solvent appropriate for the surface to be cleaned. For solvents which evaporate quickly, apply more solvent than for solvents which evaporate slowly.

2) Wipe the area with a solvent dampened wiping cloth.

a) When cleaning bonded surfaces with solvent saturated wiping cloth, thoroughly scrub the bonding surfaces. Wipe dry with cloth before solvent evaporates.

b) Keep solvent cloth saturated and turn frequently to avoid reusing a dirty area.

3) Wipe dry with another clean wiping cloth before the solvent evaporates. It is important to wipe the surface immediately after cleaning because if the solvent evaporates before it is wiped up, the oil, and grease residue will remain on the surface. Although the DS 108 solvent has a slow evaporation rate, wipe dry as soon as possible as delay will result in a longer lingering odour.

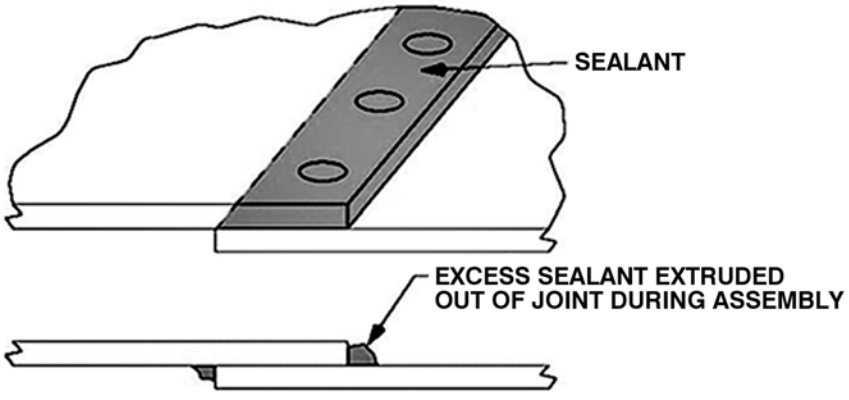
4) If more cleaning is needed, apply fresh solvent to a clean portion of the wiping cloth and repeat Para 3.A.1) through to Para 3.A.3).

5) Areas to be sealed which contain acrylic plastic (plexiglass) parts must be cleaned with NAPHTHA spirits.

B. Application of Sealant

1) Faying Surface Sealing

a) A faying surface seal is a thin, even layer of sealant between two overlapping surfaces preventing leakage of air or liquid from a sealed area to an unsealed area (refer to Figure 1).



Faying Surface Seal
Figure 1

b) Fay seal as follows:

1 If necessary, mask off adjacent areas using masking tape and/or cardboard.

2 If using a 3" mohair roller, pour a small quantity of sealant onto a clean piece of polyethylene plastic sheet and saturate the roller surface with sealant. If using a standard 1/2" bristle brush, pour the sealant into a clean wax free container and saturate the brush with sealant.

3 Apply a uniform coat of sealant to the faying surface of one part. Completely cover the faying surface leaving a smooth layer approximately 0.005" thick if using the brush method or 0.004" thick if using the roller method. Avoid applying the sealant too heavily or over-brushing applied sealant.

4 Draw parts together using Cleco fasteners or slave bolts in at least every fourth fastener hole. Install temporary fasteners with a washer under the head and nut to prevent marking the part surfaces.

5 Re-tighten wing-nut Clecos or slave bolts approximately 5 minutes after initial tightening.

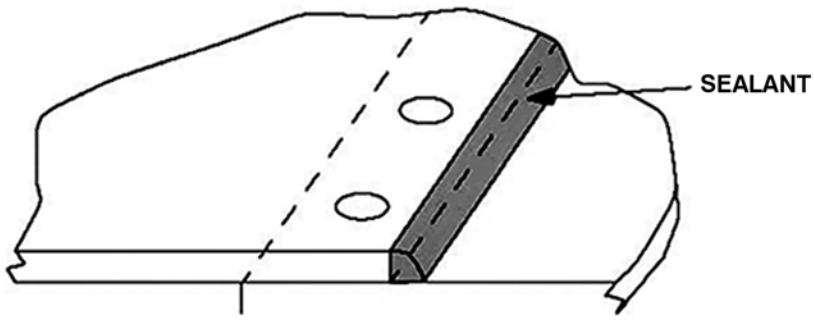
6 Install permanent fastners starting at the centre of the pattern or line and working outwards, removing Clecos or slave bolts as installation progresses. Do not remove temporary fasteners until installing permanent fasteners. Remove sealant extruded into fastener holes before inserting fasteners or wipe sealant from fastener ends, locking grooves or threads after insertion.

7 Torque or drive all fasteners at least twice within the assembly life of the sealant. Allow a minimum of 5 minutes between torquing operations.

8 Wipe sealant from fastener ends. Remove and fair the excess sealant extruded along both sides of the overlap after assembly using a sealant scraper before the sealant cures (refer to Figure 1). Minimum visible sealant shall remain along the entire mating surfaces.

2) Fillet Sealing

a) A fillet seal is a bead of sealant applied as a dam along a seam, on either the pressure or weather side of the structure, preventing the passage of air or liquid through the joint (refer to Figure 2).



General Description of Fillet Seal
Figure 2

b) Apply fillet seals as follows:

1 Load a sealant gun with sealant. For overhead fillet sealing, use sealant that has approximately half of its pots life expired.

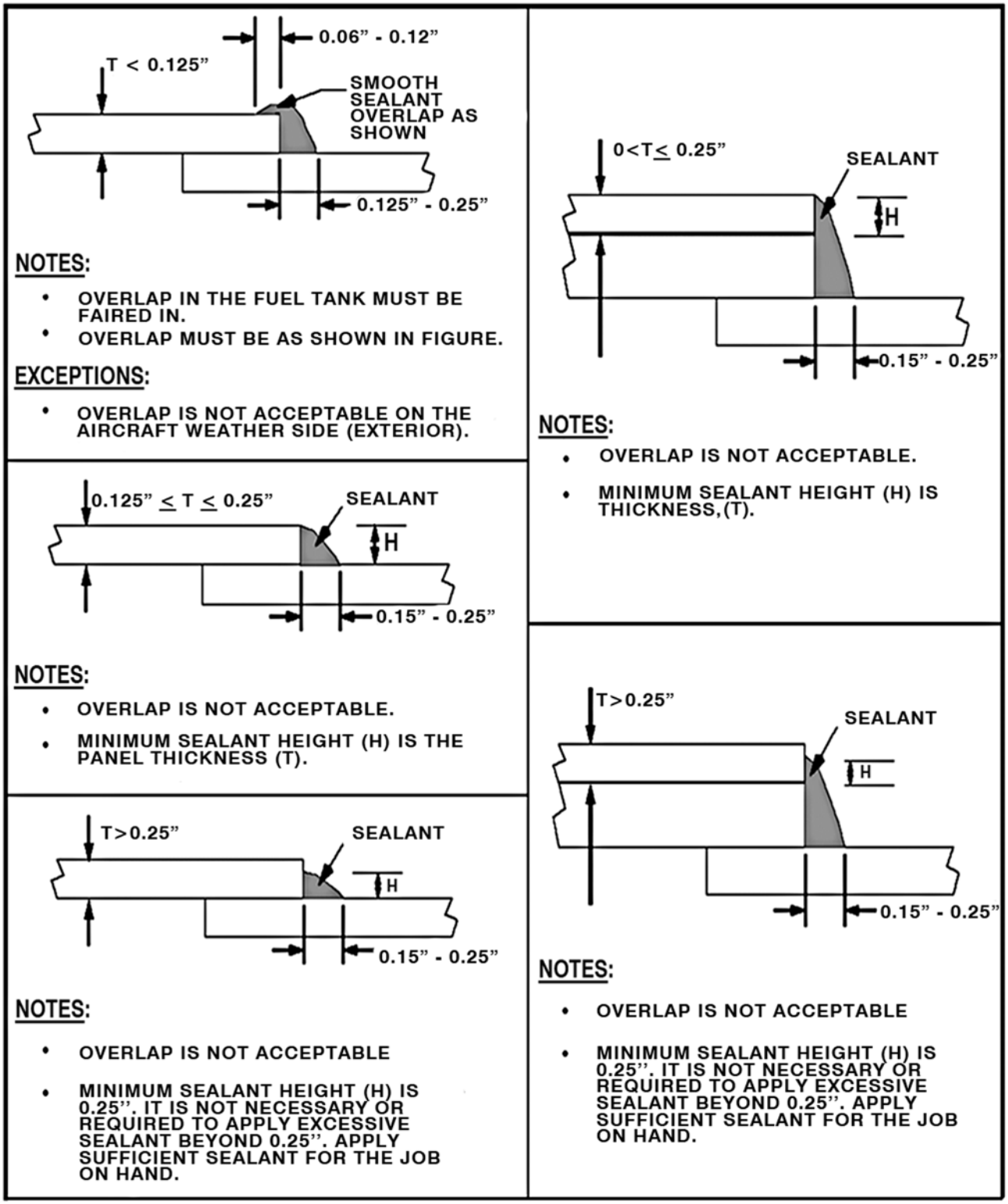
2 Hold the flared or standard nozzle at 90° to the direction of travel and 45° to the surface of the parts.

3 Squeeze the gun trigger and move the nozzle along the seam in approximately 3 foot increments at a rate that maintains a slight build-up of sealant in front of the nozzle. This will force the sealant into the root of the angle and exclude air from the bead. If required, cut nozzles back to obtain larger holes.

4 Ensure that applied sealant beads do not extend across the manufactured heads of installed fasteners (refer to Figure 3).

5 After applying each 3 foot increment, examine the seal for air bubbles. Open any air bubbles and fill them with fresh sealant. When opening the bubbles, make the cavities large enough to permit the fresh sealant to fill them.

6 If a sealant bead overlaps onto the structure in a fuel area, fair the sealant bead into the structure as shown in Figure 3.

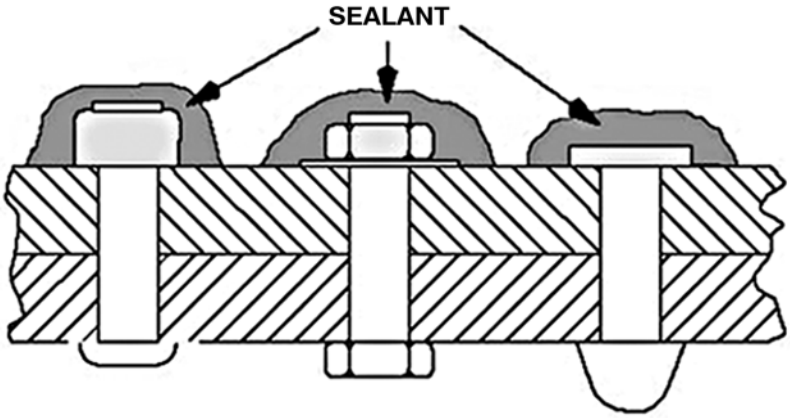


Acceptable Application of Fillet Seals
Figure 3

3) Dome Sealing Fasteners

a) Unless specified dome sealing fasteners that penetrate pressure or fluid light structures, dome seal on the pressure side.

b) Dome seals must completely cover the fastener head or end (including attachment hardware, nuts, washers, locking collars, etc.) and overlap the adjacent structure by approximately 1/4" (refer to Figure 4).

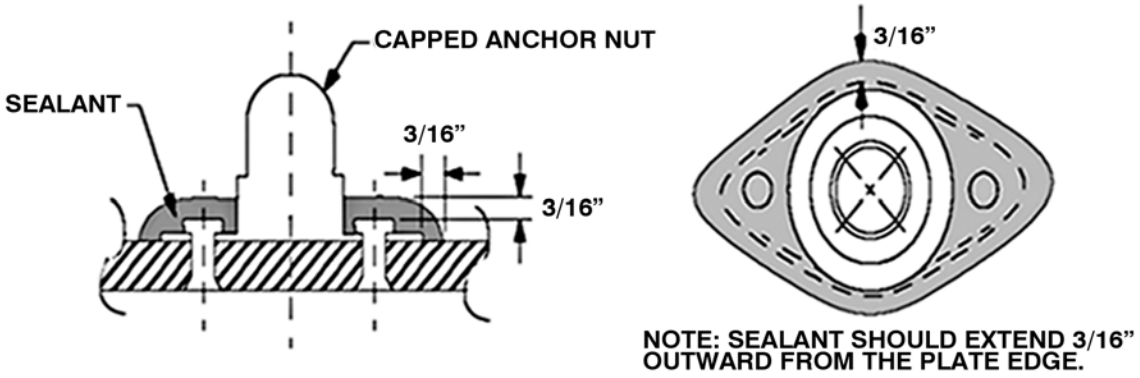


Typical Dome Sealing Of Fasteners
Figure 4

c) If using modified nozzles for dome sealing rivets and lockbolts, ensure the nozzle is full of sealant and free of voids before applying the first dome seal in a series.

d) It is acceptable to use a modified flared nozzle for dome sealing NAS 1473A–3 and NAS 1473A–4 capped anchor nuts. Ensure the nozzle is full of sealant and free of voids before applying the first dome seal in a series.

e) Dome seal anchor nuts approximately as follows (refer to Figure 5).



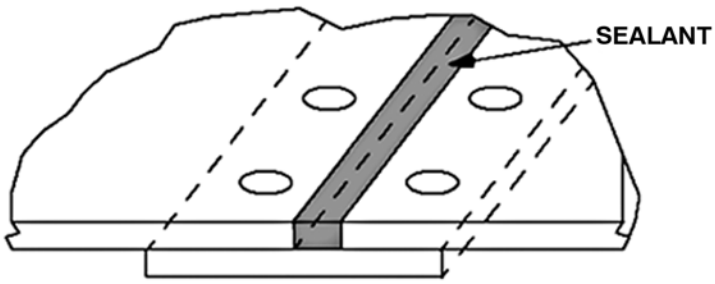
Dome Sealing Anchor Nut
Figure 5

1 Hold the sealant gun perpendicular to the work with the modified flared nozzle completely covering the anchor nut.

2 Apply the dome seal to completely cover the anchor nut plate. Ensure that the sealant dome is at least 0.1875” (3/16”) above each rivet shop head and also extends 0.1875” (3/16”) beyond the edge of the anchor nut plate as shown in Figure 5.

4) Butt Joint Sealing

a) A butt joint seal is a bead of sealant applied to the gap between the parts that butt together, such as skin panels, providing an aerodynamically smooth surface (refer to Figure 6).



Butt Seal Joint
Figure 6

b) Avoid applying butt joint seals between exterior skin panels within 72 hours before painting the aircraft exterior.

c) Apply butt joint seals as follows:

1 Load a sealant gun within sealant. Use a standard nozzle. If necessary, cut the nozzle back to obtain a larger hole.

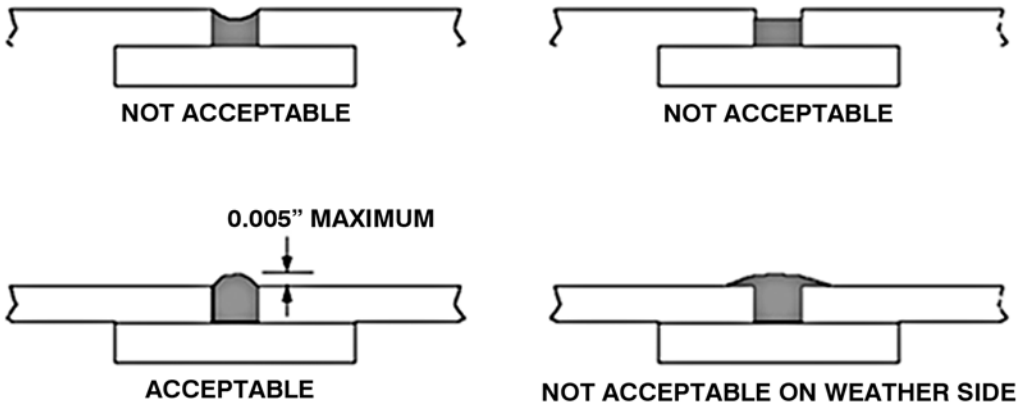
2 Hold the nozzle in line with the joint, at approximately 45° to the part surface, and place the nozzle tip in the bottom of the joint.

3 Squeeze the trigger and move the nozzle back along the joint in approximately 3 foot increments at a rate that allows the sealant to continuously fill the joint flush with the adjacent surfaces. Avoid air entrapment in the sealant bead. The cured sealant must be flush (0” to + 0.005”) with the surrounding structure (refer to Figure 7). Ensure sealant does not overlap onto adjacent skin surfaces on the weather side of a structure when aerodynamic smoothness is a requirement.

4 After applying each 3 foot increment of sealant, examine the bead for air bubbles. Open any air bubbles and fill them with fresh sealant. When opening bubbles, make the cavities large enough to permit the fresh sealant to fill them easily.

5 If necessary, smooth out the sealant bead with a sealant spatula.

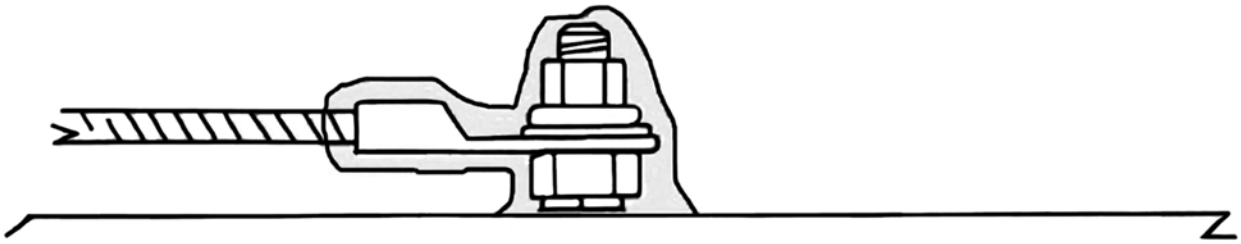
d) Refer to Figure 7 for acceptable and unacceptable butt joint sealing configurations.



Butt Seal Configurations
Figure 7

5) Dome Sealing Electrical Connections

a) Dome seals must completely cover the electrical connection (including attachment hardware and entire terminal lug) and overlap the adjacent structure by a minimum of 1/4" (refer to Figure 8 for details).



Dome Sealing Connections
Figure 8

b) Use P/S 870 Class A sealant and apply by brush. P/S 870 Class B sealant may be used for any touchups.

C. Resealing

Seams and joints which show signs of leakage can be resealed by laying a fillet of sealant along the weather edge of the defective seam or joint. Use applicator gun to pressure feed the sealant into the leak path. Allow to cure, as in Para 4.

4. Sealant Curing

A. Curing of the mixed sealant is as per manufacturers recommendation.

1) The curing rate of mixed sealants varies greatly with changes in temperature or humidity. Sealant curing is extremely slow when the ambient temperature is less than 60°F.

Note Lower temperature and relative humidity extends cure time. Higher temperature and relative humidity shortens application life.

2) Tack-free cure is the time required for the sealant to cure sufficiently such that shop swarf, chips, etc. will not stick to the sealant. To prevent contamination of the seal, do not do further work on sealed areas until the sealant is tack-free.

5. Clean-Up

A. Remove excess sealant from structures and parts, before the sealant material has cured, with DS108 or MEK (except as noted in Para 5.B).

B. Acrylic plastic (plexiglass) parts must be cleaned with NAPHTHA.

6. Maintenance of Equipment

Clean all equipment with methyl ethyl ketone before the sealant materials have cured.

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Sealant and Adhesives – General Data

1. Sealant and Adhesives – List of Manufacturers

Table 1

Sealant/Adhesives Prefix	Manufacturers	
PR	PPG	
	PPG Aerospace North America	Sales Office Pheonix 22665 Dobson Rd. Suite 200 Mesa AZ 85202
	Europe	France Victoria Center Building A2 20 Chenin Laporte 313000 Toulouse France.
	Far East	Hong Kong Suites 1010–1015, 10F Cityplaza One 1111, King's Road Taikoo Shing Hong Kong
GC	Goal Chemical Sealants Corp. refer to PPG	
PRO-SEAL	Pro-Seal refer to PPG	
CS	Chemseal Corp.	
	North America	Chemseal Corp. 11120 Sherman Way Sun Valley California U.S.A. 01352
	No Overseas Outlets	
3C	Churchill Chemical Corp.	
	North America	Churchill Chemical Corp 344 Franklin Street. Melrose Mass. U.S.A. 02176
	No Overseas Outlets	
Loctite	Loctite Corp.	
	North America	Loctite Corp. 705N Mountain Rd. Newington Ct U.S.A. 06111

Table 1 (continued)

Sealant/Adhesives Prefix	Manufacturers	
	Europe	Loctite Deutschland GMbh Arabellastrasse 17 8000 Muechen 81 West Germany
	Far East	Loctite (Australia) PTY LTD 56– 58 Alexander Ave. Taren Point NSW 2229
	Postal Address	P.O. Box 361 Miranda 2228 Australia
3M and EC	3M	
	North America	Minnesota Mining & Manufacturing Co. 3M Center St. Paul Minnesota U.S.A. 55101
	Europe	3M Europe S.A. 106 BOUL de la Woluwe B–1200 Brussels Belgium
	Far East	3M Australia Ltd. 950 Pacific Highway P.O. Box 99 Tymble NSW 2073 Australia
PSA 529/SR18	PSA 529/SR18	
	North America	Silchem Products 526 Gordon Baker Road Willowdale Ontario Canada M2H 3B4
	Europe	General Electric Plastics B.V. Postbus 117 4600 AC Bergew OP Zoom Netherlands
	Far East	GE Plastics Australia Ltd Silicone Products Div. 112–114 Wetherill St Silverwater NSW 2141 Australia
	Dow Corning North America	Dow Corning (Canada) Inc. 6747 Tampobello Road Mississauga Ontario Canada L5N 2M1

Table 1 (continued)

Sealant/Adhesives Prefix	Manufacturers	
	Europe	Dow Corning Ltd Barry Glamorgan South Wales CX6 7YL U.K
	Far East	Dow Corning Ltd 4 Ray Road Epping N.S.W. 2121 Australia
	Ciba–Geigy Corp.	
	North America	Ciba–Geigy Corp. Three Skyline Drive Hawthorn New York U.S.A. 10532
	Europe	Ciba–Geigy GMbh Post Fach 1160/1180 D–7867 Wehr–Baden West Germany
	Far East	Ciba–Geigy (Aust) Ltd. P.O. Box 332 Thomastown Victoria 3074 Australia

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Environmental Sealing (Dual Wrap Kit) – General Data

1. General

WARNING **SEALANTS MAY CONTAIN TOXIC COMPOUNDS. REFER TO MSDS FOR PROPER PERSONAL PROTECTION EQUIPMENT (PPE) TO USE WHEN WORKING WITH MATERIALS. ALWAYS WEAR GLOVES AND EYE PROTECTION AND AVOID SKIN CONTACT. IF SEALANT OR ANY SEALANT COMPOUND CONTACTS SKIN, CLEAN THOROUGHLY.**

The dual wrap sealing kit is for cold application on harnesses, connectors and/or related connector accessories (backshells, clamps etc.) requiring moisture sealing and abrasion/corrosion protection, for example, fuel gallery.

Note The dual wrap sealing methodology can be applied to installed connector(s), backshell(s) and connector accessories (harness, clamps, shield terminations etc.) all together or to individual components, as required, for environmental sealing.

2. Materials and Equipment

WARNING **USE ONLY FLAME RETARDANT (FR) (FOR EXAMPLE HT3326–5FR–50) ON AIRCRAFT APPLICATIONS DUE TO ITS FLAME RETARDANT CHARACTERISTICS.**

A. Materials

- Stretch Seal Tape, AD89503FR (Flame Retardant)
- Self Fusing Tape, AD59163FR (Flame Retardant)
- Sealant (Potting Compound), HT3326–5FR (Flame Retardant)
- Isopropyl Alcohol (IPA) to Federal Specification TT–I–735, Grade A or B
- Sealant Dispenser (e.g. UG981108–01 or equivalent)

3. Surface Preparation

Prior to installation, ensure the harness bundle and components that will come in contact with the Connector Sealing Kit are free of dirt, oil, grease and other contamination. Remove residue from previously applied adhesives or sealants. Clean harness and components, as required, with isopropyl alcohol.

If possible, un-mate the connector and orient the connector face in a downward position.

4. Preparation of Potting Compound

Remove the injectable polyurethane sealant from the package. Load the cartridge into the sealant dispenser (PN UG981108–01 or equivalent) and follow the package instructions for priming and dispensing the potting compound/sealant HT3326–5FR.

5. Removal/Installation

A. Installation of Connector Sealing Kit

The connector sealing kit consists of Flame Retardant (FR) Stretch Seal Tape (PN AD89503FR), FR Self Fusing Tape (PN AD59163FR) and Polyurethane FR Sealant (PN HT3326–5FR).

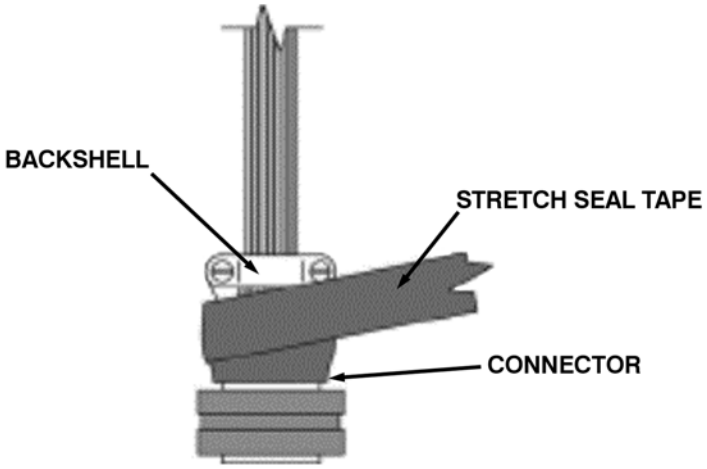
Note HT3326–5FR sealant must be stored and applied at temperatures >55°F. Shelf life of the sealant in the sealed container is six months or as specified in MSDS. The cure time at ambient temperature of 77°F (25°C) is ≤ 4 hours. Any opened cartridge must be used within 24 hours of opening the original packaging. Dispose unused cartridge/sealant as per manufacturer's instructions.

B. Installation of Stretch Seal Tape and Sealant

Install the Stretch Seal Tape (AD89503FR) and polyurethane sealant (HT3326–5FR) as follows:

1) Remove the Stretch Seal Tape from package and unroll a small length of tape. Pull back a small amount of the release film, making sure that the tape remains free of dirt or other contaminants.

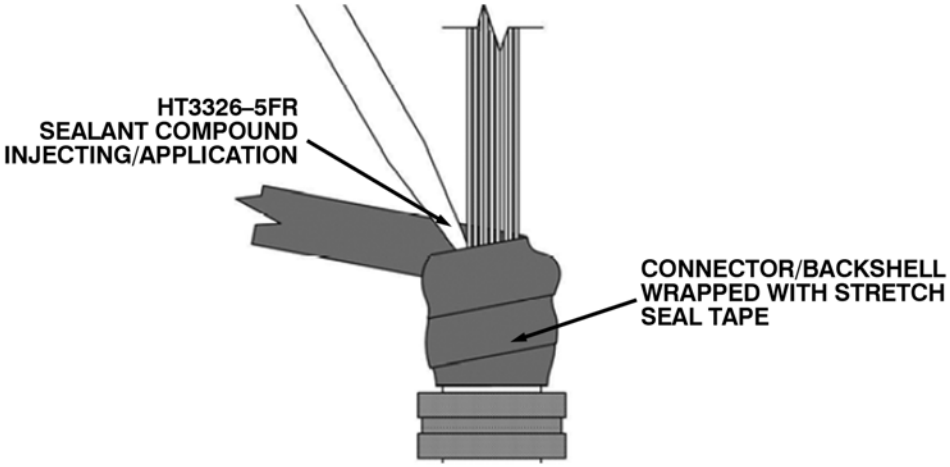
2) Begin wrapping the Stretch Seal Tape around the connector and backshell to be protected, behind the coupling ring (nut) or mating flange/surface. Stretch the tape 25 to 50% to ensure good compression around the connector body and rear accessories (refer to Figure 1). Remove release film as required.



Stretch Seal Tape around Connector and Backshell
Figure 1

3) Continue to wrap the Stretch Seal Tape with a 50% overlap.

4) Stop wrapping the Stretch Seal Tape and hold in place. Immediately inject potting HT3326–5FR (Flame Retardant) compound into the wire bundle behind the connector/backshell rear accessory or through any openings in the rear accessory (refer to Figure 2).

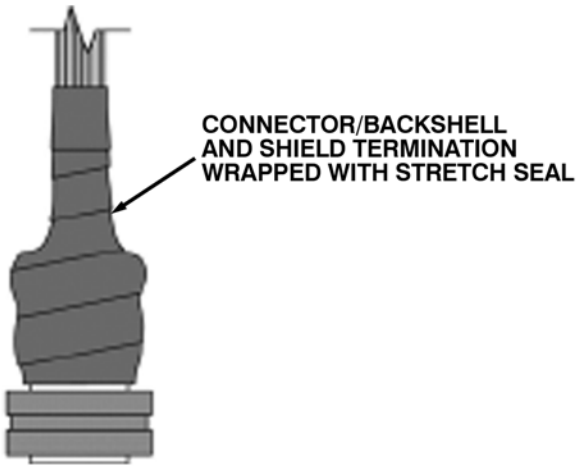


Injecting HT3326–5FR Compound
Figure 2

5) Alternate wrapping the Stretch Seal Tape and injecting the potting compound, until the connector rear accessory is covered, and at least one complete wrap of Stretch Seal Tape has been installed on the harness bundle.

   Note If shield termination is there continue wrapping the Stretch Seal Tape over the shield termination until the back of the connector/backshell is covered, and one complete wrap of Stretch Seal Tape has been installed on the harness bundle.

6) Cut off any excess Stretch Seal Tape. Follow the instructions provided with the potting compound for disposal.



Complete Stretch Seal Tape with Sealing Compound
Figure 3

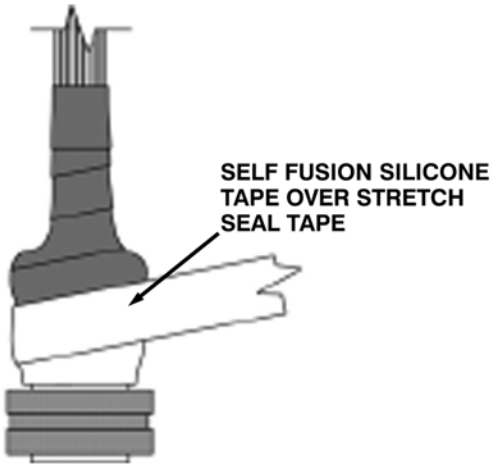
C. Installation of Self Fusing Silicone Tape

Install Self Fusing Silicone Tape (AD59163FR) over the stretch seal tape as follows:

1) Remove the Self Fusing Silicone Tape from the package. Pull back a small amount of the release film, making sure that the tape remains free of dirt or other contaminants.

2) Begin to wrap the connector behind the top of coupling ring (nut) over the stretch seal tape with one complete revolution on itself.

3) After the first full wrap is complete, continue wrapping with a 50% overlap over the top of the Stretch Seal Tape. Use the line in the middle of Self Fusing Silicone Tape as a guide and remove the release film as required (refer to Figure 4).



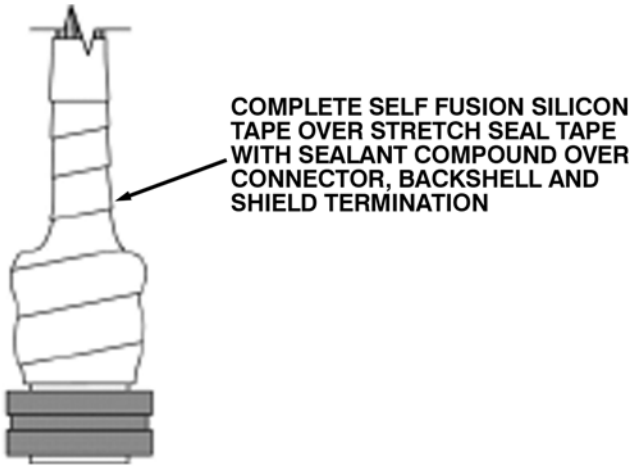
Installation of Self Fusion Silicone Tape
Figure 4

4) Stretch the Self Fusing Silicone Tape 50 to 300% during application to ensure a tight fit when complete. Stretching the tape will aid in the fusing process.

5) Clean any excess potting compound (sealant) that is forced out of the Stretch Seal Tape from the wire bundle.

6) Continue to wrap the Self Fusing Silicone Tape until at least one complete wrap past the end of the stretch seal tape. Make one complete wrap over the top of the last wrap. The first and last wrap must always be a complete wrap directly over the top of the previous wrap.

7) Cut off any excess Self Fusing Silicone Tape.



Complete Environmental Sealing Wrap
Figure 5

D. Environmental Sealing Kit Removal

1) Cut the Self Fusing Silicone Tape with a sharp blade or similar tool, taking care not to cut the harness underneath. Peel the tape away from the harness bundle.

2) Remove the Stretch Seal Tape. Remove the polyurethane sealant by pulling with fingers or other blunt instrument, so as not to damage the wires.

3) Use isopropyl alcohol to remove any remaining residue.

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Environmental Sealing (Self Leveling Green Sealant) – General Data

1. General

WARNING **SEALANTS MAY CONTAIN TOXIC COMPOUNDS. REFER TO MSDS FOR PROPER PERSONAL PROTECTION EQUIPMENT (PPE) TO USE WHEN WORKING WITH MATERIALS. ALWAYS WEAR GLOVES AND EYE PROTECTION AND AVOID SKIN CONTACT. IF SEALANT OR ANY SEALANT COMPOUND CONTACTS SKIN, CLEAN THOROUGHLY.**

Self Leveling Green (SLG) sealant is used for connector accessories (backshells, clamps etc.) requiring moisture sealing and abrasion/corrosion protection, for example, fuel gallery.

Note The SLG sealing methodology can be applied to installed backshell(s) and connector accessories (harness, clamps, shield terminations etc.) all together or to individual components, as required, for environmental sealing.

2. Materials and Equipment

WARNING **USE ONLY FLAME RETARDANT (FR) (FOR EXAMPLE HT3326–5FR–50) ON AIRCRAFT APPLICATIONS DUE TO ITS FLAME RETARDANT CHARACTERISTICS.**

A. Materials

- Sealant (Potting Compound), HT3326–5FR (Flame Retardant)
- Isopropyl Alcohol (IPA) to Federal Specification TT–I–735, Grade A or B
- Sealant Dispenser (e.g. UG981108–01 or equivalent)

3. Surface Preparation

Prior to installation, ensure the components that will come in contact with the SLG sealant are free of dirt, oil, grease and other contamination. Remove residue from previously applied adhesives or sealants. Clean harness and components, as required, with isopropyl alcohol.

If possible, un-mate the connector and orient the connector face in a downward position.

4. Preparation of Potting Compound

Remove the injectable polyurethane sealant from the package. Load the cartridge into the sealant dispenser (PN UG981108–01 or equivalent) and follow the package instructions for priming and dispensing the potting compound/sealant HT3326–5FR.

5. Removal/Installation

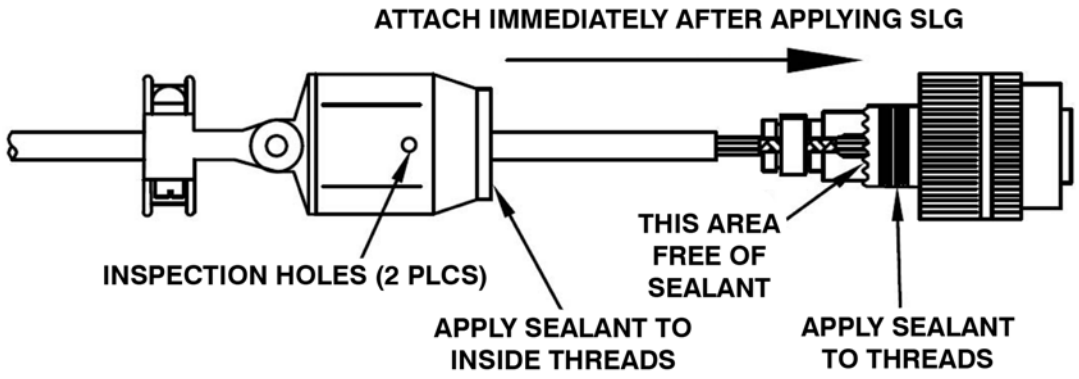
A. Installation of SLG Sealant

Note HT3326–5FR sealant must be stored and applied at temperatures >55°F. Shelf life of the sealant in the sealed container is six months or as specified in MSDS. The cure time at ambient temperature of 77°F (25°C) is ≤ 4 hours. Any opened cartridge must be used within 24 hours of opening the original packaging. Dispose unused cartridge/sealant as per manufacturer's instructions.

1) Apply a thin layer of HT3326–5FR–50 SLG sealant onto the threads of the backshell and the threads on the plug and then immediately attach the two parts together.

CAUTION **DO NOT ALLOW THE SLG SEALANT TO CONTACT THE FAYING SURFACES BETWEEN THE PLUG AND THE SHIELD ADAPTER, AS THIS WILL INTERFERE WITH THE ELECTRICAL BONDING.**

2) After attaching the backshell, verify that the shield adapter is properly seated by looking through the inspection holes.

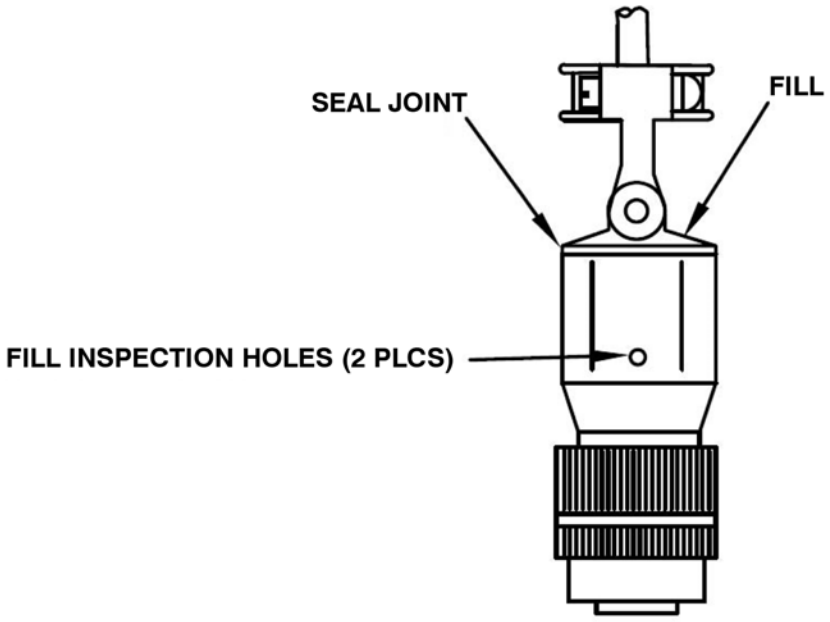


Application Locations of the SLG Selant
Figure 1

3) Position the backshell vertically and inject SLG sealant into the back openings as shown in Figure 2. Fill the cavity so that the sealant completely covers the shield adapter.

4) Unplug the connectors and supply SLG sealant into the inspection holes and along the joint shown in Figure 2.

Note Do not apply SLG sealant overtop the plug, as this will interfere with the plug's rotation.



Final Application Locations of the SLG Sealant
Figure 2

5) Dispose of unused cartridge/sealant as per manufacturer's instructions.

B. Removal of SLG Sealant

1) To avoid damaging the wires, remove the polyurethane sealant by pulling with fingers or a blunt instrument.

2) Use isopropyl alcohol to remove any remaining residue.

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PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR20-13**

Insert in 20-60-01 before Page 1

INSPECTION, CLEANING AND LUBRICATION OF 7X19 AND 7X7 CONTROL CABLES

REASON FOR ISSUE:

To include additional inspection/cleaning procedures for the flight cables.

ACTION:

In Chapter 20, ignore existing Section 20-60-01 and insert revised section, as shown on Pages 2 to 8 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data

1. General

WARNING **THIS PROCEDURE USES MATERIALS THAT MAY BE DANGEROUS TO THE PERSONNEL'S HEALTH AND/OR THE ENVIRONMENT. SUFFICIENT PRECAUTIONS MUST BE TAKEN AT ANY STEP OF USE. IF NOT, THIS CAN CAUSE DAMAGE TO THE ENVIRONMENT AND DANGEROUS EFFECTS ON THE PERSONNEL'S HEALTH AND SAFETY.**

This standard practice specifies the procedures and requirements for wear and damage inspection of 7x19 and 7x7 control cables such as used for flight controls; cleaning and lubrication procedures are also provided. The control cables are of 7x19 construction type (7 strands, each strand with 19 wires) or 7x7 (7 strands, each strand with 7 wires). Control cable wear or breakage usually occurs where the cable passes through fairleads, over pulleys and quadrants. At manufacture, the wire rope is treated with an internal lubricating compound. This lubrication compound decreases internal friction, helps to prevent corrosion and adds to the life of the cable. For carbon steel non-jacketed cables, added lubrication and protection from corrosion is applied externally when the cable assembly is made and or installed.

A. Materials and Equipment

1) Materials

- IPA or Acetone
- Lubricant – MIL–PRF–81322 (Aeroshell 22)
- DS108, Diestone DLS, or Diestone HFP

2) Equipment

- Cloths – lint free
- Cloth – course weave
- 10 power magnifying glass
- Fibre brush

B. Safety Precautions

1) Use a cloth of sufficient thickness for protection of your hands when inspecting control cables for broken wires. Broken wires can cause injuries.

2) Sufficient ventilation must be provided when using solvent in confined areas. When applying solvent, operator will wear protective respiratory equipment.

3) Protective gloves will be worn to avoid skin contact with solvent. If contact occurs, wash the area thoroughly with soap and water.

4) Refer to the applicable manufacturer's Material Safety Data Sheet (MSDS) for specific safety data on any of the materials specified herein.

2. Cleaning

CAUTION **DO NOT USE VAPOR DEGREASING, STEAM CLEANING, OR SOLVENTS TO REMOVE CORROSION PREVENTIVE COMPOUNDS FROM THE OUTER SURFACES OF THE CONTROL CABLES. THESE PROCEDURES WILL ALSO REMOVE THE INTERNAL LUBRICANT.**

A. Clean the cables as follows:

1) Use a clean, dry, lint-free cloth to remove all dirt, debris, and old lubricant from the surface of the control cable.

2) When cables remain installed in the aircraft, all accessible lengths of the cable must be cleaned so that all cable strands over the full length of travel at fairleads, pulleys, quadrants, and cable drums can be inspected. Control must be moved through their full range to allow cleaning of the cable where it contacts fairleads, pulleys, quadrants, and cable drums.

3) When cables are removed from the aircraft, clean the full length of the cable.

4) If corrosion preventive compound is present locally on surface of cables, pulleys, quadrants or cable drums, use a small amount of DS108, Diestone DLS, or Diestone HFP on a clean, lint-free cloth to remove compound.

Note Corrosion preventive compounds can be opaque and may prevent through inspection of cable.

3. Inspection/Check

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. Cable Damage

1) No bruising, kinking, untwisting, bends or distortion on any control cable is permitted. Install new control cable when such defects are found.

B. External Wear

Refer to Figure 1 for details.

1) It is not practical to measure individual wires to determine wear; however, minor wear of individual wire appearing on the surface of any control cable is permissible.

Note Whenever cable wear is found, the cause for wear must be determined and corrected.

2) If wear on individual wire exceeds 40% of the original wire size, remove and discard cables and install new cables.

Note An indication of wear beyond 40% is the apparent blending of worn areas on the adjacent wires.

C. Broken Wires

Refer to Figure 2 for details.

1) Do an inspection for broken wires as follows:

Note Broken wires do not always extend from the lay of the cable. When broken wires or corrosion are suspected, remove the cable and carefully bend and twist the cable with your hands to make sure that all broken wires are found.

a) Rub a cloth along the length of the cable in both directions. Broken wires catch the cloth.

b) Use special care in areas where sections of the cable go around pulleys, quadrants, drums and through fairleads or pressure seals. Move controls through their full range to expose all cable surfaces adjacent to fairleads, pulleys, quadrants, and/or cable drums.

c) Pay close attention to the sections of the cable that are worn or have corrosion.

2) When broken wires are found, relieve cable tension and carefully examine the cable with a 10 power magnifying glass to make the full level of the damage known.

3) Replace a control cable when one or more of the conditions in Table 1 — Cable Broken Wire Limits are exceeded.

Note Flight control cables with broken wire(s) occurring in a wear area shall be considered unserviceable.

Table 1 Cable Broken Wire Limits

Cable Type	Location	Allowable Broken Wire
7x7 other than Fight Control Cable	Any	3 per one inch (25.4 mm) cable span
7x19 other than Fight Control Cable	Any	6 per one inch (25.4 mm) cable span
7x7 Fight Control Cable	Any	3 per one inch (25.4 mm) cable span, not to occur in consecutive inch spans
7x19 Fight Control Cable	At critical point	3 within the length of cable travel at pulleys, quadrants, drums, fairleads, fuselage cut-outs and within 12 inches (304.8 mm) of a cable end or terminal
	Away from critical point	6 per one inch (25.4 mm) cable span, not to occur in consecutive one inch (25.4 mm) spans.

D. Corrosion

CAUTION DO NOT USE METALLIC WOOL, WIRE BRUSHES, OR ABRASIVE MATERIALS TO REMOVE SURFACE CORROSION FROM CABLES. THESE MATERIALS WILL CAUSE CONTAMINATION THAT WILL RESULT IN SUBSEQUENT INTERNAL WEAR AND CORROSION.

1) Carefully examine the cable external surfaces for corrosion. Use special care when the conditions that follow occur:

a) There are cable runs through or near areas such as battery and baggage compartments, lavatories, and wheel wells.

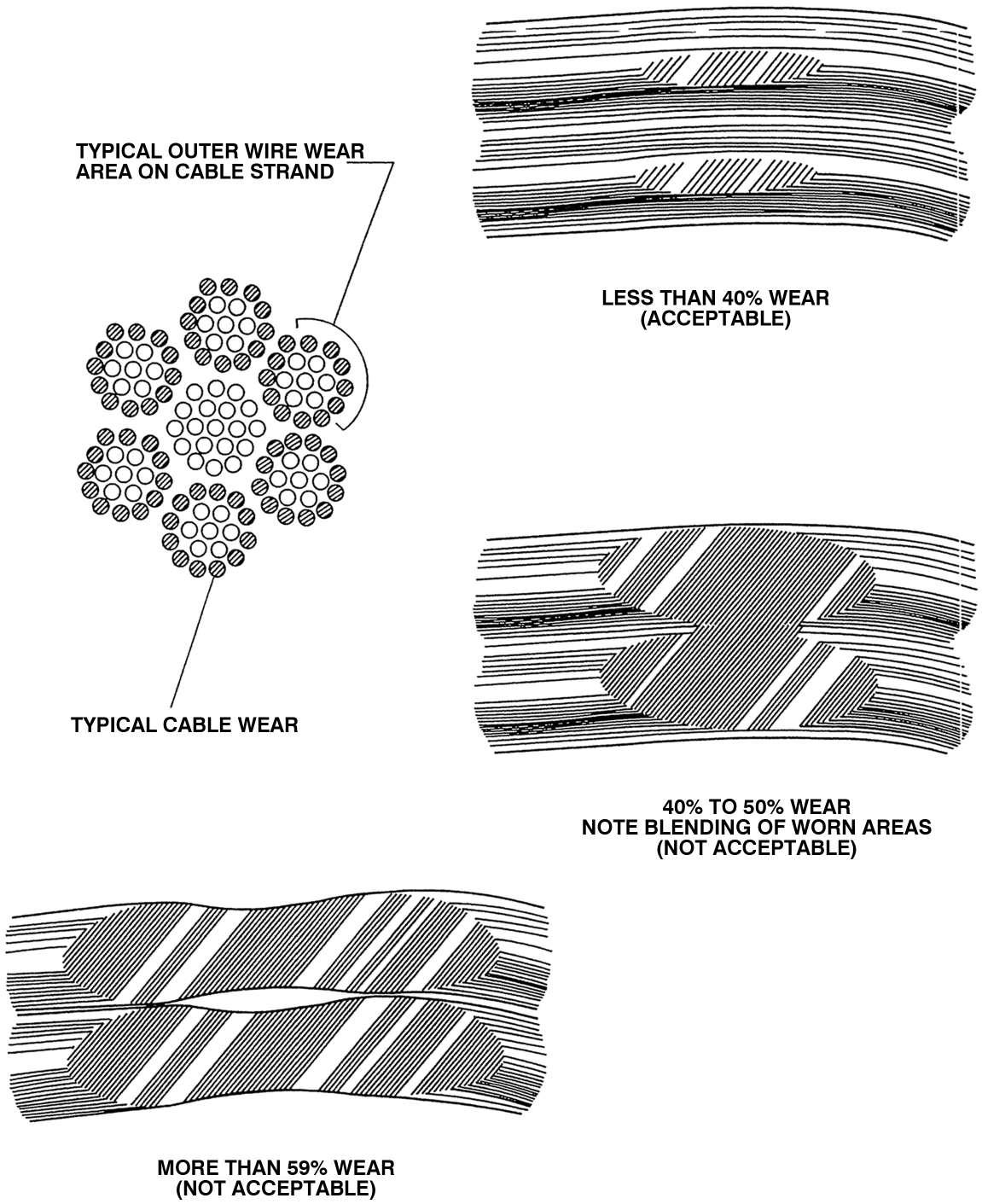
b) Broken wires are found in sections where wear is not caused by airframe components such as pulleys, pressure seals or fairleads.

c) Cleaning or metal brightening procedures are done adjacent to cable runs.

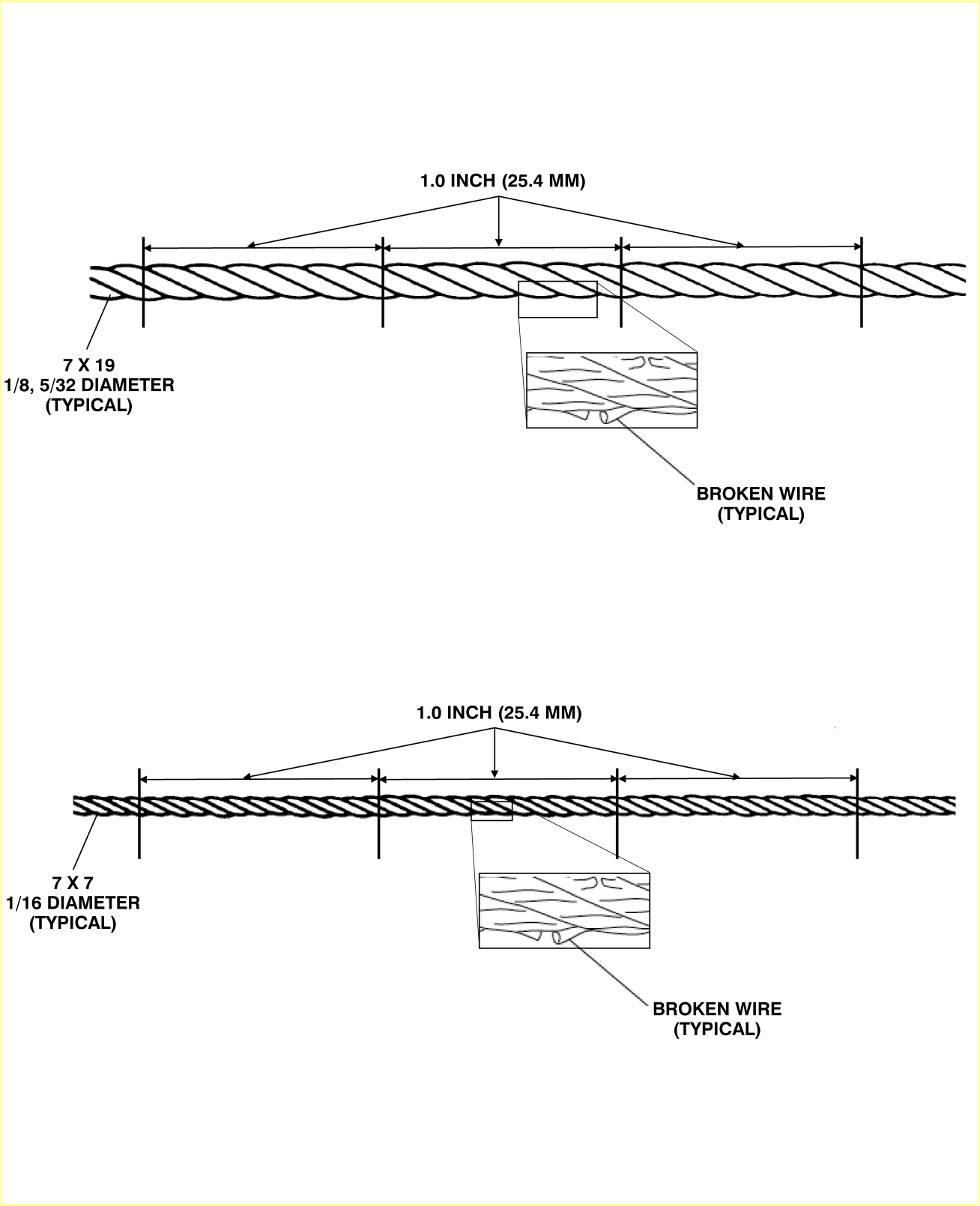
2) If external surface corrosion is found (white powder or a dark red), release the tension or remove the cable and twist it to visually examine the internal surfaces for corrosion products (white powder or a dark red).

3) Remove corrosion from the external surfaces with a coarse weave cloth or a fibre brush. After removing external corrosion and full cleaning to remove all debris, and corrosion particles, apply lubricant as per Para 4.

4) Replace control cable when internal corrosion is found.



Control Cable Wear Limits – Individual Wires
Figure 1



Control Cable Broken Wire Limits
Figure 2

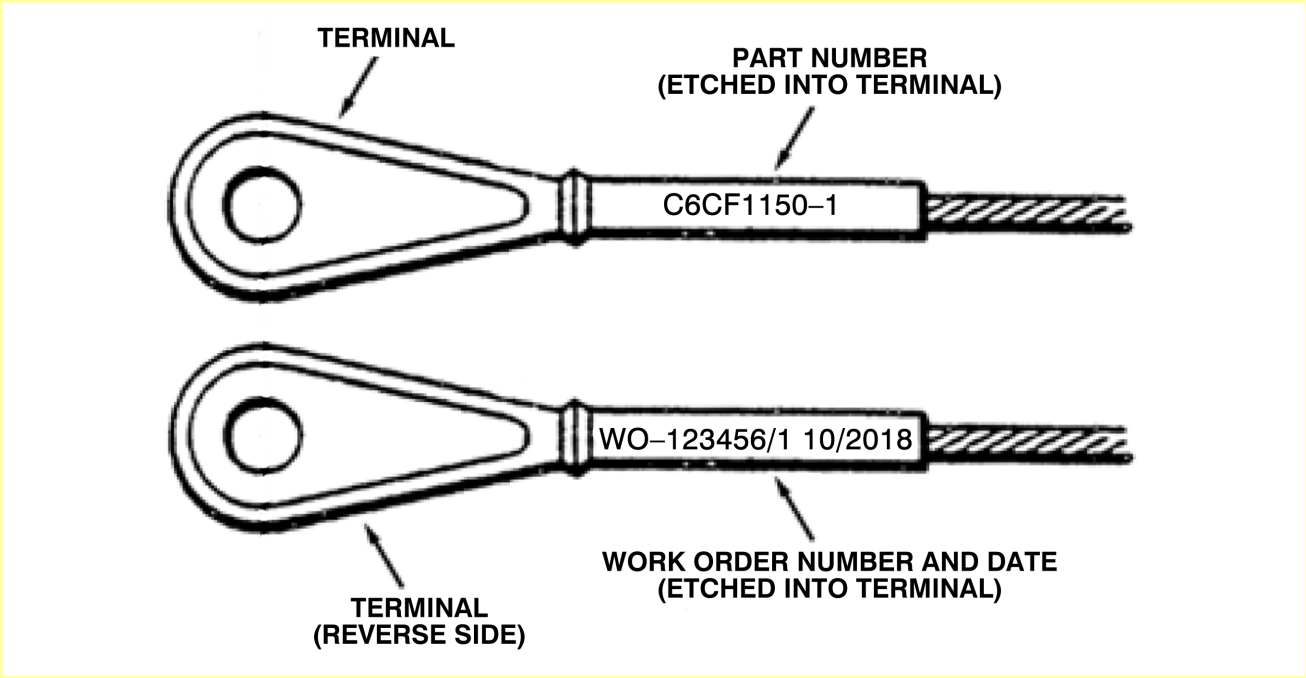
4. Lubrication

A. Examine the full length of carbon steel non-jacketed cables for a sufficient external coating of MIL–PRF–81322 (Aeroshell 22) lubricant. Use special care in those areas of the aircraft where frequent maintenance and servicing tasks are done.

Note This does not apply to corrosion resistant steel cables (composition B) and jacketed cables (Type 2). Lubrication attracts abrasive particles to corrosion resistant steel cables resulting in wear on pulleys, fairleads and pressure seals. Jacketed cables do not require lubrication. Cable composition can be confirmed by comparing the cable part number found on the cable end or terminal to the information in PSM 1-64-4. Refer to Figure 3.

B. When lubricant is removed as a result of inspection or removal and installation procedures on carbon steel non-jacketed cables, apply a thin layer of MIL–PRF–81322 (Aeroshell 22).

Note Apply lubricant MIL–PRF–81322 (Aeroshell 22) at regular times during the life of the carbon steel non-jacketed cable.



Control Cable Identification (Typical)
Figure 3

5. Requirements

Refer to Figure 4 for details.

A. Examine the airframe components.

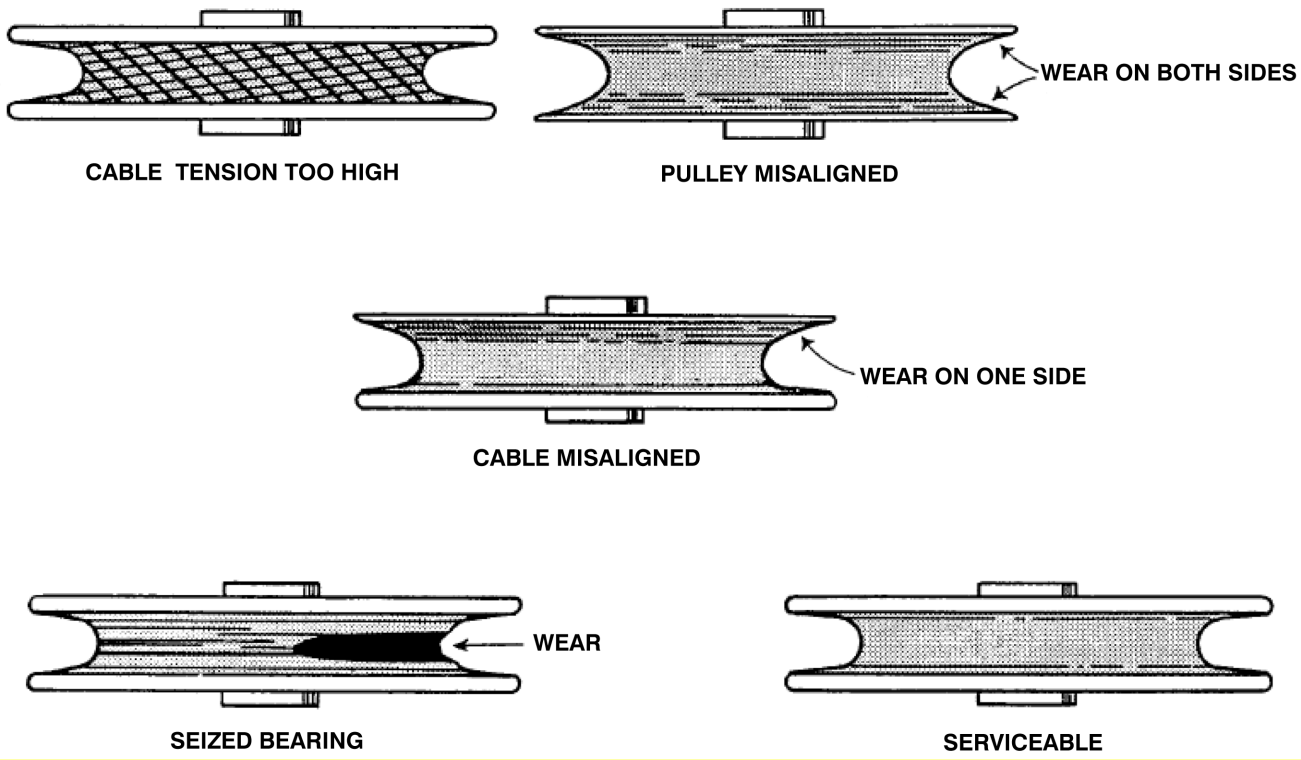
1) Examine the airframe components (such as pulleys, fairleads and guard pins) that are related to cables that have damage. Wear patterns on related airframe components are frequently an indication of the conditions that follow:

a) The cable tension is too high.

b) The pulley bearing is seized.

c) The airframe components not aligned correctly.

2) Correct all of the unsatisfactory conditions that can cause cable damage.



Pulley Wear Patterns (Typical)
Figure 4

NOTE: Please see the TEMPORARY REVISION that revises this page.

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A. Materials and Equipment

1) Materials

- IPA or Acetone
- Lubricant – MIL–PRF–81322 (Aeroshell 22)
- DS108, Diestone DLS, or Diestone HFP

2) Equipment

- Cloths – lint free
- Cloth – course weave
- 10 power magnifying glass
- Fibre brush

B. Safety Precautions

1) Use a cloth of sufficient thickness for protection of your hands when inspecting control cables for broken wires. Broken wires can cause injuries.

2) Sufficient ventilation must be provided when using solvent in confined areas. When applying solvent, operator will wear protective respiratory equipment.

3) Protective gloves will be worn to avoid skin contact with solvent. If contact occurs, wash the area thoroughly with soap and water.

4) Refer to the applicable manufacturer's Material Safety Data Sheet (MSDS) for specific safety data on any of the materials specified herein.

2. Cleaning

CAUTION **DO NOT USE VAPOR DEGREASING, STEAM CLEANING, OR SOLVENTS TO REMOVE CORROSION PREVENTIVE COMPOUNDS FROM THE OUTER SURFACES OF THE CONTROL CABLES. THESE PROCEDURES WILL ALSO REMOVE THE INTERNAL LUBRICANT.**

A. Clean the cables as follows:

1) Remove corrosion preventive compound (MIL–C–16173 Grade 2) with a clean lint free cloth that is soaked with DS108, Diestone DLS, or Diestone HFP.

3. Inspection/Check

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

Refer to Figure 1 for details.

A. Cable Damage

1) No bruising, kinking, untwisting, bends or distortion on any control cable is permitted. Install new control cable when such defects are found.

B. External Wear

1) It is not practical to measure individual wires to determine wear; however, minor wear of individual wire appearing on the surface of any control cable is permissible.

2) If wear on individual wire exceeds 40% of the original wire size, remove and discard cables and install new cables.

Note An indication of wear beyond 40% is the apparent blending of worn areas on the adjacent wires.

C. Broken Wires

1) Do an inspection for broken wires as follows:

Note Broken wires do not always extend from the lay of the cable. If necessary, remove the cable and carefully bend and twist the cable with your hands to make sure that all broken wires are found.

a) Rub a cloth along the length of the cable. Broken wires catch the cloth.

b) Use special care in areas where sections of the cable go around pulleys, quadrants, drums and through fairleads or pressure seals.

c) Check sections of the cable that are worn or have corrosion.

2) When broken wires are found, carefully examine the cable with a 10 power magnifying glass to make the full level of the damage known.

3) Replace a control cable when one or more of the conditions that follow are found:

a) More than three broken wires in a one inch (25.4 mm) span of 7x7 cable. If limit is exceeded, install new cable.

b) More than six broken wires in a one inch (25.4 mm) span of 7x19 cable.

D. Corrosion

CAUTION **DO NOT USE METALLIC WOOL, WIRE BRUSHES, OR ABRASIVE MATERIALS TO REMOVE SURFACE CORROSION FROM CABLES. THESE MATERIALS WILL CAUSE CONTAMINATION THAT WILL RESULT IN SUBSEQUENT INTERNAL WEAR AND CORROSION.**

1) Carefully examine the cable external surfaces for corrosion. Use special care when the conditions that follow occur:

NOTE: Please see the TEMPORARY REVISION that revises this page.

a) There are cable runs through or near areas such as battery and baggage compartments, lavatories, and wheel wells.

b) Broken wires are found in sections where wear is not caused by airframe components such as pulleys, pressure seals or fairleads.

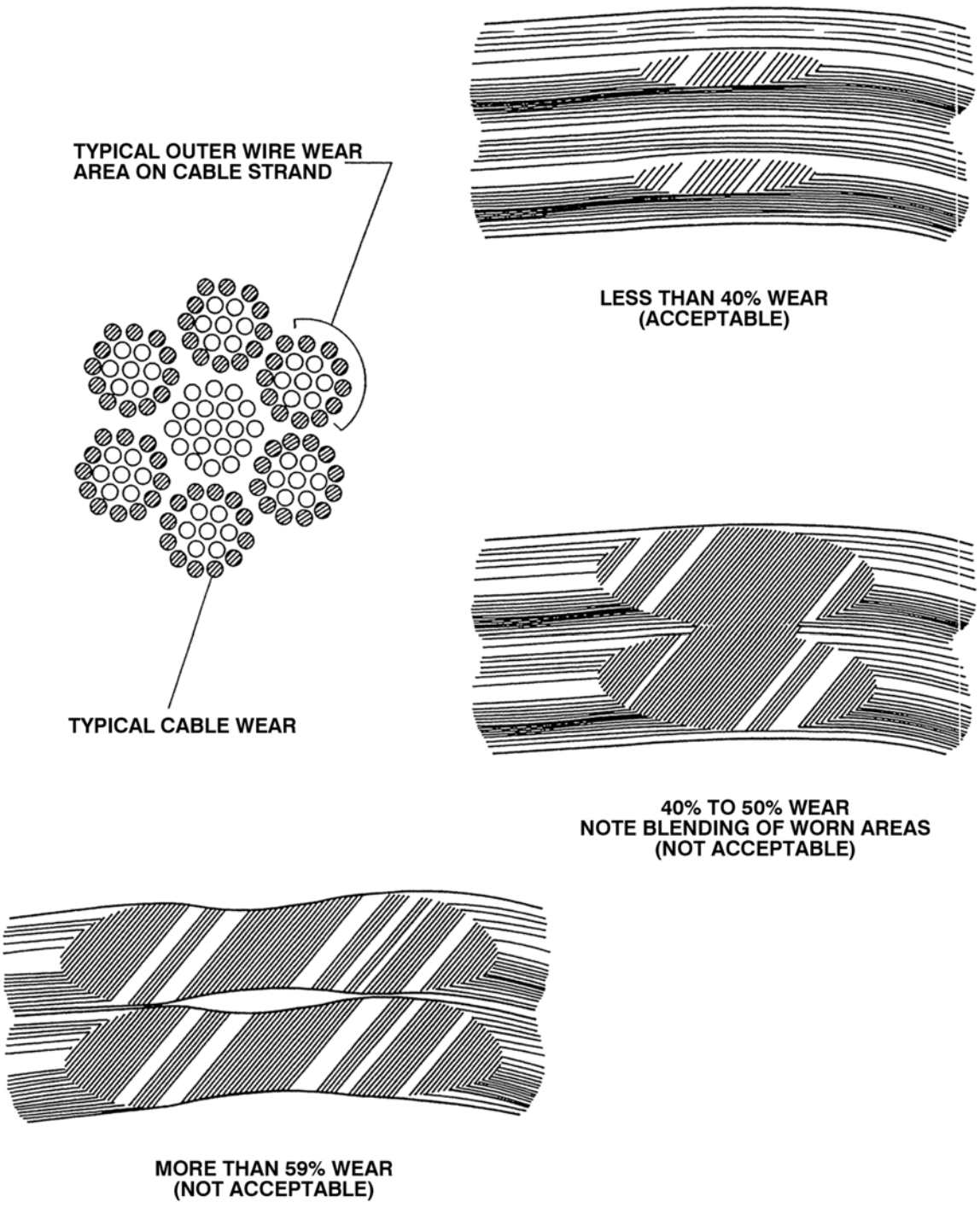
c) Cleaning or metal brightening procedures are done adjacent to cable runs.

2) If external surface corrosion is found (white powder or a dark red), release the tension or remove the cable and twist it to visually examine the internal surfaces for corrosion products (white powder or a dark red).

3) Remove corrosion from the external surfaces with a coarse weave cloth or a fibre brush.

4) Replace control cable when internal corrosion is found.

NOTE: Please see the TEMPORARY REVISION that revises this page.



Control Cable Wear Limits – Individual Wires
Figure 1

NOTE: Please see the TEMPORARY REVISION that revises this page.

4. Lubrication

A. Examine the full length of carbon steel non-jacketed cables for a sufficient external coating of MIL–PRF–81322 (Aeroshell 22) lubricant. Use special care in those areas of the aircraft where frequent maintenance and servicing tasks are done.

Note This does not apply to corrosion resistant steel cables (composition B) and jacketed cables (Type 2). Lubrication attracts abrasive particles to corrosion resistant steel cables resulting in wear on pulleys, fairleads and pressure seals. Jacketed cables do not require lubrication.

B. When lubricant is removed as a result of inspection or removal and installation procedures on carbon steel non-jacketed cables, apply a thin layer of MIL–PRF–81322 (Aeroshell 22).

Note Apply lubricant MIL–PRF–81322 (Aeroshell 22) at regular times during the life of the carbon steel non-jacketed cable.

5. Requirements

A. Examine the airframe components.

1) Examine the airframe components (such as pulleys, fairleads and guard pins) that are related to cables that have damage. Wear patterns on related airframe components are frequently an indication of the conditions that follow:

a) The cable tension is too high.

b) The pulley bearing is seized.

c) The airframe components not aligned correctly.

2) Correct all of the unsatisfactory conditions that can cause cable damage.

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Handling of Static Sensitive Electronic Components and Devices – General Data

1. Scope

This standard specifies the procedure and requirements for the protection of static sensitive electronic components and devices during receipt verification, transportation, storage, stores handling, assembly, assembly verification, installation and removal.

2. Equipment

A. Static awareness labels, self-adhesive, as per JEDEC–14 standard (e.g., 3M No. 7101), as shown in Figure 1.

B. Static shielding bags, transparent plastic, 6" X 10", capable of preventing static damage to static sensitive components or devices (e.g., 3M No. 2100–6–10).

C. Conductive foam, non-corrosive, high density, 1/4" thick (e.g. 3M No. 2910–1/4–12–12).

D. Static free cushioning material, plastic honeycomb cell construction to PPP–C–1842, Type III.

E. 3M #8020 static free work station grounding kit, components as listed below. Set up static free work stations as shown in Figure 2. As an alternative to the set-up shown in Figure 2, static free work stations may be set up as shown in Figure 3. It is acceptable for the wrist strap ground cord to be connected to the same ground lug as the table mat ground cord or directly to an electrical connection to earth ground. As an alternative to the 3M #8020 static free work station grounding kit, it is acceptable to equip static free work stations with static dissipative table mats, floor mats, wrist straps and grounding cords with a resistance 10^6 – 10^9 ohms; it is not acceptable to use conductive (10^5 or less ohms) or insulated (10^{10} or more ohms) table mats, floor mats, wrist straps or ground cords.

- 8210 Series Static Table Mat
- 8220 Series Static Floor Mat
- 2066 CHARGE-GUARD Wrist Strap (with ground cord)
- 3040 Ground Cord
- 3043 Inter-Connecting Ground Cord

F. Tote-Boxes, conductive plastic.

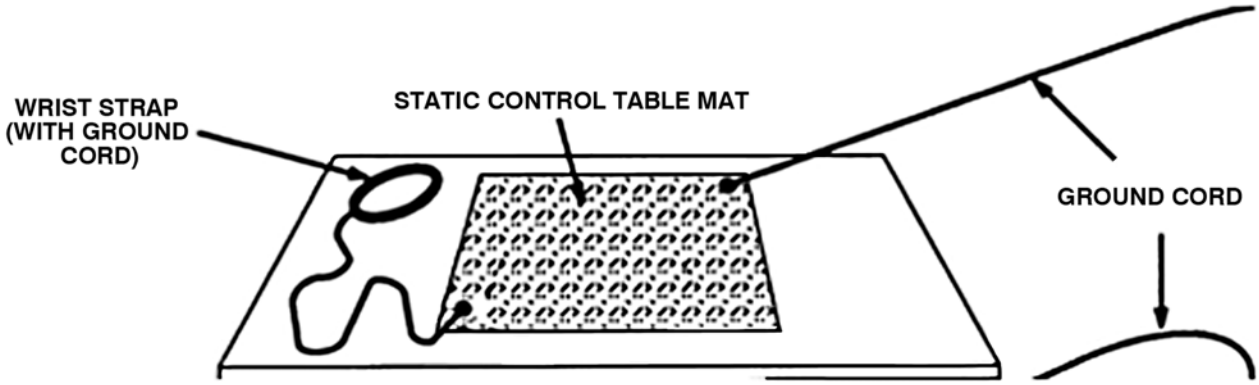
G. Connector caps, conductive plastic.

H. Resistivity meter (e.g., ACL model 375 or ACL model 475).

I. Wrist strap tester (e.g., Bystat Strapcheck BSL–9010).



Static Awareness Label
Figure 1



Set-Up of Static Free Work Stations
Figure 2



Alternative Set-Up of Static Free Work Stations
Figure 3

3. Procedure

A. General

1) Components or devices which can be damaged or destroyed by an electrostatic discharge (ESD), or shock, are termed static sensitive. Consider all components which are to be assembled to printed circuit boards (transistors, integrated circuit (IC) chips, etc.) static sensitive components. Consider all completed or partially completed circuit boards and avionics black boxes are static sensitive devices. Printed circuit boards on which no components have been assembled are not considered static sensitive.

2) A static electronic shock is caused by the instantaneous discharge of an electrostatic potential or voltage which is built through friction against a non-conducting surface, such as a nylon rug or a plastic chair. An ESD shock too small to even notice may be enough to destroy the electronic logic of an IC chip.

3) To prevent ESD damage to static sensitive components and devices it is necessary to protect from static shock at all times. During transport and storage, the static sensitive components and devices must be shielded from electrostatic fields by being packaged in ESD protective material. During receipt verification, stores handling, assembly and installation or removal of static sensitive components or devices, personnel must be grounded at all times by means of a conductive wrist strap. When installing or removing avionics boxes, personnel must be grounded to the aircraft structure by means of a wrist strap or other grounding equipment at all times.

B. Storage

1) Store static sensitive components in their original boxes, bags or static shielded tubes, identified by a static awareness label.

2) Place completed or partially completed printed circuit boards to be stored within a suitable static shielding bag identified and sealed by means if a static awareness label.

3) Ensure all connectors are capped with ESD protective caps and package avionics black boxes to be stored in static free cushioning material. Identify the package as static sensitive by means of a static awareness label.

C. Stores Handling

1) Unless it is necessary to split a lot or batch of components or devices, static sensitive components or devices may be issued from stores with no special handling procedures, except that the package must not be opened. If it is necessary to split a lot or batch, repackage the components or devices for issue as follows:

a) Place the original package of components or devices on a conductive table mat at a static free work station. Personnel filling the order must be grounded to the static free work station by means of a conductive wrist strap before opening the container in which the components or devices are packaged.

b) Repackage static sensitive components and devices as follows:

- Individually repackage static sensitive components, other than IC chips, in a suitable static shielding bag.
- Transfer IC chips onto conductive foam pads by placing the chip onto the foam pad and gently pushing down to press the chip leads partially into the foam and then place the foam pad in a static shielding bag. Take extreme care at all times when handling IC chips to avoid touching or bending the chip leads.
- Place printed circuit boards on which components have been assembled within a static shielding bag. Printed circuit boards on which no components have been assembled do not require ESD protective packaging.
- For avionics boxes, cap all connectors with ESD protective caps and package in static free cushioning material.

c) Seal static shielding bags or static shielding material and identify as static sensitive by means of a static awareness label placed in a readily visible location.

d) Place the identified static shielding bag or static cushioning material bag in a conductive plastic tote-box.

D. Installation and Removal of Avionics Boxes

1) Take care to avoid touching the connector contact pins when installing or removing avionics boxes.

2) Immediately after removing installed avionics boxes, fit conductive plastic caps over the connectors and package the avionics in static free cushioning material. Identify the static free cushioning material as containing a static sensitive component or device.

Electrical Bonding and Grounding of Aircraft Structures – General data

1. General

Aircraft electrical bonding is the process by which electrical conductivity is achieved between the component and parts of the aircraft, to ensure electrical continuity of the structure.

Grounding is the electrical connection of a conducting object to primary structure to provide a path for current.

Bonding and grounding connections as specified in these procedures are made in order to ensure the proper operation of the electrical equipment, to suppress the electromagnetic interference resulting from static discharge, electrical shock and lightning strikes and to ensure the aircraft is electrically stable.

2. Types of Bonding Installation

The types of bonding installation required should be specified within the appropriate installations of the PSM 1-64-2. Refer to Table 1 — Type of Bonding Installations. If the type of bonding installation required is not specified within the associated section of the PSM 1-64-2, contact Viking Air Ltd., Technical Support.

Table 1 Type of Bonding Installations

TYPE OF BONDING (AS SPECIFIED IN THE PSM 1-63-2 SECTIONS)	**SURFACE TO BE PREPARED (ELECTRICAL BOND CONTACT AREA)**
Bonding through Direct Contact	refer to Figure 1
Bonding through Attachment Hardware	Nuts/Bolts refer to Figure 2 Screws refer to Figure 3 Rivets Inner surface of rivet hole
Bonding of Connectors	refer to Figure 4
Bonding through Bonding Jumpers (e.g., ground studs)	refer to Figure 5
Bonding of Conductive Plumbing Line Fittings which Pass through the Structure	refer to Figure 6

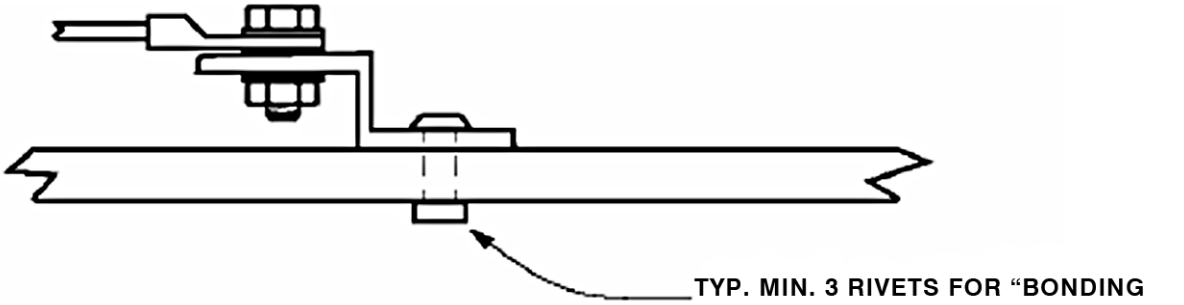
Note Electrically bond externally mounted equipment (i.e. antennas, light housings, data sensors, static wicks, etc.) through direct contact.

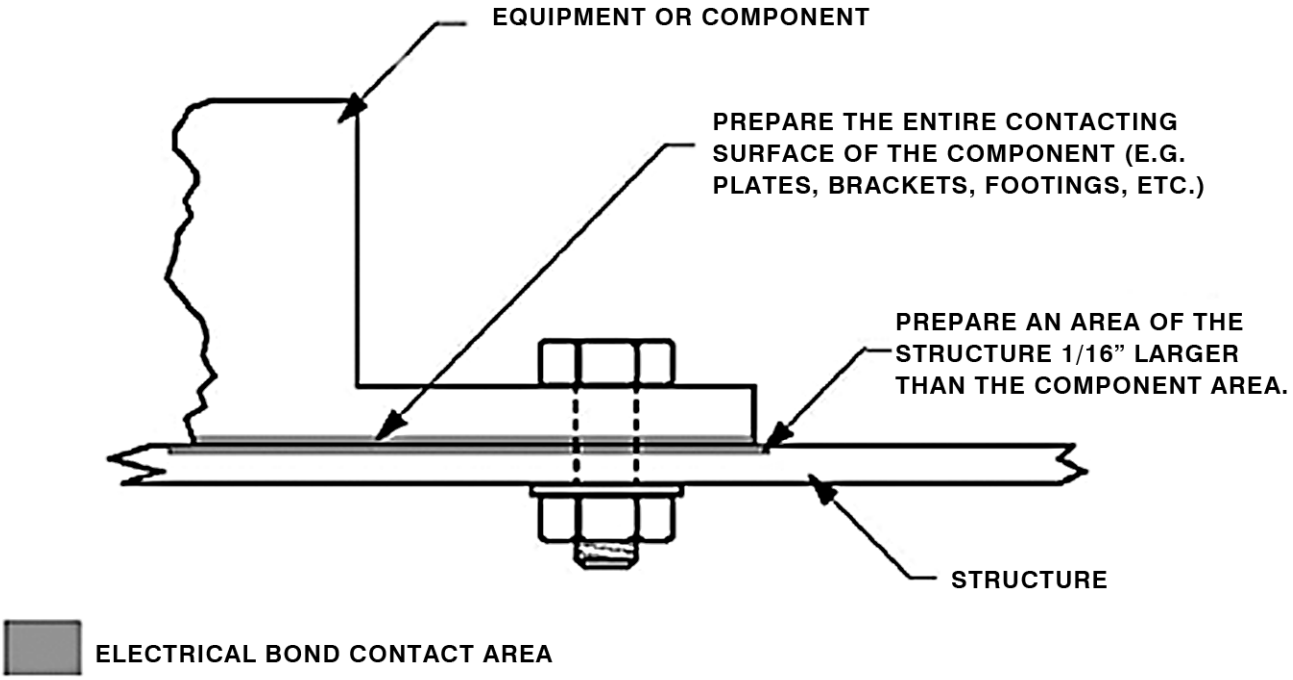
Note After bonding, touch-up the bond points according to Para 10. and seal the bonding and grounding hardware according to Para 11.

Table 1 Type of Bonding Installations (continued)

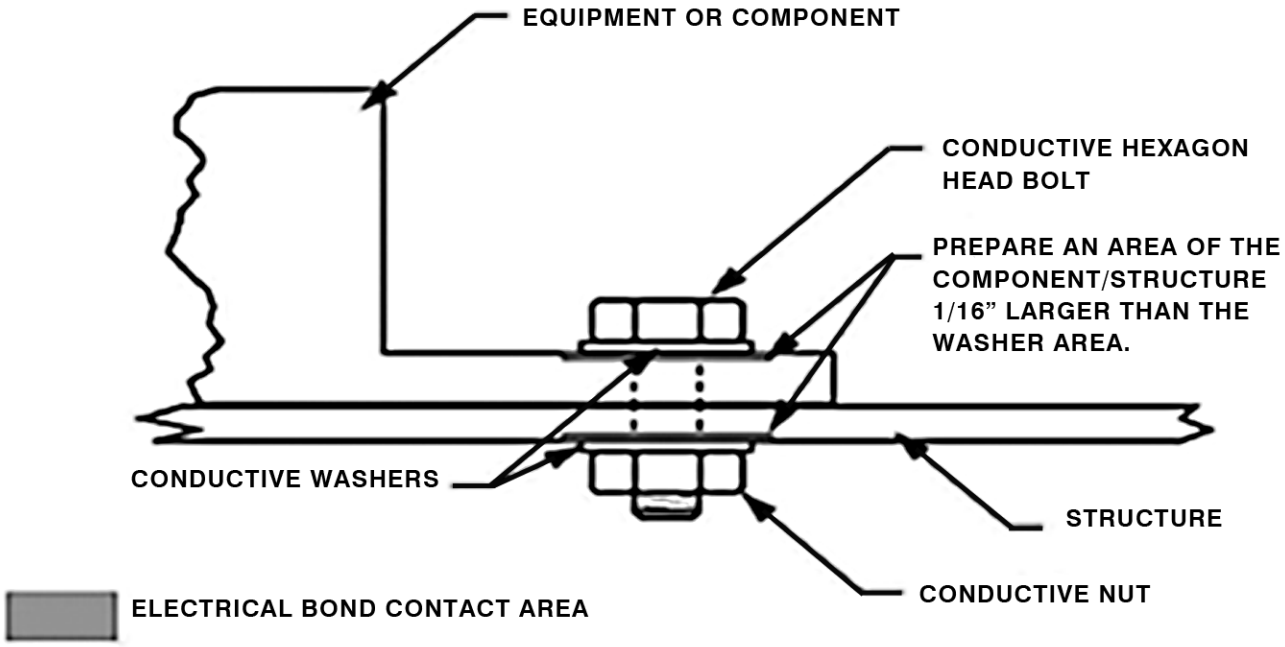
Note "Bonding through Attachment Hardware" for riveted joints must be with a minimum of three rivets if the rivet holes are bare of all insulating finishes, possesses an alodine finish and have been driven (i.e., are not pull type). However, there are high current applications in which more than three rivets are required or where rivets are not acceptable. Specific section of PSM 1-64-2 will specify.

Note In some applications a combination of bonding types is required and will be specified. For example, if a bonding jumper is terminated at a "Z" bracket, as shown below, the instructions for the jumper would be "Bonding through Bonding Jumper" while the instructions for the bracket would be "Bonding through Attachment Hardware" or "Bonding through Direct Contact". Typ. min. 3 rivets for "Bonding through Attachment Hardware".

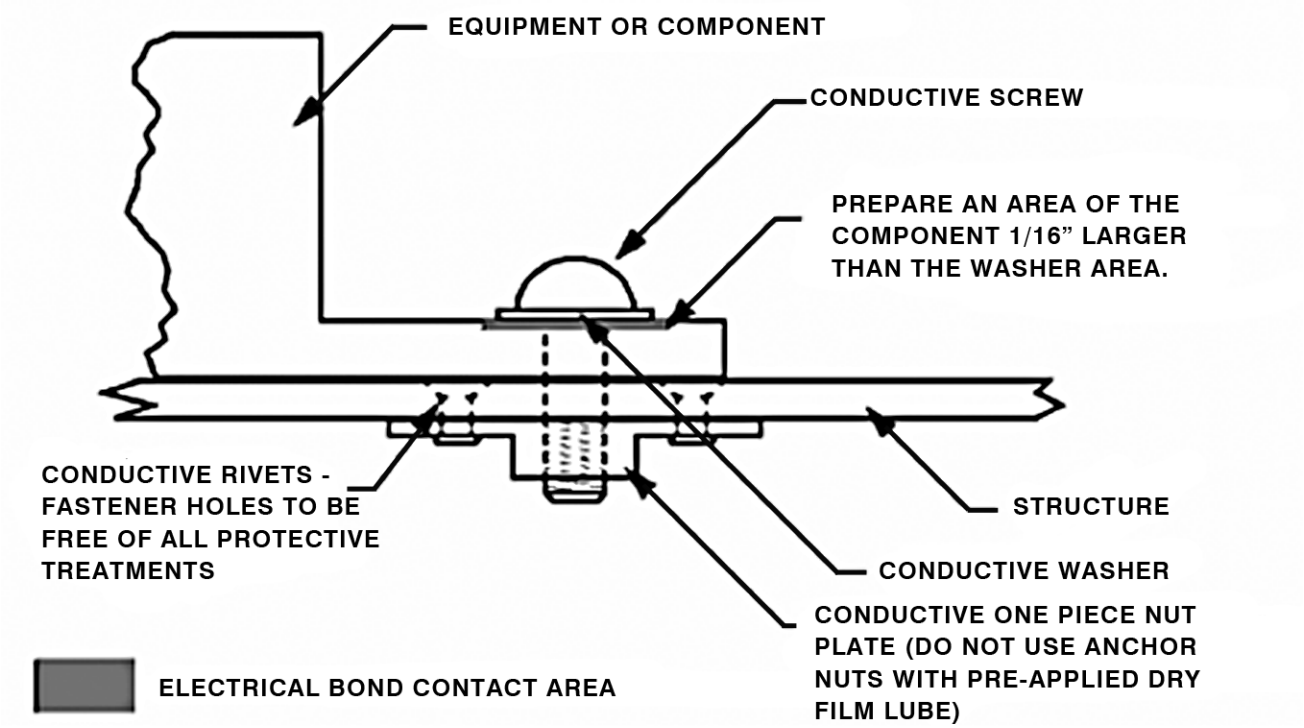




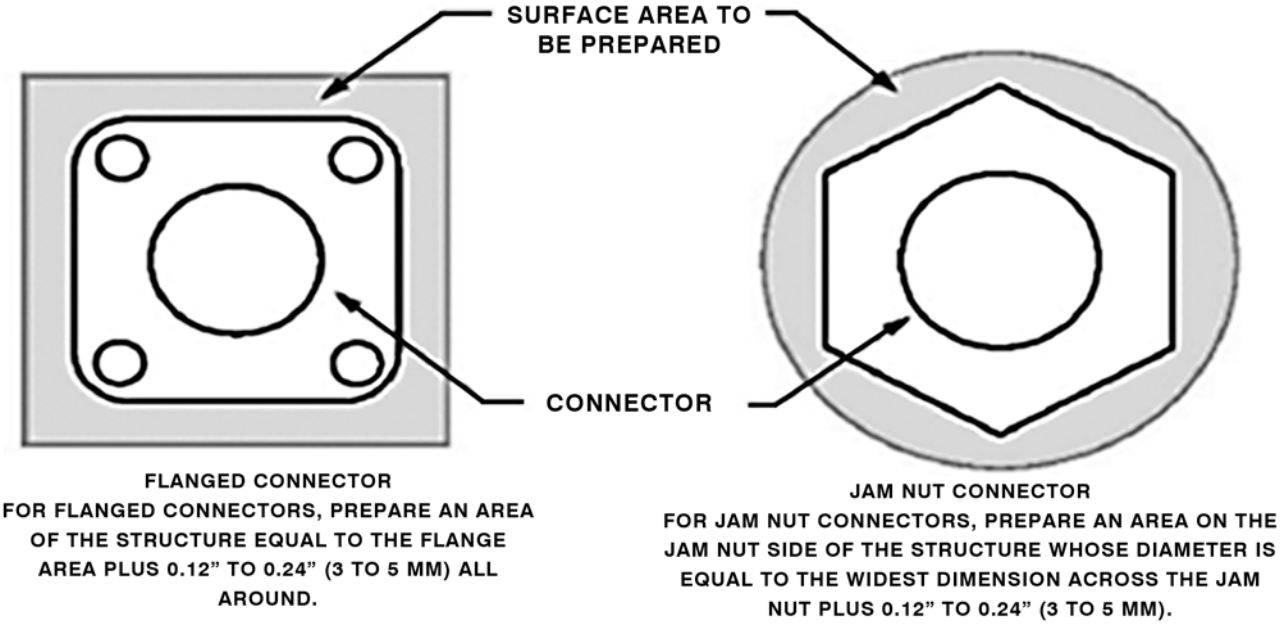
Surfaces to be Prepared for Bonding
Figure 1



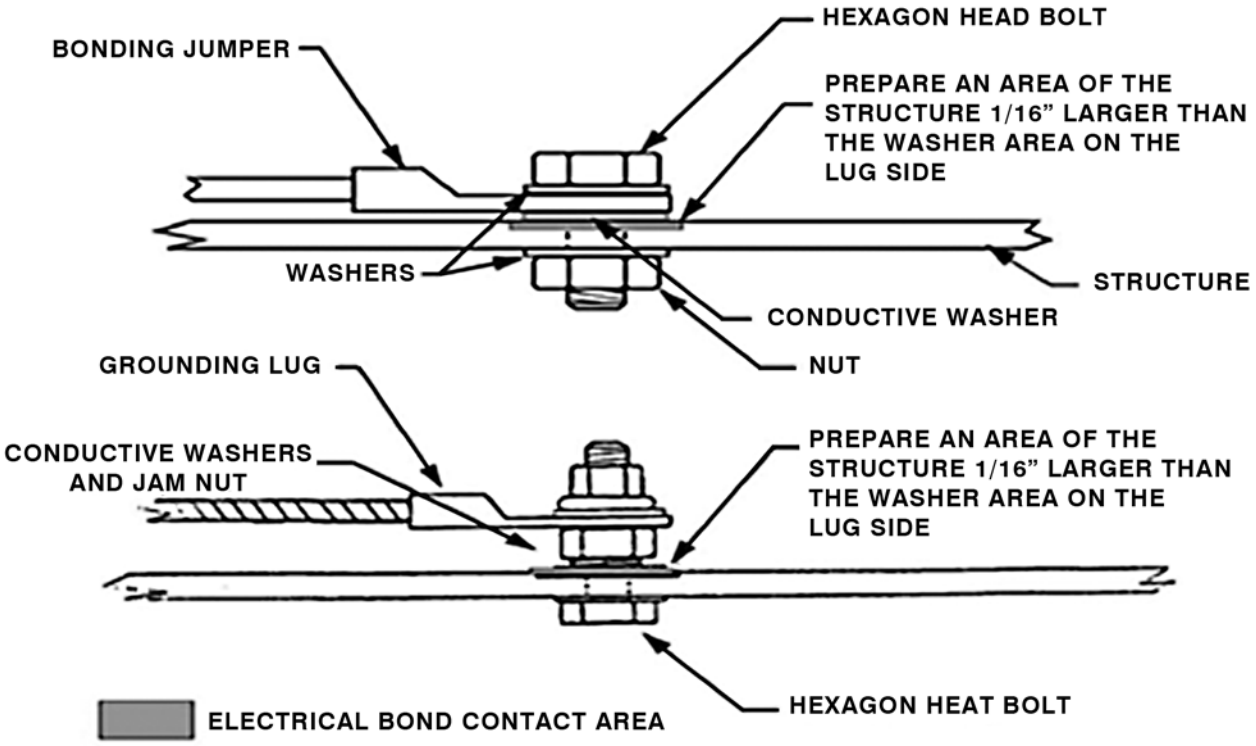
Surfaces to be Prepared with Conductive Nuts and Bolts
Figure 2



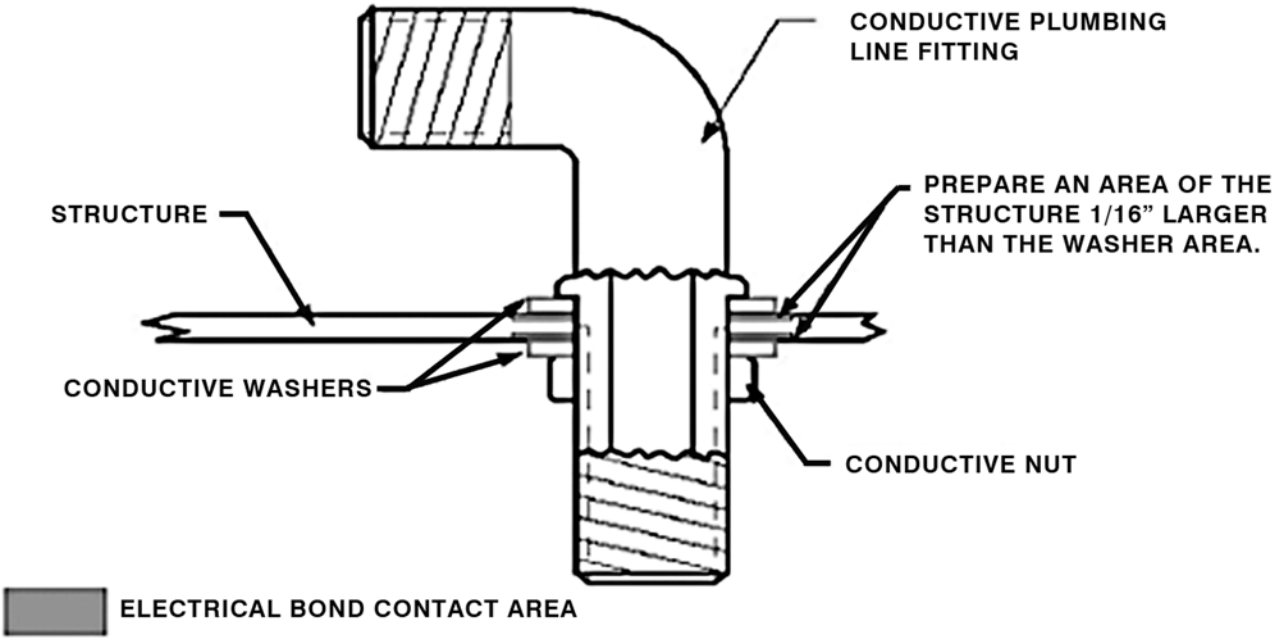
Surfaces to be Prepared for Bonding with Conductive Screws and Anchor Nuts
Figure 3



Surfaces to be Prepared for Bonding of Electrical Connectors
Figure 4



Surfaces to be Prepared for Bonding through Bonding Jumpers (Typical)
Figure 5



Electrical Bonding of Plumbing Lines Fittings to the Structure
Figure 6

3. Requirements

A. Check that all bonding point surfaces were thoroughly cleaned before the installation of clamps and bonding jumpers. This ensures a satisfactory bond.

B. Check pipe lugs for cracks, bonding terminals for insecure wire braiding, and clamps and connections for looseness.

C. Ensure that all bond connections are tightened to the torque value specified in Table 2 — Bonding Hardware Torque Values.

D. Ensure that the method and materials used for bonding and ground connections are in conformance with the standards and procedures in 20-80-00, Installation of Aircraft Antennas – General data.

E. Ensure that bonding jumpers are installed so that the wire does not come into contact with surrounding structure, equipment, fuel pipes, or other bonding jumpers (refer to 20-75-01, Application of Bonding Jumpers – General data, Figure 2). If contact is unavoidable, use a CSP 476 bonding jumper (complete with insulating sleeve). If there is a jumper other than the CSP 476, contact Viking Air Ltd.

F. For cable harnesses with individually shielded wires but without overbraid shields, verify proper shield termination according to PSM 1-64-2W, as applicable, before the backshell is closed.

G. Ensure that touch-up primer and, if applicable, paint top-coat, has been applied to any aircraft structure remaining exposed after installation of all bonding and grounding hardware.

4. Installation of Bonding Hardware and Ground Studs

A. Bonding Hardware

Install bonding hardware and connections according to the specified type of bonding installation. If the type of bonding installation required is not specified within the associated section of the PSM 1-64-2, contact Viking Air Ltd.

When installing the bonding hardware, including jumpers, check that all of the nuts or attachment screws are fully tightened.

Table 2 Bonding Hardware Torque Values

SCREW SIZE	CONVENTIONAL AND BLIND TYPE STUDS (INCH POUNDS)	PEM STUDS (INCH POUNDS)
#6	N/A	5 - 9
#8	15 - 20	15 - 18
#10	25 - 30	25 - 30
1/4	40 - 45	N/A
5/16	80 - 85	N/A
3/8	110 - 125	N/A
1/2	140 - 155	N/A

B. Ground Studs

1) Install all blind type bonding and grounding terminal studs in accordance with manufactures recommendation.

2) Install all PEM studs for use as bonding and grounding terminal studs in accordance with manufactures recommendation.

3) Torque all ground stud jam nuts and securing nuts to the values shown in Table 2 — Bonding Hardware Torque Values.

5. Installation of MS90298 Fuel Nozzle Grounding Receptacles

A. Prepare the inner surface if the skin around the hole is drilled as specified in Para 6.

B. Install the receptacle, including the inner and outer washers, and tighten the nut securely so that the receptacle does not rotate and the inner washer is firmly in contact with the skin.

C. Carry out the bonding resistance measurement test in accordance to Para 8.C.

D. Stake the receptacle nut at 3 points, approximately 120° apart, and seal the interior hardware and exterior hardware as specified in Para 11.

6. Surfaces to be Prepared for Electrical Bonding

A. Except as noted below, the electrical bond contact area refers to an area approximately 1/16” larger all around than the area of the component being bonded or, in the case of loop type clamps on tubing, approximately 50% greater than the clamp contact area.

Note Voi-Shan E-M grounding studs utilize the fastener hole to establish the electrical conductivity. Therefore, for Voi-Shan E-M grounding studs, the electrical bond contact area is limited to the fastener hole.

B. If surface is covered with a high resistance film then:

1) Strip the high resistance film from the electrical bond contact area as per Para 7.

Note Strip high resistance films on the grounding lug side to provide an acceptable electrical bond contact area.

Note After stripping any high resistance films, test the electrical bond contact area for consistent electrical conductivity and then chemical conversion coat, Class 3, within 8 hours of stripping the high resistance films.

2) Test the electrical bond contact area for consistent electrical conductivity according to Para 8.C.

3) If the electrical conductivity is not consistent over the electrical bond contact area then repeat Para 6.B.1) and Para 6.B.2) as required.

4) Touch up the electrical bond contact area with chemical conversion coating Class 3 as per 20-30-02, Chemical Conversion Coatings (Alodine) – Manual Application).

5) Solvent clean the electrical bond contact area as per Para 7.

6) Install component as per the applicable section PSM 1-64-2.

C. If the surface is not covered with a high resistance film then:

1) Solvent clean the electrical bond contact area as per Para 7.

2) Install component as per the applicable section in PSM 1-64-2.

D. The following finishes and surface conditions are satisfactory for bonding connections and do not need to be removed. Solvent clean these finishes according to Para 7. immediately before installation:

1) Plated surfaces

2) Graphite (powder, flake or stick)

3) Chrome pickle or selenious acid treatment on magnesium alloys

4) Conductive Chemical conversion coating (e.g. MIL–C–5541 Class 3)

E. If specified in the associated sections of PSM 1-64-2, coating the shank of a conductive bolt with F13, ensure that no F13 gets under the bolt head.

7. Surface Preparation

A. Removal of High Resistance Films

1) Materials and Equipment

- SD 8383 spot facing tool
- Abrasive paper, aluminium oxide, 180 – 240 grit
- Tack rag

2) Removal of High Resistance Films

Except as noted below, mechanically remove oxides and anodic films, dichromate treatment, paint, primer, etc. Using either an SD 8383 spot facing tool or abrasive paper. Keep the amount of metal removed during mechanical cleaning to a minimum.

a) On structure other than aluminium alloy, it is also acceptable to use a stainless steel wire brush to mechanically remove oxides and anodic films, dichromate treatment, paint, primer, etc. Do not use stainless steel wire brushes on aluminum alloys.

b) On aircraft fluid lines which have been F19 primed (without bond point masking), locally strip the primer from the bond contact area using chemical paint stripper according to manufactures recommendations, use masking tape to limit the extent of stripping.

c) If mechanical stripping is impractical, strip paints, dyes or stains from bond contact areas chemically according to manufactures recommendations. Use masking tape to limit the extent of stripping. When removing non-conductive coatings, do not use the same abrasive tool on different kinds of metal (i.e. don't use a toll on aluminum parts that have been used on steel parts).

3) After mechanically abrading the surface, use a tack rag to remove any dust or adhering particles.

4) Treat the bond contact areas of magnesium alloy parts which have been mechanically cleaned with 10% selenious acid solution according to manufactures recommendations.

5) Thoroughly clean all bond contact surfaces (including terminal lugs of bonding jumpers) by solvent wiping them according to Para 7.B immediately before installation.

B. Manual Solvent Cleaning

Note Solvent clean area about 50% greater than sealant contact area.

1) Materials and Equipment

- Clean Lint-free wiping clothes
- Protective Gloves (e.g. DSC 422–1 and DSC 422–2)
- Approved Respirator
- Approved Safety glasses

2) Manual Solvent Cleaning

Note Solvent clean an area about 50% than sealant contact area.

WARNING KEEP SOLVENT AWAY FROM FIRE AND OTHER SOURCES OF IGNITION. PROVIDE SUFFICIENT VENTILATION WHEN WORKING IN CONFINED SPACES. AVOID SKIN CONTACT WITH SOLVENT.

3) Apply small amount of solvent to a clean wiping cloth. Refer to 20-40-01, Silicone Adhesives – General Data, Table 1 for the solvent appropriate for the surface to be cleaned. For solvents which evaporate quickly, apply more solvent for solvents which evaporate slowly.

4) Wipe the area with a solvent dampened wiping cloth.

a) When cleaning bonded surfaces with solvent saturated wiping cloth, thoroughly scrub the bonding surfaces. Wipe dry with a cloth before solvent evaporates.

b) Keep solvent cloth saturated and turn frequently to avoid reusing a dirty area.

5) Wipe dry with another clean wiping cloth before the solvent evaporates.

Note It is important to wipe the surface immediately after cleaning because if the solvent evaporates before it is wiped up, the oil, and grease residue will remain in the surface. Although the DS 108 solvent has a slow evaporation rate, wipe dry as soon as possible.

6) Check for a water free surface.

7) If more cleaning is needed, apply fresh solvent to a clean portion of the wiping cloth and repeat Para 7.B.1) through to Para 7.B.6).

8. Resistance Measurement

A. General

1) If possible, take resistance measurements or mask areas where resistance measurements are to be taken before applying non-conductive coatings.

2) If non-conductive coatings have been applied to the area where the resistance measurement is to be taken, remove an area of coating sufficient to accommodate the ohmmeter probe according to Para 7.A before taking the measurement.

3) If the protective coating has been removed from a part or an area of the part was masked to accomplish resistance measurement, touch up the area as follows after taking the measurement:

B. Equipment

- Approved Safety glasses
- Multimeter, 1 megohm range.
- 500 volt Insulation Resistance Tester (e.g., Biddle Instruments P/N 210801–2)
- Spot facing tool (e.g., SD 8383)

Note Any of the following ohmmeters may be used, provided hardware and training is available:

- BCD M1 Milliohm meter
- Biddle Instruments digital low resistance ohmmeter, DRLO #247001
- SD 8778–1 10 amp constant current ohmmeter test set
- Ducter Instruments/Megger Instruments BT51 digital low resistance ohmmeter
- Ducter D203 Micro-Ohmmeter by Megger Instruments
- HP 4328A Milliohm meter
- HIOKI 3220 or 3540 Milliohm meter
- Eaton Corp. model 584 resistance tester
- Eutron S.A.S. Micro-Ohmmeter BVM 3–10
- AOIP Measures OM 21–1 Ohmmeter
- AEMC Micro-Ohmmeter, Model 5600

C. Testing of Stripped or Masked Electrical Bond Contact Areas

1) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing both probes of a BCD M1 Milliohm meter on the stripped/masked surface and measuring the resistance. The resistance between the two points must be no more than 2.5 milliohms unless stated otherwise.

Note It is acceptable to use one of the ohmmeters specified in Para 7.A as an alternative to the BCD M1 Milliohm meter.

2) Check the electrical conductivity between several random pairs of points within the electrical bond contact area to ensure consistency.

3) If the electrical conductivity is not consistent over the electrical bond contact area, remove the non-conductive coating as specified in Para 7. and re-test the area.

9. Electrical Component Bonding

All Rack Mount/Remote Mount units shall be electrically bonded to the airframe. This is accomplished by ensuring that the mating surfaces between the LRU mounting tray (or LRU mounting feet if a tray is not used) provides a low impedance electrical path. The mating surfaces must be free of all paint and other non-conductive elements and should be burnished to ensure a good bond.

All units must be properly grounded to the airframe ground. Resistance measurements to be made between the unit mounting tray or unit mounting feet if a tray is not used (or other acceptable/known unit ground) and the applicable mounting surface. Maximum resistance per each electrical bonding interface is 2.5 milliohms not to exceed 10 milliohms from unit to primary structure.

Panel/Pedestal Mount Units cannot be readily grounded to the airframe via the panel mounting screws of DZUS fasteners. Grounding of these units is accomplished by using the shield grounding braid or bonding jumper bonded to the airframe.

10. Touch-up of Bond Points and Resistance Measurements Points

A. Except as noted below, if the protective coating has been removed from a part or an area of the part that was masked to accomplish electrical bonding or resistance measurement, solvent clean the affected area according to Para 7. and touch-up the area with F19 primer applied by brush within 24 hours of installing the bonding hardware:

Note Touch-up integral fuel tanks with F21 Type II primer applied by brush.

Bond points which are to be potted with sealant as specified in Para 11. do not require primer touch-up.

B. Ensure that primer refinishing provides complete coverage of the affected area and cover an area approximately twice that of the area cleaned for bonding/resistance measurement.

C. If the affected area requires paint touch-up, brush coat the area with the top-coat paint system matching the existing.

D. Touch-up any unused bonding points provided at the ends of primed fluid lines by:

1) Solvent clean the affected area according to Para 7.

2) Touch-up the area with F19 primer applied by brush within 24 hours of installing the bonding hardware.

11. Sealing of Bonding and Grounding Hardware

Refer to Figure 1 and Figure 2 for the typical areas to be sealed on the various types of bonding assemblies. For clamps which secure/support overbraid harness and provide electrical bonding and grounding, pot the clamp attachment point to the structure; however, it is not necessary to pot or seal in the area where the clamp contacts overbraid the harness.

A. Materials

- Sealant spatulas (e.g. SD9164, Mk1, Mk2, Mk3 and Mk4)
- PR1776M Sealant
- Humiseal 1B15 Sealant if no sealant has been specified
- Neozapon red dye only if Humiseal 1B15 is used
- Humiseal 1B31LOC or Humiseal 1B31EPA

Note 1 Humiseal 1B31LOC and Humiseal 1B31EPA contain no MEK, and for this reason are preferred over Humiseal 1B15 sealant. Humiseal 1B31LOC and Humiseal 1B31EPA do not need to be dyed red, they can be inspected using UV lighting tools.

2 Before use, Humiseal 1B15 sealant must be dyed red by mixing in 4 - 6 grams of BASF Corp. 335 Neozapon red dye per U.S. quart of sealant. Humiseal 1B15 sealant can be purchased with red dye pre mixed or, sealant and dye separately, to be mixed prior to use. The purposes of dying the Humiseal 1B15 sealant is to make it visible for inspection purposes, so some variation in the shade and intensity of red colour is acceptable.

B. Equipment

- Protective Gloves (e.g. DSC 422–1 and DSC 422–2)
- Approved Safety glasses

C. Preparation

1) Solvent clean an area about 50% greater than sealant contact area, in accordance to Para 7.

2) When installing the bonding hardware, including bonding jumpers, check that all of the nuts or attachment screws are fully tightened.

3) Pot the electrical bonding or grounding attachment points and hardware on the fuselage as well as those inside the fuel tanks with sealant. If the sealant is not specified then pot as follows:

a) Inside the fuselage (baggage compartments, electrical bays, avionics bays, electrical box assemblies, cockpit, instrument panel, cabin, tailcone) pot with either PR1776M or Humiseal sealant (Humiseal 1B31LOC, Humiseal 1B31EPA or Humiseal 1B15).

CAUTION DO NOT USE ANY TYPE OF HUMISEAL SEALANT IN AREAS SUBJECT TO HIGH FREQUENCY EXPOSURE TO SKYDROL HYDRAULIC FLUID.

b) Inside fuel tanks, pot with PR1776M.

CAUTION DO NOT USE ANY TYPE OF HUMISEAL SEALANT INSIDE FUEL TANKS.

c) For exterior and high moisture areas (landing gear, empennage, hydraulic bay, fuel gallery, fuel filling provisions, windscreens, wire antennas, inside radome), pot with PR1776M sealant.

d) Unless specified, it is acceptable to seal electrical bonding attachment points and hardware that are part of an electrical box or panel with either PR1776M or Humiseal.

4) If Humiseal sealant is used:

a) Before using Humiseal 1B15 sealant, dye the sealant red per Para 11.A, Note 2.

b) Before sealing with Humiseal, clean the area using LPS CFC Free cleaner. Gentle air drying with compressed air is permitted.

c) Apply Humiseal coating using a paint brush of a size suitable for the area to be coated. Humiseal 1B15 may be brushed with a small addition of Humiseal Thinner 503. The consistent application of the film depends on component density and the operator's technique.

5) If PR1776M sealant is used prepare in accordance to manufacturer specifications.

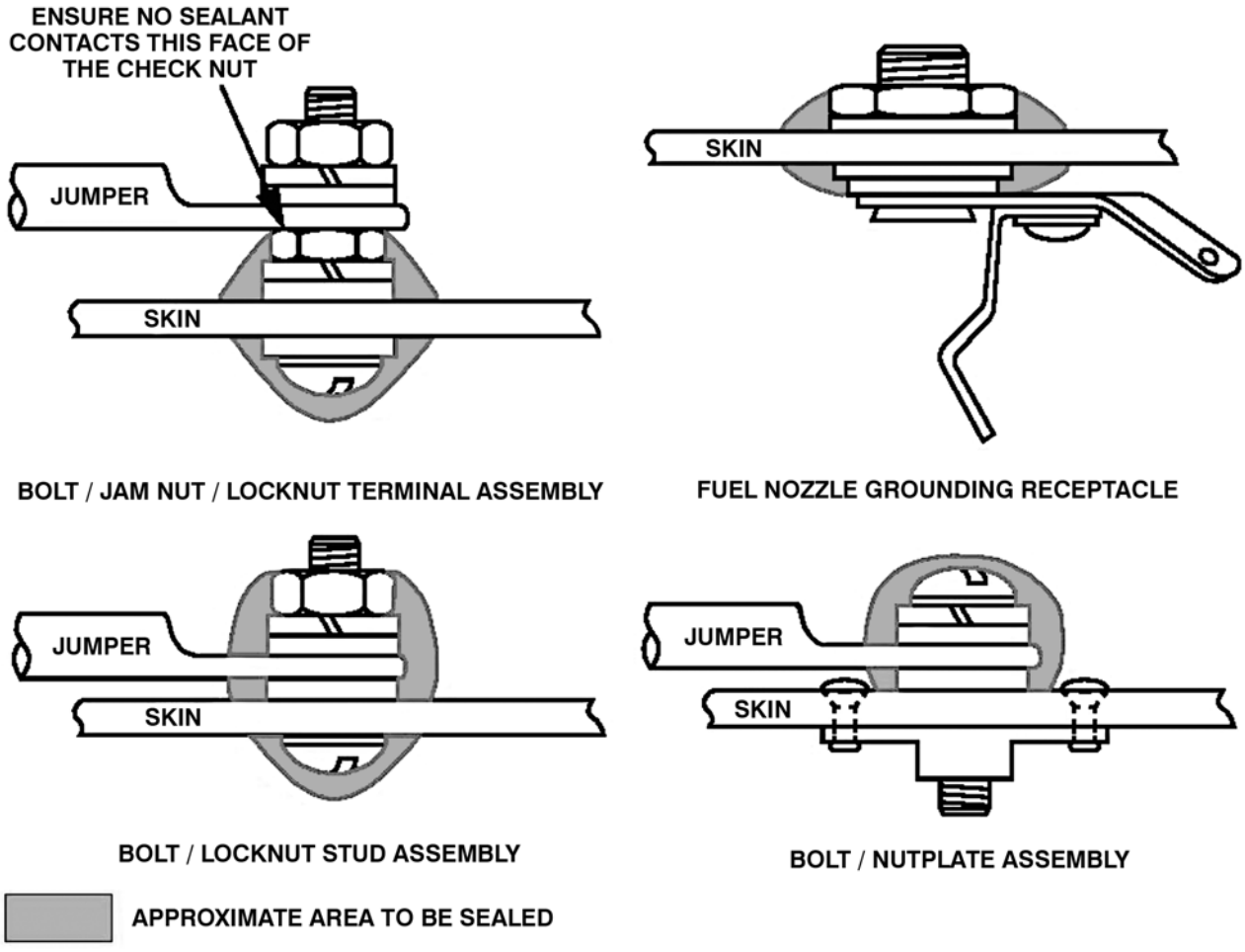
D. Application of Sealant

1) Apply sealant to the electrical bonded assemblies using the procedures listed in 20-50-01, Weather Sealing – General Data. For the final application of sealant refer to the electrical bonded assemblies as shown in Figure 7 and Figure 8 for areas NOT exposed to high levels of moisture, and Figure 9 and Figure 10 for areas exposed to high levels of moisture.

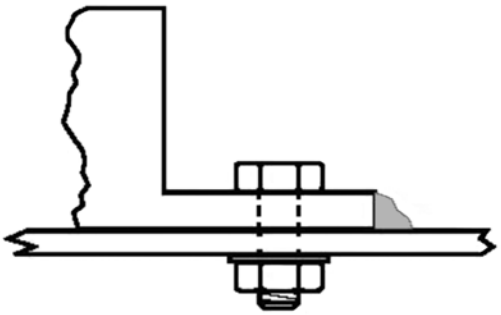
2) When installing bonding hardware on hardware metallic tubes, including clamps, bonding jumpers and hardware, seal the clamp around the circumference of the tube, screw, nut, washers and the ring portion of the terminal.

3) Extend the sealant approximately 1/8” beyond the area stripped of protective coating.

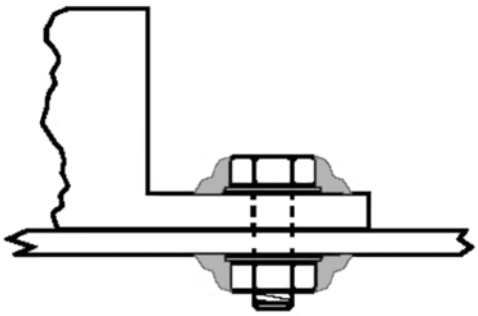
4) For M81714/27 stud mounted ground modules installed outside the fuselage and in high moisture areas, seal the stud with PR1776M sealant. Touch up bare metal using F19 primer.



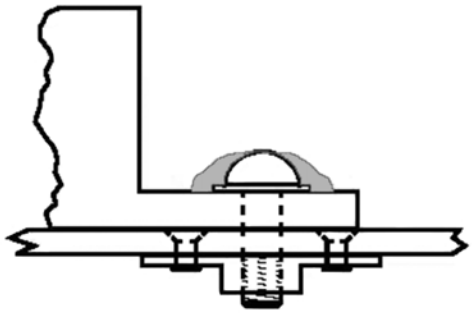
Sealing of Electrical Bonding Assemblies – Indirect Bonding (NOT exposed to high levels of moisture)
Figure 7



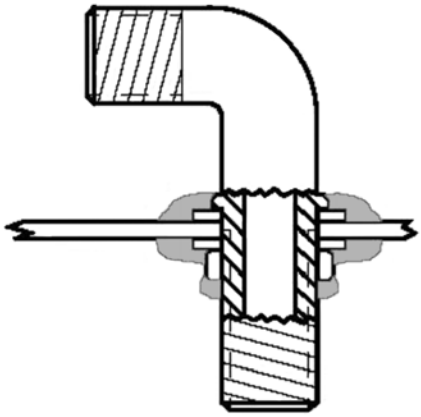
DIRECT CONTACT BONDING



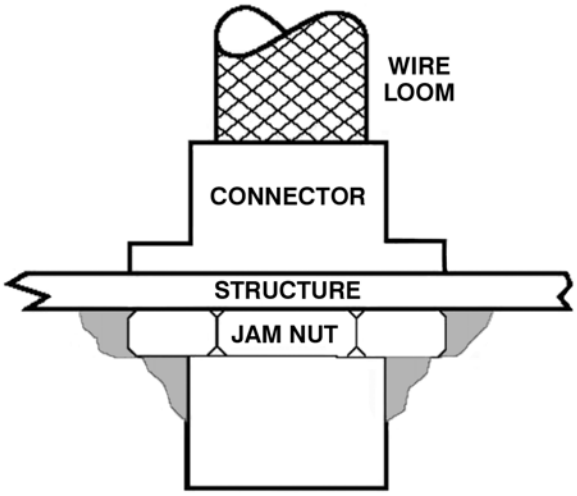
BONDING WITH CONDUCTIVE
NUTS AND BOLTS



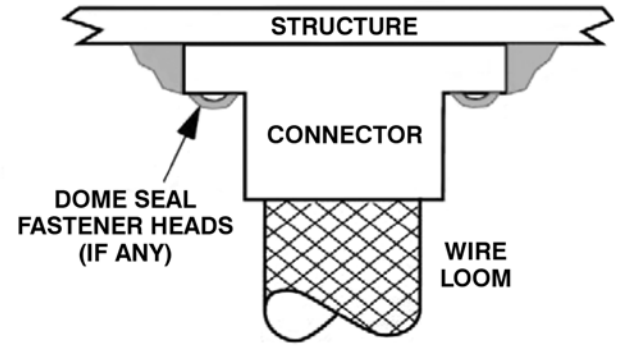
BONDING WITH CONDUCTIVE SCREWS



BONDING OF PLUMBING LINE
FITTINGS TO THE STRUCTURE



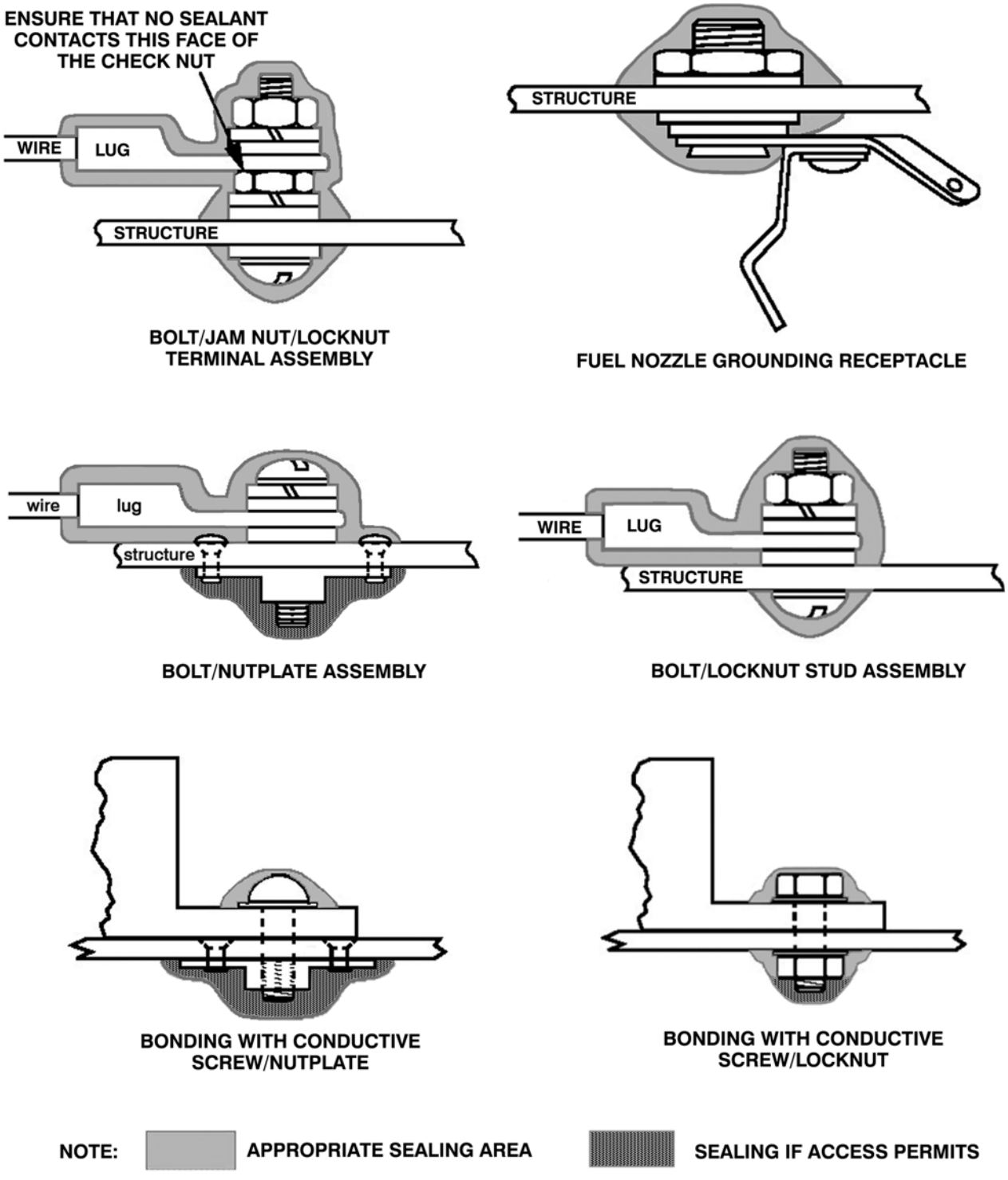
BONDING OF JAM NUT
ELECTRICAL CONNECTORS



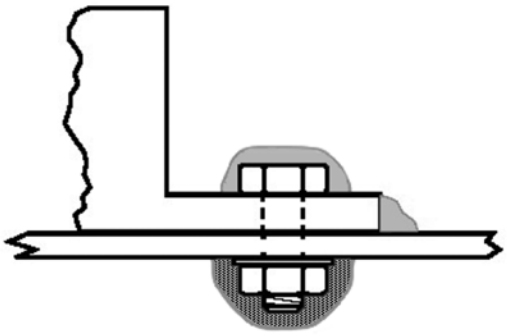
BONDING OF FLANGED
ELECTRICAL CONNECTORS

APPROXIMATE AREA TO BE SEALED

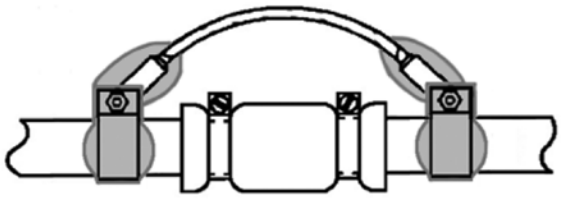
Sealing of Electrical Bonding Assemblies – Direct Bonding (NOT exposed to high levels of moisture)
Figure 8



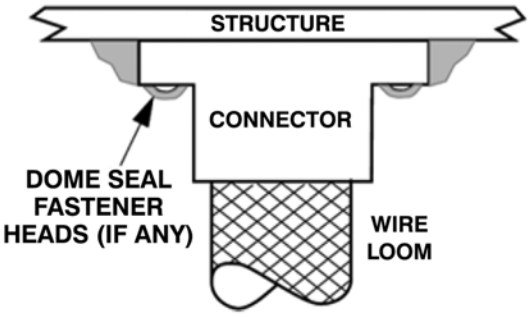
Sealing of Electrical Bonding Assemblies – Indirect Bonding (exposed to high levels of moisture)
Figure 9



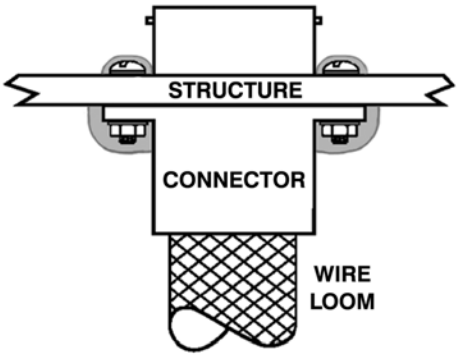
DIRECT CONTACT BONDING



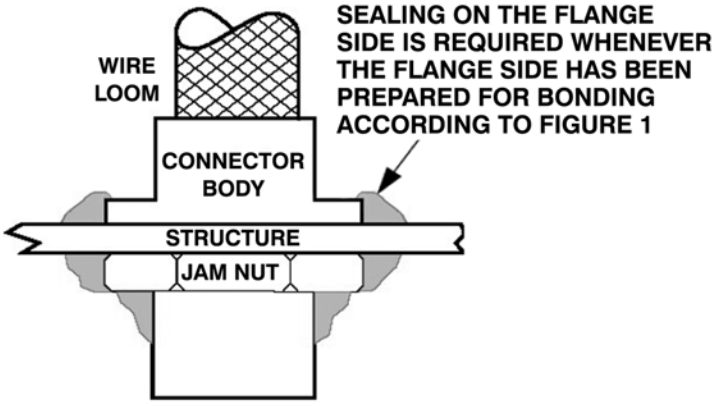
LOOP CLAMP BONDING



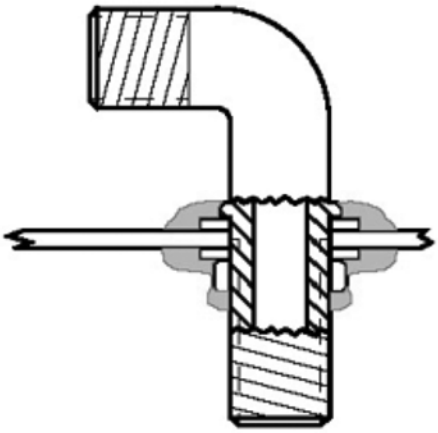
FLANGE CONNECTOR BONDING



FLANGE CONNECTOR BONDING



JAM NUT CONNECTOR BONDING



PLUMBING LINE FITTING BONDING

NOTE: APPROPRIATE AREA TO BE SEALED SEALING IF ACCESS PERMITS

Sealing of Electrical Bonding Assemblies – Direct Bonding (exposed to high levels of moisture)
Figure 10

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Application of Bonding Jumpers – General data

1. General

This standard specifies the procedures and requirements for a typical installation of bonding hardware.

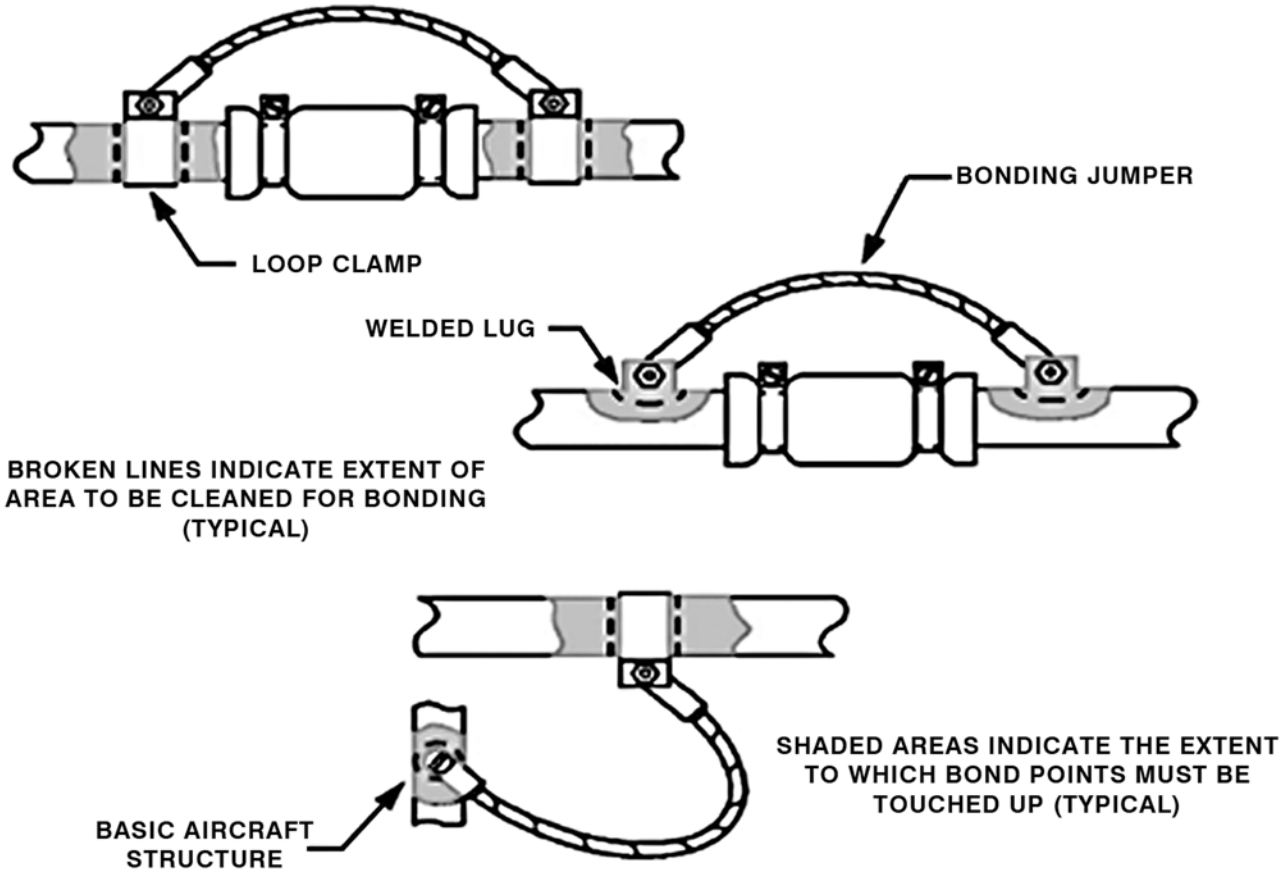
2. Application of Bonding Jumpers

A. Install the bonding connections as directly as possible without applying strain to the bonding jumpers or welded lugs. Refer to Figure 1.

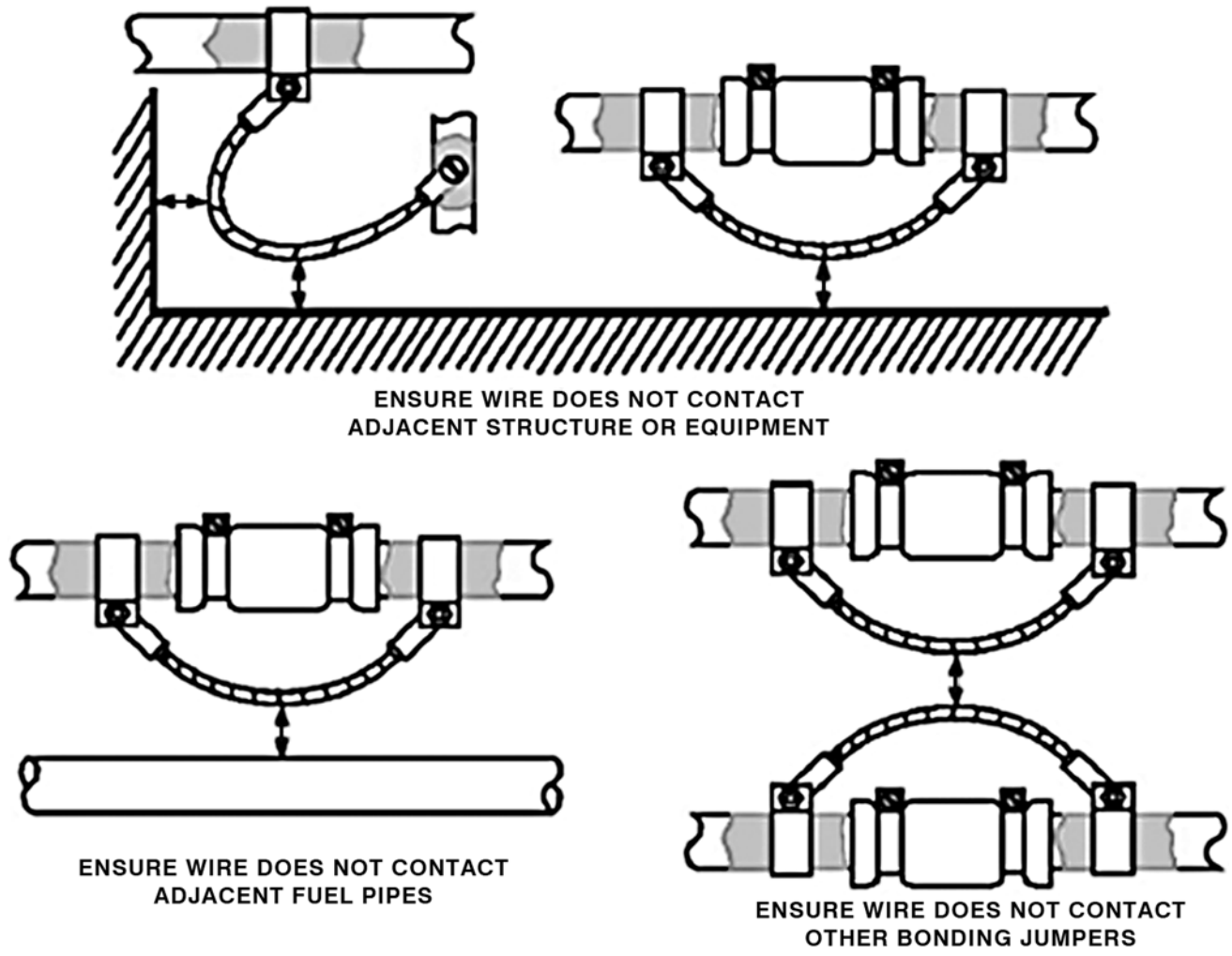
B. Install bonding jumpers so that the wire does not come into contact with surrounding structure, equipment, fuel pipes or other bonding jumpers. Refer to Figure 2. If contact is unavoidable a CSP 476 bonding jumper complete with insulating sleeve must be used.

C. If possible, locate bonding connections in protected areas in order to avoid contact with electrical wiring and control boxes or inference with moveable components of the aircraft.

D. When installing a loop type clamp on a tubular member, take care not to crimp or indent the tube.



Typical Application of Bonding Jumpers
Figure 1



Typical Bonding Jumper Clearance Requirements
Figure 2

Electrical Bonding of Plumbing Lines – General data

1. General

   This standard specifies the procedures and requirements for the typical electrical bonding for plumbing lines.

2. Electrical Bonding of Plumbing Lines (Typical)

   A. Unless otherwise specified, bond metal plumbing lines to the structure with a mechanically secure connection.

   B. If a plumbing line has intervening non-conductive joints (e.g. T–joints with anodized threads), bond each individual section of more than 3 inches either to the structure or the adjacent plumbing line section.

   C. If two or more components are bonded in a series, bond both ends of the circuit to the structure to ensure that the loss of an intermediate bond connection will not leave any component isolated from ground.

   D. Regard tube connections which include an electrical bonding feature (e.g. Wiggins W700 series) as inherently bonded and therefore an integral (conductive) part of the plumbing line.

   E. Bond plumbing line sections to the structure at intervals not exceeding 144 inches.

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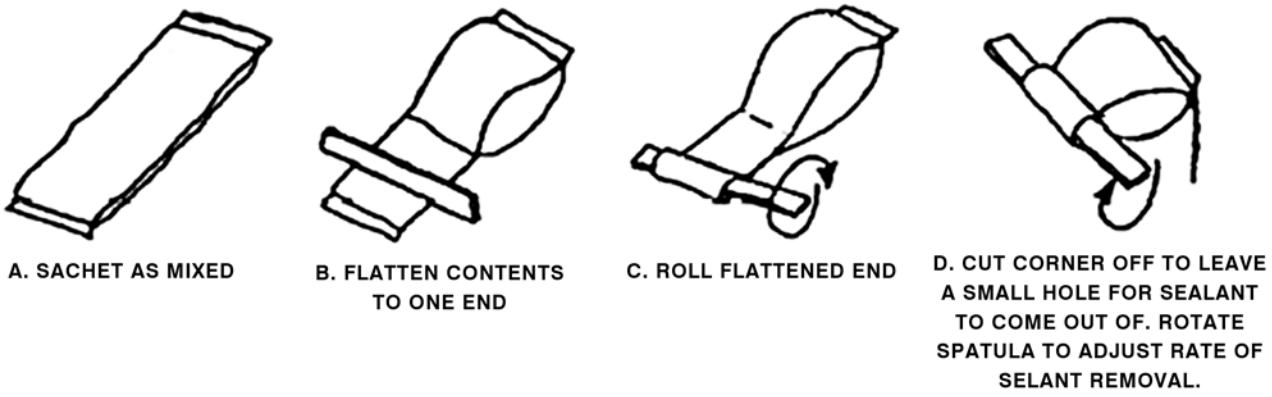
Installation of Aircraft Antennas – General data

1. General

This standard specifies the procedures and requirements for the typical installation of aircraft antenna. To be used when no antenna installation is provided in the applicable sections of the PSM 1-64-2.

2. Materials and Equipment

- Felt pad impregnated with release agent, HR Smith gasket wipes (Product No. 10–900–SS).
- Sealant, MIL–PRF–81733 prepared in accordance to 20-50-01, Weather Sealing – General Data.
- Sealant, Courtaulds Aerospace PR1428 prepared according to 20-50-01, Weather Sealing – General Data. If using sealant which comes in a plastic sachet, after mixing the sealant squeeze it all to one end of the sachet and then cut a small hole in the corner as shown below:
- Sealant spatulas (e.g. SD9164, Mk1, Mk2, Mk3 and Mk4)
- Protective Gloves (e.g. DSC 422–1 and DSC 422–2)
- Approved Safety glasses



Plastic Sachet Sealant
Figure 1

3. Requirements

A. Check that all bonding point surfaces were thoroughly cleaned before the installation of clamps and bonding jumpers. This ensures a satisfactory bond.

B. Check pipe lugs for cracks, bonding terminals for insecure wire braiding, and clamps and connections for looseness.

C. Ensure that all bond connections are tightened to the torque value specified in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data Table 2.

D. Ensure that the method and materials used for bonding and ground connections are in conformance with the standards and procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

E. Ensure that bonding jumpers are installed so that the wire does not come into contact with surrounding structure, equipment, fuel pipes, or other bonding jumpers (refer to 20-75-01, Application of Bonding Jumpers – General data Figure 2). If contact is unavoidable, use a CSP 476 bonding jumper (complete with insulating sleeve). If there is a jumper other than the CSP 476, contact Viking Air Ltd.

F. For cable harnesses with individually shielded wires but without overbraid shields, verify proper shield termination according to PSM 1-64-2W, as applicable, before the backshell is closed.

G. Ensure that touch-up primer and, if applicable, paint top-coat, has been applied to any aircraft structure remaining exposed after installation of all bonding and grounding hardware.

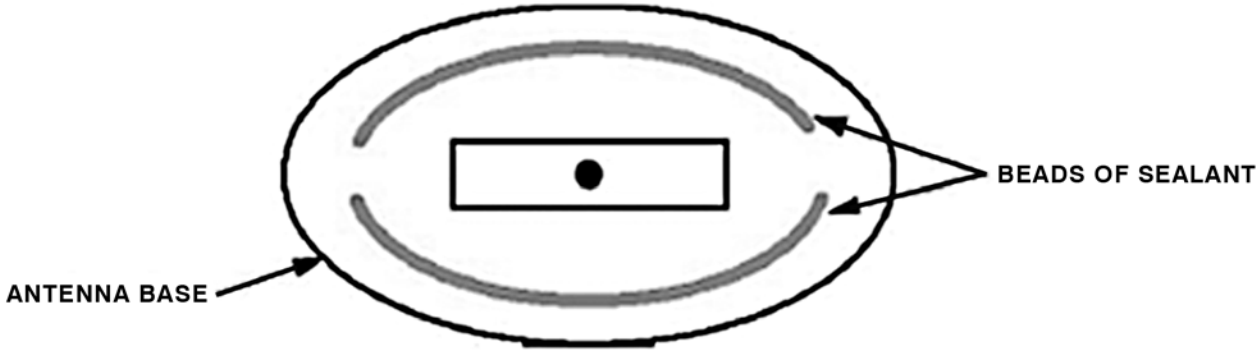
4. Installation Procedures

Note If the antenna is being mounted on the upper surface of the airframe and is fitted with drainage holes, ensure that the drainage holes at the base and sides are kept free of sealant to allow drainage from the shell. Plug and fillet seal any drainage holes on top of the antenna using the plugs provided.

Note If the antenna is being mounted on the lower surface of the airframe and is fitted with drainage holes, ensure that the drainage holes at the tip (i.e., lowest point of inverted antenna) are kept free of sealant to allow drainage from the shell. Plug and fillet seal any drainage holes on the base or sides of the antenna using the plugs provided.

A. For wet-fit and dry-fit gaskets, apply release agent to the antenna base and the mounting area of the airframe using a gasket wipe. Ensure that the surfaces are well covered with release agent, but not to excess.

B. If the engineering drawing specifies use of wet-fit gasket, apply PR1428 sealant to the antenna base and the mounting area of the airframe approximately as shown below and then spread the sealant evenly over the complete surface of the antenna base and the mounting area of the airframe to a thickness of approximately, but not more than, 0.010" using a wooden spatula. Do not apply sealant if installing a dry-fit gasket.

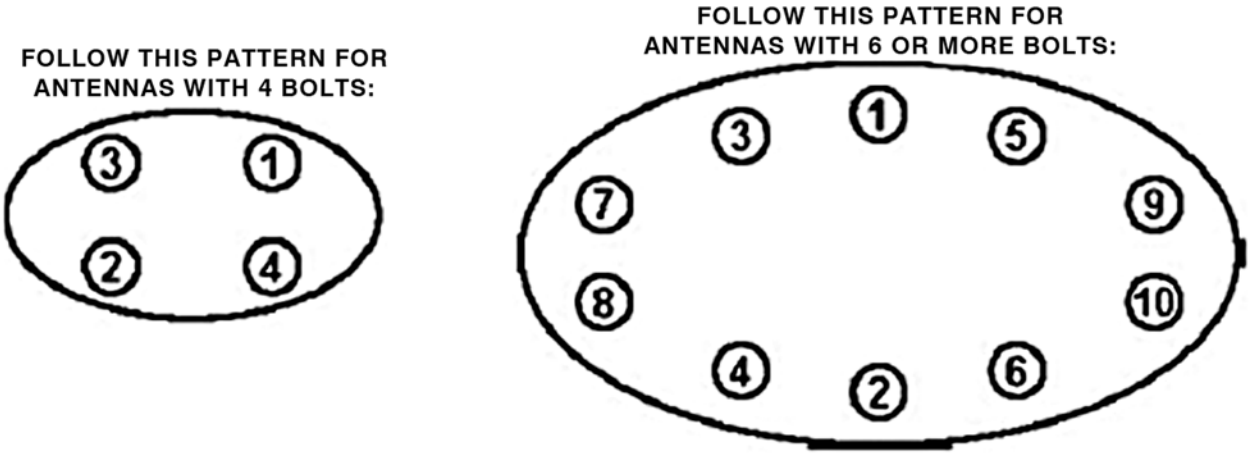


Base of the Antenna
Figure 2

1) Carefully place the gasket on the antenna base, ensuring that the mounting holes in the gasket align with the antenna mounting holes. If a H.R. Smith conductive dry fit sealing gasket is to be used, remove all the release paper from the mating side of the gasket to expose the adhesive surface before applying the gasket.

2) Fit the antenna/gasket assembly on the aircraft and secure it with the fixing bolts. For H.R. Smith conductive dry fit sealing gaskets, remove all the release paper from the mating side of the gasket to expose the adhesive surface before applying in the antenna/gasket assembly.

3) Tighten the bolts, in the pattern shown in Figure 3:



Antenna Bolt Pattern
Figure 3

4) Solvent clean the exposed area of the aircraft structure around the antenna in accordance to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data Para 6.

5) Touch-up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer refinishing provides complete coverage of the affected area.

6) If the affected area requires paint touch-up, brush coat the area with the top-coat paint system matching the existing.

7) Use MIL–PRF–81733 sealant to fillet seal around the antenna base in accordance with 20-50-01, Weather Sealing – General Data and seal the securing screws in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data Para 10. The aircraft will be ready to fly approximately 24 hours after sealing.

8) Carry out the antenna electrical bonding check in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data Para 8.

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CHAPTER 21

AIR CONDITIONING

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CHAPTER 21 – AIR CONDITIONING

LIST OF EFFECTIVE PAGES

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Air Conditioning – General Data

1. General

The heating and ventilating system utilizes bleed air and fresh air from a scoop intake in the fuselage to provide heated, ram, or mixed air to maintain appropriate temperatures in the cabin and flight compartment. Heated air is circulated through ducts beneath the flooring to outlets at the base of each cabin and flight compartment floor. Two tubes, one for each side of the windshield, deliver heated air to the windshield defrost outlets. The same ducts also circulate cooling ram air or a mixture of heated air and ram air in the cabin and flight compartment. Two additional tubes, one for each outboard side of the flight compartment, deliver unconditioned ram air to the gaspers located on each outboard side of the instrument panel.

Additional ducts deliver ram air to the cabin through individual passenger gaspers located in the cove moulding ducts on each side of the cabin upper walls. A fan is installed in the air intake duct for forced ram air ventilation when on ground. the system provides for manual temperature control by means of a manually controlled motorized hot air valve and manual control of both ram air inflow and cabin/flight compartment air distribution balance. An exhaust vent is installed in the cabin roof.

Refer to Figure 1 for details.

A. Heating System

Compressed and heated bleed air from the engine compressors passes through pipes to the main section of the heating system where, depending on temperature selection, it is controlled and mixed with secondary air (ram, fan pressure or recirculated air). The warm mixed air then passes through a silencer, and is ducted to outlets at floor level in the cabin, pilot's foot warmers, windshield and flight compartment outlets. (Refer to 21-40-00, Heating System – General Data). The heating system is controlled manually, the control switches being mounted on the co-pilot's sub-panel on the instrument panel in the flight compartment. (Refer to 21-60-00, Environmental Control System (Mod 6/2014) – General Data).

B. Cooling System

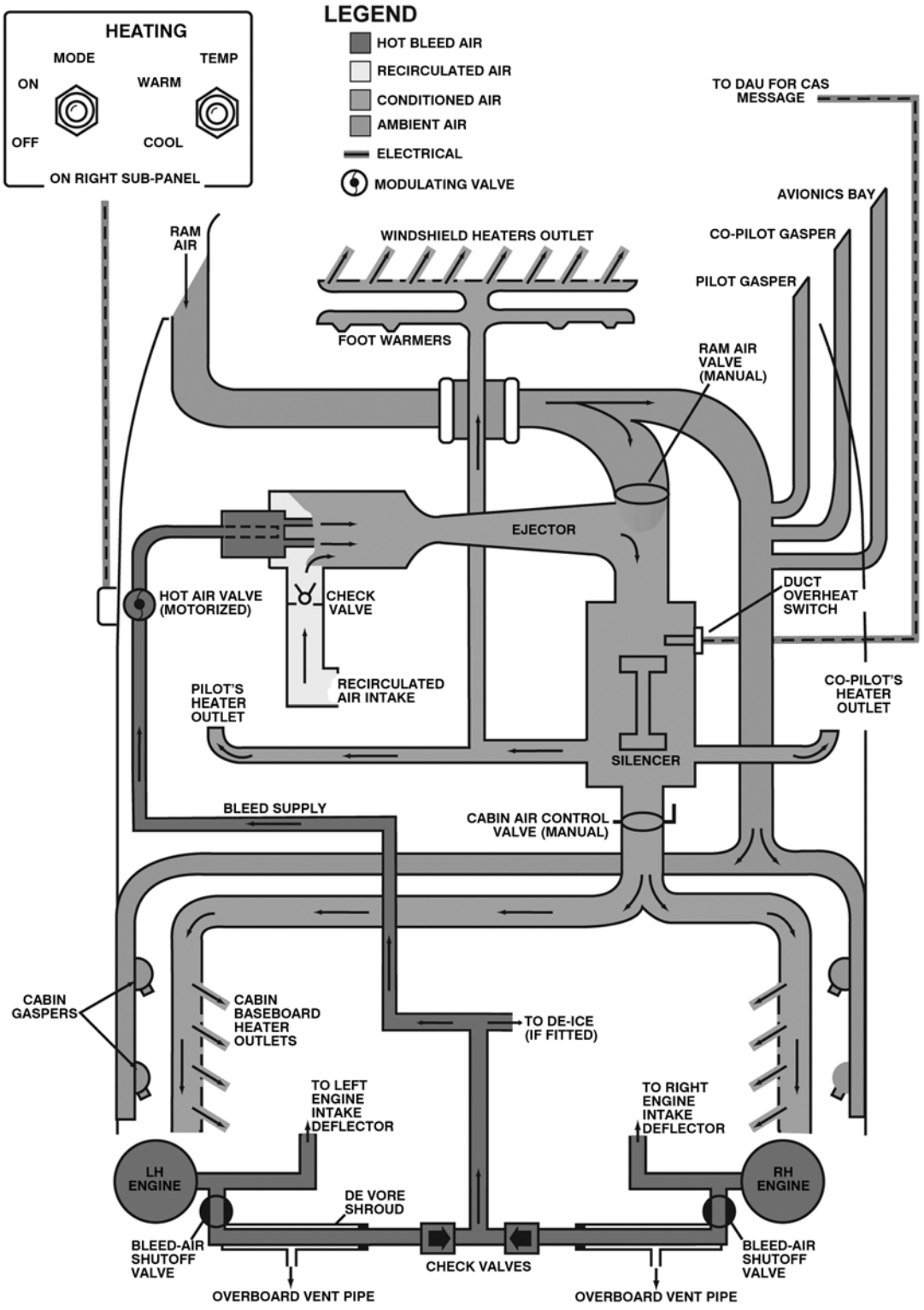
Ram air, ducted from a scoop on the left side of the fuselage nose, is supplied to the heating system, and two gaspers located on both sides of the cabin above the windows. The ram air valve, manually controlled by a handle in the flight compartment, controls the amount of ram air to the heating system. To provide a supply of cool air when the aircraft is stationary, an electric fan is installed in the main ram air duct. Additional flight compartment ventilation can be obtained by opening the flight compartment windows. Flight compartment ventilation is provided by two gaspers in the instrument panel.

C. Air Exhausting

A portion of the exhaust air vents through a vent assembly mounted in the cabin roof. The vent outlet is covered externally by a fairing which is open at the rear end. A shielded vent prevents rain entry when aircraft is parked in adverse weather conditions (Mod 6/1633). A suction, induced by the vent assembly, extracts the stale air from the aircraft interior through a grille in the cabin ceiling. Stale air also passes through vents in the luggage compartment, and leaves the aircraft at the tail cone.

D. Temperature Control System

The temperature control system controls the opening and closing of the hot air valve, installed in the heating system bleed air supply line, to maintain a selected cabin temperature. The control switches are mounted on the co-pilot's subpanel on the instrument panel in the Environmental section of the flight compartment. (Refer to 21-60-00, Environmental Control System (Mod 6/2014) – General Data).



Air Conditioning – Schematic
Figure 1

Avionics/Equipment Cooling (Mod 6/2014) – General Data

1. General

The distribution system provides circulation of cool air for the equipment installed in the forward avionics compartment and around the Data Acquisition Unit (DAU), located in the fuselage bulkhead.

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Avionics Cooling Fan – General data

1. General

To provide cool air circulation in the forward avionics compartment, located in the nose between stations 37.21 to 44.00. Cool air is drawn into the compartment through an intake and circulated around the compartment by two electronic fans installed on the top avionics shelf.

2. Description

Two Sandia SAFE 128 avionics cooling fans are installed in the forward avionics compartment to provide air circulation for the avionics. The fans include a fault detection output which provides a ground when the fans are operating above a preset RPM threshold. Two relays, located inside the junction box, are triggered by the ground signal from each fan output. If either relay opens, a ground is provided to the MAU, which produces an Avionics Fan Fail CAS.

The avionics cooling fans are mounted on the upper avionics shelf around station 54, one on the pilot's side, another on the co-pilot's side.

28 Volt DC power is provided from the left bus through a 1 amp AVIONICS FAN (E7) circuit breaker on avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

3. Adjustment/Test

A. Avionics Cooling Fan

The unit may be tested by manually slowing the fans, one at a time, by applying a light pressure to the center hub of the fan until the Avionics Fan Fail CAS appears. Confirm the CAS clears when the fan returns to full speed.

4. Removal/Installation

A. Removal

1) Ensure the aircraft electrical system is OFF.

2) Pull circuit breaker AVIONICS FAN (E7) (left bus) on the avionics circuit breaker panel and fit safety clip.

3) Gain access to the forward avionic shelves through the forward baggage compartment.

4) Remove top and middle covers attached to bulkhead station 37.21 and 44.00 respectively for full access.

5) Disconnect the electrical connections.

6) Remove four mounting screws.

7) Remove fan.

Note Stow connectors of removed unit.

B. Installation

1) Ensure the aircraft electrical system is OFF.

2) Install fan on the upper avionics shelf.

3) Set unit in position and install four mounting screws.

4) Connect the electrical connectors. Ensure connection is sound and secure.

5) Reinstall top and middle covers attached to bulkhead station 37.21 and 44.00 respectively. Ensure covers are mounted correctly and secure.

6) Remove safety clips and reset circuit breaker AVIONICS FAN (E7) (Left Bus) on the avionics circuit breaker panel.

7) Close forward baggage compartment door.

5. Troubleshooting

A. Avionics Fan Fault

1) Ensure AVIONICS FAN (E7) circuit breaker is closed.

2) Visually inspect the connector for damaged wiring or contacts. Ensure electrical connector is fully seated.

3) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

4) If repair or service is required, send to appropriately rated avionics repair facility.

DAU Cooling Fan – General data

1. General

Air is circulated around the Data Acquisition Units by a three port cooling fan installed just aft of the DAUs.

2. Description

A Sandia ACF 328 cooling fan is installed in the ceiling to provide air circulation for the DAUs. Power to the fan is controlled by the thermal switch which is located on the DAU maintenance panel around station 244.00. The thermal switch is set to activate when the temperature reaches 110 degrees Fahrenheit (43 degrees Celsius).

The DAU cooling fan is mounted in the aircraft ceiling around station 270.00. Hoses run from the fan ports, one hose runs between the DAUs, the other two hoses run on either side of the DAUs.

28 Volt DC power is provided from the left bus through a 3 amp DAU COOLING FAN (U2) circuit breaker on station 332.00 circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

3. Adjustment/Test

A. DAU Cooling Fan Test

1) While monitoring temperature, apply heat to the thermal switch.

2) The switch should close at a temperature of 110 ± 8 degrees Fahrenheit (43 degrees Celsius). Confirm fan is operating.

3) While monitoring temperature, allow the thermal switch to cool.

4) The switch should open at a temperature of 100 ± 5 degrees Fahrenheit. Confirm fan is no longer running.

B. Thermal Switch Adjustment

1) Remove thermal switch if required.

2) Rotate set screw until the pointer reads 110 degrees Fahrenheit.

4. Removal and Installation

A. DAU Cooling Fan

1) Removal of DAU Cooling Fan

a) Ensure the aircraft electrical system is OFF.

b) Open the DAU COOLING FAN (U2) circuit breaker on the station 332.00 circuit breaker panel and fit safety clip.

c) Gain access to the ceiling compartment aft of bulkhead 262.

d) Disconnect electrical connector.

e) Loosen (3) hose clamps and remove hoses from fan ports.

f) Remove the (9) mounting screws.

g) Remove unit from the bracket.

Note Stow connector of removed components.

2) Installation of DAU Cooling Fan

a) Ensure the aircraft electrical system is OFF.

b) Open the DAU COOLING FAN (U2) circuit breaker on the station 332.00 circuit breaker panel and fit safety clip.

c) Set unit in position and install (9) mounting screws.

d) Install hoses to fan ports and tighten (3) hose clamps.

e) Reconnect electrical connector. Ensure connection is secure.

f) Install interior as required.

g) Remove safety clip and close DAU COOLING FAN (U2) circuit breaker on the station 332.00 circuit breaker panel.

B. Thermal Switch

1) Removal of Thermal Switch

a) Ensure the aircraft electrical system is OFF.

b) Open the DAU COOLING FAN (U2) circuit breaker on the station 332.00 circuit breaker panel and fit safety clip.

c) Gain access to the ceiling compartment aft of bulkhead 240.

d) Remove (2) electrical terminals. Protect terminals from short circuits to airframe or each other.

e) Remove (2) mounting screws.

f) Remove switch from panel.

2) Installation of Thermal Switch

a) Ensure the aircraft electrical system is OFF.

b) Ensure switch is set to the proper temperature, refer to Para 3.B.

c) Set switch in position and install (2) mounting screws.

d) Remove protection and install (2) electrical terminals. Ensure terminals are tight.

e) Install interior as required.

f) Remove safety clip and close DAU COOLING FAN (U2) circuit breaker on the station 332.00 circuit breaker panel.

5. Troubleshooting

A. DAU Cooling Fan

1) Ensure DAU COOLING FAN (U2) circuit breaker is closed.

2) Visually inspect the connector for damaged wiring or contacts. Ensure electrical connector is fully seated.

3) Ensure thermal switch is operating normally, refer to Para 4.B.

4) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

5) If repair or service is required, send to appropriately rated avionics repair facility.

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Heating System – General Data

1. General

The heating system utilizes heated and compressed bleed air from tappings on both engine compressors, mixes the bleed air with secondary air (ram, fan pressure, or recirculated air) and distributes it to outlets in the cabin and flight compartment.

2. Description

Refer to Figure 1 for details.

The heating system consists of an interconnected bleed pipe assembly incorporating two solenoid-operated shut-off valves and two check valves, hot air valve, expansion chamber, ejector, silencer, cabin air control valve, ducts and outlets. For system control refer to 21-60-00, Environmental Control System (Mod 6/2014) – General Data.

A. Bleed Air Pipe Assembly

The pipes from each engine compressor casing are routed to their respective shut-off valves, (located in the engine nacelles), and from the shut-off valves through the wings to join a tee pipe in the cabin roof; check valves are installed in each end of the tee pipe. From the tee pipe, a common pipeline routed forward, above the cabin ceiling and down the rear left-hand side of the cabin/flight compartment bulkhead, to connect to the hot air valve under the flight compartment floor. A “De Vore” protection is installed over the bleed air pipes from the shut-off valves through the wings to the cabin roof. Pipes are insulated with fiberglass and are sealed to exclude fluids.

B. Hot Air Valve

The hot air valve, connected between the bleed air pipe assembly and a pipe connected to the expansion chamber, is located below the pilot's seat under the flight compartment floor. The valve is of the motorized butterfly type which takes approximately 30 seconds to move through its full 90 degree range of travel, thus providing control sensitivity.

C. Expansion Chamber

The expansion chamber, which reduces bleed air noise, is a duct into which protrudes a pipe connected to the inlet end of the duct; the pipe from the hot air valve connects to this end. The outlet end of the expansion chamber incorporates a group of nozzles which protrude into the ejector and a flange which bolts to the ejector.

D. Ejector

The ejector, located laterally across the fuselage below the flight compartment floor, forms the mixing chamber for recirculated air. The nozzles, protruding into the ejector and the ejector diffuser, induce a suction which draws in recirculated air from the flight compartment when ram or fan pressure air is not being used as the secondary air source. The recirculated air duct, connected between an intake in the flight compartment footwell and the rear of the ejector, incorporates an integral silencer and check valve.

E. Silencer

The silencer is located longitudinally below the flight compartment, its bifurcated inlet connects to the ram air fan duct and ejector outlet. The silencer outlet, which incorporates the cabin air control valve, connects to the distribution ducting. Two smaller diameter outlets on the silencer provide for the connection of flight compartment, windshield and foot warmer ducts. Provision is also made on the silencer for the installation of temperature control system components.

F. Cabin Air Control Valve

The cabin air control valve is located in the silencer outlet duct and controls the supply of warm (with hot air valve and engine bleed valves closed) or the cooling air engine supply to the cabin. The valve is manually operated by a handle located in the cockpit central pedestal. Closing the air control (by pulling the handle fully out), allows all available flow to be directed into the flight compartment.

G. Ducts

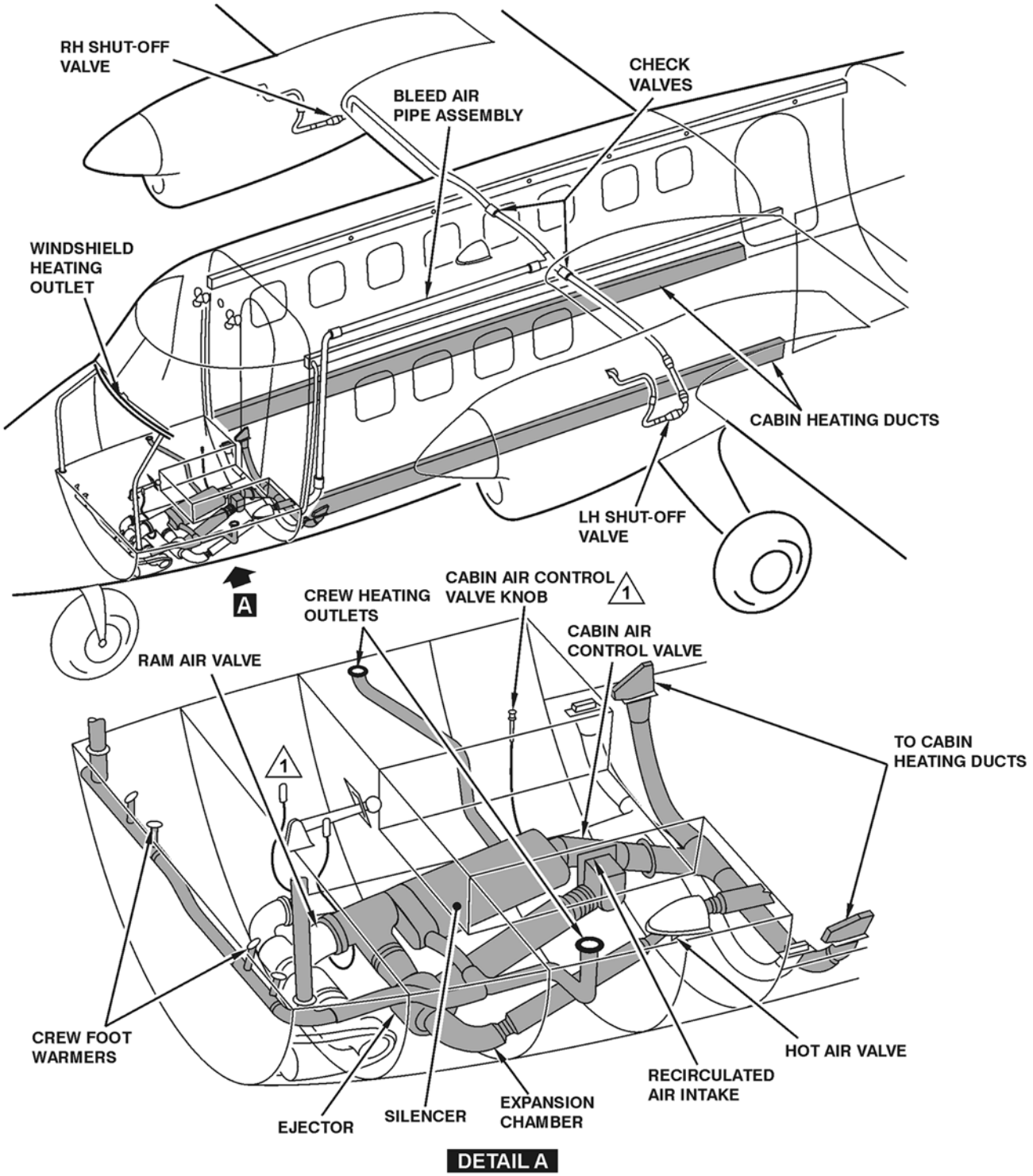
The heating system ducts comprise an installation of aluminum alloy and polycarbonate tubes which lead from the silencer to the pilot's and co-pilot's foot warmers, the windshield heater and cabin heater outlets.

H. Outlets

The heating system outlets connect to the terminating ends of the windshield, crew and cabin heating system ducts. The outlets are of polycarbonate or nylon material and formed to provide a suitably diffused air flow. The windshield outlets are integrated with the glare shield above the instrument panel, and the crew foot warmer outlets are at floor level and direct their flow towards the rudder pedals. The cabin outlet ducts are box section ducts which extend from the flight compartment/cabin bulkhead rearwards at the base of each wall.

I. Cabin Temperature Indication

The cabin air temperature will be displayed through CAS on the environmental panel (refer to 21-60-00, Environmental Control System (Mod 6/2014) – General Data).



Heating System Installation
Figure 1

3. Operation

With the engines running, bleed air from the engine compressors is controlled by the solenoid-operated shut-off valves which are energized by selecting switches on the co-pilot panel. The switches are identified BLEED AIR, LEFT and RIGHT, and since they are normally ON bleed air from the shut-off valves passes through the check valves, which prevent reverse flow in the event of unequal engine compressor casing pressures, and through the common supply pipeline to the hot air valve. The hot air valve, receiving signals from the temperature control system, opens or closes to regulate the amount of bleed air into the ejector. A flow limiting orifice, which limits the bleed air flow to 4 percent of the two-engine bleed flow-rate, is located immediately upstream of the hot air valve. Bleed air noise is reduced by the expansion chamber just prior to entering the ejector.

Depending on temperature selection, position of ram air valve handle, and if ram air or fan pressure air is available, the ejector will either draw in recirculated air for mixing with the bleed air, or eject hot air to mix with ram or fan air before entering the silencer. With the cabin air control valve open, the main flow of warm air will pass through the cabin ducts to the cabin outlets, where it is distributed through a pattern of round perforations in the inward facing side of each outlet duct. A limited supply of warm air is provided to the flight compartment, foot warmer and windshield heating outlets. If, however, the cabin air control valve is closed, all available warm air will be distributed to the latter and the supply to the cabin outlets is shut-off.

4. Adjustment/Test

A. Test Heating System Bleed Air Pipe Installation

Equipment required:

- Source of compressed air 100 psi
- Pressure test rig consisting of a pressure tank capable of withstanding 160 psi and having a volume of 10 – 12 cu. ft., a 0 – 100 psi pressure gauge, and two shut-off valves to isolate the rig from the aircraft and the compressed air source.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Set appropriate circuit breakers for system under test.

3) If a heat exchanger package is fitted, remove cabin roof upholstery in location of station 177.00 to provide access to bleed air system pipeline. Disconnect heat exchanger pipe from bleed pipe and blank off bleed pipe tapping.

4) Disconnect and blank off bleed pipe at first or second coupling downstream from engine (between take-off point and shut-off valve) in left nacelle, using a Marman style blanking disc.

WARNING BEFORE ATTEMPTING TO DISCONNECT PIPING, RELEASE ALL AIR PRESSURE FROM THE SYSTEM.

Note A suitable blanking disc 1.710 inches diameter x 0.065 ± 0.015 inch thick, may be manufactured from aluminum alloy.

5) Disconnect the left intake deflector pipe and connect test rig pressure line to bleed air pipeline.

6) Select LEFT, BLEED AIR switch ON and HEATING MODE switch to OFF. Apply a pressure of 40 psi and check all connections for obvious leaks. Rectify leaks as necessary.

Note Air will be leaking from bleed shut-off valve vents, this is normal and acceptable. If leak rate is high it may be necessary to replace valve to meet the requirements of Para 4.A.7).

7) Build up pressure to 80 psi and close shut-off valve on test rig to isolate rig from compressed air source; record time taken for system pressure to fall to 70 psi. If time taken is:

a) 2 minutes or more, system is good.

b) 30 seconds to 2 minutes, system is acceptable but at the earliest opportunity leaks must be rectified.

c) 30 seconds or less, system is unserviceable and leaks must be rectified immediately.

8) Select LEFT, BLEED AIR switch OFF, open hot air valve by selecting warm, and HEATING MODE switch to ON. Pressurize system to 70 psi.

9) Select LEFT, BLEED AIR switch ON and note that test rig pressure drops rapidly.

10) Disconnect test rig, select HEATING MODE switch to OFF and ensure LEFT, BLEED AIR switch is at ON.

11) Reconnect left intake deflector pipe to bleed air pipe. Remove blank and reconnect bleed air pipe coupling (between take-off point and shut-off valve).

12) Repeat Para 4.A.4) through Para 4.A.11) for right-hand side, using RIGHT, instead of LEFT, BLEED AIR switch.

13) If a heat exchanger package is fitted, remove blank and connect heat exchanger pipe to bleed pipeline tapping. Replace cabin roof upholstery.

14) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Bleed Air Shut-off Valve – Maintenance Practices

1. Removal/Installation

   A. Remove Bleed Air Shut-off Valve

      1) Gain access to appropriate engine nacelle.

      2) Disconnect electrical connections from shut-off valve.

      3) Remove valve and pipe insulation as necessary.

      4) Remove the two clamps securing shut-off valve to bleed pipes and remove valve.

   B. Install Bleed Air Shut-off Valve

      1) Examine replacement shut-off valve for signs of damage.

      2) Install valve (solenoid end of valve towards engine) using the two clamps and a new metal O-ring. Tighten clamps and torque to 35 to 45 pound–inches.

      3) Install pipe and valve insulation and wirelock.

      4) Connect electrical connections to valve.

      5) Test bleed air system for leaks. (Refer to 21-40-00, Heating System – General Data).

      6) Complete temperature control system function test. (Refer to 21-60-00, Environmental Control System – General Data).

2. Adjustment/Test

   A. Operational Test – Solenoid

      Note To pressure test the operation of the bleed air shut-off valve, refer to 21-40-00, Heating System – General Data.

      1) Ensure applicable BLEED AIR (L or R) circuit breaker is in.

      2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      3) Select BLEED AIR switch to LEFT or RIGHT as applicable and listen for sound (audible click) of bleed air shut-off valve solenoid operating.

      4) Select BLEED AIR switch to OFF and note similar sound of valve solenoid operating in other direction.

      5) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Ejector – Maintenance Practices

1. Removal/Installation

   A. Remove Ejector

      1) Gain access to compartment, under flight compartment floor, on the left-hand side of fuselage. (Refer to 12-00-00, Servicing – General Data).

      2) Slacken and remove recirculated air duct clamps and sleeves. Remove recirculated air duct.

      3) Slacken and remove ejector diffuser outlet clamp and sleeve.

      4) Slacken and remove clamps and sleeve from hot air valve outlet, and remove pipe between hot air valve and expansion chamber inlet.

      5) Remove bolts securing ejector brackets to aircraft structure.

      6) Remove ejector, complete with expansion chamber, from aircraft.

      7) Remove expansion chamber from ejector.

   B. Install Ejector

      1) Secure expansion chamber to ejector using a new insulation gasket, bolts and washers.

         Note Ensure expansion chamber is located correctly in relation to ejector.

      2) Position ejector, complete with expansion chamber, in aircraft.

      3) Install bolts to secure ejector brackets to aircraft structure.

      4) Install pipe between expansion chamber inlet and hot air valve outlet. Install hot air valve outlet sleeve and clamp, torque clamp to 35 to 45 pound–inches.

      5) Install ejector diffuser outlet to silencer duct, sleeve and clamp. Tighten clamp.

      6) Install recirculated air duct using sleeves and clamps. Tighten clamps.

      7) At subsequent engine ground run check ducts and pipe joints for air leaks.

      8) Secure access panel. (Refer to 12-00-00, Servicing – General Data).

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Hot Air Valve – Maintenance Practices

1. Removal/Installation

   A. Remove Hot Air Valve

      1) Remove hot air valve cover under pilot's seat.

      2) Gain access to compartment under flight compartment floor, through access panel on left-hand side of fuselage. (Refer to 12-00-00, Servicing – General Data).

      3) Remove cover under pilot's seat.

      4) Disconnect electrical wiring from hot air valve.

      5) Remove hot air valve inlet and outlet clamps and sleeves.

      6) Remove hot air valve from aircraft.

   B. Install Hot Air Valve

      1) Examine replacement hot air valve for signs of damage.

      2) Position valve in aircraft, and secure with sleeves and clamps. Torque clamps to 35 to 45 pound–inches.

      3) Connect electrical wiring to hot air valve.

      4) Test environmental control system controls, and function test environmental control system during engine ground run (refer to 21-60-00, Environmental Control System (Mod 6/2014) – General Data). Check disturbed clamp joints for air leaks during initial stage of ground run.

      5) Install cover under pilot's seat and secure nose fuselage access panel. (Refer to 12-00-00, Servicing – General Data).

2. Adjustment/Test

   A. Operational Test

      Note To test the operation of the hot air valve, refer to 21-60-00, Environmental Control System (Mod 6/2014) – General Data, Para 4.A.

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Cabin Air Control Valve – Maintenance Practices

1. Adjustment/Test

A. Function Test Cabin Air Control Valve

1) Open right-hand access panel on underside of nose section. (Refer to 12-00-00, Servicing – General Data).

2) Close cabin cool air gaspers.

3) Pull RAM AIR valve control handle to OPEN.

4) Push CABIN AIR valve control handle to the valve open position. Check that cabin air control valve shaft handle contacts left-hand flange of stop plate, and that a distance of 3.5 inches exists between center of control cable fork end attachment pin and end of cable sleeve.

5) Pull CABIN AIR valve control handle to CLOSE. Check that cabin air control valve shaft handle moves through a range of approximately 90°, and that control cable fork end butts against end of cable sleeve.

6) Operate main duct fan. (Refer to 21-50-11, Main Duct Fan – Maintenance Practices).

7) Check that no air flows through cabin baseboard heater outlets.

8) Push CABIN AIR valve control handle to the valve open position. Check that air flows through cabin baseboard heater outlets.

9) Switch off main duct fan.

10) Close right-hand access panel. (Refer to 12-00-00, Servicing – General Data).

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Cooling System – General Data

1. General

The cooling system is essentially the means of supplying ram air to the heating system, and to the cabin gaspers installed along both sides of the cabin at each seat position. Additionally, the cooling system supplies air to the gaspers in the cockpit.

2. Description

Refer to Figure 1 for details.

Components of the cooling system are: ram air scoop, main duct, main duct fan, fan duct, ram air valve, ducts and gaspers.

A. Ram Air Scoop and Main Duct

The ram air scoop is a protruding intake on the left-hand side of the fuselage nose which provides the total supply of ram air. The polycarbonate main duct connects the ram air scoop to the main duct fan.

Note For aircraft incorporating Mod 6/2387, the air scoop has rubber washers installed. Torque the screws to compress the rubber washers (which are underneath the metal washers) to approximately one-half of their original thickness. For air scoop without rubber washers (Pre Mod 6/2387) torque the #10 screws to 10 – 16 in–lbs.

B. Main Duct Fan

An electrically operated fan is installed between the main duct and fan duct, below the flight compartment floor. The fan electrical circuit (refer to 21-60-00, Environmental Control System (Mod 6/2014) – General Data) is protected by a 20–amp VENT FAN circuit breaker on the overhead circuit breaker panel, and the fan is controlled by a switch mounted on the flight compartment pedestal. Access to the fan is through a side panel, below the flight compartment left door.

C. Fan Duct

The Y–shaped fan duct connects to the main duct fan, heating system silencer inlet, and the cabin air duct. A ram air valve is installed in the heating system silencer branch of the duct.

D. Ram Air Valve

The butterfly type ram air valve, located in the fan duct, is manually operated by a control cable in the flight compartment, to regulate ram air flow to suit heating system requirements.

E. Cooling System Ducts

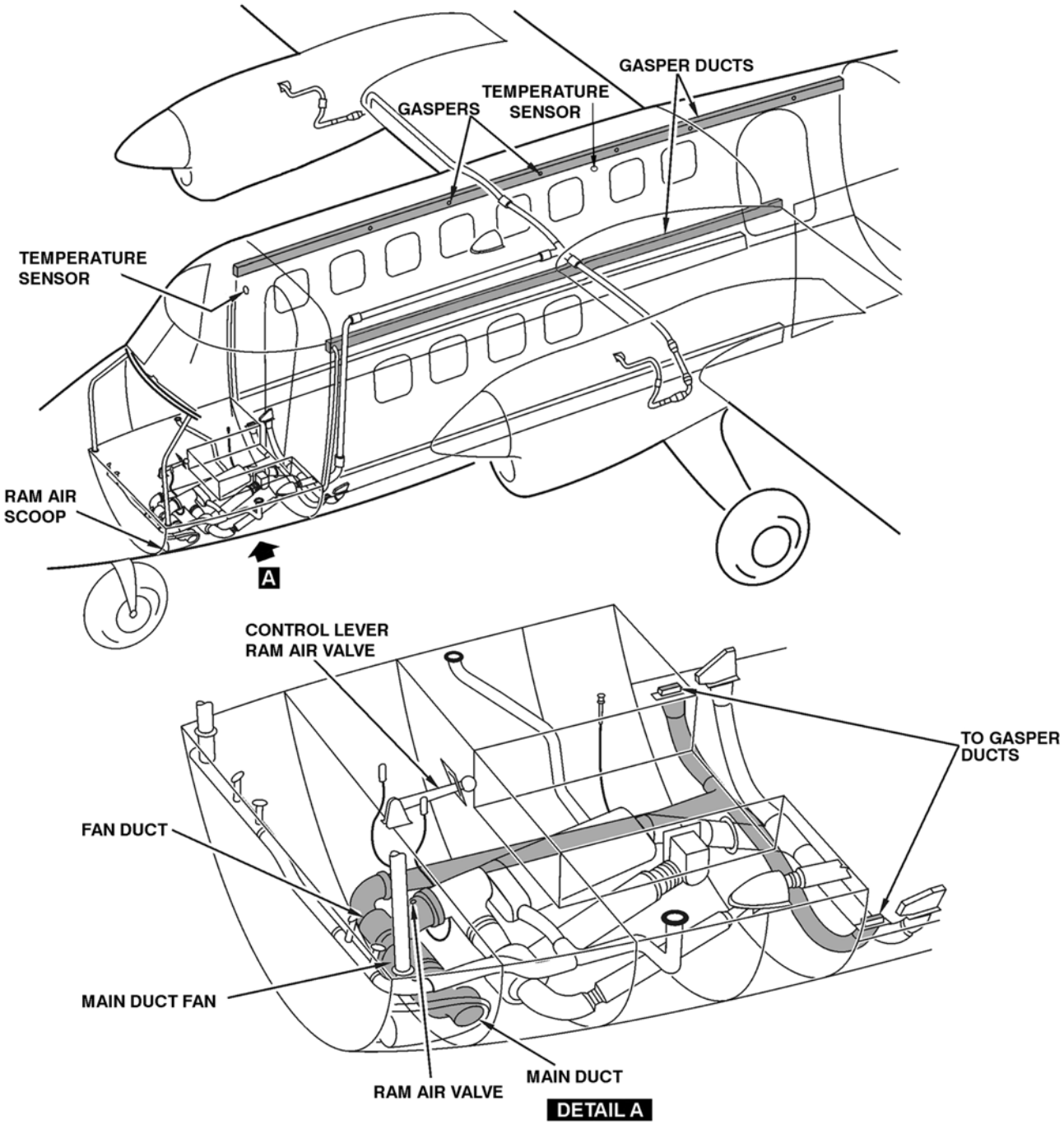
The cooling system ducting connects to the fan duct and is routed under the flight compartment floor. The cooling system ducting provides cooling for the cockpit gaspers, to the avionics bay forward of the cockpit instrument panel, and up both sides of the cabin behind the flight compartment/cabin bulkhead to join box section ducts running along both sides of the cabin above the cabin windows. The box section ducts house the cabin gaspers.

F. Cooling System Outlets

The cooling system outlets, or gaspers, are installed in the cockpit instrument panel as well as each seat position on the box section ducts.

3. Operation

In flight, ram air is ducted from the scoop on the left-hand side of the fuselage nose, through the fan, into the fan duct. The fan duct splits to supply ram air to the heating system and the cooling system. When the aircraft is stationary, the main duct fan may be switched on to draw in outside air to supply cooling air.



Cooling System Installation
Figure 1

Main Duct Fan – Maintenance Practices

1. Removal/installation

   A. Remove Main Duct Fan

      1) Remove ejector (refer to 21-40-21, Ejector – Maintenance Practices).

      2) Disconnect fan electrical wiring.

      3) Slacken and remove sleeves and clamps securing ram air fan duct to fan, silencer duct and ventilation duct.

      4) Remove bolts, washers and nuts securing ram air fan duct bracket to flight compartment floor structure.

      5) Move ram air fan duct sufficiently to allow fan to be withdrawn from ram air intake duct. Disconnect ram air valve control at valve handle, if necessary.

      6) Remove bolts and washers securing fan to support brackets.

      7) Remove fan from aircraft.

   B. Install Main Duct Fan

      1) Examine ram air duct to fan seal, change if damaged.

         Note If seal requires changing, bond seal to duct sleeve using Dow Corning A4000 (refer to 20-50-02, Sealant and Adhesives – General Data.

      2) Examine replacement main duct fan for signs of damage.

      3) Position main duct fan in aircraft, ensure inlet end of fan enters ram air duct without damage to seal.

      4) Install bolts and washers to secure fan to support brackets.

      5) Install ram air fan duct and secure to fan, silencer duct and ventilation duct, using sleeves and clamps. Tighten clamps.

      6) Reconnect and adjust ram air valve control, if removed.

      7) Connect fan electrical wiring.

      8) Install ejector (refer to 21-40-21, Ejector – Maintenance Practices).

2. Adjustment/Test

   A. Operational Test

      1) Ensure VENT FAN circuit breaker is in.

      2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      3) Pull RAM AIR valve control handle to OPEN.

      4) Select VENT FAN switch to ON.

      5) Position hand over windshield heater outlets and check for airflow.

      6) Select VENT FAN switch to OFF.

      7) Push RAM AIR valve control handle to CLOSED.

8) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3. Inspection/Check

A. Check Brushes for Wear

1) Remove main duct fan (refer to Para 1.A).

2) Lift off caps from brush and spring assemblies.

3) Lift out brush and spring assemblies.

Note Mark each assembly and corresponding holder to ensure exact original positioning if brush and spring assemblies do not require replacement.

4) Inspect brush and spring assemblies for brush wear, cracks and chipping.

Note The brushes are to be replaced if cracked, chipped, or when wear exceeds 20% of useful brush length. (Useful brush length (100%) 7/16 inch).

5) Install brush and spring assemblies.

6) Fit caps over installed assemblies.

7) If new brushes are installed, carry out Para 3.A.8) and Para 3.A.9), otherwise omit.

8) Apply 8 to 14 volts to motor terminals and operate fan until brushes are 75 to 90% seated.

WARNING BEFORE ENERGIZING BLOWER OR MOTOR, ASSEMBLY MUST BE SECURED AND GUARD PROVIDED AROUND BLOWER END TO PROTECT PERSONNEL.

9) Increase voltage to 27 volts and operate fan for a further 20 minutes.

10) Install main duct fan (refer to Para 1.B).

Ram Air Control Valve – Maintenance Practices

1. Adjustment/Test

   A. Function Test Ram Air Control Valve

      1) Open right-hand access panel on underside of nose section. (Refer to 12-00-00, Servicing – General Data).

      2) Close cabin cool air gaspers.

      3) Pull CABIN AIR valve control handle to CLOSED.

      4) Push RAM AIR valve control handle to CLOSED. Visually check that ram air valve shaft lever contacts left-hand flange of stop plate.

      5) Pull RAM AIR valve control handle to OPEN. Visually check that ram air valve shaft lever has moved through a range of approximately 90° and contacts right-hand flange of stop plate.

      6) Operate main duct fan. (Refer to 21-50-11, Main Duct Fan – Maintenance Practices).

      7) Check that air flows through pilot's and co-pilot's foot warmer, and windshield heater outlets.

      8) Push RAM AIR valve control handle to CLOSED.

      9) Check that air ceases to flow through flight compartment outlets.

      10) Switch off main duct fan.

      11) Close right-hand access panel. (Refer to 12-00-00, Servicing – General Data).

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Cooling/Demisting Fans (S.O.O. 6322) – General Data

1. General

One or two electrically-driven fans are installed in the flight compartment on the canopy at station 106 (approximately), and may be used to direct air onto the pilot's and co-pilot's windshields, for demisting purposes or for additional cooling.

The fans are 12 Volt DC; each connected to the 28 Volt DC right bus by a 25 ohm series connected resistor.

Each fan is attached to the aircraft canopy by two screws, and is controlled by a switch labeled FLT COMP FANS, located on the upper forward side of the station 111 sloping bulkhead, to the LH side of the bulkhead opening.

28 Volt DC power is provided from the right bus through a 5 amp FLT COMP FANS (B1) circuit breaker on the main circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

2. Adjustment/Test

A. Operational Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Select FLT COMP FANS switch to ON.

3) Observe operation of pilot's and co-pilot's fans.

Note Fan blades and motors should be smooth running without irregular noise, rattling, or vibration.

4) Select FLT COMP FANS switch to OFF.

5) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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STA 111 Bulkhead Cabin Fans (S.O.O. 6378) – General Data

1. General

Provisions have been made for the installation of two electrically-driven fans in the cabin, on the aft face of the STA 111 bulkhead. The fans can be used to direct air around the passenger cabin for cooling purposes. The fans are located on each side of the cockpit door. A formed sheet metal stand-off assembly is riveted to the bulkhead. Each fan is attached to the stand-off assembly with three (3) machine screws. A Lone Star 122253–10A EMI filter is installed in the circuit. The filter is mounted on the pulley bracket box, installed on the forward side of the STA 111 bulkhead on the left side just above and behind the Pilot.

The fans are operated by a CABIN FANS STA 111 switch located in the flight compartment, above and behind the Pilot on the STA 111 bulkhead adjacent the EMI filter. Each fan has an integral OFF–LOW SPEED–HIGH SPEED switch within the base of the fan assembly, to allow for passenger adjustment.

28 Volt DC power is provided from the right bus through a 15 amp CABIN FANS (B14) circuit breaker on the main circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

2. Adjustment/Test

A. Operational Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Select CABIN FANS STA 111 switch to ON.

3) Move the integral switch located at the base of each fan through the OFF, LOW SPEED and HIGH SPEED positions, observing the operation at each setting.

Note The fan blades and motors should be smooth running without irregular noise, rattling, or vibration.

4) Select CABIN FANS STA 111 switch to OFF.

5) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3. Removal/Installation

A. Remove STA 111 Bulkhead Cabin Fans

1) Ensure aircraft power is OFF. (Refer to 24-00-00, External Power – General Data).

2) Pull the CABIN FANS (B14) circuit breaker from the main circuit breaker panel and fit safety clip.

3) Remove three (3) mounting screws and washers on the base of each fan.

4) Before disconnecting the electrical leads from the terminal on the fan assembly, discharge the EMI filter through a +A terminal either by:

a) Connecting the +A terminal to an aircraft ground through a 150 ohm, 7.5 watt resistor for at least 30 seconds; or,

b) Connecting a wire to an aircraft ground and discharging the EMI filter by touching the other end of the wire to the +A terminal.

Note Do not re-engage circuit breaker until leads are reconnected to fan assembly terminals.

Note If leads need to be subsequently removed for further maintenance, repeat discharge procedures in Para 3.A.4) prior to lead disconnection.

5) Carefully pull the fan from the stand-off assembly, and disconnect the electrical wiring.

6) Remove the fan assembly from the aircraft.

B. Install STA 111 Bulkhead Cabin Fans

1) Ensure aircraft power is OFF. (Refer to 24-00-00, External Power – General Data).

2) Ensure CABIN FANS (B14) circuit breaker is pulled and fitted with safety clip.

3) Examine the fan assembly for damage, replace if necessary.

4) Place the fan assembly near the stand-off assembly and connect the electrical wiring.

Note If leads need to be subsequently removed for further maintenance, repeat discharge procedures in Para 3.A.4), prior to lead disconnection.

5) Install three (3) mounting screws and washers. Ensure the fan assembly is secure on the stand-off assembly.

6) Remove the safety clip and close CABIN FANS (B14) circuit breaker on the main circuit breaker panel.

7) Do an operational test as per Para 2.A.

4. Servicing

A. Fan Failure

Note The fan is not repairable. If a fan is unserviceable, it must be replaced.

1) Ensure aircraft power is OFF. (Refer to 24-00-00, External Power – General Data).

2) Remove the three (3) screws fastening the fan base to the stand-off assembly.

3) Pull the wiring out and disconnect the wiring splices.

4) Replace the fan assembly with a new part.

5) Connect the new fan to the wiring splices and re-install screws.

5. Troubleshooting

A. Fan Fault

1) Ensure CABIN FANS (B14) circuit breaker is closed.

2) Ensure the integral switch at the bottom of each fan is in the LOW or HIGH position.

3) Ensure the CABIN FANS STA 111 switch is ON.

4) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

STA 332 Blower (S.O.O. 6379) – General Data

1. General

Provisions have been made for the installation of a single electrically-driven centrifugal (squirrel-cage) style blower, on the STA 332 bulkhead. The blower directs air forward into the passenger compartment, to aid in air circulation.

The blower is mounted to a formed sheet metal panel that is fastened by screws to the aft face of the STA 332 bulkhead. On aircraft with a closed STA 332 bullhead, air is moved through a sheet metal duct and exits through a screen. On aircraft with an open STA 332 bulkhead, the screen is fastened directly to the filler panel and no duct is required. The blower has two speeds: high speed provides 28 volt DC directly to the blower motor, and low speed is provided through a 3 ohm resistor located on the blower housing.

The blower fan is operated by a CABIN BLOWER STA 332 switch located in the flight compartment, above and behind the Pilot on the STA 111 bulkhead.

28 Volt DC power is provided from the right bus through a 15 amp CABIN FANS (B14) circuit breaker on the main circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

2. Adjustment/Test

A. Operational Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Select CABIN BLOWER STA 332 switch to HIGH and LOW, observing the operation at each setting.

Note The fan motor should be smooth running without irregular noise, rattling, or vibration.

3) Select CABIN BLOWER STA 332 switch to OFF.

4) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3. Removal/Installation

A. Remove STA 332 Blower

1) Ensure aircraft power is OFF. (Refer to 24-00-00, External Power – General Data).

2) Pull the CABIN FANS (B14) circuit breaker from main circuit breaker panel and fit safety clip.

3) Remove the outlet screen mounting screws from the forward side of STA 332 bulkhead.

4) If installed, remove the duct assembly by removing the four (4) screws attaching the duct assembly to the blower assembly and filler panel assembly.

5) Remove two (2) screws securing blower assembly to filler assembly (upper holes).

6) Carefully pull the blower from the mounting assembly, and disconnect the blower electrical wiring.

7) Remove the blower assembly from the aircraft.

B. Install STA 332 Blower

1) Ensure aircraft power is OFF. (Refer to 24-00-00, External Power – General Data).

2) Ensure CABIN FANS (B14) circuit breaker is pulled and fitted with safety clip.

3) Examine the blower assembly for damage, replace if necessary. Ensure the outlet screen is clean of debris.

4) Place the blower assembly near the mounting bracket and connect the electrical wiring.

5) Install two (2) screws in upper holes of filler panel to secure blower assembly.

6) If equipped, install the duct assembly by installing the four (4) screws attaching the duct assembly to the blower assembly and filler panel assembly.

7) Install the mounting screws for the outlet screen at the forward side of STA 332 bulkhead.

8) Ensure the blower assembly is secure on mounting bracket.

9) Remove safety clip and close CABIN FANS (B14) circuit breaker on the main circuit breaker panel.

10) Do an operational test as per Para 2.A.

4. Servicing

A. Blower Failure

Note The blower fan is not repairable. If a blower fan is unserviceable, it must be replaced.

1) Ensure aircraft power is OFF. (Refer to 24-00-00, External Power – General Data).

2) Disconnect the wiring splices at the blower.

3) Remove the outlet screen.

4) If the aircraft has a closed STA 332 bulkhead, remove the duct.

5) Remove the screws fastening the blower fan assembly to the filler panel.

6) Replace the blower fan assembly with a new part.

7) Connect the new blower fan to the wiring splices and re-install screws, duct (if equipped) and outlet screen (refer to Para 3.B).

5. Troubleshooting

A. Fan Fault

1) Ensure CABIN FANS (B14) circuit breaker is closed.

2) Ensure the CABIN BLOWER STA 332 switch is in the LOW or HIGH position.

3) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

Environmental Control System (Mod 6/2014) – General Data

1. General

The temperature control system provides manual regulation of temperature inside the aircraft, by controlling the flow of engine compressor bleed air entering the heating system.

2. Description

Components of the temperature control system are: a manual control panel, outside air temperature sensor, duct temperature sensor, duct overheat switch and duct overheat caution message displayed through CAS.

A. Control Panel

The temperature control system control panel is located on the instrument and identified ENVIRONMENTAL. For heating there are two switches – a MODE switch with an upper ON and lower OFF position; and a TEMP switch which is a 3 position switch, spring-loaded to the center neutral position, with momentary WARM and COOL positions at the upper and lower points.

If the MODE switch is placed in the OFF position, the bleed air valve that supplies bleed air to the ejector assembly will motor to the closed position. If the MODE switch is placed in the ON position, the pilot can then adjust the position of the valve as desired by holding the TEMP switch in either the up (WARM) or down (COOL) position.

B. Outside Air Temperature Sensor

The outside air temperature sensor is a probe located in the cooling system left hand side of the aircraft near the forward baggage compartment door at station 347.00 to 376.00.

C. Cabin Temperature Sensor

The cabin temperature sensor is located in a plastic housing mounted on the mid cabin, approximately at station 271.90, in a multi purpose bracket that houses one of the cabin dome lights and PA speakers.

D. Duct Overheat Switch

The duct overheat switch is a probe located in the heating system silencer.

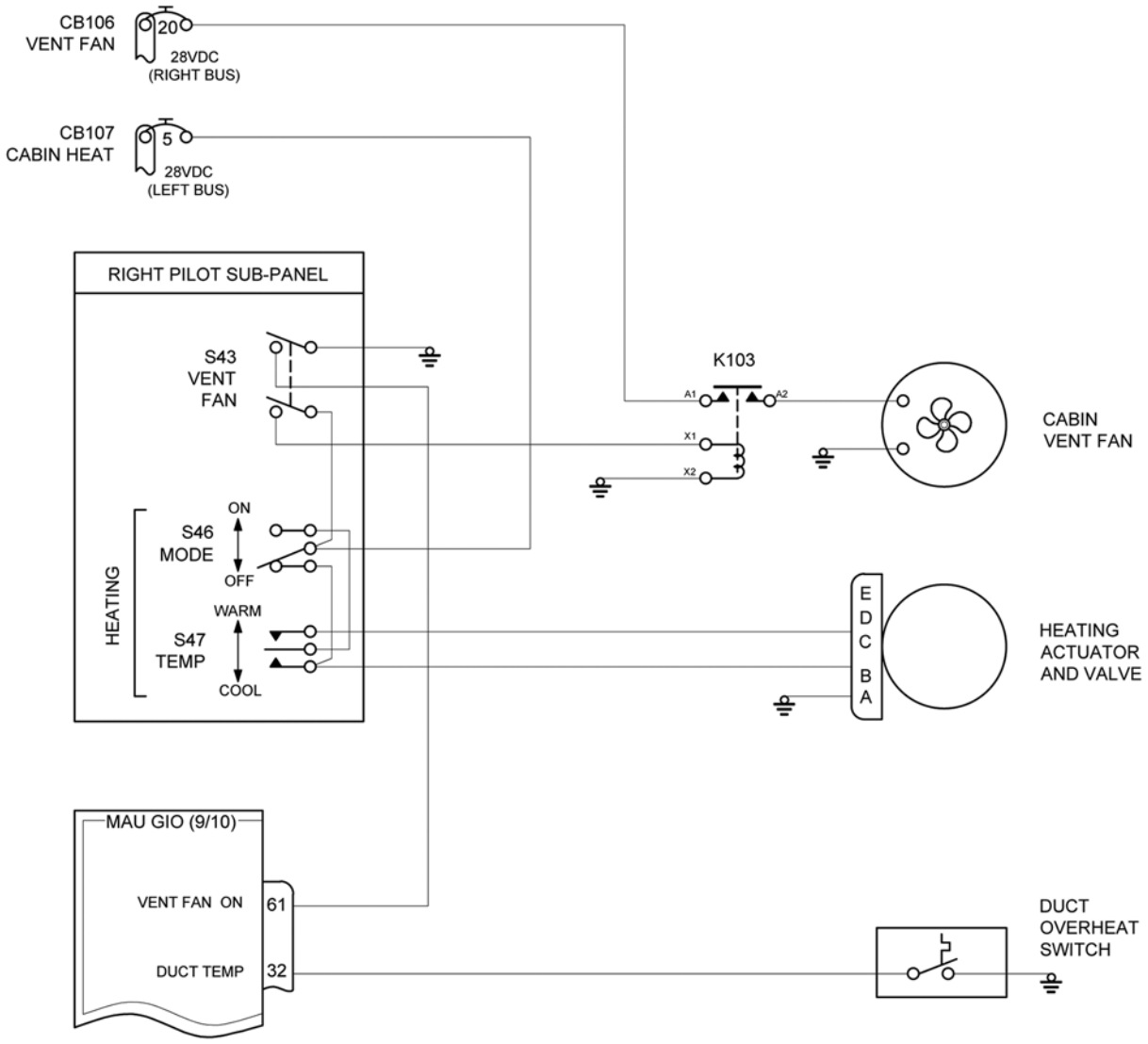
E. Duct Overheat Caution Message

The duct overheat caution message is displayed on the CAS message window and is identified Duct Overheat.

3. Operation

Refer to Figure 1 for details.

The hot air valve is driven to the desired position by the selection of the HEATING–MODE and HEATING–TEMP switches, which routes 28 Volt DC direct to the appropriate hot air valve actuator windings, in the COOL–WARM positions.



Temperature Control Systems – Electrical Schematic
Figure 1

4. Adjustment/Test

A. Test Environmental Control System Controls

Equipment required:

- Source of external electrical power.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) On the instrument panel, select the HEATING–MODE switch to ON and HEATING–TEMP switch to WARM.

3) Check audibly that the hot air valve actuator, located under the floor below the pilot's seat, runs for 30 ± 8 seconds.

4) Select the HEATING–TEMP switch to COOL, and check that hot air valve actuator again runs for 30 ± 8 seconds.

5) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Function Test Temperature Control System

1) With engines running (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data) set one engine to at least 0 percent power (if test is carried out in flight, set engines for level flight).

2) Set LEFT and RIGHT– BLEED AIR switches ON, and check that the cabin air control handle, located in the cockpit center pedestal, is fully pushed in (valve open position).

3) Confirm that temperature output varies appropriately when the HEATING–TEMP switch is selected WARM/COOL.

4) Select HEATING–TEMP switch and BLEED AIR switches to OFF. Stop engines (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

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Cabin Temperature Sensor – Maintenance Practices

1. Adjustment/Test

A. Operational Test – Cabin Temperature Sensor

Note Cabin temperature is not essential and the following procedure need only be done to enable adequate passenger comfort.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Observe cabin temperature as displayed on CAS as compared to an available thermometer. Alternatively open cabin doors and compare to OAT reading.

3) Replace sensor as required.

4) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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CHAPTER 22

AUTO FLIGHT

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CHAPTER 22 – AUTO FLIGHT

LIST OF EFFECTIVE PAGES

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Auto Flight – General Data

1. General

The Automatic Flight Control System (AFCS) is the integration of Mod 6/2303 Flight Director (FD) and S.O.O. 6221 Autopilot Systems. The FD displays pitch and bank angles required for the aircraft to follow a selected path. The FD can be coupled with an optional three-axis Autopilot. When integrated the AFCS controls the aileron, elevator, and rudder servos to maintain the desired aircraft attitude. The optional Autopilot (S.O.O. 6221) includes a yaw damper and electric pitch trim servo system.

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Automatic Flight Control System (AFCS) (Mod 6/2303 and S.O.O. 6221) – General Data

1. General

Refer to Figure 1 for component locations.

The Automatic Flight Control System (AFCS) is used to guide the flight path of the aircraft by generating Flight Director (FD) commands that can be coupled to an optional three-axis Autopilot (AP), if S.O.O. 6221 is incorporated. The optional three-axis autopilot controls the aileron, elevator, and rudder servos to maintain the aircraft in a desired attitude. When equipped with the AP, the aircraft also has electric pitch trim and a yaw damper system.

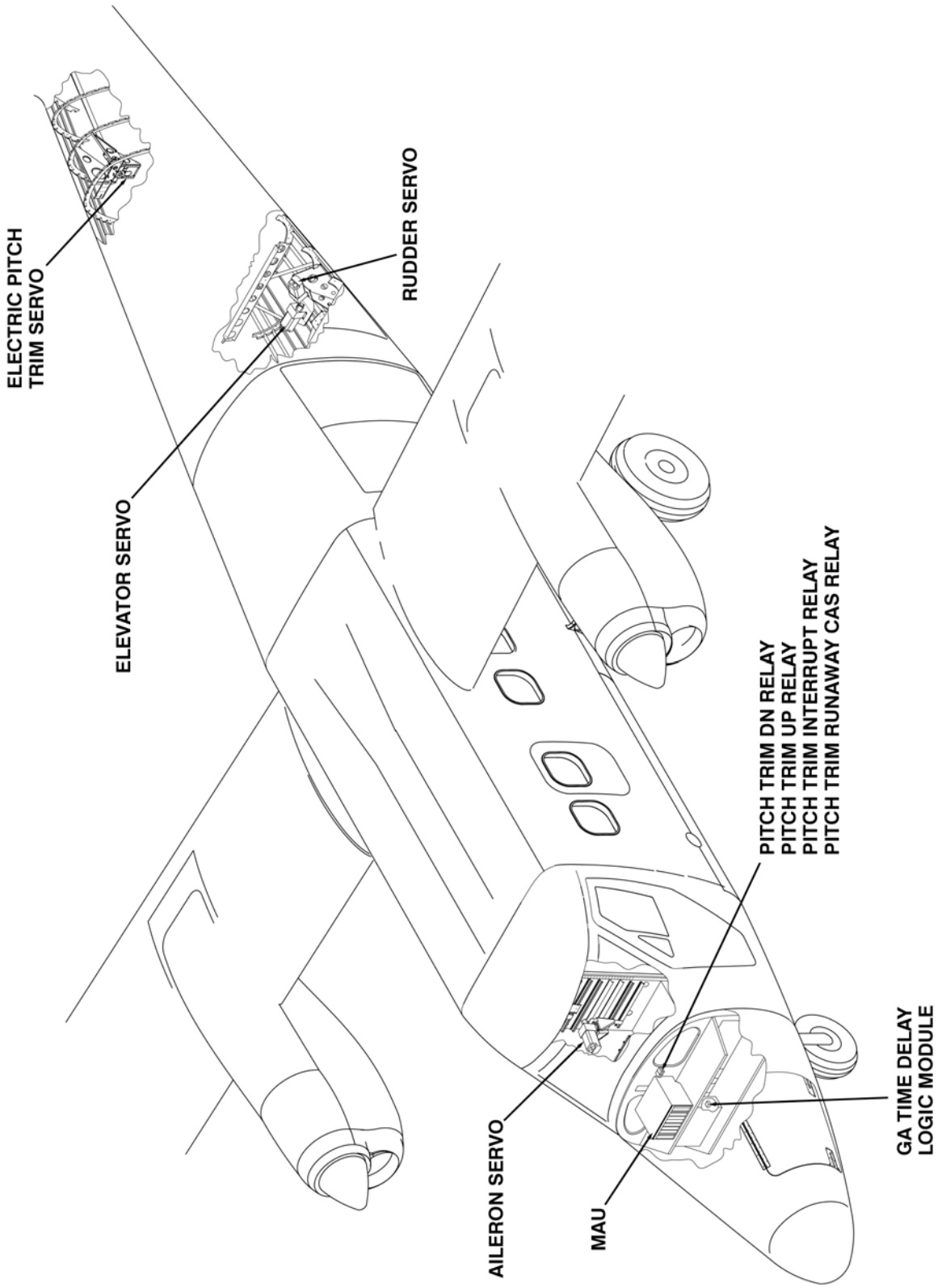
The AFCS consists of the following primary components:

- MAU AIOP Modules
- Flight Controller (FC)
- Control Wheel Switches
- Electric Pitch Trim Interrupt Relay
- Autopilot Servos (S.O.O. 6221)
- Electric Pitch Trim (S.O.O. 6221)
- Electric Pitch Trim Disable Switch (S.O.O. 6221)
- Electric Pitch Trim Up/Down Relays (S.O.O. 6221)
- GA Time Delay Logic Module (S.O.O. 6221)
- Pitch Trim Runaway CAS Relay (S.O.O. 6221)

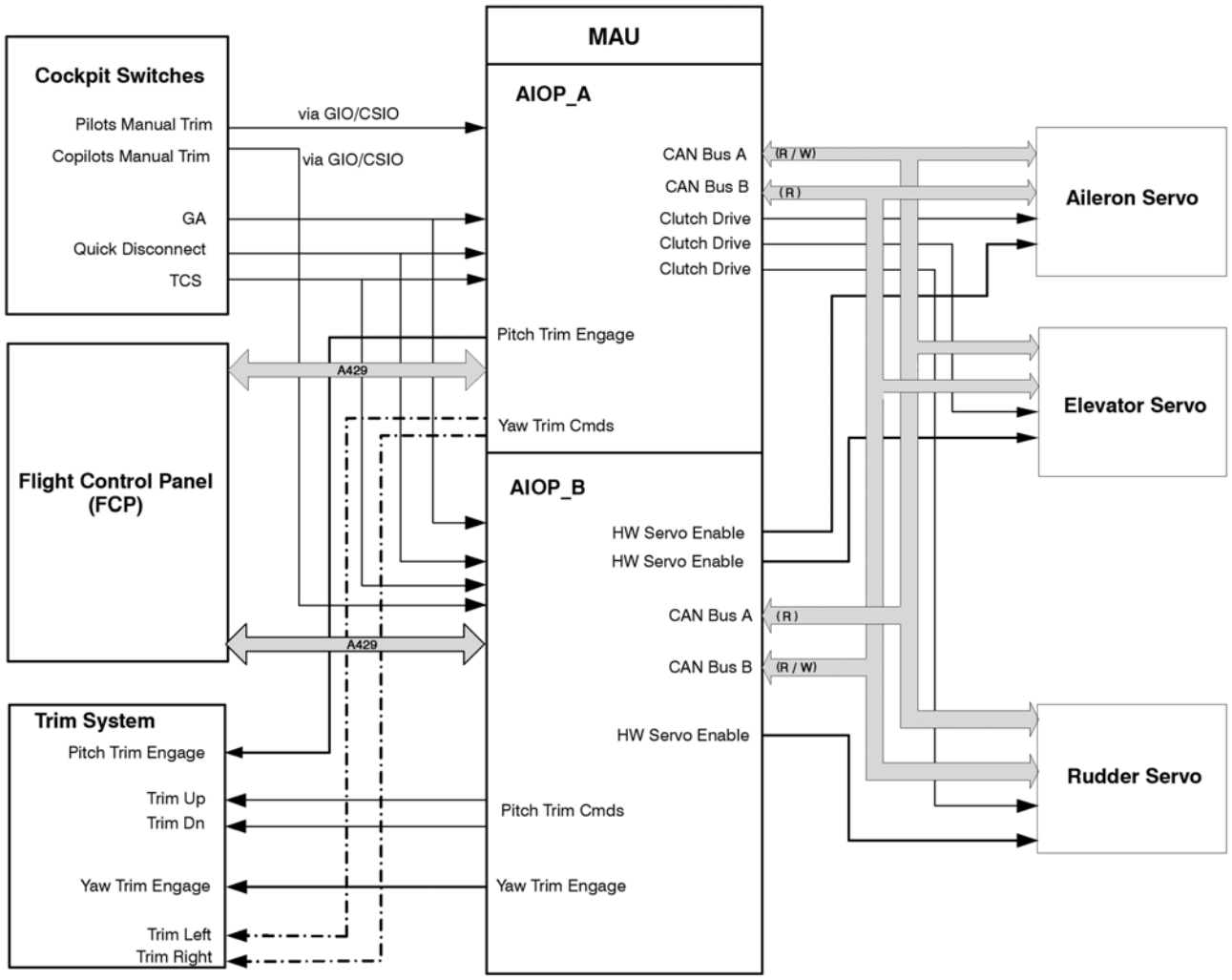
The AP only receives vertical and lateral steering commands from the FD. The FD commands are used to generate required control deflections that are actuated by aileron, elevator and rudder servos, and automatic actuation of the electric pitch trim system. The servos are connected to the MAU via dual Controller Area Network (CAN) data buses. Commands received by the servo are processed within the servo to check for agreement prior to servo action. Both servo channels must agree in their monitoring of motor current, clutch solenoid engagement and motor position. The aileron, elevator and rudder servos incorporate internal electric clutch mechanisms that release when commanded by the MAU or when electrical power is removed (CB open or power failure). The pitch trim servo incorporates a slip-clutch mechanism which requires scheduled maintenance.

The AP controls the primary servos and trim servo through the MAU as shown in Figure 2. The FD generates AP commands utilizing data from the dual-channel ADAHRS, two GPS receivers, the Flight Management System (FMS), two MMDRs, a single Flight Controller (FC), engine data provided to Apex through two Data Acquisition Units (DAUs) and the aircraft's stall warning system. Figure 3 details a functional block diagram of the Apex AFCS.

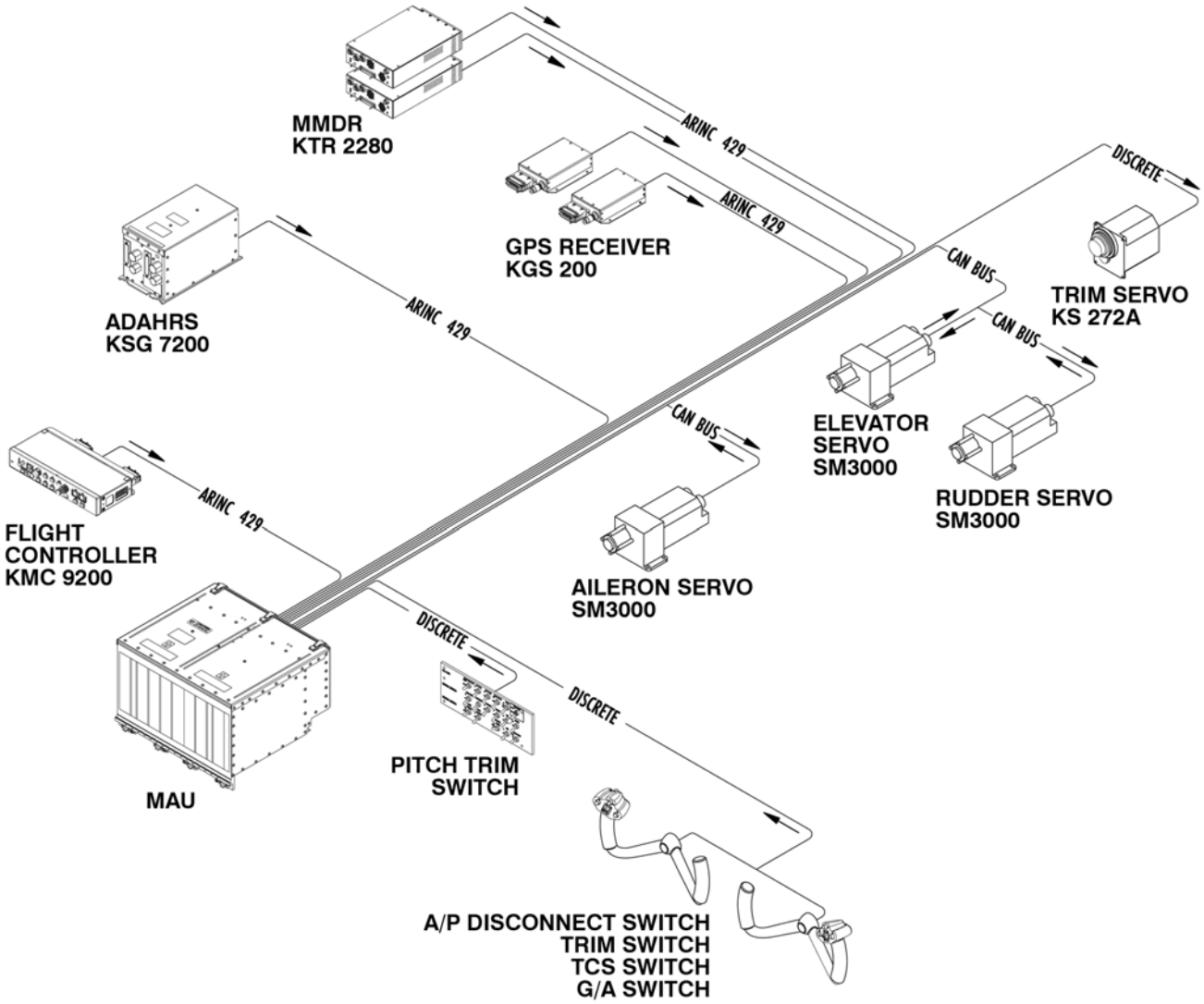
The AFCS architecture operates around two parallel channels (A and B) within the Apex Modular Avionics Unit (MAU), both channels are required to be functional for normal AFCS operation. An AP disengage occurs if a miscompare is detected between AP channel commands.



Location of Components
Figure 1



Apex AFCS Architecture
Figure 2



Apex Automatic Flight Control System – Block Diagram
Figure 3

2. Description

A. Flight Controller (FC) (KMC 9200)

Refer to Figure 4 for details.

The KMC 9200 Flight Controller (FC) is used to select and activate various functions, including mode selection of the Flight Director (FD). Incorporating S.O.O. 6221 will add Autopilot (AP) coupled to the FD steering commands. The FC provides selection for all AFCS functions except for the following functions:

- Go Around (GA) mode
- Touch Control Steering (TCS)
- Quick Disconnect (AP DISC)

The functions via the Flight Controller include:

- L/R – Selects which PFD (Left or Right) is used for FD coupling. At power-up, the default setting is Left
- FD, AP, YD – Engage or disengage the Flight Director (FD) and with S.O.O. 6221 Autopilot (AP)/Yaw Damper (YD)
- Nose DN/Nose UP – Adjust target values in modes: PIT, SPD, VS
- HDG/T, NAV, BL, VS, SPD, ALT, VNAV, APR – Engage or disengage each associated lateral and vertical mode of the FD

Refer to 31-13-00, Apex Controllers – General data for Flight Controller (FC) details.



KMC 9200 Flight Controller
Figure 4

B. Flight Control Primary Servos (SM3000)

The aileron (roll), elevator (pitch) and rudder (yaw) fail passive, primary servos, move the associated aircraft control surfaces in response to signals from the AIOP (Actuator Input Output Processor) located within the MAU. Communication between the MAU and primary servos is over bidirectional CAN Buses, motor enable discrete and clutch enable discrete.

Refer to 22-10-11, Primary Servos (S.O.O. 6221) – Maintenance Practices for details on Primary Servos.

Refer to 22-10-12, Primary Servo Capstan (S.O.O. 6221) – Maintenance Practices for details on Capstan.

C. Pitch Trim Servo (KS 272A)

The pitch trim servo controls the elevator trim tab in response to signals from the AIOP within the MAU when autopilot is enabled, or directly from the Pitch Trim Up / Down Relays when the servo is manually operated with the control wheel pitch trim switches. (Refer to 22-20-00, Pitch Trim System – General Data).

D. Autopilot and Flight Director Switches

Refer to Figure 5 for details.

Four AFCS control wheel switches are located on the outboard side of the Pilot and Co-pilot control wheels. To accommodate AFCS switches, the radio XMIT switch has been relocated to the forward face of the hand wheel switch housing utilizing a different switch type from the legacy design.

1) Autopilot Disconnect Switch (AP only)

The Autopilot Disconnect switch marked AP DISC (red) (installed with AP) is located top-centre on the hand wheel switch cluster. AP DISC is a momentary switch which disconnects autopilot, yaw damper and the trim servo when pressed. Autopilot, yaw damper will remain disconnected after the AP DISC switch is released.

2) Go-Around Switch

The Go-around switch marked G/A (green) is located on the outboard side of the hand wheel switch cluster. G/A is a momentary switch which commands the FD into a wings level pitch up attitude and disengages the autopilot.

3) Touch Control Steering Switch

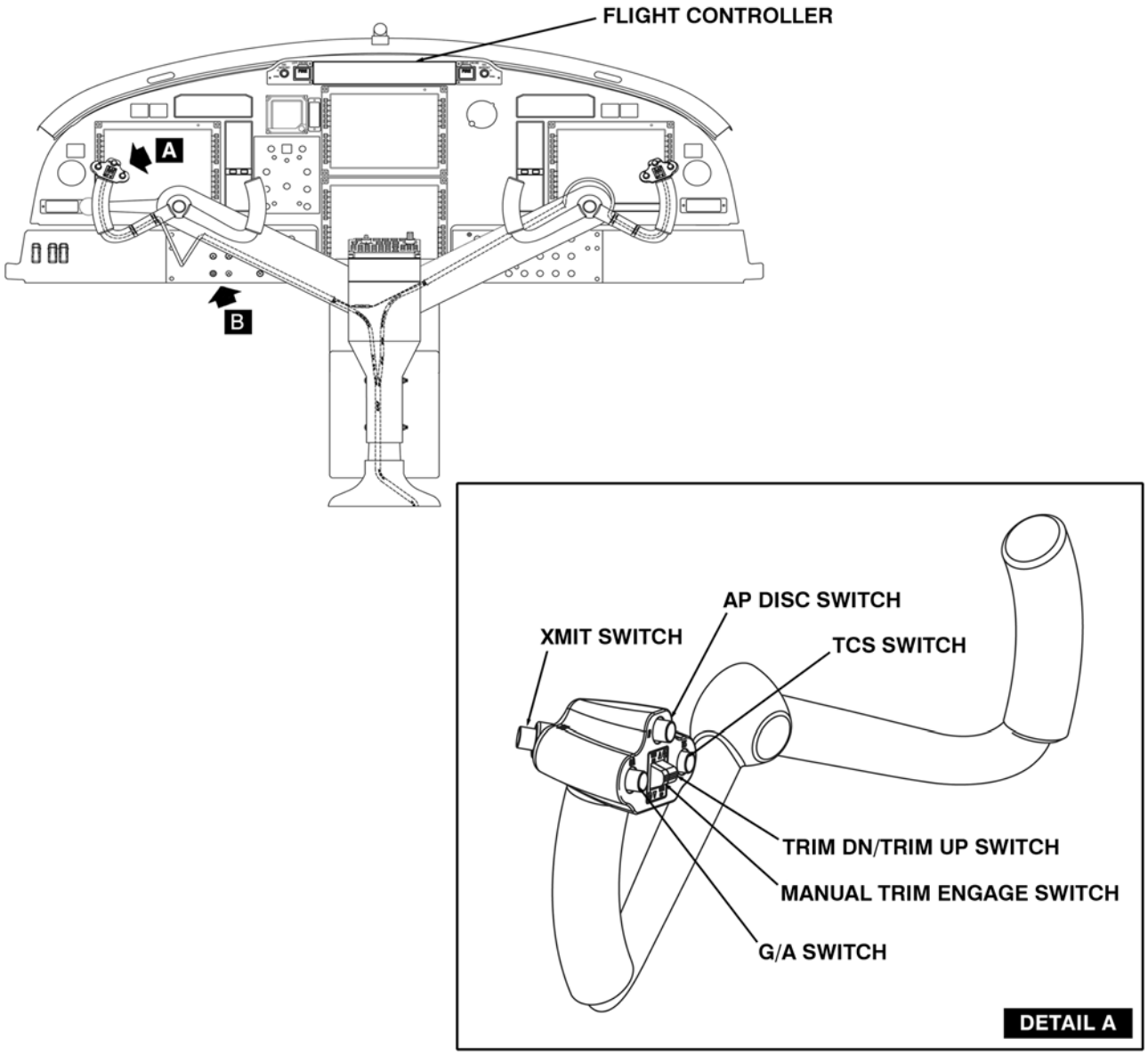
Touch control steering switch marked TCS (black) is located on the inboard side of the hand wheel switch cluster. TCS temporarily disconnects flight control servos and trim servo from the control surfaces when pressed. The system will maintain the aircraft flight condition (attitude, heading) when switch is released.

4) Trim Switches (AP only)

Trim switches marked TRIM DN and TRIM UP (installed with AP) is located in the centre of the hand wheel switch cluster. The trim switch manually controls the elevator trim servo. The switch has split controls, trim engage and trim direction. Both controls must be engaged to actuate the trim servo. Activation of the trim engage switch, while autopilot is engaged, will disengage autopilot.

5) Pitch Trim Switch (AP only)

A pitch trim disable switch marked PITCH TRIM is located on the pilot's switch panel. The switch has two positions, DISABLE and NORMAL. The switch must be in the NORMAL position (down) for the autopilot to function or for manual electronic control of the elevator trim servo. Selecting DISABLE will disconnect autopilot as well as disabling the manual electric pitch trim.



Autopilot Controls
Figure 5

3. Adjustment/Test

A. Flight Director (FD) Operational Test (Mod 6/2303)

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionic system to initialize.

3) Select the Flight Controller FD pushbutton, ensure that the LED above the button is ON.

4) On the Pilot and Co-pilot PFD, ensure the Flight Director bars are displayed and agree with the aircraft attitude.

5) On the Pilot and Co-pilot PFD, ensure that a green ROL and green PIT are displayed as active modes.

6) On the Flight Controller, press and hold the NOSE UP pushbutton, ensure the Flight Director bars move up.

7) On the Flight Controller, press and hold the NOSE DN pushbutton, ensure the Flight Director bars move down.

8) On the Flight Controller, press the HEADING rotary control to centre the heading bug.

9) On the Flight controller, press the HDG/T pushbutton, ensure that the LED above the button is ON.

10) On the Pilot and Co-pilot PFD, ensure the PFD heading bug is displayed in magenta.

11) Use the Flight Controller heading rotary knob to move the heading bug 20 degrees left of the aircraft heading, ensure the Flight Director bars follow the heading bug to the left.

12) Use the Flight Controller heading rotary knob to move the heading bug 20 degrees right of the aircraft heading, ensure the Flight Director bars follow the heading bug to the right.

13) On the Flight Controller, press the HDG/T pushbutton, ensure that the LED above the button is OFF.

14) Select the Pilot Go-Around (G/A) switch, ensure green GA indication on the Pilot and Co-Pilot PFD and the Flight Director bars on the Pilot and Co-pilot PFD move up slightly above the current aircraft attitude.

15) Select the Pilot Touch Control Steering (TCS) switch, ensure an amber TCS and green PIT indication on the Pilot and Co-pilot PFD and FD bars return to centre.

16) Select the Co-pilot Go-Around (G/A) switch, ensure green GA indication on the Pilot and Co-Pilot PFD and the Flight Director bars on the Pilot and Co-pilot PFD move up slightly above the current aircraft attitude.

17) Select the Co-pilot Touch Control Steering (TCS) switch, ensure an amber TCS and green PIT indication on the Pilot and Co-pilot PFD and FD bars return to centre.

18) Select the Flight Controller FD pushbutton, ensure that the LED above the button is OFF.

19) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Automatic Flight Control System (AFCS) Operational (Routine) Test (S.O.O. 6221)

1) Ensure that AP SERVOS (P1) (Left Bus) and PITCH TRIM (N1) (Left Bus) circuit breakers on the avionics circuit breaker panel are closed.

2) Ensure the PITCH TRIM switch is in the NORMAL position.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Wait several minutes for the aircraft electrical and avionics systems to initialize. During initialization, a Power-on Built In Test (PBIT) test will be completed on the Pitch Trim servo system.

5) Confirm AP 1 Fail and YD 1 Fail CAS messages extinguish and are not displayed after the avionics systems initialize.

6) Select the Pilot's control wheel switch to the TRIM UP position, confirm the trim wheel travels TRIM UP.

7) Select the Pilot's control wheel switch to the TRIM DN position, confirm the trim wheel travels TRIM DN.

8) Select the Co-pilot's control wheel switch to the TRIM UP position, confirm the trim wheel travels TRIM UP.

9) Select the Co-pilot's control wheel switch to the TRIM DN position, confirm the trim wheel travels TRIM DN.

10) Select and hold the Pilot's control wheel switch to either the TRIM UP or TRIM DN position and press and hold the AP DISC simultaneously, confirm the control wheel stops. Release the AP DISC, confirm the trim wheel travels resumes. Release the trim switch.

11) Select and hold the Co-pilot's control wheel switch to either the TRIM UP or TRIM DN position and press and hold the AP DISC simultaneously, confirm the control wheel stops. Release the AP DISC, confirm the trim wheel travels resumes. Release the trim switch.

12) Select and hold the Pilot's control wheel trim switch to either the TRIM UP or TRIM DN position and select the PITCH TRIM switch to the DISABLE position, confirm the control wheel stops. Release the trim switch. Confirm the AP 1 Fail CAS message illuminates.

13) Select the PITCH TRIM switch to the NORMAL position, confirm the AP 1 Fail CAS message extinguishes.

14) Select the Flight Controller YD pushbutton, ensure the following:

    a) LED above the Flight Controller YD pushbutton is ON.

    b) On the Pilot and Co-pilot PFD, a green YD is displayed.

    c) The yaw servo is engaged by confirming the rudder pedals are firm.

        1 On the Pilot control wheel select the AP DISC. Ensure the following:

            1 LED above the Flight Controller YD pushbutton is OFF.

            2 On the Pilot and Co-pilot PFD, a green YD is not displayed.

            3 The yaw servo is disengaged by confirming the rudder pedals can be moved freely.

15) Select the Flight Controller YD pushbutton, ensure the following:

    a) LED above the Flight Controller YD pushbutton is ON.

    b) On the Pilot and Co-pilot PFD, a green YD is displayed.

    c) The yaw servo is engaged by confirming the rudder pedals are firm.

        1 On the Co-pilot control wheel select the AP DISC. Ensure the following:

            1 LED above the Flight Controller YD pushbutton is OFF.

            2 On the Pilot and Co-pilot PFD, a green YD is not displayed.

            3 The yaw servo is disengaged by confirming the rudder pedals can be moved freely.

16) Select the Flight Controller YD pushbutton, ensure the following:

a) LED above the Flight Controller YD pushbutton is ON.

b) On the Pilot and Co-pilot PFD, a green YD is displayed.

c) The yaw servo is engaged by confirming the rudder pedals are firm.

1 On the Pilot control wheel select the GA button. Ensure the following:

1 LED above the Flight Controller YD pushbutton is OFF.

2 On the Pilot and Co-pilot PFD, green YD is not displayed.

3 The yaw servo is disengaged by confirming the rudder pedals can be moved freely.

4 On the Pilot and Co-pilot PFD, a green ROL and GA is displayed.

5 On the Pilot and Co-pilot PFD, the flight director bars move up slightly above the current aircraft attitude.

6 Press the Pilot TCS to cancel the GA mode.

17) Select the Flight Controller YD pushbutton, ensure the following:

a) LED above the Flight Controller YD pushbutton is ON.

b) On the Pilot and Co-pilot PFD, a green YD is displayed.

c) The yaw servo is engaged by confirming the rudder pedals are firm.

1 On the Co-pilot control wheel select the GA button. Ensure the following:

1 LED above the Flight Controller YD pushbutton is OFF.

2 On the Pilot's and Co-pilot's PFD, green YD is not displayed.

3 The yaw servo is disengaged by confirming the rudder pedals can be moved freely.

4 On the Pilot's and Co-pilot's PFD, a green ROL and GA is displayed.

5 On the Pilot's and Co-pilot's PFD, the flight director bars move up slightly above the current aircraft attitude.

6 Press the Co-pilot TCS to cancel the GA mode.

18) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. AFCS Function Test (S.O.O. 6221)

WARNING **WHEN OPERATING THE AUTOPILOT, ENSURE THE FLIGHT CONTROLS ARE CLEAR AND ALL RIGGING PINS ARE REMOVED.**

Note If any power interruption has occurred prior to completion of the MAU's internal self-test, AP 1 Fail CAS message may persist longer than 2 minutes. If this occurs, an OFF/ON cycle of the aircraft electrical power is required to allow the MAU to reboot without power interruptions.

Note When the Autopilot is engaged, ensure it does not drive the servos against the stops. Autopilot is to be engaged for short periods of time for all ground tests. If at any time the Autopilot begins to drive a servo against the stop, disengage Autopilot and discontinue the test. After the flight controls, rudder pedal, and trims are reset to the center position, the Autopilot can be reengaged and the test steps continued.

1) AFCS Function Test

a) Ensure that AP SERVOS (P1) (Left Bus) and PITCH TRIM (N1) (Left Bus) circuit breakers, on the avionics circuit breaker panel are closed.

b) Ensure the PITCH TRIM switch is in the NORMAL position.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Wait several minutes for the aircraft electrical and avionics systems to initialize. During initialization, a Power-on Built In Test (PBIT) test will be completed on the Pitch Trim servo system.

e) Confirm AP 1 Fail and YD 1 Fail CAS messages extinguish and are not displayed after the avionics systems initialize.

f) Attach a pitot/static test set (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data) and increase airspeed to >60 knots to enable autopilot functionality.

g) Ensure the flight controls move freely in roll, pitch and yaw. Set the flight controls and rudder pedals to the center position.

h) On the Flight Controller select the L/R pushbutton. Ensure L/R can be selected to either the Pilot position (left) or Co-pilot position (right). Upon completion, select the Pilot (L) position.

i) Select the Flight Controller FD pushbutton, ensure that the LED above the button is ON.

j) On the Pilot and Co-pilot PFD, ensure the flight director bars are displayed and agree with the aircraft attitude.

k) On the Pilot and Co-pilot PFD, ensure that a green ROL and green PIT are displayed as active modes.

l) On the Flight Controller, press the HEADING rotary control to center the heading bug and press the HDG/T pushbutton. Ensure that the LED above the HDG/T pushbutton is ON.

m) On the Pilot and Co-pilot PFD, ensure the PFD heading bug is displayed in magenta.

n) Select the Flight Controller AP pushbutton, ensure that the LED above the pushbutton is ON and that the LED above the YD pushbutton is ON.

o) On the Pilot and Co-pilot PFD, ensure green AP and YD are displayed.

p) Ensure that the autopilot servos engage and the flight controls become firm.

q) Use the Flight Controller heading rotary knob to move the heading 5 degrees left of the aircraft heading, ensure the Flight Director bars and flight controls follow the heading bug to the left.

r) Use the Flight Controller heading rotary knob to move the heading 5 degrees right of the aircraft heading, ensure the Flight Director bars and flight controls follow the heading bug to the right.

s) On the Flight Controller, press the HDG/T pushbutton, ensure that the LED above the button is OFF.

t) On the Flight Controller, repeatedly select the NOSE UP pushbutton, ensure the Flight Director bars move up and flight controls move aft.

u) On the Flight Controller, repeatedly select the NOSE DN pushbutton, ensure the Flight Director bars move down and flight controls move forward.

v) On the FC select the AP pushbutton, ensure the LED above the AP is OFF. Ensure the LED above the YD is ON.

w) On the Pilot and Co-pilot PFD, ensure AP flashes red and red inverse and the aural warning cavalry charge persists.

x) Ensure that the autopilot roll and pitch servos disengage and the flight controls move freely in roll and pitch.

y) On the Pilot or Co-pilot control wheel select the AP DISC, ensure that the red flashing AP clears, and the cavalry charge aural mutes.

z) On the FC, ensure the green LED above the YD button is OFF and the YAW servo has disengaged by moving the rudder pedals freely.

aa) Reduce the airspeed on the Pitot/Static test set to bring the aircraft into GND mode.

ab) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) AFCS Engage/Disengage Test

a) Ensure that AP SERVOS (P1) (Left Bus) and PITCH TRIM (N1) (Left Bus) circuit breakers on the avionics circuit breaker panel are closed.

b) Ensure the PITCH TRIM switch is in the NORMAL position.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Wait several minutes for the aircraft electrical and avionics systems to initialize. During initialization, a PBIT test will be completed on the Pitch Trim servo system.

e) Confirm AP 1 Fail and YD 1 Fail CAS messages are not displayed after the avionics systems initialize.

f) Attach a pitot/static test set (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data) and increase airspeed to >60 knots to enable autopilot functionality.

g) Set the flight controls and rudder pedals to the center position. On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

4 The autopilot servo clutches engage and the flight controls are firm.

h) On the Pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure the following:

1 The cavalry charge aural warning is continually heard.

2 The LED above the Flight Controller AP pushbutton is OFF.

3 The LED above the Flight Controller YD pushbutton is OFF.

4 The Pilot and Co-pilot PFD AP flashes red and red inverse.

5 On the Pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure on the Pilot and Co-pilot PFDs the flashing red AP and green YD are not displayed. Ensure the cavalry charge is muted.

6 The autopilot servo clutches disengage and the flight controls can be moved freely in roll, pitch and yaw. Set the flight controls and rudder pedals to the center position.

i) On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

4 The autopilot servo clutches engage and the flight controls are firm.

j) On the Co-pilot control yoke, press and release the AP DISC. Ensure the following:

1 The cavalry charge aural warning is continually heard.

2 The LED above the Flight Controller AP pushbutton is OFF.

3 The LED above the Flight Controller YD pushbutton is OFF.

4 The Pilot and Co-pilot PFD AP flashes red and red inverse.

5 On the Co-pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure on the Pilot and Co-pilot PFDs the flashing red AP and green YD are not displayed. Ensure the cavalry charge is muted.

6 The autopilot servo clutches disengage and the flight controls can be moved freely in roll, pitch and yaw. Set the flight controls and rudder pedals to the center position.

k) On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

4 The autopilot servo clutches engage and the flight controls are firm.

l) On the Pilot control yoke, press, briefly hold, and release the pitch trim switch to either the TRIM DN or TRIM UP position. Ensure the following:

1 The cavalry charge aural warning is continually heard.

2 The LED above the Flight Controller AP pushbutton is OFF.

3 The LED above the Flight Controller YD pushbutton is ON.

4 The Pilot and Co-pilot PFD AP flashes red and red inverse.

5 On the Pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure on the Pilot and Co-pilot PFDs the flashing red AP and green YD are not displayed. Ensure the cavalry charge is muted. Ensure the LED above the Flight Controller YD pushbutton is OFF.

6 The autopilot servo clutches disengage and the flight controls can be moved freely in roll, pitch and yaw. Set the flight controls and rudder pedals to the center position.

m) On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

4 The autopilot servo clutches engage and the flight controls are firm.

n) On the Co-pilot control yoke, press, briefly hold, and release the TRIM DN or TRIM UP position. Ensure the following:

1 The cavalry charge aural warning is continually heard.

2 The LED above the Flight Controller AP pushbutton is OFF.

3 The LED above the Flight Controller YD pushbutton is ON.

4 The Pilot and Co-pilot PFD AP flashes red and red inverse.

5 On the Co-pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure on the Pilot and Co-pilot PFDs the flashing red AP and green YD are not displayed. Ensure the cavalry charge is muted. Ensure the LED above the Flight Controller YD pushbutton is OFF.

6 The autopilot servo clutches disengage and the flight controls can be moved freely in roll, pitch and yaw. Set the flight controls and rudder pedals to the centre position.

o) On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

4 The autopilot servo clutches engage and the flight controls are firm.

p) On the Pilot's switch panel, select the PITCH TRIM switch to the DISABLE (up) position. Ensure the following:

1 The cavalry charge aural warning is continually heard.

2 The LED above the Flight Controller AP pushbutton is OFF.

3 The LED above the Flight Controller YD pushbutton is ON.

4 The Pilot and Co-pilot PFD AP flashes red and red inverse.

5 On the Pilot or Co-pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure on the Pilot and Co-pilot PFDs the flashing red AP and green YD are not displayed. Ensure the cavalry charge is muted. Ensure the LED above the Flight Controller YD pushbutton is OFF.

6 The autopilot servo clutches disengage and the flight controls can be moved freely in roll, pitch and yaw.

7 Confirm AP 1 Fail message is displayed.

q) On the Pilot's switch panel, select the PITCH TRIM switch to the NORMAL (down) position. Ensure the following:

1 Confirm AP 1 Fail messages is not displayed.

r) Set the flight controls and rudder pedals to the center position. On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

s) On the avionics circuit breaker panel, open the PITCH TRIM (N1) circuit breaker. Ensure the following:

1 The cavalry charge aural warning is continually heard.

2 The LED above the Flight Controller AP pushbutton is OFF.

3 The LED above the Flight Controller YD pushbutton is ON.

4 The Pilot and Co-pilot PFD AP flashes red and red inverse.

5 On the Pilot or Co-pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure on the Pilot and Co-pilot PFDs the flashing red AP and green YD are not displayed. Ensure the cavalry charge is muted. Ensure the LED above the Flight Controller YD pushbutton is OFF.

6 The autopilot servo clutches disengage and the flight controls can be moved freely in roll, pitch and yaw. Set the flight controls and rudder pedals to the centre position.

7 Confirm AP 1 Fail message is displayed.

t) On the avionics circuit breaker panel, close the PITCH TRIM (N1) circuit breaker. Ensure the following:

1 Confirm AP 1 Fail message is not displayed.

u) Set the flight controls and rudder pedals to the center position. On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pusbutton is ON.

2 The LED above the Flight Controller YD pusbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

v) On the avionics circuit breaker panel, open the AP SERVOS (P1) circuit breaker. Ensure the following:

1 The cavalry charge aural warning is continually heard.

2 The LED above the Flight Controller AP pushbutton is OFF.

3 The LED above the Flight Controller YD pushbutton is OFF.

4 The Pilot and Co-pilot PFD AP flashes red and red inverse. Ensure amber YD is displayed.

5 On the Pilot or Co-pilot control yoke, press and release the Quick Disconnect (AP DISC). Ensure on the Pilot and Co-pilot PFDs the flashing red AP is not displayed, and amber YD is displayed. Ensure the cavalry charge is muted.

6 The autopilot servo clutches disengage and the flight controls can be moved freely in roll, pitch and yaw.

7 Confirm AP 1 Fail and YD 1 Fail messages are displayed.

w) On the avionics circuit breaker panel, close the AP SERVOS (P1) circuit breaker. Ensure the following:

1 Confirm AP 1 Fail and YD 1 Fail messages are not displayed.

2 The Pilot and Co-pilot PFD, ensure the amber YD is not displayed.

x) Reduce the airspeed on the Pitot/Static test set to bring the aircraft into GND mode.

y) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) AFCS Touch Control Steering (TCS) Test

a) Ensure that AP SERVOS (P1) (Left Bus) and PITCH TRIM (N1) (Left Bus) circuit breakers, on the avionics circuit breaker panel are closed.

b) Ensure the PITCH TRIM switch is in the NORMAL position.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Wait several minutes for the aircraft electrical and avionics systems to initialize. During initialization, a PBIT test will be completed on the Pitch Trim servo system.

e) Confirm AP 1 Fail and YD 1 Fail CAS messages extinguish and are not displayed after the avionics systems initialize.

f) Attach a pitot/static test set (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data) and increase airspeed to >60 knots to enable autopilot functionality.

g) Set the flight controls and rudder pedals to the center position. On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

4 The autopilot servo clutches engage and the flight controls are firm.

h) On the Pilot control yoke, press and hold the TCS pushbutton. Ensure the following:

1 On the Pilot and Co-pilot PFD's the AP caption changes from green AP to an amber TCS.

2 The pitch and roll servos disengage and the flight controls can be moved freely. Set the flight controls to the center position.

i) Release the Pilot TCS pushbutton. Ensure the following:

1 On the Pilot and Co-pilot PFD's the amber TCS caption changes to a green AP.

2 The pitch and roll servos re-engage and the flight controls are firm.

j) On the Co-pilot control yoke, press and hold the TCS pushbutton. Ensure the following:

1 On the Pilot and Co-pilot PFD's the AP caption changes from green AP to an amber TCS.

2 The pitch and roll servos disengage and the flight controls can be moved freely. Set the flight controls to the center position.

k) Release the Co-pilot TCS pushbutton. Ensure the following:

1 On the Pilot and Co-pilot PFD's the amber TCS caption changes to a green AP.

2 The pitch and roll servos re-engage and the flight controls are firm.

l) Reduce the airspeed on the Pitot/Static test set to bring the aircraft into GND mode.

m) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) AFCS Trim Sense Test

a) Ensure that AP SERVOS (P1) (Left Bus) and PITCH TRIM (N1) (Left Bus) circuit breakers, on the avionics circuit breaker panel are closed.

b) Ensure the PITCH TRIM switch is in the NORMAL position.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Wait several minutes for the aircraft electrical and avionics systems to initialize. During initialization, a PBIT test will be completed on the Pitch Trim servo system.

e) Confirm AP 1 Fail and YD 1 Fail CAS messages are not displayed after the avionics systems initialize.

f) Attach a pitot/static test set (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data) and increase airspeed to >60 knots to enable autopilot functionality.

g) Move the flight controls to approximately the center position.

h) On the FC, select the AP pushbutton. Ensure the following:

1 The LED above the Flight Controller AP pushbutton is ON.

2 The LED above the Flight Controller YD pushbutton is ON.

3 The Pilot and Co-pilot PFD green AP and YD are displayed.

4 The autopilot servo clutches engage and the flight controls are firm.

i) Hold the flight control yoke and gently turn it to the right with enough force to turn the yoke, but not enough force to override the servo.

j) Ensure after approximately 15 seconds, the AP Hold LH Wing Down caution CAS message displays.

k) Release the flight control yoke, ensure the AP Hold LH Wing Down caution CAS message extinguishes.

l) Hold the flight control yoke and gently turn it to the left with enough force to turn the yoke, but not enough force to override the servo.

m) Ensure after approximately 15 seconds, the AP Hold RH Wing Down caution CAS message displays.

n) Release the flight control yoke, ensure the AP Hold RH Wing Down caution CAS message extinguishes.

o) Hold the flight controls and push them forward with enough force to move the controls, but not enough force to override the servo. Ensure the following:

1 The pitch trim running indicator (green dot) shows on the MFD.

2 The pitch trim indicator drives nose UP.

3 The manual pitch trim wheel drives nose UP.

4 After approximately 10 seconds the AP Hold Nose Up caution CAS message is displayed.

p) Release the flight controls. Ensure the following:

1 The pitch trim running indicator (green dot) extinguishes on the MFD.

2 The pitch trim indicator stops moving.

3 The manual pitch trim wheel stops moving.

4 The AP Hold Nose Up caution CAS message extinguishes.

q) Hold the flight controls and pull them back with enough force to move the controls, but not enough to override the servo. Ensure the following:

1 The pitch trim running indicator (green dot) shows on the MFD.

2 The pitch trim indicator drives nose DOWN.

3 The manual pitch trim wheel drives nose DOWN.

4 After approximately 10 seconds the AP Hold Nose Down caution CAS message is displayed.

r) Release the flight controls. Ensure the following:

1 The pitch trim running indicator (green dot) extinguishes on the MFD.

2 The pitch trim indicator stops moving.

3 The manual pitch trim wheel stops moving.

4 The AP Hold Nose Down caution CAS message extinguishes.

s) On the FC, select AP to disconnect the autopilot and select the AP DISC on the control yoke to mute the cavalry charge aural alert.

t) Reduce the airspeed on the Pitot/Static test set to bring the aircraft into GND mode.

u) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. De-activation/Re-activation

A. De-activation of Autopilot (S.O.O. 6221)

Note Do not pull AP SERVOS circuit breaker (P1) on the Avionics CB Panel. This circuit breaker must be left engaged, as it is required for normal Flight Director operation.

Note Leave all electrical connectors installed on AP Servos and Pitch Trim Servo.

Note Leave bridle cables installed and tensioned on AP Servos and Pitch Trim Servo.

Note With Autopilot disabled, the following tests are not required:

- AFCS Operational (Routine) Test (S.O.O. 6221), refer to Para 3.B
- AFCS Function Test (Mod S.O.O. 6221), refer to Para 3.C
- KS 272A Pitch Trim Servo System, refer to 22-20-00, Pitch Trim System (S.O.O. 6221) – General Data, Para 3.A.

1) Ensure aircraft power is OFF.

2) Pull PITCH TRIM circuit breaker (N1) and secure with safety clip.

3) Select DISABLE position on PITCH TRIM switch (refer to Figure 6).

4) Install ELECTRIC PITCH TRIM INOP placard at the location shown in Figure 6, to the right of PITCH TRIM switch. Text height to be minimum 0.25” in height and clearly legible.

5) Install AUTOPILOT AND ELECTRIC PITCH TRIM DISABLED placard at location shown in Figure 6, on bezel of upper MFD screen, below Flight Controller. Text height to be minimum 0.40” in height and clearly legible.

6) Install Apex Option File with Autopilot Options disabled, as per Section 45-00-01, Apex Options and Settings File – Load Procedures.

7) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

8) Wait several minutes for the aircraft electrical and avionics systems to initialize.

9) Attach a Pitot/Static test set (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data) and increase airspeed to above 60 knots to AIR mode to enable autopilot functionality.

10) Confirm the Flight Controller YD does not engage when selected:

a) LED above the Flight Controller YD pushbutton is OFF.

b) On the Pilot’s and Co-pilot’s PFD, a green YD is not displayed.

c) The yaw servo is disengaged by confirming the rudder pedals can be moved freely.

d) Confirm FUNCTION UNAVAILABLE Status CAS message.

11) Confirm Flight Controller AP pushbutton does not engage when selected:

a) LED above the Flight Controller AP pushbutton is OFF.

b) LED above the Flight Controller YD pushbutton is OFF.

c) Confirm FUNCTION UNAVAILABLE Status CAS message.

12) Reduce the airspeed on the Pitot/Static test set to bring the aircraft into GROUND mode.

13) Do operational test of the Flight Director, as per Para 3.A.

B. Re-activation of Autopilot (S.O.O. 6221)

1) Install Apex Option File with Autopilot Options enabled, as per Section 45-00-01, Apex Options and Settings File – Load Procedures.

2) Remove safety clips and close PITCH TRIM circuit breaker (N1).

3) Remove ELECTRIC PITCH TRIM INOP placard at location shown in Figure 6, to the right of PITCH TRIM switch.

4) Remove AUTOPILOT AND ELECTRIC PITCH TRIM DISABLED placard at location shown in Figure 6, on bezel of upper MFD screen, below Flight Controller.

5) Select NORMAL position on PITCH TRIM switch (refer to Figure 6).

6) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

7) Wait several minutes for the aircraft electrical and avionics systems to initialize.

8) Attach a Pitot/Static test set (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data) and increase airspeed to above 60 knots to AIR mode to enable autopilot functionality.

9) Select the Flight Controller YD, ensure the following:

a) LED above the Flight Controller YD pushbutton is ON.

b) On the Pilot's and Co-pilot's PFD, a green YD is displayed.

c) The yaw servo is engaged by confirming the rudder pedals are firm.

10) Confirm Flight Controller AP pushbutton does engage when selected:

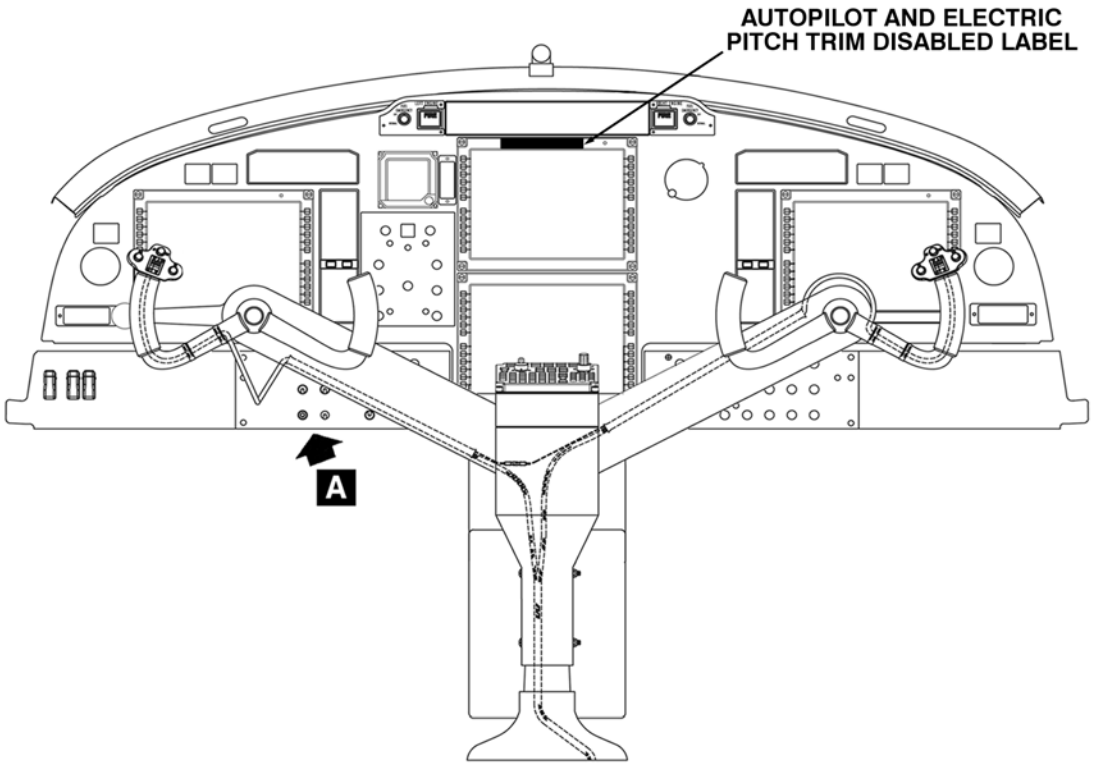
a) LED above the Flight Controller AP pushbutton is ON.

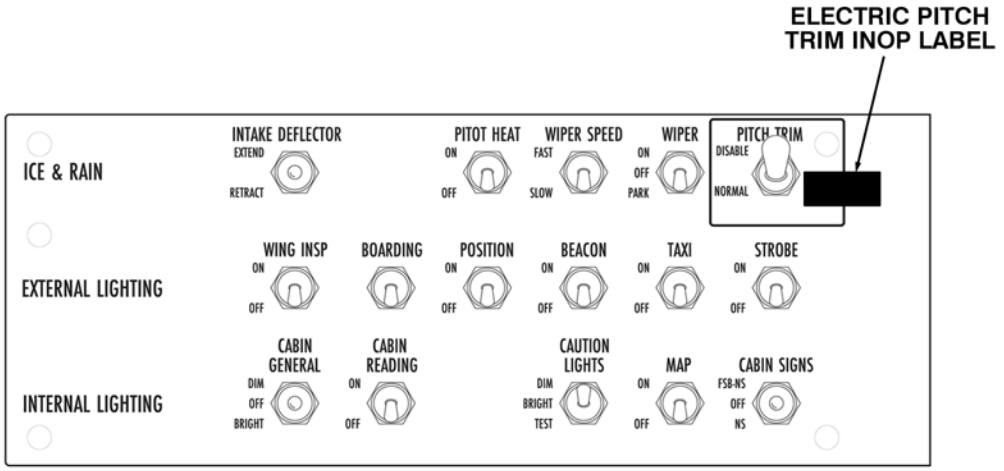
b) LED above the Flight Controller YD pushbutton is ON.

c) The yaw servo is engaged by confirming the rudder pedals are firm.

11) Do a operational test of the Flight Director, as per Para 3.A.

12) Do a function test of the AFCS, as per Para 3.C.





DETAIL A

Installation of ELECTRIC PITCH TRIM INOP and AUTOPILOT AND ELECTRIC PITCH TRIM DISABLED Placards
Figure 6

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Primary Servos (S.O.O. 6221) – Maintenance Practices

1. Description

A. Flight Control Primary Servos (SM3000)

The aileron (roll), elevator (pitch) and rudder (yaw) fail passive, primary servos, move the associated aircraft control surfaces in response to signals from the AIOP (Actuator Input Output Processor) with the MAU. Communication between the MAU and primary servos is over bidirectional CAN (Control Area Network) Buses, motor enable discrete and clutch enable discrete.

Each servo contains a DC motor, gearing and an electromechanical clutch. The electromechanical clutch is engaged when 28 volts is applied to the clutch supply. The clutch only engages or disengages the servo; torque limiting is done electronically and controlled through the Apex software. Each servo is mounted on and drives a capstan (SB3000), which actuates the associated aircraft control surfaces through servo bridle cables clamped to the control cables.

28 Volt DC power is protected by a 7.5 Amp AP SERVOS (P1) (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

2. Removal/Installation

Refer to Figure 1 and Figure 2 for details.

A. Remove Primary Servos (Rudder, Elevator or Aileron)

Note The primary servos (rudder, elevator or aileron) can be replaced without removing the capstan and disturbing the cable rigging.

1) Ensure all aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to servos as follows:

a) Rudder and Elevator

1 Remove bulkhead panel at station 406 of rear baggage compartment.

2 Remove aft baggage extension floor panels.

b) Aileron

1 Remove Co-pilot's seat. (Refer to 25-10-00, Flight Compartment – General Data, Para 3.).

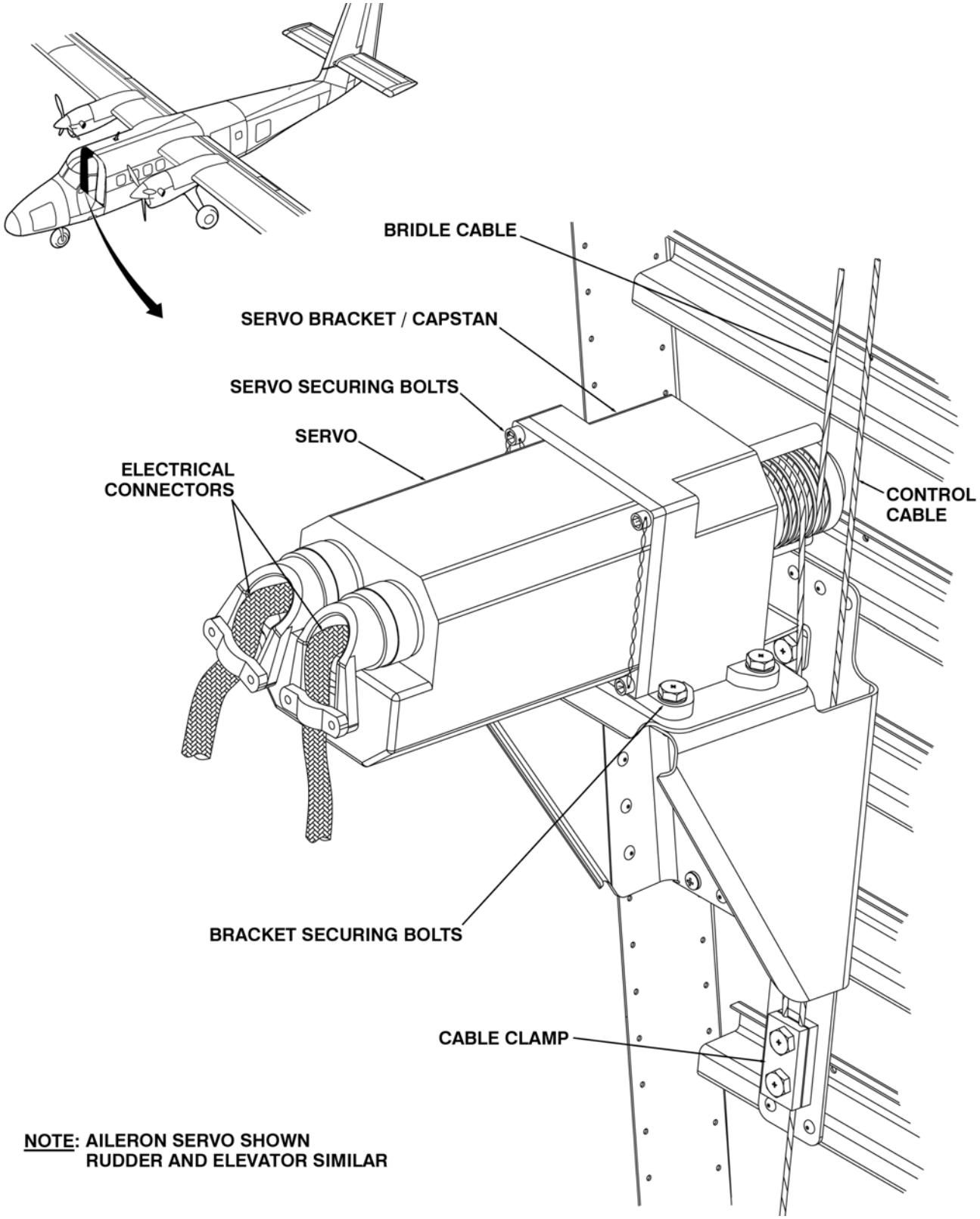
2 Remove aileron cable covers on Co-pilot side in flight compartment.

3) Disconnect electrical connectors from servos.

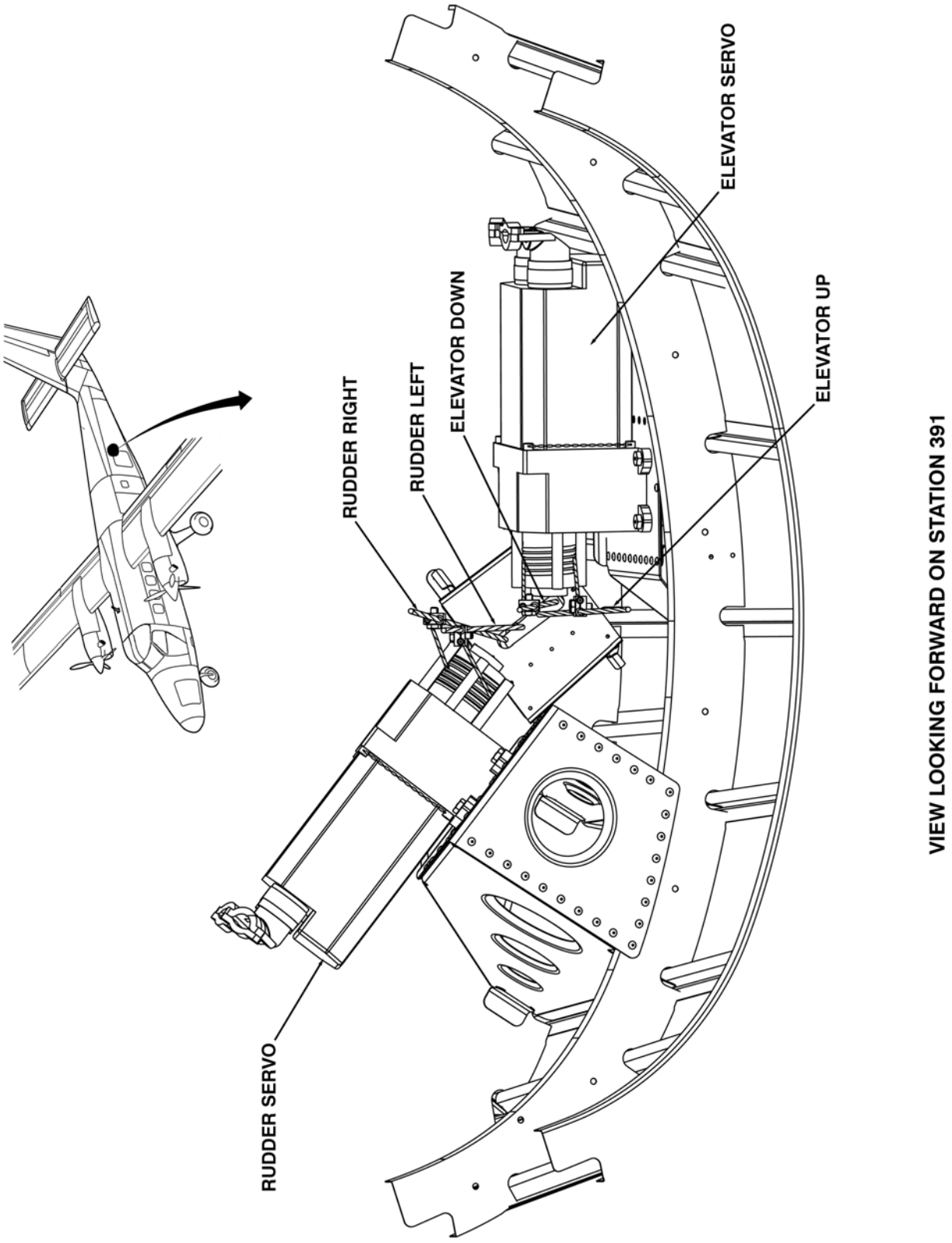
4) Remove lock wire from captive head screws securing the servo to the capstan.

5) Support servo, remove captive screws securing servo to capstan and withdraw servo, taking care not to damage splines on the servo drive shaft or capstan.

6) Remove and discard servo O-ring.



Primary Servo Installation
Figure 1



Completed Rudder and Elevator Servo Installation
Figure 2

B. Install Primary Servos (Rudder, Elevator or Aileron)

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED.**

1) Ensure all aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Servo to airframe electrical bonding is maintained through direct contact between the servo and airframe mating surfaces. Prior to installation, ensure mating surfaces are clean and free of contaminants. Refer to procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data, as required.

3) Ensure servo capstan is correctly rigged. (Refer to 22-10-12, Primary Servo Capstan (S.O.O. 6221) – General Data, Figure 1, Figure 2 and Figure 3).

4) Install new servo O-ring, apply thin coat of Krytox 283 AD grease or equivalent to internal spline, O-ring on servo body and pilot diameter on servo capstan.

5) Engage internal spline on servo drive shaft with spline on servo capstan shaft and secure with captive screws. Torque screws to 29±3 in-lbs and lockwire.

6) Connect electrical connectors to servo.

7) Do bonding check. (Refer to Para 3.A).

8) Restore aircraft electrical power. (Refer to 24-00-00, Electrical Power – General Data).

9) Function test system. (Refer to 22-10-00, Automatic Flight Controls System (AFCS) (Mod 6/2303 and S.O.O. 6221) – General Data, Para 3.C. and Para 3.F.).

10) If rudder or elevator servo has been installed, install aft baggage extension floor panels and bulkhead panel at station 406 of rear baggage compartment. If aileron servo has been installed, install aileron cable covers in flight compartment and install Co-pilot's seat.

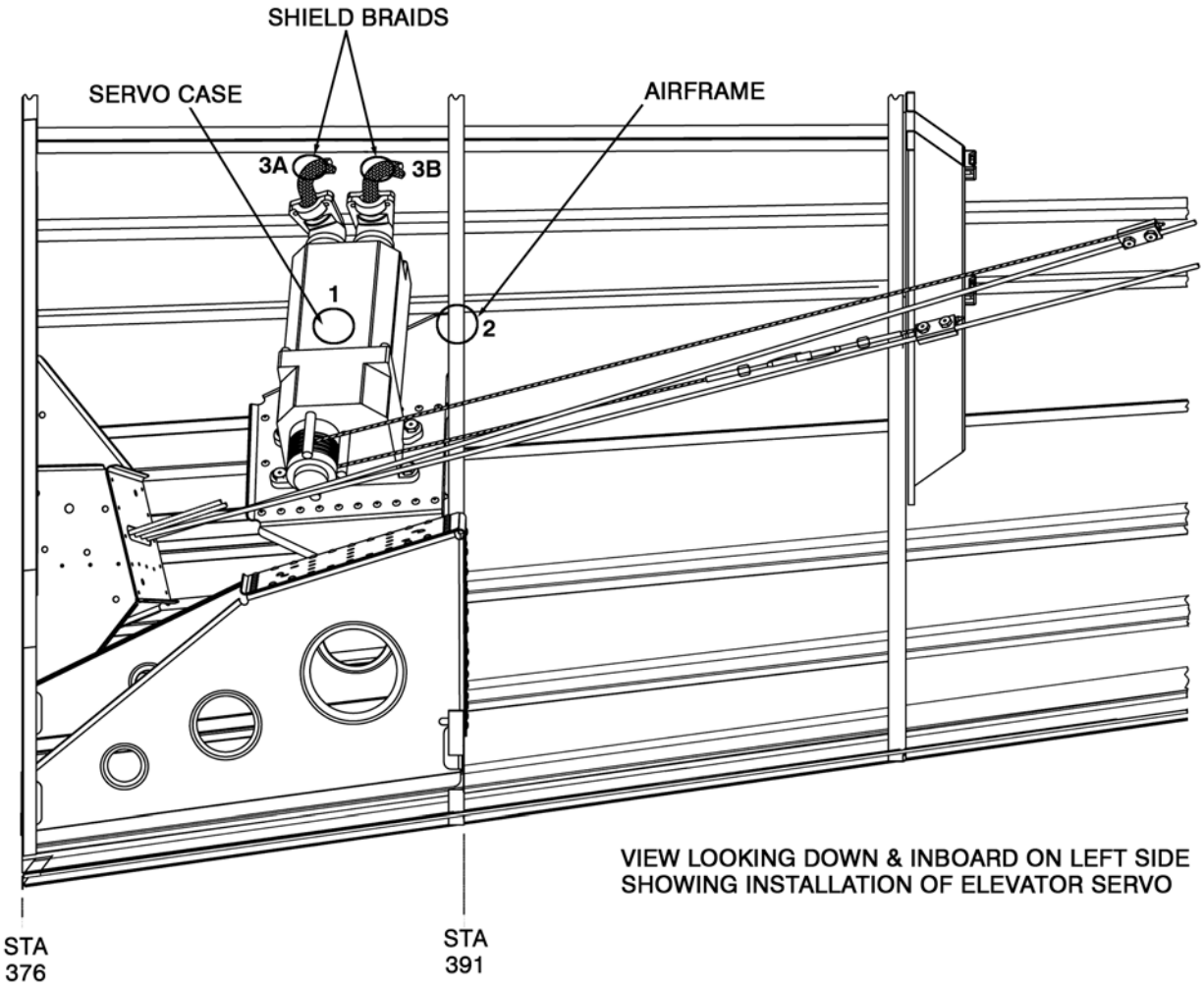
3. Adjustment/Test

A. Electrical Bonding

Refer to Figure 3 for Elevator Bonding Check.

Refer to Figure 4 for Rudder Bonding Check.

Refer to Figure 5 for Aileron Bonding Check.



CONDUCT THE FOLLOWING ELECTRICAL BONDING CHECKS:

SERVO CASE TO AIRFRAME (POINT 1 TO POINT 2) REFER TO NOTES 1 AND 2. IMPEDANCE BOND VALUE TO BE LESS THAN 2.5 MILLIOHMS. IF NOT ACHIEVABLE, REFER TO NOTE 3.

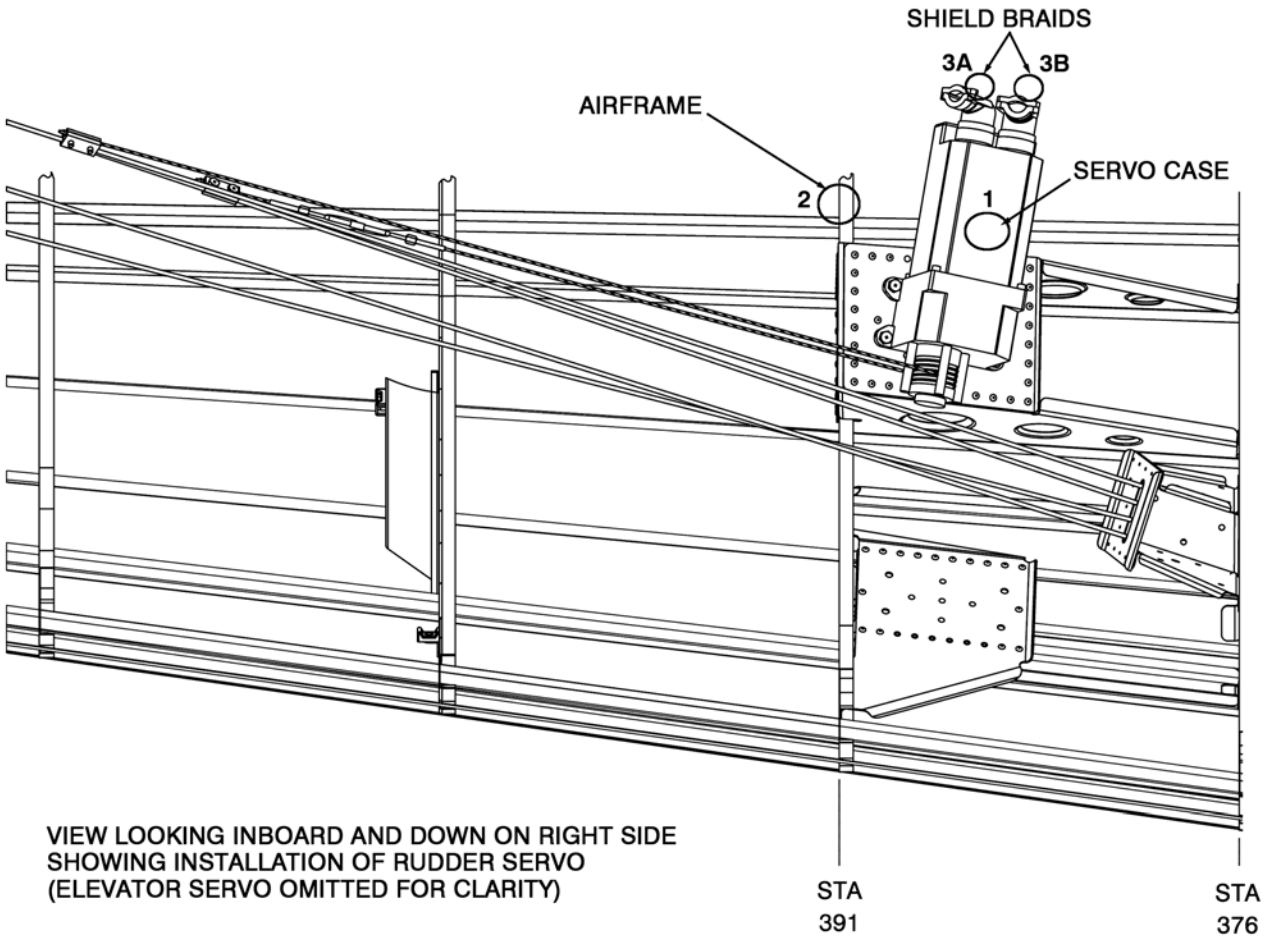
SHIELD BRAIDS TO SERVO CASE (POINT 3A TO POINT 1) REFER TO NOTE 1. IMPEDANCE BOND VALUE TO BE LESS THAN 7.5 MILLIOHMS. IF NOT ACHIEVABLE, REFER TO NOTE 3. REPEAT FOR POINT 3B TO POINT 1.

NOTE 1: SERVO PROBE CONTACT POINT SHOULD BE SELECTED ON SERVO CASE. REMOVE NON-CONDUCTIVE COATING, IF REQUIRED, TO ACHIEVE AN ACCEPTABLE CONTACT POINT. SERVO ASSEMBLY FASTENER MAY BE USED.

NOTE 2: AIRFRAME PROBE CONTACT POINT SHOULD BE SELECTED AS CLOSE AS POSSIBLE TO SERVO BRACKET, BUT NOT ON THE MOUNTING BRACKET. REMOVE NON-CONDUCTIVE COATING, IF REQUIRED, TO ACHIEVE AN ACCEPTABLE CONTACT TO AIRFRAME GROUND.

NOTE 3: IF REQUIRED, THE MAXIMUM STATED IMPEDANCE VALUES MAY BE ADJUSTED PER THE FOLLOWING; AN ACCEPTABLE IMPEDANCE BOND VALUE (MAXIMUM MILLIOHMS) MAY BE DETERMINED BY MULTIPLYING THE NUMBER OF CONTACT BOND SURFACES BY 2.5 MILLIOHMS FOR EACH BOND SURFACE LOCATED BETWEEN THE PROBE CONTACT POINTS.

Elevator Bonding Check
Figure 3



VIEW LOOKING INBOARD AND DOWN ON RIGHT SIDE
SHOWING INSTALLATION OF RUDDER SERVO
(ELEVATOR SERVO OMITTED FOR CLARITY)

CONDUCT THE FOLLOWING ELECTRICAL BONDING CHECKS:

SERVO CASE TO AIRFRAME (POINT 1 TO POINT 2) REFER TO NOTES 1 AND 2. IMPEDANCE BOND VALUE TO BE LESS THAN 2.5 MILLIOHMS. IF NOT ACHIEVABLE, REFER TO NOTE 3.

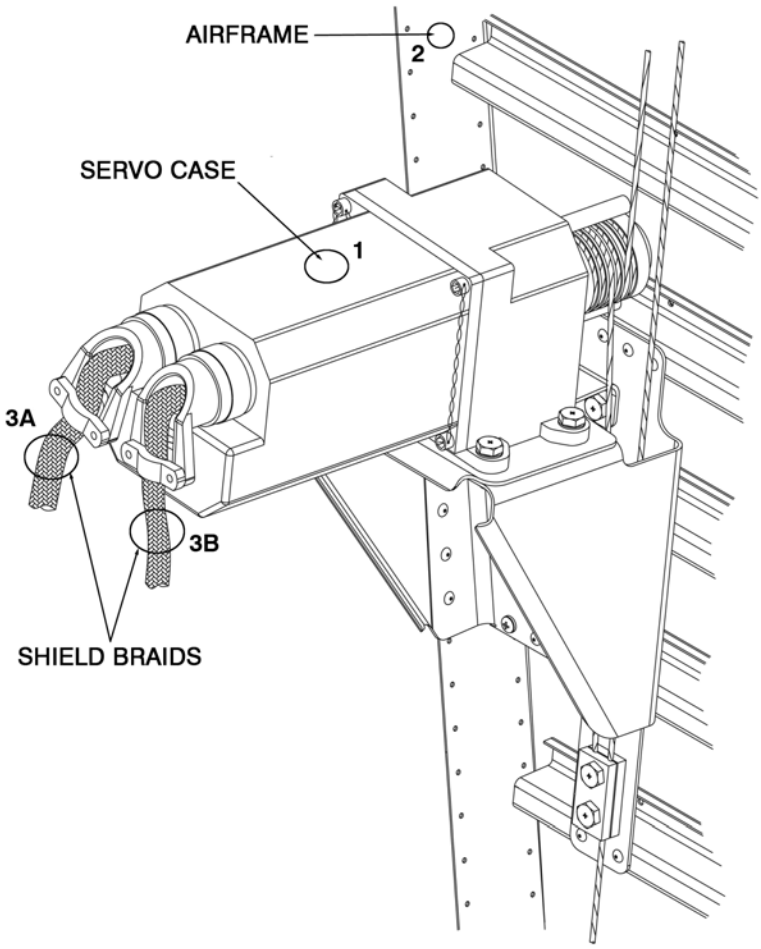
SHIELD BRAIDS TO SERVO CASE (POINT 3A TO POINT 1) REFER TO NOTE 1. IMPEDANCE BOND VALUE TO BE LESS THAN 7.5 MILLIOHMS. IF NOT ACHIEVABLE, REFER TO NOTE 3. REPEAT FOR POINT 3B TO POINT 1.

NOTE 1: SERVO PROBE CONTACT POINT SHOULD BE SELECTED ON SERVO CASE. REMOVE NON-CONDUCTIVE COATING, IF REQUIRED, TO ACHIEVE AN ACCEPTABLE CONTACT POINT. SERVO ASSEMBLY FASTENER MAY BE USED.

NOTE 2: AIRFRAME PROBE CONTACT POINT SHOULD BE SELECTED AS CLOSE AS POSSIBLE TO SERVO BRACKET, BUT NOT ON THE MOUNTING BRACKET. REMOVE NON-CONDUCTIVE COATING, IF REQUIRED, TO ACHIEVE AN ACCEPTABLE CONTACT TO AIRFRAME GROUND.

NOTE 3: IF REQUIRED, THE MAXIMUM STATED IMPEDANCE VALUES MAY BE ADJUSTED PER THE FOLLOWING; AN ACCEPTABLE IMPEDANCE BOND VALUE (MAXIMUM MILLIOHMS) MAY BE DETERMINED BY MULTIPLYING THE NUMBER OF CONTACT BOND SURFACES BY 2.5 MILLIOHMS FOR EACH BOND SURFACE LOCATED BETWEEN THE PROBE CONTACT POINTS.

Rudder Bonding Check
Figure 4



CONDUCT THE FOLLOWING ELECTRICAL BONDING CHECKS:

SERVO CASE TO AIRFRAME (POINT 1 TO POINT 2) REFER TO NOTES 1 AND 2. IMPEDANCE BOND VALUE TO BE LESS THAN 2.5 MILLIOHMS. IF NOT ACHIEVABLE, REFER TO NOTE 3.

SHIELD BRAIDS TO SERVO CASE (POINT 3A TO POINT 1) REFER TO NOTE 1. IMPEDANCE BOND VALUE TO BE LESS THAN 7.5 MILLIOHMS. IF NOT ACHIEVABLE, REFER TO NOTE 3. REPEAT FOR POINT 3B TO POINT 1.

NOTE 1: SERVO PROBE CONTACT POINT SHOULD BE SELECTED ON SERVO CASE. REMOVE NON-CONDUCTIVE COATING, IF REQUIRED, TO ACHIEVE AN ACCEPTABLE CONTACT POINT. SERVO ASSEMBLY FASTENER MAY BE USED.

NOTE 2: AIRFRAME PROBE CONTACT POINT SHOULD BE SELECTED AS CLOSE AS POSSIBLE TO SERVO BRACKET, BUT NOT ON THE MOUNTING BRACKET. REMOVE NON-CONDUCTIVE COATING, IF REQUIRED, TO ACHIEVE AN ACCEPTABLE CONTACT TO AIRFRAME GROUND.

NOTE 3: IF REQUIRED, THE MAXIMUM STATED IMPEDANCE VALUES MAY BE ADJUSTED PER THE FOLLOWING; AN ACCEPTABLE IMPEDANCE BOND VALUE (MAXIMUM MILLIOHMS) MAY BE DETERMINED BY MULTIPLYING THE NUMBER OF CONTACT BOND SURFACES BY 2.5 MILLIOHMS FOR EACH BOND SURFACE LOCATED BETWEEN THE PROBE CONTACT POINTS.

Aileron Bonding Check
Figure 5

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Primary Servo Capstan (S.O.O. 6221) – Maintenance Practices

1. Removal/Installation

A. Primary Servo Capstan (Rudder, Elevator and Aileron)

For Rudder refer to Figure 1 for details.

For Elevator refer to Figure 2 for details.

For Aileron refer to Figure 3 for details.

1) Removal of Primary Servo Capstan

a) Ensure the aircraft electrical power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove locking clips and unscrew turnbuckle to release servo bridle cable tension.

c) Disconnect cable clamps.

d) Disconnect electrical connectors from servos.

e) Supporting the servo and servo capstan, remove bolts securing servo capstan bracket to aircraft structure, disengage capstan bracket from mounting.

Note Inspect bracket mounting pads and mounting structure foot print for corrosion.

2) Installation of Primary Servo Capstan

WARNING INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED.

a) Ensure the aircraft electrical power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Servo to airframe electrical bonding is maintained through direct contact between the servo and airframe mating surfaces. Prior to installation, ensure mating surfaces are clean and free of contaminants. Refer to procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data, as required.

c) Ensure capstan has been correctly rigged.

d) Mount capstan to supporting aircraft structure.

e) Torque mounting bolts to 55±5 in-lbs above running torque.

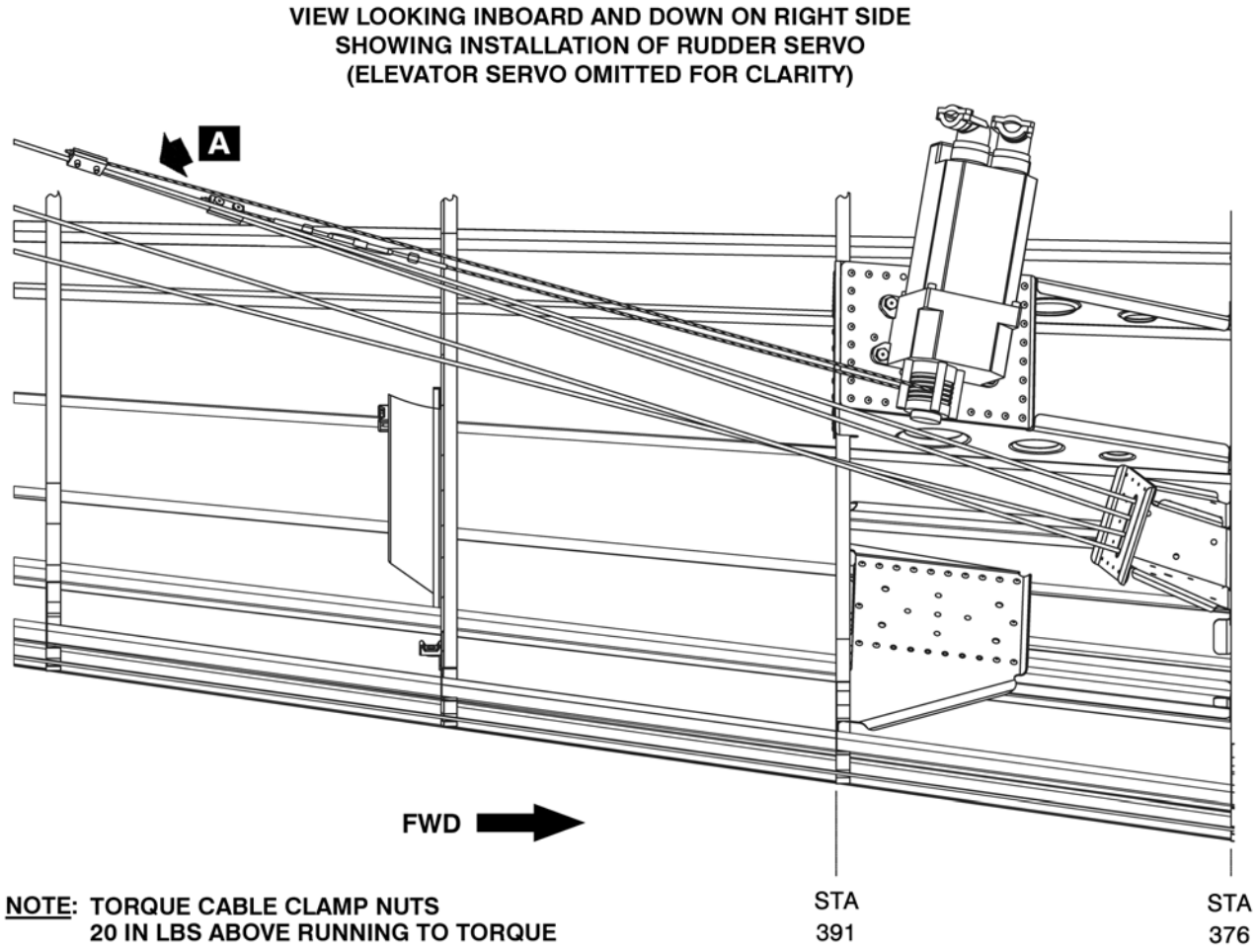
f) Connect servo cable to turnbuckle and to clamp. (Refer to Figure 1, Figure 2 and Figure 3).

g) Torque clamp nuts to 20 in-lbs above running torque.

h) Rig bridle cables in accordance with Para 2.

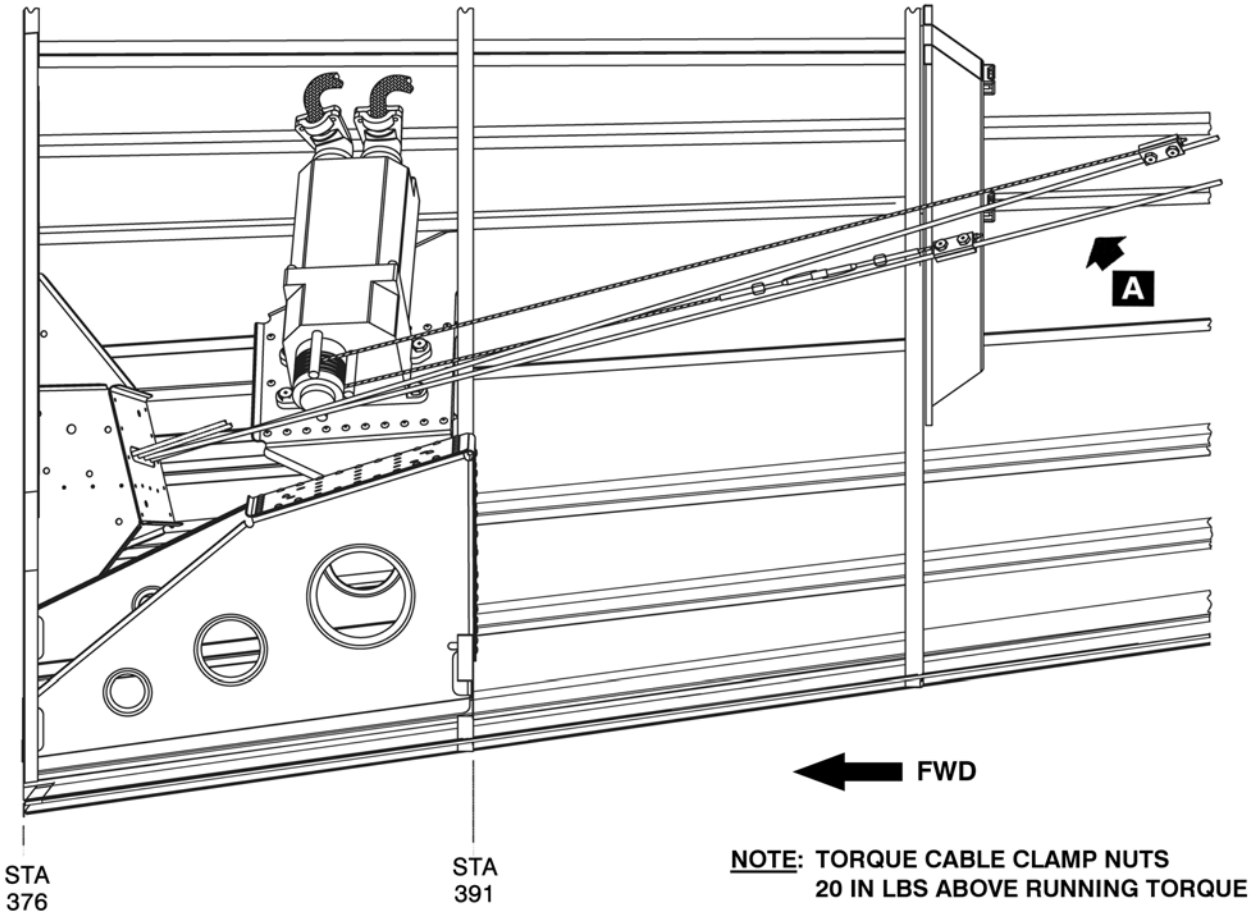
i) Connect electrical connectors to servos.

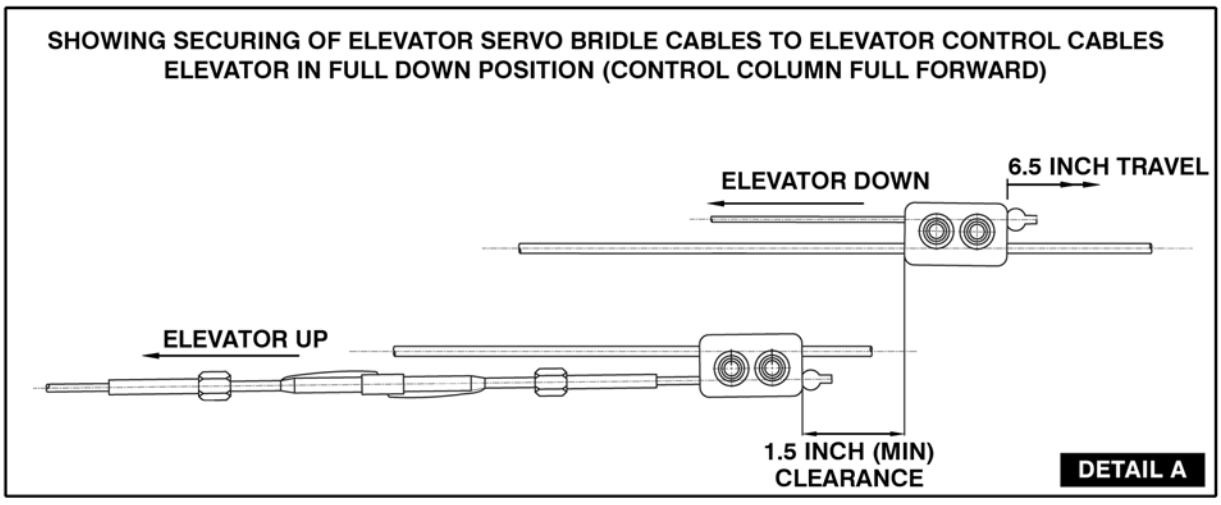
j) Do bonding check. (Refer to 22-10-11, Primary Servos (S.O.O. 6221) – Maintenance Practices, Para 3.A.).



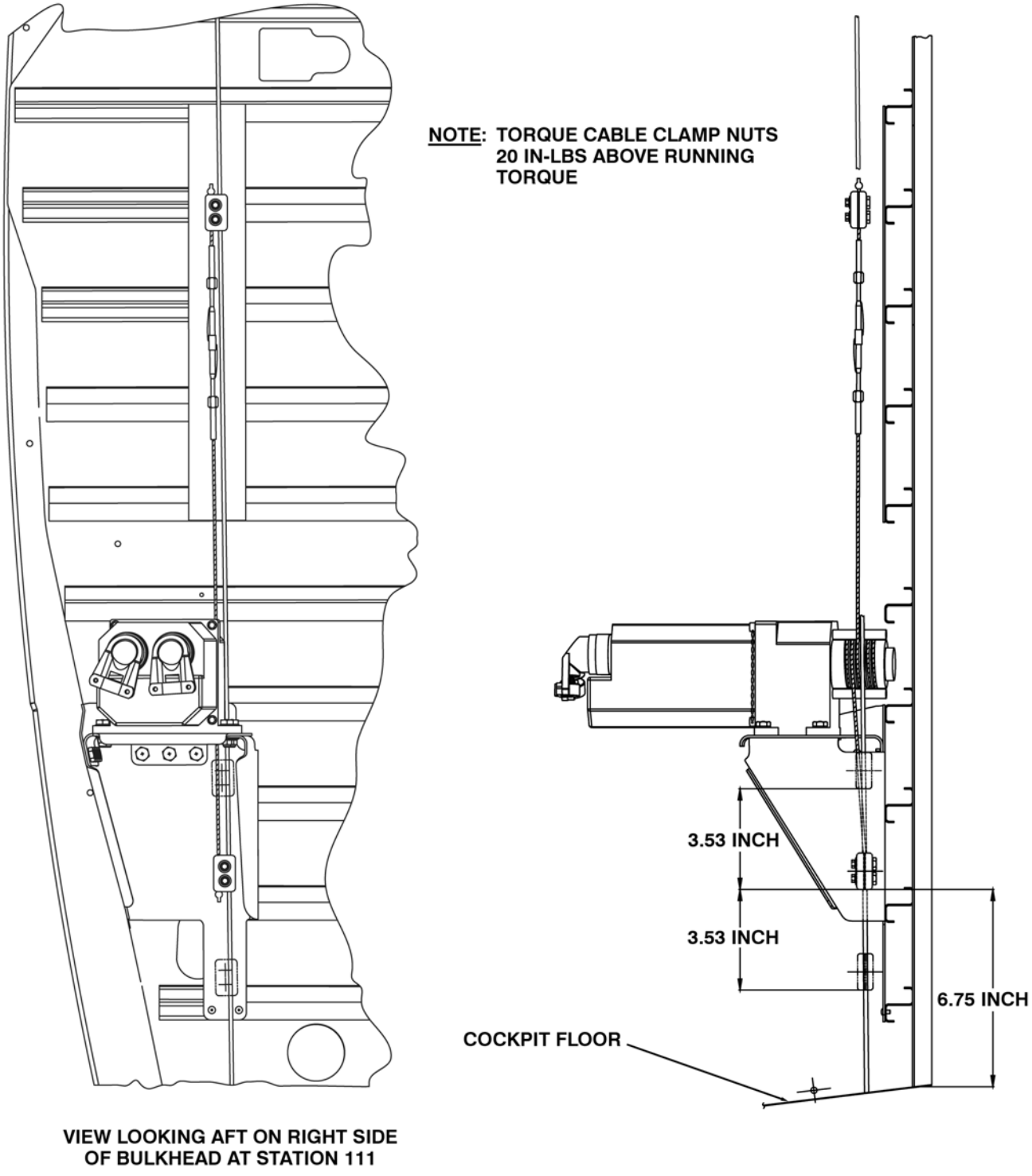
Rudder Servo Capstan Installation
Figure 1

VIEW LOOKING DOWN & INBOARD ON LEFT SIDE
SHOWING INSTALLATION OF ELEVATOR SERVO
(RUDDER SERVO OMITTED FOR CLARITY)





Elevator Servo Capstan Installation
Figure 2



Aileron Servo Capstan Installation
Figure 3

k) Function test system. (Refer to 22-10-00, Automatic Flight Controls System (AFCS) (Mod 6/2303 and S.O.O. 6221) – General Data, Para 3.C. and Para 3.F.).

l) If rudder or elevator servo has been installed, install aft baggage extension floor panels and bulkhead panel at rear of rear baggage compartment. If aileron servo has been installed, install aileron cable covers in flight compartment and install Co-pilot seat.

B. Primary Servos Bridle Cable

Refer to Figure 4 for details.

1) Removal of Primary Servo Bridle Cable from Capstan

Note The Primary Servos (Rudder, Elevator and Aileron) can be replaced without removing the capstan and disturbing the cable rigging.

a) Ensure the aircraft electrical power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove primary servo capstan bracket.

c) Disengage tabs of locking washer from cable keepers.

d) Detach cable keepers and locking washer from servo bracket. Discard locking washers.

e) Remove and retain bridle cable securing screws from capstan and detach bridle cables from capstan.

Note Inspect capstan cable grooves and bridle cables for wear.

2) Installation of Primary Servo Bridle Cable on Capstan

a) Ensure the aircraft electrical power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Install bridle cables onto capstan and engage swaged cable terminal ends with capstan anchor points on capstan.

Note To ensure cable ball is captured, use screw MPC P/N 99945680061 only.

c) Install bridle cable securing screws, torque to 5 ± 0.5 in-lbs and lock wire with .02 diameter anneal monel wire. Ensure lockwire does not extend beyond the outer diameter of the capstan.

d) Wrap each bridle cable around capstan for approximately 2 wraps.

e) Assemble locking washers onto threaded end of cable keepers and install onto capstan. Torque cable keepers to 18±2 in-lbs. Engage one tab of locking washers with cable keeper flat.

Note Gap between cable keepers and capstan shall not exceed 0.04 inch.

f) Rig in accordance with Para 2.

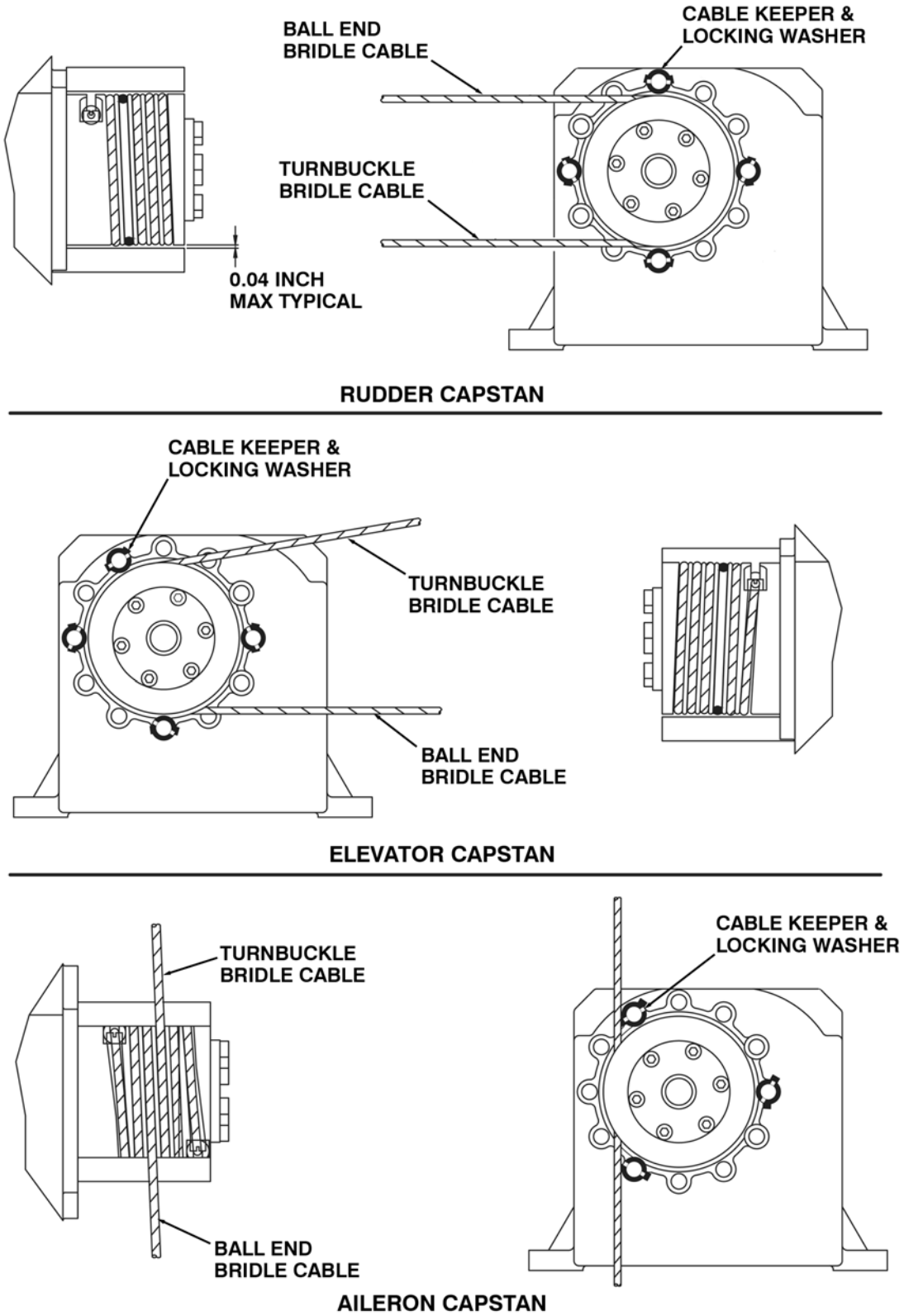
Note The aircraft must be allowed to stabilize at a temperature >55°F for a minimum of 12 hours prior to adjustment of the bridle cable tension.

Cable tension to be in accordance with Table 1 — Servo Bridle Cable Tension.

g) Check cable tensions. (Refer to 27-00-00, Flight Controls – General Data, Para 2.B.

h) Connect electrical connectors to servo.

i) Function test system. (Refer to 22-10-00, Automatic Flight Control System (Mod 6/2303 and S.O.O. 6221) – General Data, Para 3.C. and Para 3.F.).



Installation of Servo Bridle Cables
Figure 4

2. Adjustment/Test

Note Ensure aircraft conditioned at minimum temperature of 55°F (13°C).

A. Rig Rudder Servo Capstan

Refer to Figure 1 for details.

1) Ensure that the rudder control cable is correctly rigged. Refer to 27-20-00, Rudder Control System – General Data.

2) Hold rudder control in full right position.

3) Tension rudder control servo bridle cable tension to Table 1 — Servo Bridle Cable Tension. Install locking clips on turnbuckle.

4) Hold right rudder pedal in fully forward position and check that distance between the cable clamps is not less than 2 inches.

5) Operate rudder controls over full range and check that there is adequate clearance between cable clamps and structure.

B. Rig Elevator Servo Capstan

Refer to Figure 2 for details.

1) Ensure that the elevator control cable is correctly rigged. Refer to 27-30-00, Elevator Control System – General Data.

2) Hold elevator controls in full down position.

3) Tension elevator control servo bridle cable tension to Table 1 — Servo Bridle Cable Tension. Install locking clips on turnbuckle.

4) Hold at control column in full forward position and check that the distance between the cable clamps is not less than 1.5 inches.

5) Operate elevator controls over full range and check that there is adequate clearance between cable clamps and structure.

C. Rig Aileron Servo Capstan

Refer to Figure 3 for details.

1) Ensure that the aileron control cable is correctly rigged. Refer to 27-10-00, Aileron Control System – General Data.

2) Lock aileron controls in neutral position.

3) Tension aileron servo bridle cable tension to Table 1 — Servo Bridle Cable Tension. Install locking clips on turnbuckle.

4) Check that the distance between the lower edge of the servo cable clamp and floor is 6.75 inches.

5) Unlock aileron controls.

6) Operate aileron controls over full range and check that there is adequate clearance between cable clamps and structure.

Table 1 Servo Bridle Cable Tension

Item	Rudder	Elevator	Aileron
Bridle Cable Tension	25 lbs +5 lbs –0 lbs	25 lbs +5 lbs –0 lbs	15 lbs +2 lbs –0 lbs

Pitch Trim System (S.O.O. 6221) – General Data

1. General

The autopilot Pitch Trim System commands elevator trim to offload loads on the elevator servo.The pitch trim servo commands are either derived from within the AIOP of the MAU using Automatic Pitch Trim commands, or manual electric pitch trim commands derived from the Pilot and Co-pilot flight control yoke pitch trim switch and relays.

The Pitch Trim System includes the following main elements:

- Automatic Pitch Trim commands (AIOP of the MAU derived commands)
- Flight Controls Split Trim Switch (Manual Electric Pitch Trim)
- PITCH TRIM Switch installed on the pilot switch panel
- KS 272A Pitch Trim Servo (includes KM 275 capstan)

2. Description

A. Flight Controls Split Trim Switch

Trim switch marked TRIM DN and TRIM UP manually controls the pitch trim servo. The trim switch has two levers (refer to Figure 1, Detail A), trim engage and trim direction. Both levers must be engaged to actuate the trim servo. Activation of the trim engage switch while auopilot is engaged will disengage autopilot and automatic pitch trim.

B. Pitch Trim Switch

A two position pitch trim lever lock master switch labelled PITCH TRIM is located on the pilot's switch panel (refer to Figure 1, Detail B). The switch has two positions DISABLE (up) and NORMAL (down). The switch must be in the NORMAL position for the autopilot to function or for manual electronic control of the elevator trim servo. Selecting DISABLE will disconnect autopilot as well as disabling the manual electric pitch trim.

C. Pitch Trim Servo (KS 272A)

The pitch trim servo contains a DC motor and drive engagement solenoid. The pitch trim servo is mounted on and drives a capstan which actuates the elevator trim tab through servo bridle cables clamped to the trim tab control cables. Torque limiting is controlled by a slip clutch within the servo capstan.

28 Volt DC power is protected by a 5 Amp PITCH TRIM (N1) (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

3. Adjustment/Test

A. Manual Electric Pitch Trim Operational Test

1) Ensure that PITCH TRIM (N1) (Left Bus), circuit breaker on the avionics circuit breaker panel is closed.

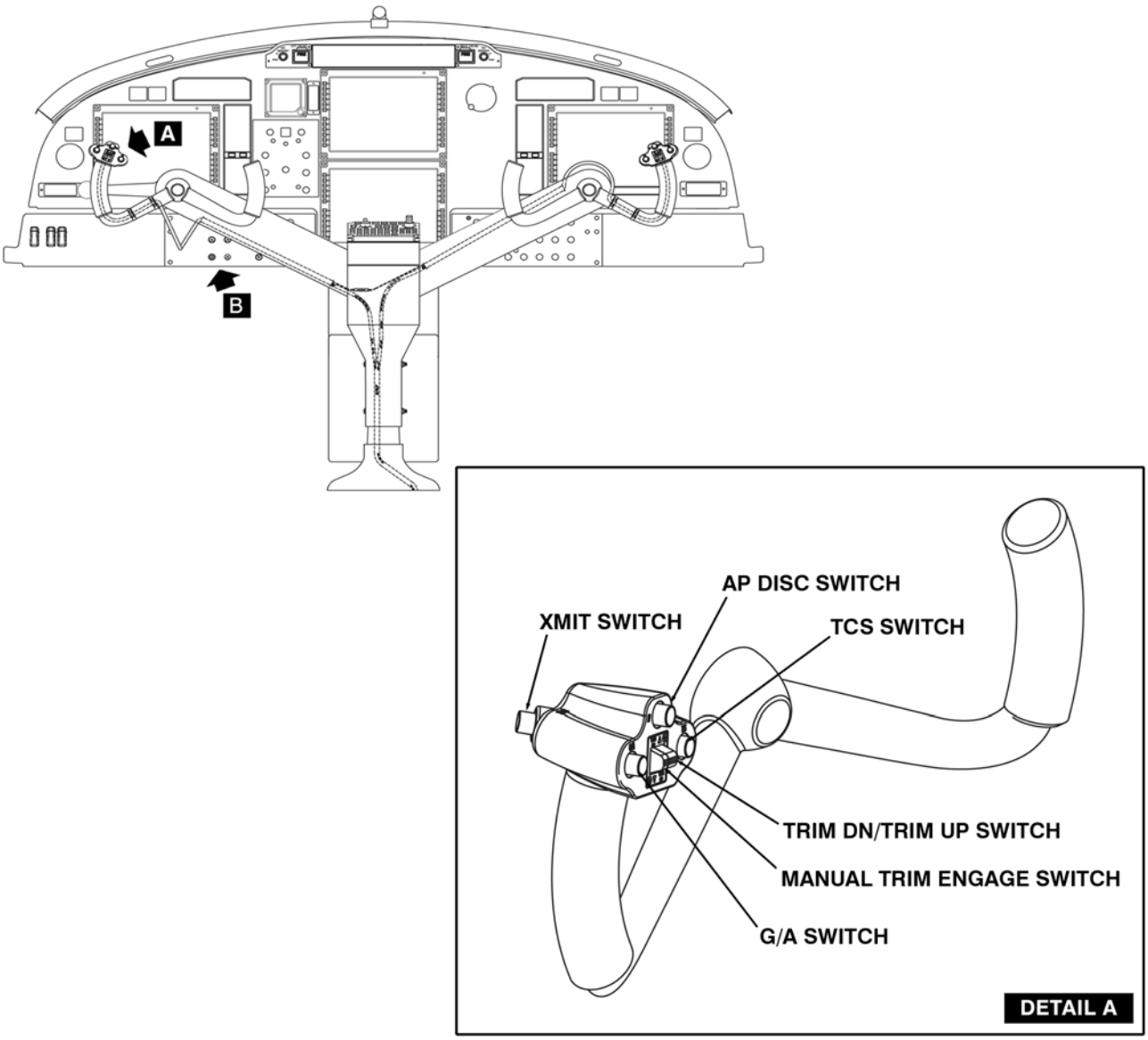
2) Ensure the PITCH TRIM switch is in the NORMAL position.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Wait several minutes for the aircraft electrical and avionics systems to initialize. During initialization a Power-on Built In Test (PBIT) will be completed on the Pitch Trim Servo System, confirm the AP 1 Fail CAS message extinguishes after the avionics system initializes.

5) Select the Pilot's control wheel switch to the TRIM UP position, confirm the trim wheel travels TRIM UP.

6) Select the Co-pilot's control wheel switch to the TRIM UP position, confirm the trim wheel travels TRIM UP.



Pitch Trim System Controls
Figure 1

7) Select the Pilot's control wheel switch to the TRIM DN position, confirm the trim wheel travels TRIM DN.

8) Select the Co-pilot's control wheel switch to the TRIM DN position, confirm the trim wheel travels TRIM DN.

9) Select and hold a control wheel trim switch to either the TRIM UP or TRIM DN position and press and hold the AP DISC simultaneously, confirm the control wheel stops. Release the AP DISC, confirm the trim wheel travel resumes. Release the trim switch.

10) Select and hold the trim switch to either the TRIM UP or TRIM DN position and select the PITCH TRIM switch to the DISABLE position, confirm the control wheel stops. Release the trim switch. Confirm the AP 1 Fail CAS message illuminates.

11) Select the PITCH TRIM switch to the NORMAL position, confirm the AP 1 Fail CAS message extinguishes.

12) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Bench Test KM 275 Capstan

CAUTION **ANY DISASSEMBLY/ASSEMBLY OF THE SERVO MUST BE DONE AT A STATIC-SAFE WORKSTATION. THE TECHNICIAN AND BENCH SHOULD BE GROUNDED.**

1) Remove the pitch trim servo from aircraft, refer to Para 4.A.1).

2) Remove the capstan from the aircraft, refer to Para 4.B.1).

3) Remove the cable guard from the capstan.

4) Put capstan on test stand.

5) Place the adapter tool over the capstan and insert the positioning pin (from the straight-up position) to secure the adapter tool, as shown in Figure 2.

6) Insert a torque wrench (Snap On P/N TEP–6FUA or equivalent) as shown in Figure 2.

7) Apply 28 VDC (1 Amp max) power to the test stand jacks.

8) Using the capstan rotation switch on the test stand, turn the test stand motor on in the clockwise direction and read the torque on the wrench. Repeat this step in the counterclockwise direction.

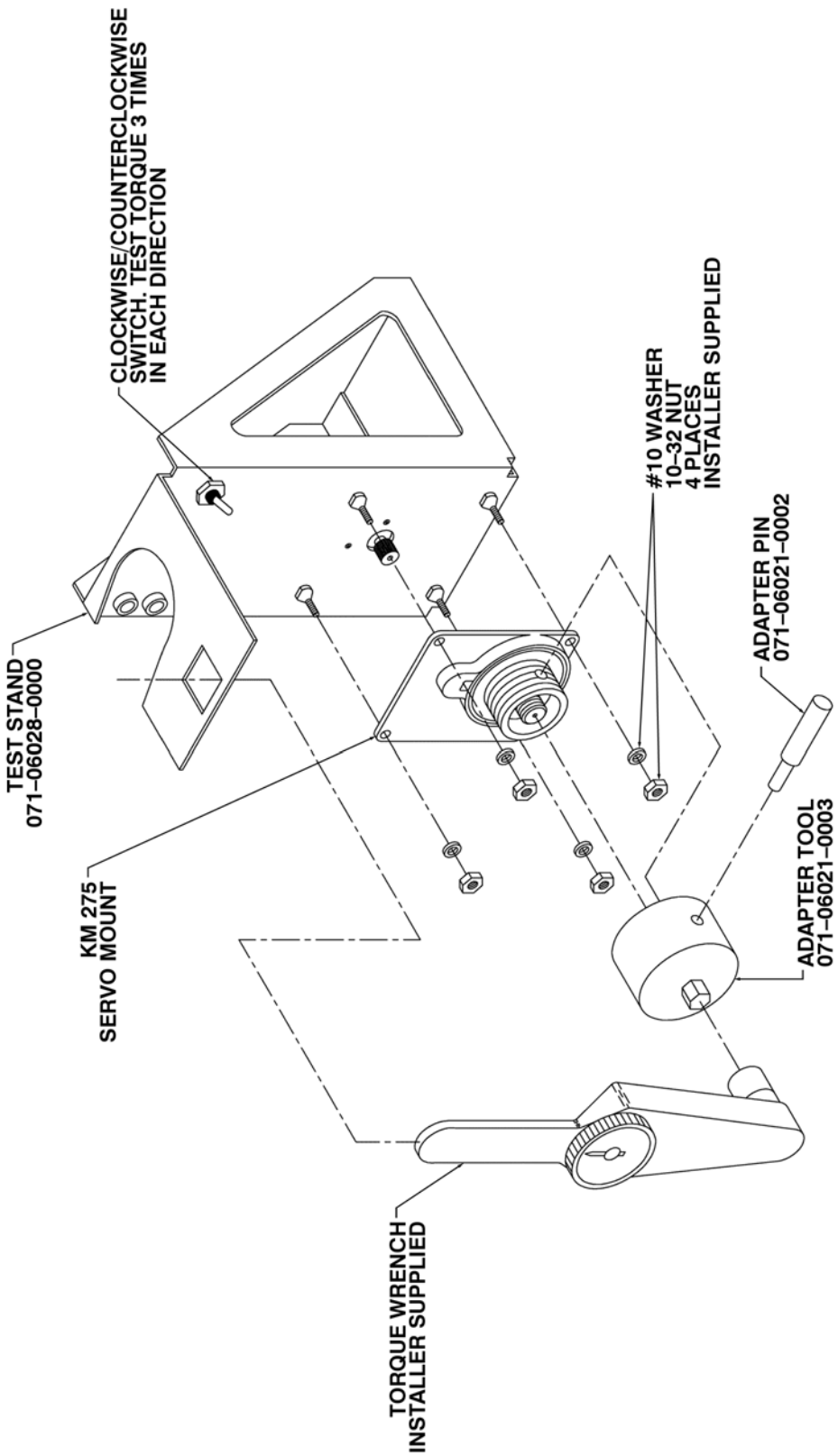
Note The desired reading is the average of the maximum and minimum readings obtained from the clockwise and counterclockwise rotations. The test is to be repeated 3 times in each direction and then average the 6 readings to determine the final torque reading. If the level determined falls below the desired value, rotate the clutch adjust nut clockwise. If the level determined falls above the desired value, rotate the clutch adjust nut counterclockwise. After any adjustment, repeat the torque test starting at Para 3.B.7).

Table 1 Trim Servo Clutch Setting

Item	Pitch Trim
Clutch Setting	22 in–lbs

9) Record the slip clutch torque reading, aircraft type, axis and date on the decal affixed to the servo mount body.

10) Reinstall the capstan, refer to Para 4.B.2).



KMC 275 Capstan Clutch Adjustment Bench Setup
Figure 2

C. Bench Test KS 272A Pitch Trim Servo

CAUTION **ANY DISASSEMBLY/ASSEMBLY OF THE SERVO MUST BE DONE AT A STATIC-SAFE WORKSTATION. THE TECHNICIAN AND BENCH SHOULD BE GROUNDED.**

1) Remove the pitch trim servo from aircraft, refer to Para 4.A.1).

2) Remove the two screws that hold the cover of the unit, and carefully slide the cover off of the wiring harness.

3) Place the servo (without cover) on the bench so that the inner parts of the unit will not be damaged.

4) Manually move the solenoid with light pressure. The plunger should move freely in and out of the solenoid body.

   Note There should be no dirt, contamination or corrosion around the solenoid plunger. This could impede the actuation of the solenoid.

5) The release spring should pull the plunger out of the solenoid and against its stop freely and without hesitation.

6) Inspect the wiring for evidence of wear or damage to the insulation. This could cause wire shorts.

7) Inspect the entire servo for any loose hardware or other abnormalities.

8) Replace the cover by carefully sliding the cover back on over the wiring harness onto the unit in its original position.

9) Install the screws using low-strength thread-locking compound such as Loctite 222 or Loctite 242.

10) Reinstall the pitch trim servo, refer to Para 4.A.2).

D. Electrical Bonding

Refer to Figure 3 for details.

Conduct the following electrical bonding check:

Note If required, the maximum stated impedance values may be adjusted per the following: an acceptable impedance bond value (maximum milliohms) may be determined by multiplying the number of contact bond surfaces by 2.5 milliohms for each bond surface located between the probe contact points.

1) Servo Case to Airframe (Point 1 to Point 2)

   Remove non-conductive coating, if required, to achieve an acceptable contact point. Impedance bond value to be less than 2.5 milliohms. If not achievable, refer to Note.

2) Ground Module and Wire Shielding

   a) Ensure all DC power is disconnected (refer to 12-00-00, Servicing - General Data, Para 2.).

   b) Disconnect aircraft connector P278 from KS 272A Trim Servo wiring pigtail.

   c) If required, insert contact P/N M39029/31–228 or equivalent into P278 pin H for milliohmmeter attachment test point. Measure aircraft side of wiring harness between P278 pin H and airframe (point 2), impedance value to be less than 20 milliohms.

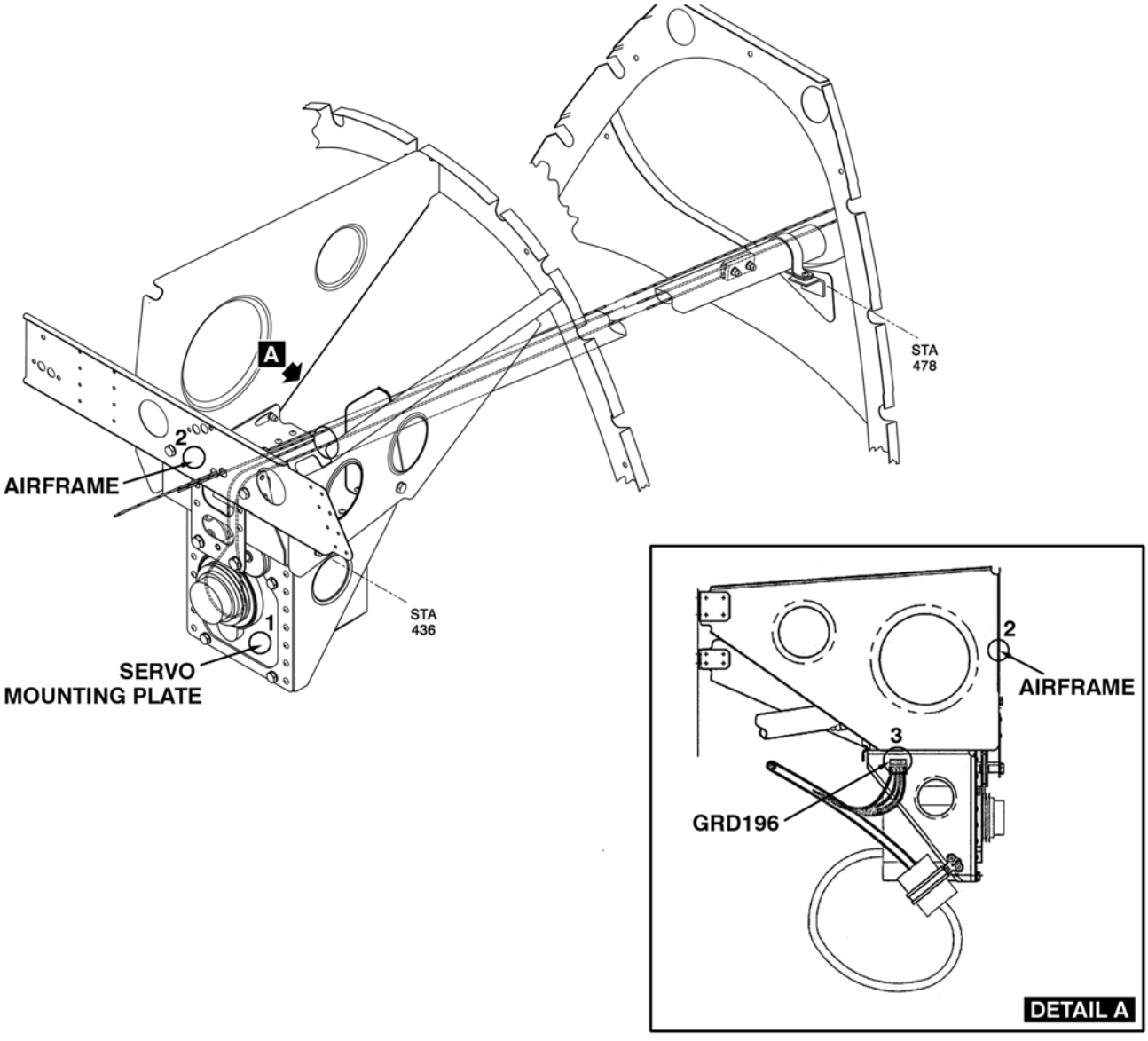
   d) If required, insert contact P/N M39029/31–228 or equivalent into P278 pin J for milliohmmeter attachment test point. Measure aircraft side of wiring harness between P278 pin J and airframe (point 2), impedance value to be less than 20 milliohms.

e) At wiring harness near P278, visually inspect all shield jumper wires between solder sleeves. Ensure all wires are secure in the solder sleeves.

f) At wiring harness near P278, visually inspect all shield drain wires between wiring harness P278 and GRD189. Ensure all shield drain wires are secure in the solder sleeves and GRD189.

g) Reconnect KS 272A Trim Servo connector P278 to KS 272A Trim Servo wiring pigtail.

h) Do Manual Electric Pitch Trim Operation Test, refer to Para 3.A.



Pitch Trim Servo Bonding Check
Figure 3

4. Removal/Installation

A. Pitch Trim Servo (KS 272A)

Refer to Figure 4 for details.

Note The KS 272A servo can be replaced without removing the capstan and disturbing the cable rigging.

1) Removal of the KS 272A Pitch Trim Servo

a) Ensure aircraft power is OFF. (Refer to 24-40-00, External Power – General Data).

b) Gain access to servo by removing bulkhead panel at rear of baggage compartment.

c) Disconnect servo electrical connector from receptacle.

d) Remove the two (2) bolts holding the pitch trim servo to the support bracket.

e) Holding the pitch trim servo securely, disengage the servo from the capstan.

f) Remove pitch trim servo from aircraft.

2) Installation of the KS 272A Pitch Trim Servo

WARNING INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED.

a) Ensure aircraft power is OFF. (Refer to 24-40-00, External Power – General Data).

b) Servo to airframe electrical bonding is maintained through direct contact between the servo and airframe mating surfaces. Prior to installation, ensure mating surfaces are clean and free of contaminants. As required, refer to procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures –General data.

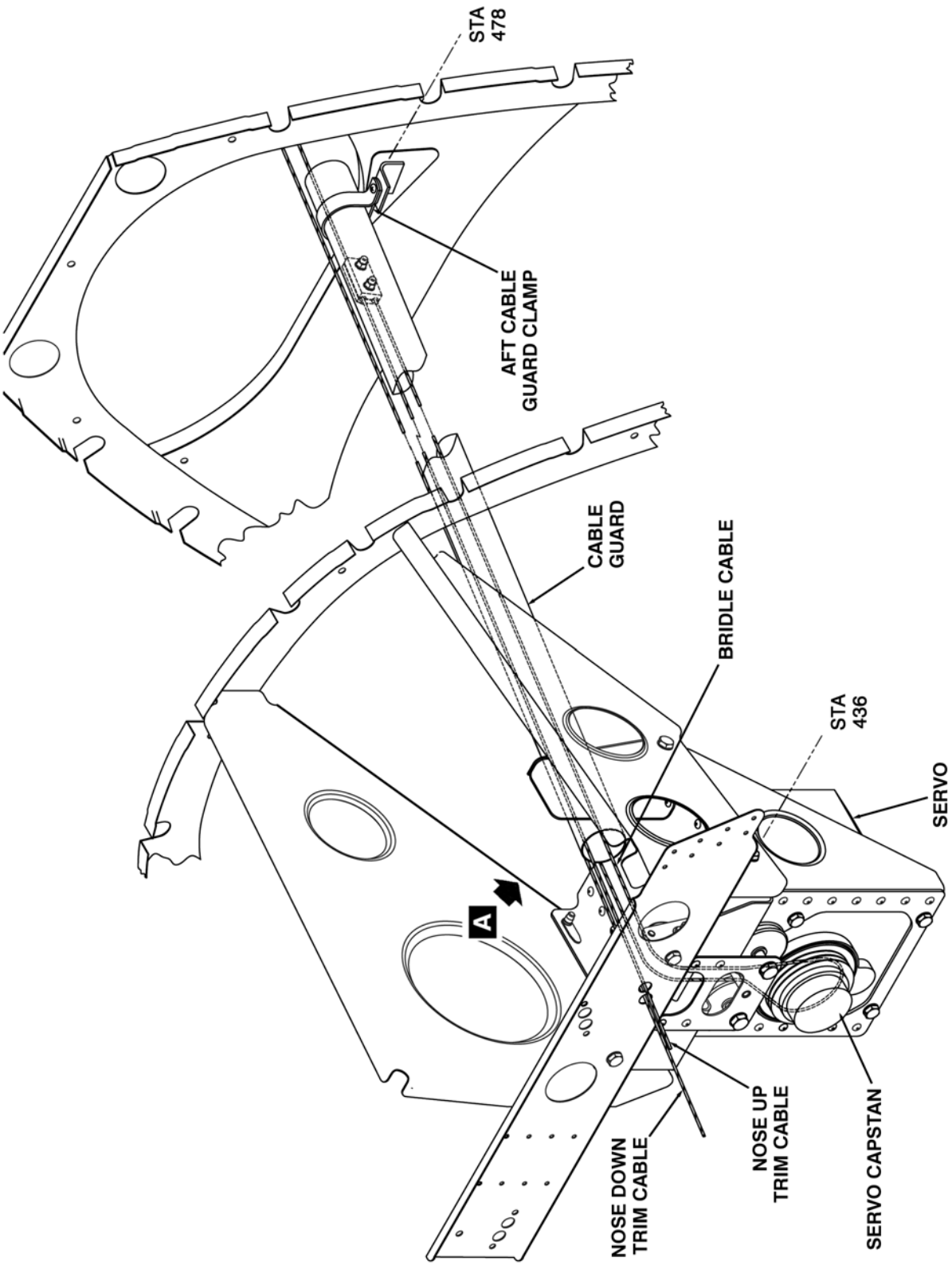
c) Engage servo drive shaft pinion with the teeth of the capstan gear.

d) Install the two (2) bolts to secure the servo to the support bracket.

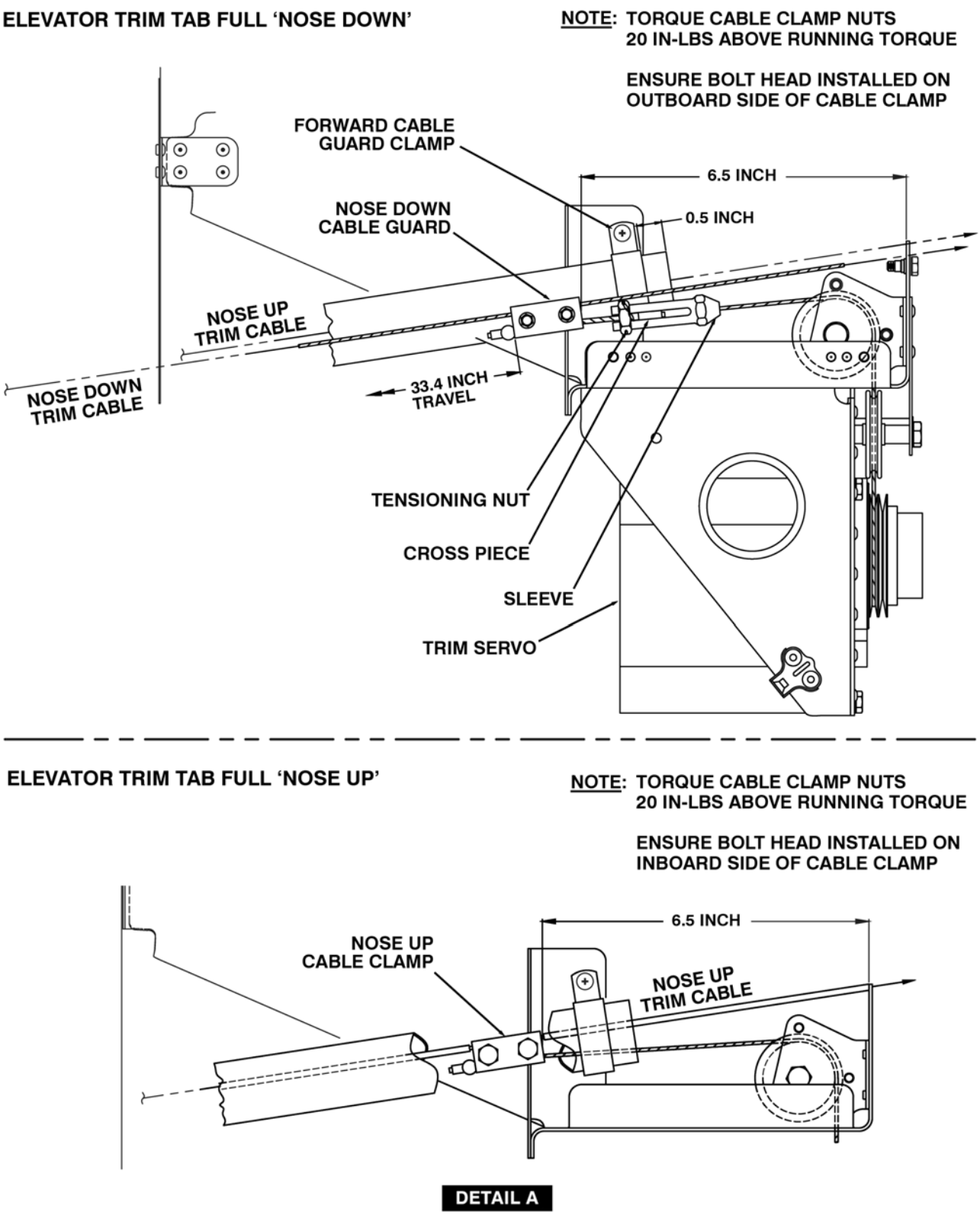
e) Connect servo electrical connector to the receptacle, and ensure the locking clips are engaged.

f) Do bonding check, refer to Para 3.D.

g) Function test system, refer to Para 3.A.



Pitch Trim Servo Installation
Figure 4 (Sheet 1 of 2)



Pitch Trim Servo Installation
Figure 4 (Sheet 2 of 2)

B. Capstan (KM 275)

1) Removal of the KM 275 Capstan

a) Ensure aircraft power is OFF. (Refer to 24-40-00, External Power – General Data).

b) Remove the pitch trim servo, refer to Para 4.A.1).

c) Remove the cable guard from capstan pulley.

d) Remove locking wire from bridle cable tensioning nut and cross piece. Unscrew nut and disconnect cross piece from sleeve. Ensure enough slack in bridle cable to disengage cable from capstan.

e) Support capstan and remove the two (2) bolts securing the capstan to support bracket.

f) Remove capstan from aircraft.

2) Installation of the KM 275 Capstan

a) Ensure pitch trim capstan slip clutch has been calibrated in accordance with Para 3.B.

b) Ensure aircraft power is OFF. (Refer to 24-40-00, External Power – General Data).

c) Install the two (2) bolts securing the capstan to the support bracket.

d) Install servo bridle cable, refer to Para 4.C.2).

C. Pitch Trim Servo Bridle Cable

1) Removal of the Pitch Trim Servo Bridle Cable

a) Ensure aircraft power is OFF. (Refer to 24-40-00, External Power – General Data).

b) With elevator trim tab in full nose down position, remove locking wiring from bridle cable tensioning sleeve and cross piece. Unscrew tensioning nut and disconnect cross piece from sleeve.

c) Disconnect ‘nose down’ cable clamp.

d) With elevator trim tab in full nose up position, loosen forward and aft cable guard clamps and slide cable guard aft. Remove ‘nose up’ cable clamp.

e) Tension elevator trim cable. (Refer to 27-33-00, Elevator Trim Tab System – General Data).

f) Reposition cable guard and tighten securing clamps.

2) Installation of the Pitch Trim Servo Bridle Cable

a) Ensure aircraft power is OFF. (Refer to 24-40-00, External Power – General Data).

b) Loosen forward and aft cable guard clamps and slide cable guard aft.

c) Loop bridle cable around aft capstan pulley and clamp bridle cable to section of nose up trim cable with nylon sleeving removed. Torque clamp nuts to 20 in-lbs above running torque.

d) Reposition cable guard and tighten securing clamps.

e) Clamp bridle cable to nose down trim cable.

f) Ensure that the pitch trim control cables are correctly rigged. (Refer to 27-33-00, Elevator Trim Tab System – General Data).

g) Lock elevator controls in the neutral position.

h) Tension trim bridle cables in accordance with Table 2 — Trim Servo Bridle Cable Tension.

Note The aircraft must be allowed to stabilize at a temperature >55°F for a minimimum of 12 hours prior to adjustment of the bridle cable tension.

Table 2 Trim Servo Bridle Cable Tension

Item	Pitch Trim
Bridle Cable Tension	15 lbs + 2 lbs – 0 lbs

i) Lockwire tensioning nut and cross piece.

j) Install capstan cable guard.

k) Unlock elevator controls.

l) Manually cycle the electric pitch trim, ensure proper operation.

m) Function test system, refer to Para 3.A.

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CHAPTER 23

COMMUNICATIONS

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CHAPTER 23 – COMMUNICATIONS

LIST OF EFFECTIVE PAGES

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Communications – General Data

1. General

The communications system comprise of two Very High Frequency Communication (VHF COM) systems, two VHF COM antenna's, and the pilot's and co-pilot's headset communication system. The VHF COM systems are integrated transceivers of the Multi-Mode Digital Radio (MMDR). Provisions have also been made for HF Communications system and for passenger address system, if installed (refer to 44-00-00, Cabin Systems – General Data).

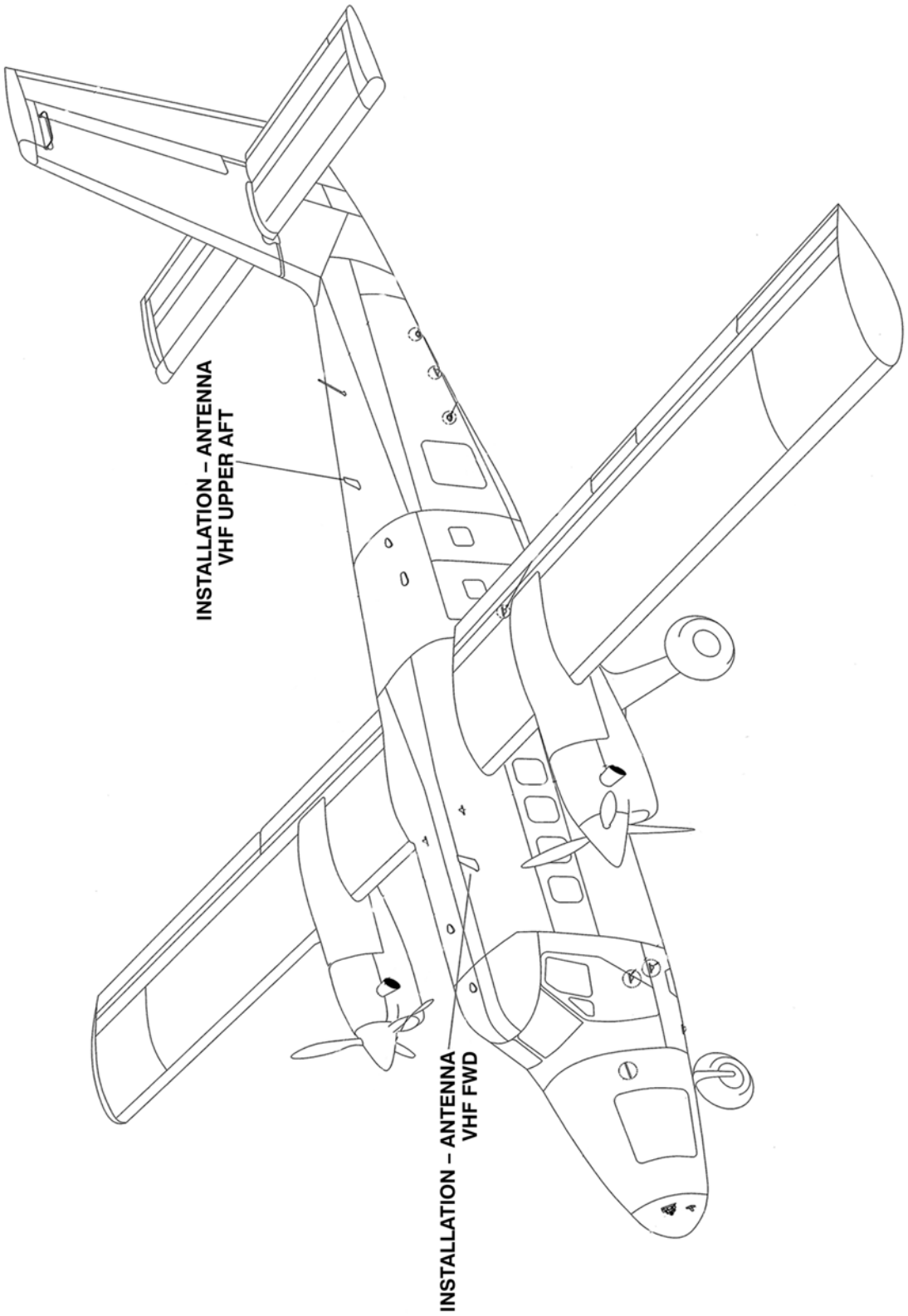
The communications systems are supplied with 28 Volt DC from the aircraft electrical system through the circuit breakers refer to PSM 1-64-2W for electrical details.

2. Antenna

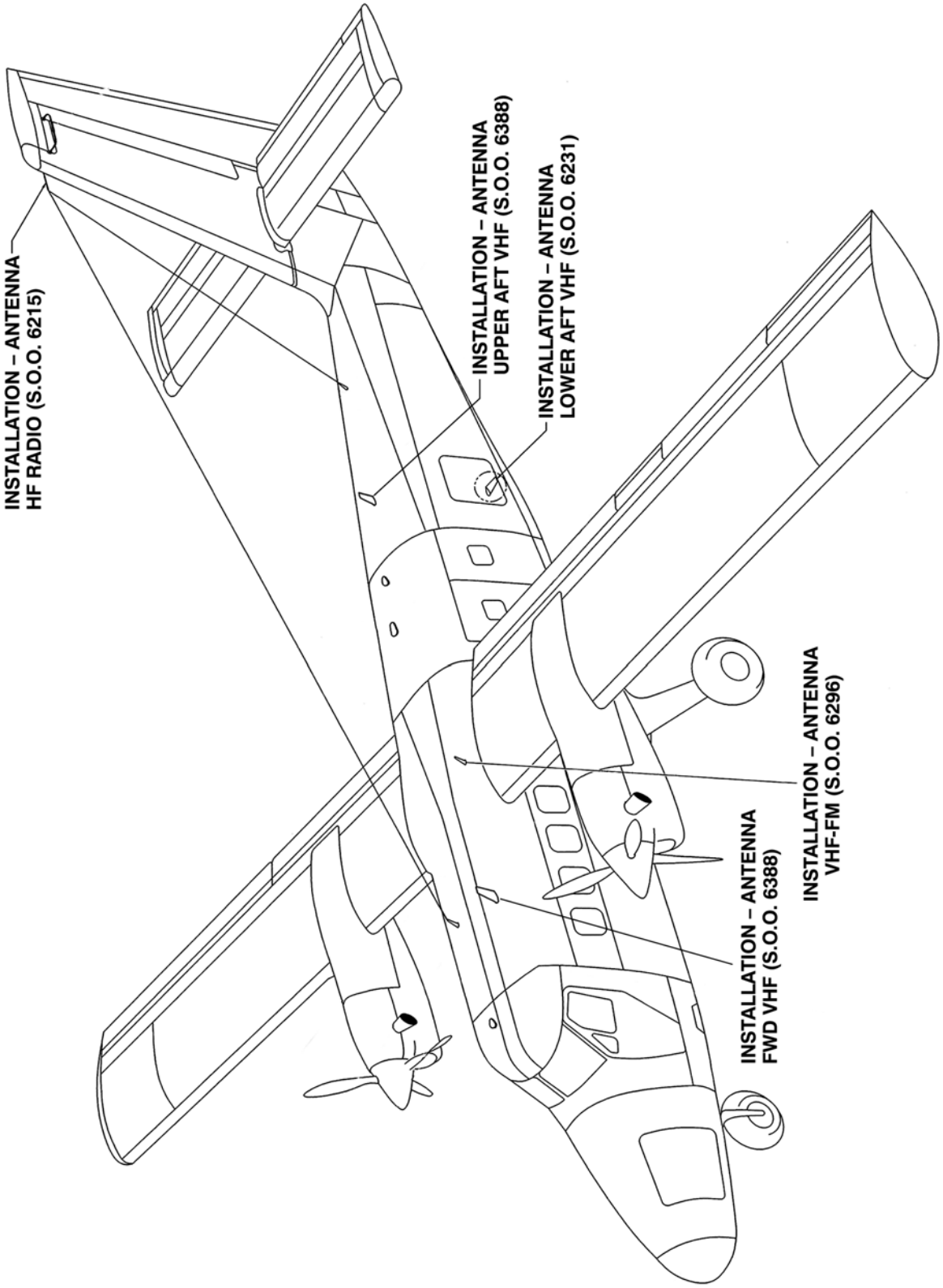
Location for the Communication antennas are listed in Table 1 — Antenna Locations.

Table 1 Antenna Locations

ITEM	ANTENNA	PRODUCT	FUNCTION	TYPE
1	VHF Upper Aft	Comant CI–108–1–L	Main VHF communications system – Provides air-to-air and/or air-to-ground voice (AM) in the VHF band	
2	VHF Forward	Comant CI–108–1–L	Main VHF communications system – Provides air-to-air and/or air-to-ground voice in the VHF band	
3	VHF Upper Aft	Comant CI–108–1–L	S.O.O. 6231 Alternate VHF communications system – Provides air-to-air and/or air-to-ground voice in the VHF band	
4	HF Radio with Coupler		S.O.O. 6215 HF communications system – Provides air-to-air and/or air-to-ground voice in the VHF band	
5	VHF-FM	Comant CI–295–200	S.O.O. 6296 VHF-FM – Provides voice communications in the 138.000 to 174.000 MHz frequency range of the VHF band.	



General Arrangement – Avionics Antennas
Figure 1



General Arrangement – Customer Option Antennas
Figure 2

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Speech Communications – General Data

1. General

The speech communication system consists of a dual VHF communication system (refer to 23-12-00, VHF Communication System – General data), a HF communication system (refer to 23-11-00, HF Communication System (S.O.O. 6215) – General data) (if installed), and a two-way telephone (refer to 23-15-00, Satellite Communication System (S.O.O. 6298) – General Data) (if installed). The VHF communication system is an integral part of the Multi-Mode Digital Radio (MMDR), (refer to 34-51-00, Multi-Mode Digital Radio – General data).

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HF Communication System (S.O.O. 6215) – General data

1. General

The High Frequency (HF) transceiver system is used for long distance voice and data communication to other aircraft and/or ground stations.

2. Description

A. High Frequency (HF) Communication System

The HF radio is controlled by a PS 440 control head that is mounted on the instrument panel, directly below the clock, on the right hand side of the multi-function displays. The audio functions of the HF radio are fully integrated into the KMA 29 audio control panel. In all other respects, the HF radio is entirely independent of the Apex avionics systems.

The KHF 1050 HF Communication System (also known as a Primus HF 1050) consists of three individual units: a KAC 1052 Antenna Coupler, a KPA 1052 Power Amplifier, and a KRX 1053 Receiver/Exciter. All three of these are installed in the aft avionics racks, between stations 406 and 451. Receiver and low level transmitter signals are located in the Receiver/Exciter. The Receiver/Exciter employs a master oscillator, which requires a short warm-up period when operating at cool temperatures.

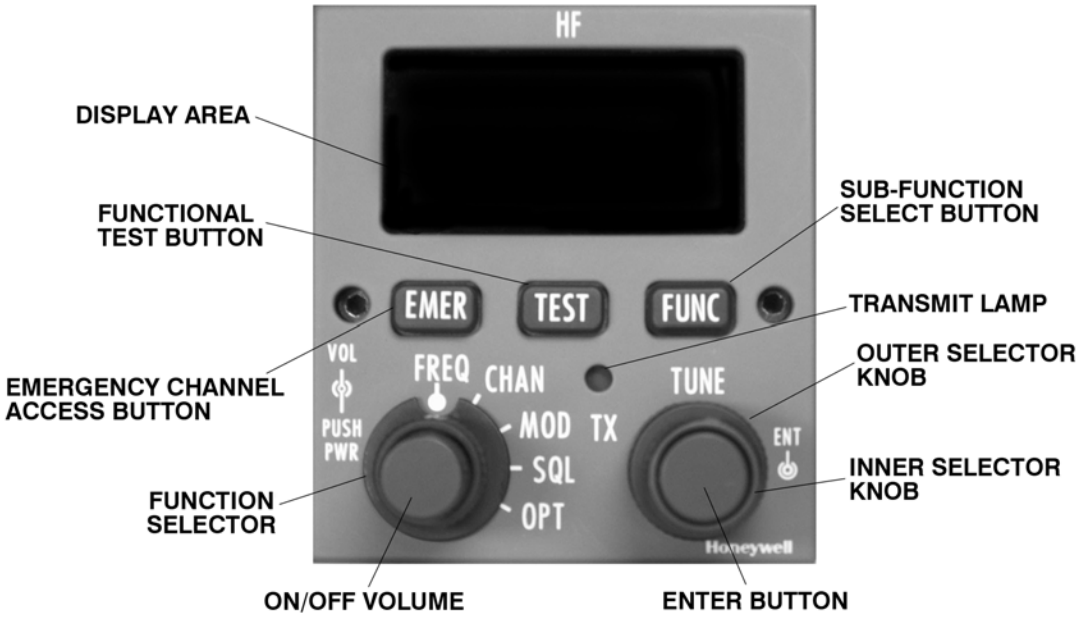
The Power Amplifier amplifies the excitation signal from the Receiver/Exciter to 200 watts peak effective power, or 50 watts of carrier power. The amplified signal is then routed to the Antenna Coupler, which matches the various impedances of the inverted V wire antenna to the 50 ohm output of the transmitter. When in receive mode, the signals from the antenna pass through the Antenna Coupler to the Receiver/Exciter via the Power Amplifier.

The inverted V antenna configuration receives and transmits signals in the frequency band from 2 MHz to 29.9999 MHz. The antenna is connected at the antenna coupler, then to an isolator at the top of the vertical stabilizer. It then connects to an antenna mast mounted on top of the aircraft.

28 Volt DC power is protected by a 5 amp RX/EX ANT CPLR (2S) (Hot Battery Bus), a 30 amp POWER AMP (2R) (Hot Battery Bus), and a 2 amp HF CTRL (4P) (Right Bus), circuit breakers located either on the avionics circuit breaker panel or on the rear circuit breaker panel located aft side of station 332.00 on the co-pilot's side (refer to 24-00-00, Electrical Power – General Data).

The HF (KHF 1050) transceiver interfaces to the following aircraft systems:

1) Dual KMA 29 Audio Control Panel. (Refer to 23-53-00, Audio Control Panel – General Data).



HF Controller – Layout
Figure 1

3. Adjustment/Test

A. Function Test HF (KHF 2100) Transceiver

1) General

These procedures are for the function test of the HF (KHF 2100) transceiver.

CAUTION **SOME ELECTRICAL POWER UNITS (EPU) MAY REGULATE POORLY WHEN SUBJECTED TO STRONG RF FIELDS. IF FLUCTUATIONS IN THE EPU ARE NOTED WHEN TRANSMITTING, DISCONTINUE USING THE EPU. WHEN AN EPU CANNOT BE USED, AN ALTERNATIVE METHOD OF TESTING THE HF IS WITH ENGINES RUNNING AND GENERATORS ON.**

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that the RX/EX ANT CPLR (2S), POWER AMP (2R) and the HF CTRL circuit breakers are closed.

d) On the HF controller, press the TEST button and ensure no fault codes are displayed.

e) On the HF controller, push the PWR inner concentric knob.

f) Tune to a known HF communications station. Ensure audio can be heard clearly. Adjust the volume to an acceptable level.

g) Select to a channel in which the aircraft is authorized to transmit.

h) Momentarily key the HF transceiver. Ensure the TX indicator stops flashing when the tuning process is complete.

i) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – HF Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to empennage through rear baggage compartment.

3) Locate HF Antenna aft of station 451.00 on the ceiling of the aircraft.

4) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the HF Feed Thru Assembly metal casing with the second probe on the HF ground strap, GRD 134. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

5) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

6) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structues – General data.

C. HF Coupler (KAC 1052) Pressurization Test

1) General

The KAC 1052 is pressurized with nitrogen. On an annual basis, the pressure of the KAC 1052 must be measured by using a voltmeter, a precision gauge or by the KHF 1052 Maintenance Program (if available).

2) Pressurization Test

a) Pressurization Test Procedure – Using Voltmeter

Note Pressure reading can differ by up to 0.375 volts (3 psi) due to the effects of temperature and barometric pressure at the time the readings are taken. Average multiple readings result in higher accuracy.

Note In the event that the measured voltage between Pin 7(+) and Pin 4(–) drops to 2.5 Volt DC or less between annual maintenance check, (pressure leaked to 4 psi or less), more frequent monitoring of the pressure is advised.

Note If rapid pressure loss persists, remove the HF Coupler (KAC 1052) unit from the plane and check for leaks.

1 Remove the protective cover from TEST J10522 port in front of the HF Coupler (KAC 1052).

2 Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3 Connect the voltmeter, the positive (+) lead to P10522 Pin 7 and the negative lead (–) to P10522 Pin 4. A voltmeter reading of 3.0 to 3.3 Volt DC indicates that the HF Coupler (KAC 1052) pressure is acceptable (between 8 to 10 psi).

4 If the pressure reads below 8 psi, charge the KAC 1052 unit with nitrogen through the Pressure Input and Measurement Valve to bring the pressure up to between 8 to 10 psi.

5 Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

6 Replace the protective cover on TEST J10522 port in front of the HF Coupler (KAC 1052).

7 Ensure protective cover is secure.

8 Visually inspect HF Coupler (KAC 1052) connectors to ensure they are sound and secure.

b) Pressurization Test Procedure – Using Precision Gauge

Note Pressure readings can differ by up to 3 psi due to the effects of temperature and barometric pressure at the time the readings are taken. Average multiple readings result in higher accuracy.

Note In the event that the measured pressure is 4 psi or less between annual maintenance check, more frequent monitoring of the pressure is advised.

Note If rapid pressure loss persists, remove the HF Coupler (KAC 1052) unit from the plane and check for leaks.

1 Connect the precision gauge on the Pressure Input and Measurement valve adjacent to J10522. A reading of 8 to 10 psi on the gauge indicates that the HF Coupler (KAC 1052) pressure is acceptable.

2 If the pressure reads below 8 psi, charge the KAC 1052 unit with nitrogen through the Pressure Input and Measurement valve to bring the pressure up to between 8 to 10 psi.

3 Remove the precision gauge.

c) Pressurization Test Procedure – Using KHF 1052 Maintenance Program

Note The KHF 1052 Maintenance Program is an additional software diagnostic program for the HF 1050 HF System, and is made available through Honeywell. Refer to Honeywell System Installation Manual – KHF 1050 HF Communication System, Manual No. 006-10640-0006, Rev 6 Nov/2007 or later for more details.

Note Pressure readings can differ by up to 3 psi due to the effects of temperature and barometric pressure at the time the readings are taken. Average multiple readings result in higher accuracy.

Note In the event that the measured pressure is 4 psi or less between annual maintenance check, more frequent monitoring of the pressure is advised.

Note If rapid pressure loss persists, remove the HF Coupler (KAC 1052) unit from the plane and check for leaks.

1 Remove the protective cover from TEST J10522 port in front of the HF Coupler (KAC 1052).

2 Connect the KAC 1052 to the Ground Equipment Laptop as instructed Honeywell System Installation Manual – KHF 1050 HF Communication System, Manual No. 006-10640-0006, Rev 6 Nov/2007 or later.

3 Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4 Access the ACP Maintenance page within the KHF 1050 Maintenance Program. Internal pressure is displayed in the upper right portion of the screen.

5 Confirm that the HF Coupler Pressure (KAC 1052) reading is between 8 to 10 psi.

6 To obtain multiple readings, click on Gauge Pressure, and repeat at desired intervals.

7 If the pressure reads below 8 psi, charge the KAC 1052 unit with nitrogen through the Pressure Input and Measurement valve to bring the pressure up to between 8 to 10 psi.

8 Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

9 Disconnect the KAC 1052 from the ground Equipment Laptop.

10 Replace the protective cover on TEST J10522 port in front of the HF Coupler (KAC 1052).

11 Ensure cover is secure.

12 Verify all cable connections from the HF Coupler (KAC 1052) are sound and secure.

4. Removal/Installation

WARNING INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LINE REPLACEABLE UNIT (LRU) MOUNTING TRAY OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W. BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the HF Controller (PS–440)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the HF CTRL (4P) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Release the two fasteners on the controller and remove the controller from the panel.

d) Disconnect the electrical connector from the controller.

Note Stow connectors as required.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the HF CTRL (4P) circuit breaker on the avionics circuit breaker panel is opened and fitted with safety clip.

c) Connect the electrical connector to the controller.

d) Slide the controller into the instrument panel.

e) Tighten the two fasteners on the controller.

f) Remove safety clip and close HF CTRL (4P) circuit breaker on the avionics circuit breaker panel.

g) Test the HF transceiver as per Para 3.A.

B. Removal and Installation of the Receiver/Exciter (KRX 1053)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the HF RX/EX ANT CPLR (2S) circuit breaker on the rear circuit breaker panel and fit safety clip.

c) Gain access to the rear avionics shelves through the rear baggage compartment.

d) Disconnect the electrical connector from the receiver/exciter.

e) Loosen retaining screw clamp from the front and slide the unit out of the mounting tray.

f) Remove unit from aircraft.

Note Stow connectors as required.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the HF RX/EX ANT CPLR (2S) circuit breaker on the rear circuit breaker panel is opened and fitted with safety clip.

c) Slide the unit into the mounting tray. Tighten the retaining screw clamp at the front lip of the receiver/exciter unit.

d) Reconnect the electrical connector to the receiver/exciter unit. Ensure connections are secure.

e) Remove safety clip and close HF RX/EX ANT CPLR (2S) circuit breaker on rear circuit breaker panel.

f) Test the HF transceiver as per Para 3.A.

C. Removal and Installation of the HF Antenna Coupler (KAC 1052)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the HF RX/EX ANT CPLR (2S) circuit breaker on the rear circuit breaker panel and fit safety clip.

c) Gain access to the rear avionics shelves through the rear baggage compartment.

d) Remove the nut and washer and disconnect the bonding cable from the antenna coupler.

e) Disconnect the electrical connector from the antenna coupler.

f) Loosen the two retaining screw clamps from the front and slide the unit out of the mounting tray.

g) Remove unit from aircraft.

Note Stow connectors as required.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the HF RX/EX ANT CPLR (2S) circuit breaker on the rear circuit breaker panel is opened and fitted with safety clip.

c) Slide the unit into the mounting tray. Tighten the two retaining screw clamps at the front lip of the antenna coupler unit.

d) Connect the bonding cable to the antenna coupler and tighten the nut and washer.

e) Reconnect the electrical connector to the antenna coupler unit. Ensure connections are secure.

f) Remove safety clip and close HF RX/EX ANT CPLR (2S) circuit breaker on the rear circuit breaker panel.

g) Test the HF transceiver as per Para 3.A.

D. Removal and Installation of the HF Power Amplifier (KPA 1052)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the POWER AMP (2R) circuit breaker on the rear circuit breaker panel and fit safety clip.

c) Gain access to the rear avionics shelves through the rear baggage compartment.

d) Disconnect the electrical connector from the power amplifier.

e) Loosen retaining screw clamp from the front and slide the unit out of the mounting tray.

f) Remove unit from aircraft.

Note Stow connectors as required.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the POWER AMP (2R) circuit breaker on the rear circuit breaker panel is opened and fitted with safety clip.

c) Slide the unit into the mounting tray. Tighten the retaining screw clamp at the front lip of the power amplifier unit.

d) Reconnect the electrical connector to the power amplifier unit. Ensure connections are secure.

e) Remove safety clip and close POWER AMP (2R) circuit breaker on the rear circuit breaker panel.

f) Test the HF transceiver as per Para 3.A.

E. Removal and Installation of the HF Antenna Assembly (Mod 6/2368)

1) Removal

Note Removal of the wire antenna can only be accomplished by cutting and discarding the wire.

a) Due to shock hazard from the HF transmitter, ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the RX/EX ANT ANT CPLR (2S), POWER AMP (2R), HF CTRL (4P) circuit breakers and fit safety clips.

c) Cut the wire, not less than 2 inches from the fitting and remove the brown polyethylene, exposing the wire gripped by the jaws.

d) Slide the extractor (Dayton Granger tool part number 14053 of 50ARM300–250) onto the bared wire aircraft and push the extractor against the jaws to release the antenna cable. Repeat the procedure at the opposite end of the wire. Discard the wire.

e) Disconnect the co-axial connector from the underside of the antenna feedthrough.

Note Stow connectors as required.

f) Remove the antenna cable feed-thru assemblies screws (6) and washers (6).

g) Remove the mounting assembly from the vertical stabilizer bracket.

2) Installation

a) Due to shock hazard from the HF transmitter, ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the RX/EX ANT ANT CPLR (2S), POWER AMP (2R), HF CTRL (4P) circuit breakers are opened and fitted with safety clips.

c) Install the vertical stabilizer bracket mounting assembly with the antenna wire fed through.

d) Mount the antenna cable feed-thru and tighten the screws (6).

e) Carefully inspect the antenna wire for damaged insulation. Straighten the wire carefully to eliminate all kinks.

Note Handle the wire carefully to keep it free of metallic particles or similar material that might become imbedded in the wire and thereby reduce dielectric strength.

f) Cut the wire to approximate length, allowing a few extra inches for later trimming to exact length.

g) Use Dayton Granger tool part number 14053 or 50ARM300–250. Insert the insulated wire into the tool until it bottoms out.

h) Hold wire in one hand. With other hand place thin knife blade in slot of the tool. Apply pressure on the knife blade and rotate a complete revolution around the wire to partially sever the insulation 7/17 inches from the end of the wire. The tool (part number 14053 or 50ARM300–250) is designed to prevent the knife blade from making contact with the conductor. This is important, since radial nicking of the conductor may cause early failure of the wire.

i) Hold wire in one hand and with pliers grip and twist end of insulation, removing the end of insulation. The exposed conductor must be 7/16", with no nicks or scratches.

j) Using a file, carefully remove all burrs and sharp edges from the end of the conductor to permit easy entry into the chuck jaws.

k) At fuselage feedthrough, thread the wire through the sleeve and elbow.

l) Thrust the wire end firmly and quickly into the feedthrough fitting, through the hole as far as it will go. This will permit the chuck to engage the bared end of the wire. Make sure that the entire 7/16" length of bare conductor is firmly secured by jaws.

m) To ensure firm seating of wire, give sharp tug outward from the jaws.

n) Fill the threaded cavity with Dow Corning DC–4 silicone grease. Screw elbow into insulator body, forcing the DC–4 into the elbow.

o) Fill the elbow with DC–4, and screw the sleeve into the elbow forcing the grease into the sleeve, such that the grease flows out the top opening of the sleeve. Sleeve should be screwed firmly into elbow, finger tight.

Note In order to prevent the entry of moisture, the antenna feedthrough sleeve must be packed with Dow Corning DC–4 Silicone Grease. Cleanliness, adequate quantity of “packed” DC–4 Grease, and a tight connection are of critical importance.

Note To improve finger pressure on sleeve, a small piece of emery cloth between fingers and sleeve is permitted. Do NOT use pliers.

p) Thread the wire through the tensioner on the vertical stabilizer and forward to the mast of the forward fuselage.

q) Trim wire to length. After installation with tensioners under load, the tension should be 35 ± 5 lbs.

r) Strip 7/16 inch of insulation from the wire per Para 4.E.2).h) and Para 4.E.2).i).

s) Thrust the wire end firmly and quickly into the shackle through the hole as far as it will go. This will permit the chuck to engage the bared end of the wire.

t) To ensure firm seating of wire, give sharp tug outward. When wire is properly installed, the end of insulation should pass into counter bore of chuck housing.

u) Attach the shackle to the mast.

v) Verify tension is 35 ± 5 lbs.

w) Remove safety clips and close RX/EX ANT CPLR (2S), POWER AMP (2R), HF CTRL (4P) circuit breakers.

x) Test the HF transceiver as per Para 3.A.

5. Troubleshooting

A. HF (KHF 2100) Transceiver

1) If the HF (KHF 2100) transceiver is deemed inoperative check the following:

a) Ensure RX/EX ANT CPLR (2S), POWER AMP (2R), HF CTRL (4P) circuit breakers are closed.

b) Check that the system units are securely installed and the retaining mechanism is securely tightened.

c) Visually inspect receiver/exciter, antenna coupler, power amplifier, and controller connectors for damaged wires or contacts.

d) If troubleshooting at the wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

e) If repair or service is required, send to appropriate rated avionics repair facility.

Note The HF (KHF 1050) system components contain BITE (Built-In Test Equipment) which constantly monitors the operation health of the unit. The KHF 1050 also stores detected failures in non volatile memory for later review.

Note If appropriately rated for avionics repair, fault isolation procedures are contained in the KHF 1050 CMM.

B. HF Controller (PS440 Control Head)

PS440 Displays	Possible Cause	Check	Remedial Action
PRS W (Antenna Coupler Pressure Warning) Note The HF System will continue to operate normally.	– the pressure has dropped significantly below normal – may indicate that the antenna coupler is approaching a Pressure Fault Condition.	– the pressure of the Antenna Coupler	– charge the Antenna Coupler with nitrogen to 8–10 psi through the pressure input and measure valve
PRS F (Antenna Coupler Pressure Fault) Note The HF System will operate at low-power condition, (50W), regardless of the transmit power selected by the pilot.	– the pressure has dropped to a level where there is a risk of arcing inside the antenna coupler	– the pressure of the Antenna Coupler	– charge Antenna Coupler with nitrogen to 8–10 psi through the pressure input and measure valve
PANEL (Control Panel Failure)	– if HF Controller (PS440 Control Head) panels has an internal failure		– replace with serviceable unit
FAIL RXEX (Receiver/Exciter Failure)	– there is a failure of the Receiver/Exciter Unit – failure in the connections to the unit – loss of DC power to unit – communications problems between the Receiver/Exciter and the Controller	– the connectors and pins are in good condition and are fully mated with the units – that DC Power is available at the KRX 1053 connector – continuity in the wiring between the KRX 1053 and PS440	– replace with the serviceable unit

PS440 Displays	Possible Cause	Check	Remedial Action
FAIL PA (Power Amplifier Failure)	– failure of the Power Amplifier unit – failure in the connections to unit – loss of DC power to unit	– the connectors and pins are in good and condition and are fully mated with the units – that DC Power is available at the KPA 1052 connector	– replace with serviceable unit
FAIL CPLR (Antenna Coupler Failure)	– failure of the Antenna Coupler unit – failure in connections to unit – loss of DC power to unit – communications problems between the Antenna Coupler and the Receiver/Exciter	– the connectors and pins are in good and condition and are fully mated with the units – that DC Power is available at the KRX 1053 connector – for continuity in the wiring between the KAC 1052 and KRX 1053	– replace with serviceable unit

1) If repair or service is required, send to appropriate rated avionics repair facility.

Note The HF (KHF 1050) system components contain BITE (Built-In Test Equipment) which constantly monitors the operation health of the unit. The KHF 1050 also stores detected failures in non volatile memory for later review.

Note If appropriately rated for avionics repair, shop fault isolation procedures are contained in KHF 1050 CMM.

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VHF Communication System – General data

1. General

The Very High Frequency (VHF) Communication (VHF COM) system is used for voice communication between aircraft and/or aircraft and a ground station.

2. Description

A. VHF Communication

The VHF COM receiver component of the Multi-Mode Digital Radio (MMDR) has 2,280 channels and is capable of 8.33 or 25 kHz channel spacing. A momentary action EMERGENCY COM switch is installed on the lower right sub-panel to enable instantaneous tuning of the number 1 VHF communications radio to 121.5 MHz. The transmitter has stuck microphone protection and will cease transmitting if the transmit function is active for more than 35 continuous seconds.

The COM 1 antenna is installed on the top of the fuselage between FS143.95 and FS162.35. The COM 2 antenna is installed on the top of the fuselage between FS347 and FS362.

The VHF COM system is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR). (Refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data).

3. Adjustment/Test

A. VHF Communication

1) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that MMDR 1 (F3) (Left Bus) and/or MMDR 1 (F4) (Right Bus), circuit breaker on the avionics circuit breaker panel are closed.

d) Set up VHF Communication ramp test unit, in accordance with the instructions in the test set manual.

e) Tune the VHF Communication ramp test unit to an unused frequency.

f) Tune COM 1 and COM 2 to match the ramp test unit frequency.

g) Enable a MOD TONE on the ramp test unit. Verify a tone is heard when COM 1 and COM 2 are selected on the audio control panel.

h) Tune COM 1 and COM 2 to a local frequency and do a radio maintenance check.

i) Do a VSWR check on COM 1 and COM 2 antennas using an inline type Wattmeter.

j) Cycle the EMERGENCY COM switch once to place 121.5 MHz frequency into the COM 1 active window.

k) Cycle the EMERGENCY COM switch twice to place 121.5 MHz frequency into the COM 1 standby window.

l) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – VHF Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to the ceiling compartment aft of station 143.95 for VHF Antenna and aft of station 347 for VHF Upper Aft Antenna. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

3) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the VHF antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

4) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

5) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

6) Replace ceiling compartment. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

C. Operational Test Emergency Frequency Com Tuning Switch

1) Activate the Emergency Frequency Com Tuning switch and ensure COM 1 changes to 121.5 MHz and places this in the active COM 1 channel (use window).

4. Removal/Installation

The VHF Communication system is an integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR), refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data Para 4.A for the MMDR Removal/Installation procedure.

A. COM 1 and COM 2 Antenna (CI–108–1–L)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the (6) mounting screws that secure the antenna to the fuselage.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the MMDR 1 (F3) and/or MMDR 2 (F4) circuit breaker on the avionics circuit breaker panel are opened and fitted with safety clips.

c) Solvent clean the electrical contact bond areas immediately before the installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with the mounting screws (6).

f) Solvent clean the exposed area of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00 Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

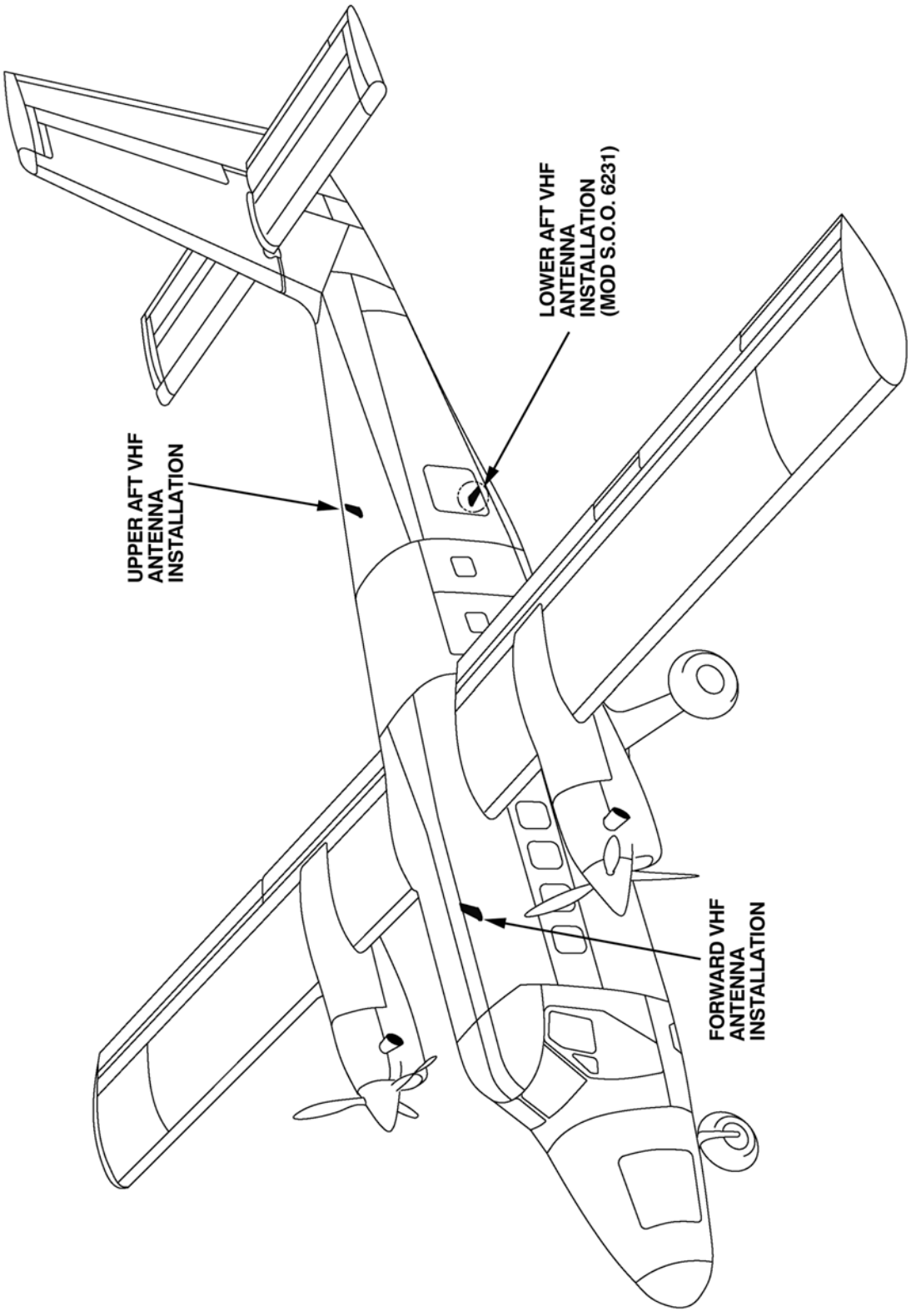
j) Check the antenna electrical bonding as per Para 3.B.

k) Remove the safety clips and close MMDR 1 (F3) and/or MMDR 2 (F4).

l) Do a function test on the VHF COM system as per Para 3.A.1).

5. Troubleshooting

The VHF Communication system is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR), refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data, Para 5.A for the MMDR Troubleshooting procedure.



VHF Antenna Locations
Figure 1

VHF-FM System (S.O.O. 6296) – General data

1. General

The Very High Frequency Modulation (VHF-FM) system is used for voice communication between aircraft and/or aircraft and a ground station in the 138.000 to 174.000 MHz frequency range of the VHF band.

2. Description

A. VHF-FM

The VHF-FM transceiver provides secondary airborne voice communications on crew selectable frequencies, operating in the 138.000 MHz to 174.000 MHz frequency range in 2.5 kHz increments, with either 25 kHz or 12.5 kHz channel spacing. The transmitter section of this unit has a minimum of 8 watts and does not exceed 10 watts output power. Output power may be reduced by a front panel switch to 1 watt, in order to reduce interference to land based systems.

The VHF-FM system includes a self-contained transceiver installed in the cockpit instrument panel, a blade style antenna installed on the cabin roof near the LH wing root, and a circuit breaker (3 amp) labeled 'VHF-FM' installed in the avionics circuit breaker panel.

The VHF-FM transceiver provides 120 operator accessible memory positions, each of which is capable of storing a transmit frequency, receive frequency, transmit frequency CTCSS (Continuous Tone-Coded Squelch System) tone or DPL (Digital Private Line) code, receive frequency CTCSS tone or DPL code, an alphanumeric identifier for each channel and in the TFM–138B, wideband (25 kHz) or narrowband (12.5 kHz) channel spacing assignment. Operating frequency and other related data are presented on a 48 character, two line LED matrix display.

Data entry and function control are done via a 12 button keypad on the front panel. Present channels may also be scrolled and scanned using the keyboard. Data may also be entered via a PC computer with the provided software and optional PIB–100 programming interface box.

3. Adjustment/Test

A. VHF-FM

1) Special Tools

- VHF-FM Ramp Test Set with antenna. (IFR AM/FM–1000, Aeroflex IFR–4000 or equivalent)
- In Line Wattmeter (Bird 43 Wattmeter with 10C or 25C element)

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that VHF-FM (Q6) (Right Bus) circuit breaker on the avionics circuit breaker panel is closed.

d) Set up VHF-FM Communication ramp test unit, in accordance with the instructions in the test set manual.

e) Tune the VHF-FM Communication ramp test unit to an unused frequency.

f) Tune the VHF-FM transceiver channel to match the ramp test unit frequency. Adjust volume and squelch to a comfortable level.

g) Enable a MOD TONE on the ramp test unit. Verify a tone is heard when the appropriate COM channel is selected on the audio control panel.

h) Do a VSWR check on the VHF-FM antenna using an inline type Wattmeter. The check should be performed with the LO/HI power switch in the HI position. Reflected power should not exceed 10% of the forward power.

i) Tune the VHF-FM transceiver to a local frequency and do a radio maintenance check. Transmit audio, receive audio and sidetone audio should be clear, undistorted, and at a comfortable level.

j) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – VHF-FM Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to the ceiling compartment aft of station 219.5. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

3) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the VHF antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be <3.0 milliohms unless stated otherwise.

4) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

5) If the electrical conductivity is not consistent over the electrical bond contact area, check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data.

6) Replace ceiling compartment. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

4. Removal/Installation

A. VHF-FM Antenna (CI–295–200)

Refer to Figure 1 for details.

1) Removal

a) Ensure the aircraft system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open VHF-FM (Q6) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Unscrew mounting screws (6) from antenna assembly.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure VHF-FM (Q6) (Right Bus) circuit breaker on the avionics circuit breaker panel is opened and fitted with a safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Move the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with the mounting screws (6).

f) Solvent clean the exposed area of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) If required, touch up the exposed area of airframe around the antenna with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data).

j) Check the antenna electrical bonding as per Para 3.B.

k) Remove the safety clips and close VHF-FM (Q6) (Right Bus) circuit breaker.

l) Do a function test on the VHF-FM system as per Para 3.A.2).

B. VHF-FM Transceiver (TFM–138B)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the VHF-FM (Q6) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Using an appropriate tool, push and turn each of the DZUS fasteners securing the unit counterclockwise until the unit disengages from the mounting.

d) Remove the transceiver unit from the mounting.

e) Remove unit from aircraft.

Note Stow the connectors of the removed components.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the VHF-FM (Q6) (Right Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clips.

c) Insert the TFM–138B unit into its mounting in the instrument panel.

d) Using an appropriate tool, push and turn each of the DZUS fasteners clockwise until the unit is secure in the mounting.

e) Remove safety clips and close VHF-FM (Q6) (Right Bus) circuit breaker on the avionics circuit breaker panel.

f) Do a function test on the VHF-FM system as per Para 3.A.2).

5. Troubleshooting

A. VHF-FM

1) If the VHF-FM Transceiver (TFM–138B) is deemed inoperative, check the following:

a) Ensure the VHF-FM (Q6) circuit breaker is closed.

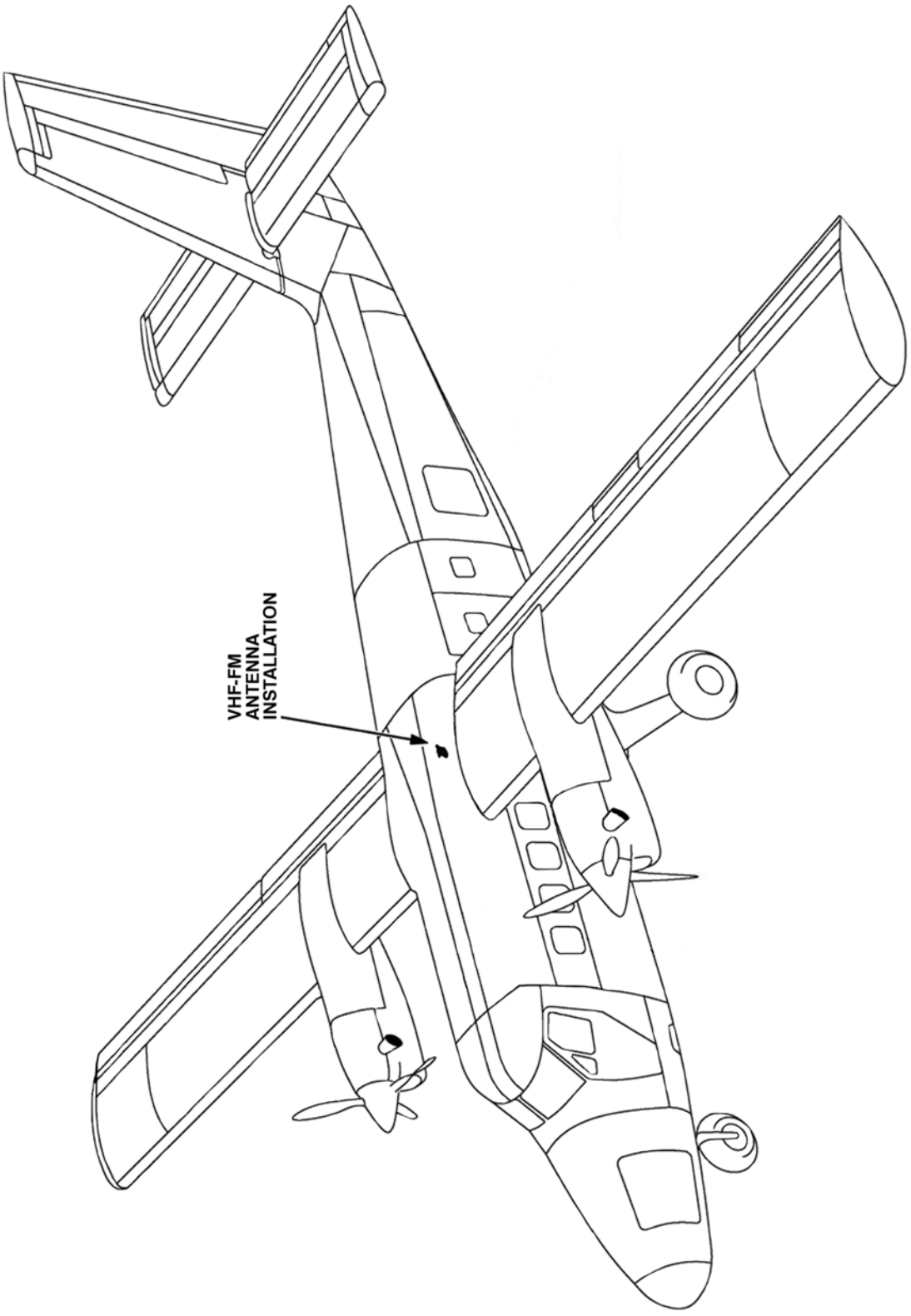
b) Visually inspect the VHF-FM unit for damaged wires or contacts.

c) Ensure that the VHF-FM connectors are properly mated and secure.

d) Ensure antenna connections are properly mated and secure.

e) If troubleshooting at the wire level is required, refer to the applicable wiring diagrams in Wiring Manual, PSM 1-64-2W.

f) If repair or service is required, send to appropriate rated avionics repair facility.



VHF-FM Antenna Location (S.O.O. 6296)
Figure 1

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Satellite Communication System (S.O.O. 6298) – General Data

1. General

Provisions are provided for the Satellite Communication System which consists of aircraft flight tracking and two-way telephone communication.

2. Description

The Satellite Communications System provides the following functions:

1 Flight-tracking which provides automatic position reporting at configurable intervals between the aircraft and ground-based personal computer, via the Iridium Low Earth Orbit (LEO) Satellite Communications Transceiver.

2 Satellite telephone audio conversion via the Iridium Satellite Communications System. Communications is through the COM 5 transmit channel on the pilot or co-pilot's audio panels. Control, including dialing is via the RC6000/A panel in the center pedestal.

3 Short text messaging via the RC6000/A panel in the center pedestal.

The system includes the following components:

1 RC6000/A control panel in the center pedestal.

2 S200 SkyNode in the forward avionics bay.

3 Low profile GPS/Iridium antenna on the upper fuselage at station 246.5.

4 A 3 Amp circuit breaker in the avionics CB panel labeled either "SATELLITE PHONE" or "SAT COM", typically located in position 4N.

The Satellite Communications System has no power switch. The system operates whenever power is present on the RH bus, and will start transmitting position reports when the GPS detects roundspeed.

As delivered, the SkyNode is set to default reporting settings. However, it is possible to modify the system configuration and settings. Some settings, such as the GPS Position Recording Interval (length of time between the SkyNode's processing and storing of a GPS position) and Connection Interval (length of time between regularly scheduled transmissions of position data), can be adjusted by the user from a Latitude Web Sentinel™ account.

Other settings such as OOOI (Out, Off, On, In) reporting, and Latitude Messaging API can be altered remotely by Latitude Technologies Corporation.

Contact Latitude Technologies Corporation for system activations and account information.

Latitude Technologies Corporation
Unit 101, 3375 Whittier Avenue
Victoria, BC V8Z 3R1
Canada

Toll Free: 1-888-966-5599
Direct and International: 1-250-475-0203
US Sales Office: 1-509-621-0229
Fax: 1-250-475-0204
Service Disruptions (24/7): 1-250-475-0203
Device Activations: 1-250-475-0203; activations@latitudetech.com

Table 1 Control Buttons

Home	The “HOME” key brings the user back to RC6000/A’s main menu screen
Menu	No function
Soft Keys	The RC6000/A has three keys that are used as contextual buttons: “S1”, “S2” and “S3”. The functions and labels of these keys change depending on the operating mode of the control panel.
Function Keys	No function
Alphanumeric Keys	There are 10 alphanumeric keys similar to those found on a telephone. These are used for dialing telephone numbers or entering text.
Arrow and Return Keys	The arrow keys and the return key (aka the enter key; “ ↵ ”) are used to navigate the RC6000/A menus. Use the arrow keys to navigate with the menus and use the return key to select a menu item.
Mark and DN/CLR	The “Mark” (“Mark Target”, also labeled “#”) and “DN/CLR” (“Down & Clear”, also labeled “*”) keys can be used as signaling inputs; pressing one of these keys triggers the SkyNode SAT COM to immediately generate and transmit a GPS position report with a special reason code that will be displayed on the Web Sentinel ground based computer. (Note 1)
May Day	The “MAY DAY” key can be used to place the SkyNode in Emergency Tracking mode. This means that the SkyNode will override the configured reporting interval and begin transmitting specially flagged GPS position reports every 10 seconds. (Notes 1, 2 and 3)

Note

1 Special feature, contact Latitude Technologies.

2 The destination to where the emergency messages are sent is determined by the SkyNode SAT COM equipment that the RC6000/A is connected with. At minimum, the SAT COM must have an active satellite network account and have been configured for emergency reporting.

3 Emergency Tracking MAY DAY mode is not part of the emergency locator transmitter. It is a supplementary capability that sends position reports to the company Web Sentinel base station at faster time intervals than for normal position reports.

3. Equipment and Materials

- Cleaner MIL–C–2576
- Isopropanol
- Digital Multimeter
- Milliohm meter (capable of measuring <3.0 milliohms)
- Personal Computer (PC) with Web Sentinel Software, internet connection and a web browser

4. Cleaning

The antenna may be cleaned only with the approved cleaning products listed in Para 3. Refer to 12-30-01, Aircraft External Cleaning – General Data for cleaning procedures. Limit cleaning to wiping with moistened cloths.

5. Adjustment/Test

A. Function Test of the SAT COM System

This test requires:

- A functioning ground based PC as described in Para 3., and radio or mobile phone communications between the cockpit and the ground based PC.

- A registered account with Latitude Technologies.
- That the forward upper panel of the forward avionics bay be temporarily removed to permit observation of the LED lights on the SkyNode.

1) General

   a) Position the aircraft to receive GPS and Iridium signals.

   b) Switch aircraft power ON.

   c) Open the SAT COM (N4) circuit breaker on the avionics circuit breaker panel and ensure that the control panel display, in the center pedestal is OFF and that the LED's on the SkyNode transceiver in the nose are OFF.

   d) Close the SAT COM (N4) circuit breaker on the avionics circuit breaker panel and ensure that the control panel display in the center pedestal illuminates and that the LED's on the SkyNode transceiver in the nose are ON (refer to troubleshooting procedures in Para 7., for a description of the LED blink patterns).

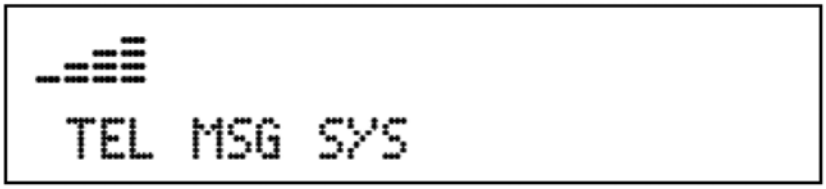
2) Power Up and Position Report Check

   a) During startup, the RC6000/A control panel will display an animated cursor moving right and left on the display's bottom line, while the top line will alternate between displaying the following two lines:

```
LATITUDE RC6000/A FW#.##
** SEARCHING FOR SkyNode **
```

   Where "#.##" is the firmware version currently installed on the RC6000/A.

   b) After the SkyNode is fully initialized and communication is established with the RC6000/A, the home screen will appear with TEL, MSG and SYS above select buttons S1, S2 and S3 respectively:



   The home screen is the starting point for access all other areas of the menu. From any other screen, or during any function operation, it is possible to return to the home screen by pressing the HOME button.

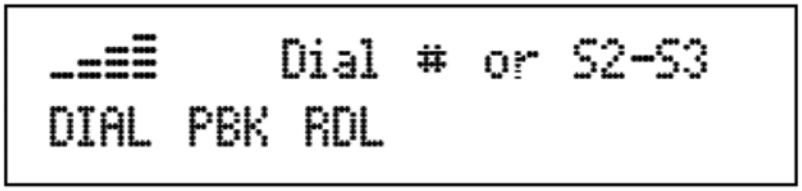
   c) There are two LED indicators on the left. The bottom LED has no function. The top LED indicates the status of short text messages received by the system. OFF if there are no messages in the inbox, BLINKING if there are messages in the inbox that are unused, and ON if there are messages that have all been read.

   d) Adjust screen brightness by pressing button "S3" (labeled "SYS"); to display the System Menu. Then press "S3" again, (labeled "BRT") from the System Menu. Adjust the screen brightness by using soft keys "S2" and "S3" or the arrow keys.

   e) After the SkyNode is fully initialized, confirm that the aircraft position is being displayed on the Web Sentinel display on the ground based PC. Confirm that the displayed position is reasonable.

3) Outbound Satellite Telephone Call Check

a) To use the satellite telephone, start in the Home Screen then press “S1” (labeled “TEL”); this will display the Main Telephone Screen.



Note When dialing a phone number, remember that all calls placed from an Iridium telephone are considered international calls and must begin with an international dialing code (for example, “001” for North America).

b) Wear a headset, and select the SAT COM transceiver (usually COM 5) on the cockpit audio panel.

c) From the Main Telephone Screen, use the number keys to dial a phone number. The number you are entering will be displayed on the top line of the display. Once you have finished dialing, press “S1” (labeled “DIAL”) to initiate the call.

Note Sending Dual-Tone Multi-Frequency (DTMF) tones; there are often instances when you must press keys during a voice call, such as remotely checking voicemail and navigating telephone menu systems. When you press a key during the call, the bottom right corner of the screen will display the text:

Sending ‘#’ tone

Where “#” is the key that was pressed (i.e. 1–9, #, *). You must wait until this text disappears before pressing another key.

d) When the call is answered, use the transmit button when talking, and release the transmit button when receiving.

e) To end a phone call that is in progress (i.e. “hang up”), press “S1” (labeled “END”).

4) Inbound Satellite Telephone Call Check

a) When a satellite call is received, the top line of the screen will display the flashing text:

***** Incoming Call *****

b) An aural “ringing” alert will be heard if the SAT COM transceiver (usually COM 5) is selected on the aircraft audio panel when the inbound call is received.

c) To answer the call select the SAT COM transceiver (usually COM 5) on the cockpit audio panel and press “S2” (labeled “ANS”) on the SAT COM control panel.

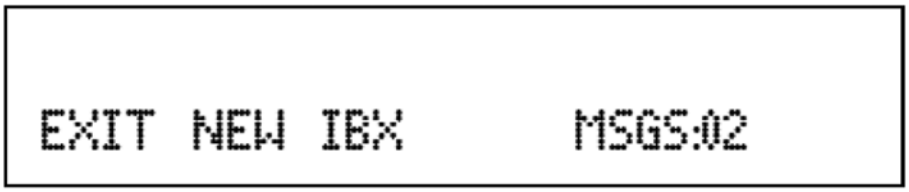
d) When the call is answered, use the transmit button when talking, and release the transmit button when receiving.

e) To end a phone call that is in progress (i.e. “hang up”), press “S1” (labeled “END”).

5) Messaging Send Check

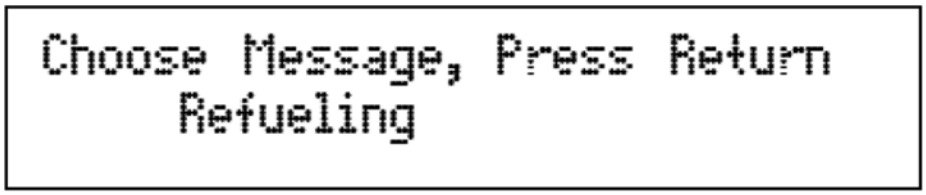
Sent messages are viewed on the ground in Web Sentinel. Forwarding to user-configured emails can also be setup in your Web Sentinel administrator account.

a) Start in the Home Screen, then press “S2” (labeled “MSG”); to display the Main Messaging Screen.



b) Press “S2” (labeled “NEW”), the screen will display the first of 9 canned messages on the bottom line. There are two types of canned message; Static and Hybrid. Canned messages are configured remotely from a Web Sentinel administrator account.

c) Use the arrow keys to navigate to your desired message, and then press return to select the message.



d) When a Static Canned Message is selected, you will be prompted to confirm that you want to send the message (soft keys “S1” and “S3” will be flashing “NO” and “YES”); simply press “YES” and the message will transmit (pressing “NO” will return you to the Main Messaging Screen).

e) Confirm that the Web Sentinel PC receives the message.

6) Messaging Receive Check

a) Send a message from the Web Sentinel PC to the aircraft.

b) When a new text message is received, the LED (green) at the left-side of the display’s top line will begin to blink, and the word “*MESSAGE*” will begin to blink on the bottom line of the display. These indications will continue until all messages in the inbox have been read.

c) Navigate to the inbox from the Main Messaging Screen by pressing “S3” (labeled “IBX”).

d) Using the arrow keys, navigate to the desired message, then press “S2” (labeled “REP”) to be taken to the New Message Screen or “S3” (labeled “DEL”) to delete the message (you will be prompted to confirm deletion).



e) A message is marked as read after being displayed for a few seconds or after one complete scroll across the screen in the case of longer messages.

6. Removal/Installation

A. Combined GPS/Iridium/Wi-Fi Antenna

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the SAT COM (N4) circuit breaker on the avionics circuit breaker panel and fit safety clip. If S.O.O. 6288 is installed, open the ECTM/FOQA (Q4) circuit breaker and fit safety clip.

c) Remove the four (4) screws securing the antenna to the airframe.

d) Carefully cut the sealant from around the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connections.

f) Disconnect the coaxial connectors.

Note Stow connectors as required.

g) Remove the antenna.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the SAT COM (N4) circuit breaker is opened and fitted with safety clips. When S.O.O. 6288 is installed, ensure ECTM/FOQA (Q4) circuit breaker is opened and fitted with safety clip.

c) Remove non-conductive material (e.g. anodic films, primer, paint, etc) from the electrical bond contact area, in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data, then apply chemical conversion coating (C10), in accordance with 20-30-02, Chemical Conversion Coatings (Alodine) – Manual Application, within 8 hours of stripping high resistance films.

d) Solvent clean the electrical contact bond areas immediately before installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data).

e) Move the antenna close to the fuselage and connect the coaxial connectors.

f) Secure the antenna to the fuselage using the four (4) mounting screws.

g) Solvent clean the exposed areas of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data).

h) Touch up exposed areas of the airframe around the antenna, if required, with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

i) Do a resistance measurement between the antenna and the airframe. Resistance must be less than 3 milliohms, in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data.

j) If the affected area requires paint touch up, brush coat the area with the top coat paint system matching the existing.

k) Fillet seal around the base of the antenna with MIL–PRF–81733 sealant, in accordance with 20-50-01, Weather Sealing – General Data. The aircraft will be ready to fly 24 hours after application of the sealant.

l) Remove the safety clip and close the SAT COM (N4) circuit breaker. When S.O.O. 6288 is installed, remove the safety clip and close the ECTM/FOQA (Q4) circuit breaker.

m) Do a function test of the SAT COM system per Para 5.A.1)

B. Dialer

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the SAT COM (N4) circuit breaker on the avionics circuit breaker panel and fit safety clip. If S.O.O. 6288 is installed, open the ECTM/FOQA (Q4) circuit breaker and fit safety clip.

c) Unscrew the built-in DZUS fasteners in the dialer unit.

d) Carefully pull the dialer from the center column until there is access to the wire connector.

e) Disconnect the wire connector.

Note Stow connectors as required.

f) Remove the dialer.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the SAT COM (N4) circuit breaker is opened and fitted with safety clip. When S.O.O. 6288 is installed, ensure the ECTM/FOQA (Q4) circuit breaker is opened and fitted with safety clip.

c) Move the dialer close to the centre column and connect the wire connector.

d) Carefully insert the dialer into the centre column and tighten the built-in DZUS fasteners.

e) Remove the safety clip and close the SAT COM (N4) circuit breaker. When S.O.O. 6288 is installed, remove safety clip and close the ECTM/FOQA (Q4) circuit breaker.

f) Do a function test of the SAT COM system as per Para 5.A.1).

C. SkyNode Unit

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the SAT COM (N4) circuit breaker on the avionics circuit breaker panel and fit safety clip. If S.O.O. 6288 is installed, open the ECTM/FOQA (Q4) circuit breaker and fit safety clip.

c) Disconnect all the connectors.

Note Stow connectors as required.

d) Remove the four (4) screws and washers holding the SkyNode in position on the top face of the upper, left avionics shelf in the FWD cargo compartment.

e) Remove the SkyNode.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the SAT COM (N4) circuit breaker is opened and fitted with safety clip. When S.O.O. 6288 is installed, ensure the ECTM/FOQA (Q4) circuit breaker is opened and fitted with safety clip.

c) Carefully position the SkyNode over the four (4) potted mounting holes on the top face of the upper, left avionics shelf in the FWD cargo compartment.

d) Install the four (4) screws and washers to secure the SkyNode in position.

e) Connect all the connectors.

f) Remove the safety clip and close the SAT COM (N4) circuit breaker. When S.O.O. 6288 is installed, remove safety clip and close the ECTM/FOQA (Q4) circuit breaker.

g) Do a function test of the SAT COM system as per Para 5.A.1).

7. Troubleshooting

A. Basic Checks

1) Ensure that the electrical system is ON (circuit breaker closed, LED on S200 SkyNode is active and control panel is illuminated).

2) Verify the SkyNode is secured and that all cables including antenna cables are in good condition and properly connected.

3) Verify that the antenna connectors at the antenna are secure and in good condition.

4) Verify that the antenna installation is clean and free of contaminants and corrosion.

5) Verify that the control panel is properly connected and properly secured into the instrument panel.

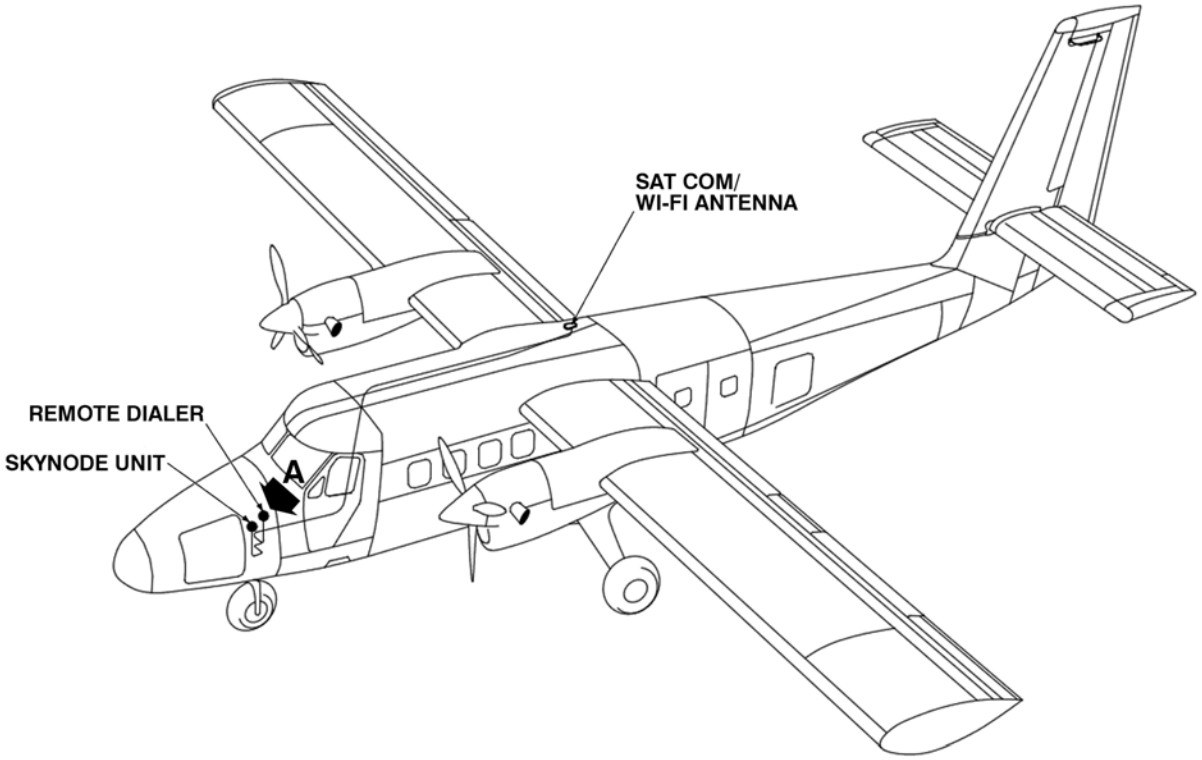
6) Refer to the Wiring Diagrams Manual (PSM 1-64-2W) for circuit breaker troubleshooting.

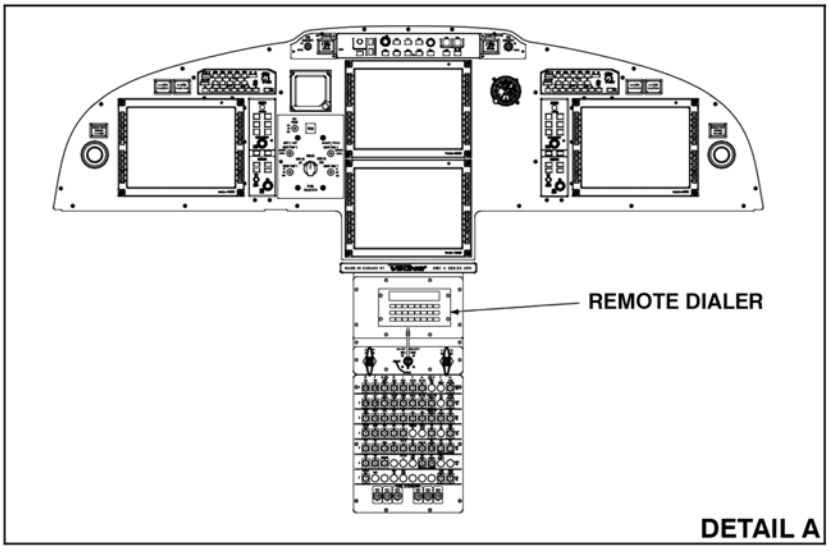
7) Investigate the LED blink pattern on the S200 SkyNode. Table 2 — Blink Pattern Meanings describes the meaning of the blink pattern.

Table 2 Blink Pattern Meanings

LED OBSERVATION	SYSTEM STATUS
LED On Solid	System running normally
Long On, Short Off	System is initializing (i.e. start up)
1 Blink Every 3 Seconds	No GPS satellite fix
2 Blinks Every 3 Seconds	No Iridium satellite fix
1 Blink in 3 Seconds, Then 2 Blinks in 3 Seconds	No Iridium network and No GPS fix
On for 1 Second Every 2 Seconds	Pass-Through Mode

8) Verify that an account with Latitude Technologies is active (refer to Contact Information in Para 2.).





Satellite Communication System Locations (S.O.O. 6298)
Figure 1

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Audio Integration – General Data

1. General

The audio integration system gives the pilot and co-pilot the ability to monitor all of the communication, navigation receivers, and microphones transmission over the selected radio transmitter. The integrating system is provided by the pilot's and the co-pilot's audio controls panels, supports the cockpit, the aural warning system, receiver monitoring, transmission, and the passenger address system (if installed). (Refer to 44-00-00, Cabin Systems – General Data).

2. Description

The audio integration system receives audio information from the following systems:

- VHF Communication system
- VHF Navigation System which include:
  - Automatic Direction Finding Systems (ADF 1 and ADF 2) (if installed)
  - Distance Measuring Equipment Systems (DME 1 and DME 2 [if installed])
  - Marker Beacon System
- HF Communication System (if installed)
- VHF-FM (if installed)
- Pilot, Co-pilot and Crew Headsets

The audio integration system also receives signals and warning tones generated by the Master Warning and Caution System through the Modular Avionics Unit. These signals and warning tones are from the speed and attitude correction system and the Crew Alerting System.

The audio integration system receives signals and warning generated directly from the Traffic Alert and Collision Avoidance System, TCAS I and TCAS II (if installed) and from the Terrain Awareness Warning System (TAWS).

Aircraft incorporating Mod S.O.O. 6229 provides a fourth (standard, non-noise cancelling) headset/mic jack, for use by an attendant sitting in the aft row of passenger seats. The jacks are installed on the aft R/H cabin side wall.

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Noise Cancelling Headsets – General Data

1. General

David Clark (H10–13XP) noise cancelling headset connections are available to the pilot and co-pilot for the aircraft audio integration through a David Clark Electronic Noise Cancelling (ENC) headset module (40594G–01) mounted on the upper outboard corners of the forward face of the sloping bulkhead at station 111.00.

2. Description

A. Electronic Noise Cancelling (ENC) Headset Module (40594G–01)

There are no speakers in the flight compartment. Critical aural warnings are delivered to the crew via the headsets, a functional headset is obligatory at each crew station. Standard jacks to accommodate use of a non-noise cancelling headset, in the event of failure of the noise cancelling headset, are provided at each crew station. The jacks are positioned such that it is not possible to install both a noise cancelling and a non-noise cancelling headset at the same time.

28 Volt DC power is protected by a 2 amp HEADSETS (G6) (Right Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data). The headset microphone and ear speakers will still function if the circuit breaker is out. The circuit breaker only provides power for the noise cancellation feature of the pilot and co-pilot headset.

The electronic noise cancelling headset module interfaces to the following aircraft system:

1) Audio Control Panel System (refer to 23-53-00, Audio Control Panel – General Data).

3. Adjustment/Test

A. Function Test

The David Clark Noise Cancelling headset module (40594G–01) can be tested with the connection of a David Clark H10–13XP headset and verifying the reduction of ambient noise when in the “on” position.

4. Removal/Installation

A. Removal

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Open HEADSETS (G6) circuit breaker on the avionics circuit breaker panel and fit safety clip.

3) Disconnect the electrical connector from the cockpit headset box assembly.

4) Remove the screws four securing the headset box assembly unit to the bulkhead.

5) Remove the screws four securing the ENC headset module faceplate to the headset box assembly.

6) Remove required wire connections from J337 connector. Refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

7) Remove the unit from aircraft.

Note Stow connectors as required.

B. Installation

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Ensure HEADSETS (G6) circuit breaker on the avionics circuit breaker panel is opened and fitted with safety clip.

3) Install required wire connections to J337 connector. Refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

4) Install the screws four securing the ENC headset module faceplate to the cockpit headset box assembly.

5) Install the screws four securing the headset box assembly unit to the bulkhead.

6) Reconnect the electrical connector to the cockpit headset box assembly. Ensure connections are secure.

7) Remove safety clip and close HEADSETS (G6) circuit breaker on the avionics circuit breaker panel.

8) Test the noise cancelling headset as per Para 3.A.

5. Troubleshooting

A. Electronic Noise Cancelling (ENC) Headset Module (45094G–01)

1) If the Electrical Noise Cancelling (ENC) Headset Module (45094G–01) is deemed inoperative check the following:

a) Ensure HEADSETS (G6) circuit breaker is closed.

b) Visually inspect the ENC module unit for damaged wires or contacts.

c) Ensure that the headset assembly connector is properly mated and secure.

d) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

e) If repair or service is required, send to appropriate rated avionics repair facility.

Audio Control Panel – General Data

1. General

The audio control panels (KMA 29) is used to make audio selections for all aircraft audio communications. It includes an automatic voice activated (VOX) intercommunication design, which eliminates the requirement for intercom squelch adjustment.

2. Description

A. Audio Control Panel (KMA 29)

Refer to Figure 1 for details.

The audio control panel supports up to five communication radios, one of which can be configured as a telephone (full duplex, with no requirement to push to talk). It also supports up to eight switched audio inputs (VHF navigation, COM, ADF, DME, marker beacon, and auxiliary). The panel also provides an interface to the Passenger Address (PA) system amplifier and the aural warning system. A marker beacon receiver is integral to the panel.

An audio control panel is installed above each PFD. Momentary push-buttons are used to select the desired COM transceivers. Pressing a button turns on the associated receiver and the green LED. The pilot can identify which receivers are selected by noting which LEDs are on. The left and right audio panels and left and right crew headsets operate independently of each other. Each pilot may transmit and receive at the same time, using different VHF or other communication radios such as HF or satellite telephone.

A marker beacon receiver provides the necessary marker beacon signals to the PFD and audio indications for an instrument Landing system (ILS). The MKR push button allows the MKR SENS button to be used to set either high or low sensitivity or mute the marker beacon audio by pressing the MUTE button on the pilots panel for five seconds. The marker beacon system may only be tested from the left audio panel. The test function on the right side (co-pilot) audio panel is inoperative.

Each audio panel has a single-button emergency reversion function that allows the pilot headset to be directly connected to the number 1 communications radio. The audio panel also supports digital data communication between the two panels, and between the audio panels and the CVR.

The KMA 29 has an automatic dimming of the pushbutton annunciator LEDs controlled by a photocell. Control of the unit backlighting is through the ‘INSTRUMENT PANEL' dimmer on the ‘DIMMING PANEL' installed on the center control column.

28 Volt DC power is protected by a 5 amp PILOT AUDIO (3E) (Left Bus) and a 5 amp COPILOT AUDIO (4E) (Right Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The KMA 29 audio control panel interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

2) VHF Communication System (refer to 23-12-00, VHF Communications System – General Data).

3) VHF Navigation System (refer to 34-52-00, VHF Navigation System – General Data).

4) HF Communication System (refer to 23-11-00, HF Communication System (S.O.O. 6215) – General Data).

5) Distance Measuring Equipment (DME) (refer to 34-54-00, Distance Measuring Equipment (DME) (Mod 6/2049) – General Data).

6) Cabin Passenger Address (PA) (refer to 44-00-00, Cabin Systems – General Data).

7) Cockpit Voice Recorder (CVR) (refer to 31-32-00, Cockpit Voice Recorder (CVR) – General Data).

8) Audio System Isolation Amplifier (refer to 23-54-00, Audio System Isolation Amplifier – General Data).

9) Noise Cancelling Headsets (refer to 23-52-00, Noise Cancelling Headsets – General Data).

10) VHF-FM (refer to 23-13-00, VHF-FM System (S.O.O. 6296) – General data).

3. Adjustment/Test

A. Function Test Audio Control Panel (KMA 29)

1) Intercommunication System (ICS) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that PILOT AUDIO (3E) and COPILOT AUDIO (4E) circuit breakers are closed on the avionics circuit breaker panel.

d) Connect a headset to the pilot, co-pilot and observers.

e) Turn the intercom system (ICS) control to the middle position.

f) Speak into the pilot, co-pilot and observer positions. Ensure intercommunication at each position is satisfactory.

g) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Receive/Transmit Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure the PILOT AUDIO (3E) and COPILOT AUDIO (4E) circuit breakers are closed on the avionics circuit breaker panel.

d) Connect a headset to the pilot, and/or co-pilot position.

e) Set MIC selector to an unused COM position. Ensure all audio selector LEDs are OFF, except the selected transmit position.

f) Tune radio to receive a valid audio signal. Select associated receive audio on the audio control panel.

g) Verify the selected pilot audio control panel receivers are present on the pilot's headset and that the selected co-pilot control panel audio are present on the co-pilot's headset.

1 COM 1

2 COM 2

3 COM 3 – if used

4 COM 4 – if used

5 COM 5 – if used

6 NAV 1

7 NAV 2

8 ADF 1 – if installed

9 ADF 2 – if installed

10 DME 1

11 DME 2 – if installed

h) Tune radio to an appropriate transmit test ground, tower, or base station.

i) Select the associated MIC on the pilot's audio control panel and/or the selected co-pilot's control panel. Do a transmit radio check for each radio.

1 COM 1

2 COM 2

3 COM 3 – if used

4 COM 4 – if used

5 COM 5 – if used

j) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Operational Test Audio Panel Emergency Switch

1) In the OFF or EMG position, the pilot is connected directly to COM 1. This allows communication capability regardless of unit condition. Any time power is removed or turned OFF, the audio selector will be placed in the fail-safe mode. In EMG mode, critical audio signals on Unswitched 1 and Unswitched 2 are also provided to the pilot headset. In EMG mode, audio signals are present in only one ear of a stereo headset.

2) Complete a radio TX/RX test. Ensure proper transmit and receive functions of COM 1.

4. Removal/Installation

These procedures are applicable to the removal and installation of the pilot's Audio Control Panel and the co-pilot's Audio Control Panel (KMA 29).

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND LIGHTNING. ENSURE INTEGRITY OF THE HIRF/LIGHTNING PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – PANEL/PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELLS/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the Audio Control Panel (KMA 29)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the PILOT AUDIO (3E) and/or the COPILOT AUDIO (4E) circuit breaker on the avionics circuit breaker panel and fit safety clips.

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING, UNIT CAN BE DAMAGED.**

c) Using a 3/32 hexagonal-head wrench, engage it into the audio control panel locking screw.

d) Turn the locking screw counterclockwise until it disengages from the audio control panel mount.

e) Pull on the edges of the audio panel, until the audio control panel is removed from the mount.

f) Remove unit from aircraft.

Note Stow connectors as required.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the PILOT AUDIO (3E) and/or the COPILOT AUDIO (4E) circuit breaker on the avionics circuit breaker panel is opened and fitted with safety clips.

c) Slide the audio panel controller into the mount on the instrument panel.

d) Push carefully on the face of the audio control panel until the locking screw is engaged at the rear of the mount.

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING, UNIT CAN BE DAMAGED.**

e) Using a 3/32 hexagonal-head wrench, engage it into the audio control panel locking screw.

f) Apply a light pressure to the edges of the unit, while turning the 3/32 wrench clockwise to tighten the locking screw.

g) Remove safety clips and close PILOT AUDIO (3E) and/or the COPILOT AUDIO (4E) circuit breaker on the avionics circuit breaker panel.

h) Test the audio control panel as per Para 3.A.

5. Troubleshooting

A. Audio Control Panel (KMA 29)

1) If the Audio Control Panel (KMA 29) is deemed inoperative check the following:

a) Ensure PILOT AUDIO (3E) and the COPILOT AUDIO (4E) circuit breakers are closed.

b) Check that the units are securely installed and the locking screw is securely tightened.

c) Visually inspect audio control panel connectors for damaged wires and contacts.

d) Ensure correct position of the PTT switch. Visually inspect PPT switch for damaged wires or contacts.

e) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

f) If repair or service is required, send to appropriate rated avionics repair facility.

2) Factory Configuration

a) Indication arrows above the microphone selectors indicate which side has which radio selected for transmit. Annunciation of which radio has been selected for transmission at the off-side position can be configured to enabled or disabled. When the offside indication is disabled, the arrow for the Mic selected will not be lit for the opposite KMA 29 position. When enabled, the pilot can see which radio the co-pilot has selected for transmit, and vice versa, by noting which of the arrows is illuminated. Each KMA 29 radio can be configured separately.

The factory configuration is that each audio panel have off-side transmit selection indication enabled. Set to approved factory default, as described below, if this feature is found to be disabled.

To toggle the offside transmit selection indication:

1 Press the right side of the “SPLIT” button three (3) times within one and a half seconds (1 ½) to toggle.

Note If the mode has been enabled, the NAV 1 indicator will blink once. If the mode has been disabled, the NAV 1 indicator will blink twice.

b) The KMA 29 contains five separate backlighting curves. The factory configuration for the backlighting curve is curve 3, which is achieved by following the procedure described below.

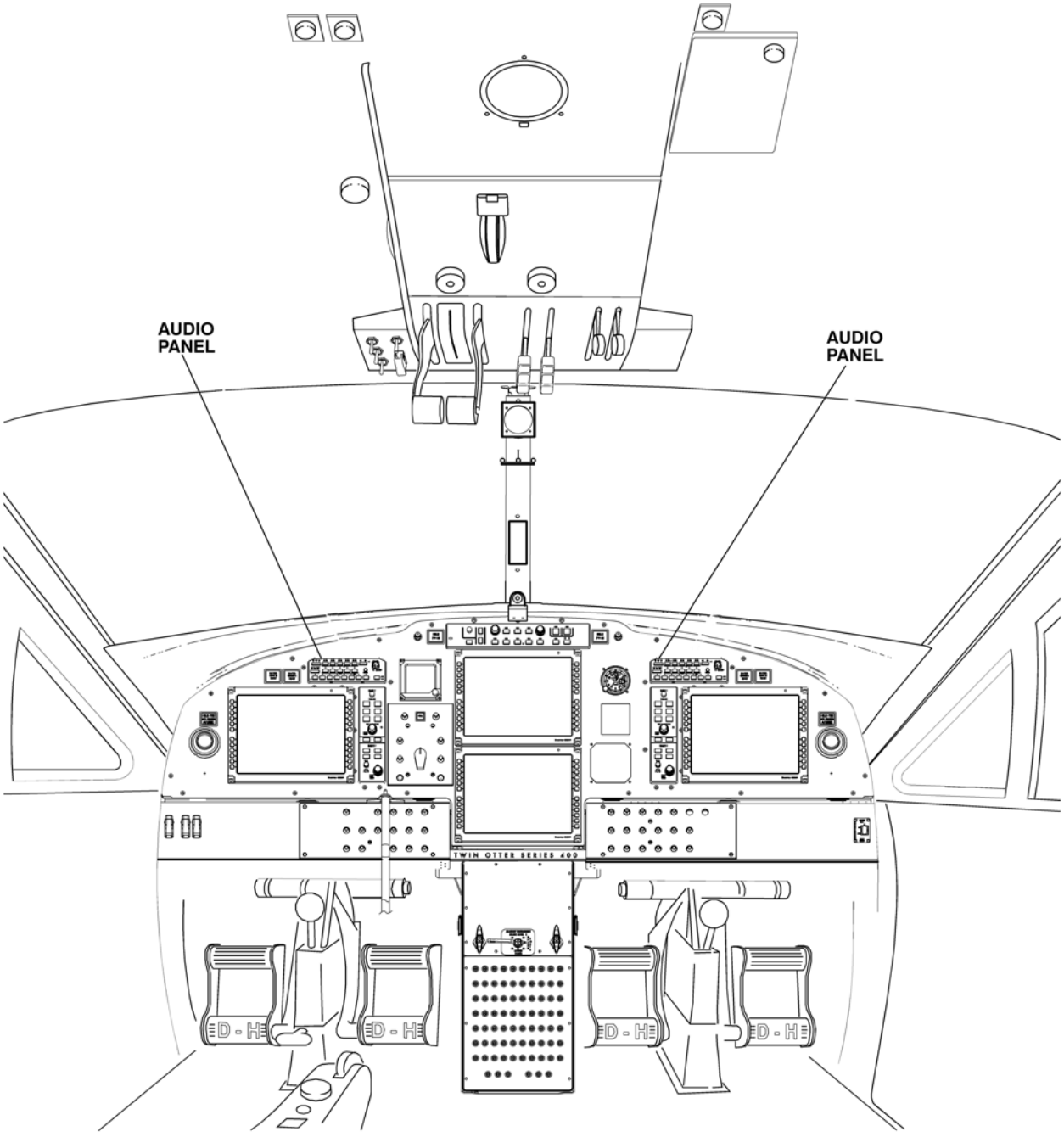
To select the factory approved backlighting curve:

1 Press and hold left split for 10 seconds.

Note Left split LED will start to blink.

2 Press COM 3 XMIT to select the approved factory default lighting curve.

Note The unit will exit this mode either after a COM is pressed or 10 seconds.



Audio Control Panels
Figure 1

Audio System Isolation Amplifier – General data

1. General

The NAT 247 audio isolation amplifier is used to mix warning audio alerts from various aircraft systems.

2. Description

A. Audio Isolation Amplifier (NAT 247)

The audio isolation amplifier is used as a mixing amplifier to integrate various aircraft audio signals into the audio control panel. The audio isolation amplifier is mounted on the port side of the middle forward avionics shelf.

28 Volt DC power is protected by a 5 amp PILOT AUDIO (3E) (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The NAT 247 audio isolation amplifier interfaces to the following aircraft systems:

1) Audio Control Panel (refer to 23-53-00, Audio Control Panel – General Data).

2) Traffic Alert and Collision Avoidance System (TCAS I and TCAS II) (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General Data).

3) Enhanced Ground Proximity Warning System (EGPWS) (TAWS) (refer to 34-44-00, Enhanced Ground Proximity Warning System (TAWS) (Mod 6/2049) – General Data).

3. Adjustment/Test

A. Function Test Audio Isolation Amplifier (NAT 247)

1) Function Test Audio Isolation Amplifier

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that PILOT AUDIO (3E) circuit breaker is closed.

d) Connect a headset to the pilot, and/or co-pilot position.

e) In the APEX system sensors page, initiate a TCAS self test, (if installed).

f) Verify TCAS system test aural alert is audible in the headset.

g) In the APEX system sensors page, initiate a self test.

h) Verify TAWS system test aural alert is audible in the headset.

i) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT/REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the Audio Isolation Amplifier (NAT 247)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the PILOT AUDIO (3E) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Disconnect the electrical connector on the front of the unit.

e) Remove the screws (4) securing the unit to the forward avionics shelf.

f) Remove the unit from aircraft.

Note Stow connectors as required.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the PILOT AUDIO (3E) circuit breaker on the avionics circuit breaker panel is opened and fitted with safety clip.

c) Align the unit mounting holes to the forward avionics shelf.

d) Secure the unit to the avionics shelf with mounting screws (4).

e) Reconnect connector to the unit. Ensure connections are secure.

f) Remove safety clip and close PILOT AUDIO (3E) circuit breaker on the avionics circuit breaker panel.

g) Test the Audio Isolation Amplifier as per Para 3.A.

5. Troubleshooting

A. Audio Isolation Amplifier (NAT 247)

1) If the Audio Isolation Amplifier (NAT 247) is deemed inoperative check the following:

a) Ensure PILOT AUDIO (3E) circuit breaker is closed.

b) Visually inspect audio isolation amplifier connector for damaged wires or contacts.

c) Ensure that the audio isolation amplifier connector is properly mated and secure.

d) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

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Static Discharging – General Data

1. General

   Static discharge wicks are installed as follows: One on each aileron, one on each outboard fore flap hinge arm assembly, two on the right hand elevator, one on the left hand elevator and three on the rudder. The wicks serve to dissipate electrostatic charges from the aircraft structure and thereby minimize radio interference from this source. The static discharge wicks consist of a rope floss of fine fibers about 13 inches long. For mechanical protection the wick is encased in a plastic sheath about one-quarter inch in diameter. One end is fixed to an electrical lug which in turn is secured to the airfoil by two screws and lock washers. At the other end of the plastic sheath, the rope floss extends about one inch so that the fibers are free to ionize in the slipstream by virtue of the impressed potential between the electrified aircraft and the surrounding space.

2. Servicing

   A. Service Wicks

      1) Cut off ragged edges from wick. Clean if necessary with lint-free cloth.

      2) Measure one inch from exposed end of rope floss, and mark plastic sheath.

      3) Carefully cut back and remove plastic sheathing to position marked in Para 2.A.2), and remove the cotton covering from around the rope floss.

      4) Ensure that the overall length of serviced wick exceeds six inches.

3. Removal/Installation

   A. Remove Static Discharge Wick

      1) Remove two screws and lock washers securing static discharge wick to airframe, and remove the wick.

   B. Install Static Discharge Wick

      1) If necessary, prepare the end of the new static discharge wick, using procedure described in Para 2.A.

      2) Secure new wick to airframe with screws and lock washers, making sure that metal to metal contact is maintained.

      3) Ensure parallel alignment with line of flight.

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CHAPTER 24

ELECTRICAL POWER

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CHAPTER 24 – ELECTRICAL POWER

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Electrical Power – General Data

1. General

Aircraft 28 Volt DC electrical power is provided by two starter-generator systems in conjunction with one 24 volt battery (main). The DC system is a single wire type installation using the airframe structure as ground return. The main battery supplies the DC power necessary when the generators are off-line and is also used to start the engines. Aircraft with Mod 6/2315 also include a small 24 volt avionics start battery. The battery is used to supplement electrical power supply to certain avionics systems during an engine start. Further information on the avionics start battery is provided in 24-30-32, Avionics Start Battery (Mod 6/2315) – General Data.

An external 28 Volt DC power source can be connected to the aircraft for engine starting and test procedures.

DC circuits are protected by thermal push-pull circuit breakers.

The main battery is a 24 volt, 48 amp-hour lead acid type located beneath the floor of the rear baggage compartment. On aircraft incorporating S.O.O. 6371, the main battery has been relocated to the nose baggage compartment. The main battery supplies power to the electrical system when the MASTER switch is selected to ON, and the POWER SOURCE switch is selected to BATTERY (Pre Mod 6/2314). When Mod 6/2314 is incorporated, the main battery supplies power to the electrical system when the BATTERY switch is selected to ON. The main battery also supplies power independently through a hot battery bus to the entrance lights, forward and rear baggage compartment lights, and the cockpit dome light and clock.

2. Removal/Installation

A. Safety Precautions

The following safety precautions should be observed prior to the removal or installation of any electrical equipment or components in the aircraft.

1) Refer to appropriate wiring diagram in the Wiring Diagram Manual, PSM 1-64-2W, prior to removal and installation of any electrical equipment or components and when disconnecting or connecting electrical wiring.

2) Ensure that MASTER and POWER SOURCE switches are set to OFF (Pre Mod 6/2314); or BATTERY, EXTERNAL and GENERATOR switches are set to OFF (Post Mod 6/2314).

3) Disconnect external power source and disconnect main battery. Pull out applicable circuit breaker and insert clip to prevent push-to-reset.

4) If units removed are not to be immediately replaced, ensure that connector ends are capped and stowed, and all loose wires are insulated and stowed.

5) After installation, remove clip and push in circuit breaker; do a function test on unit.

Removal and installation procedures for the electrical equipment is self-evident, except for the main battery (refer to 24-30-31, Main Battery Power – Maintenance Practices). Equipment grounding and bonding surfaces must be cleaned before and after installation, any bared surfaces must be refinished. When test or repair requires removal of cables from terminals, the terminal nuts must be tightened to the torque values given in Table 1 — Wrench Torque Values for Securing Terminal Nuts when cables are reconnected.

Table 1 Wrench Torque Values for Securing Terminal Nuts

TERMINAL STUD SIZE	TERMINAL LUG METAL	TORQUE-VALUE (POUND-INCHES)
1/2 inch	Copper	140 – 155
3/8 inch	Copper	110 – 125
3/8 inch (Gen Shunt Ground)	Copper	110 – 124
5/16 inch	Copper	80 – 85
AN3	Steel	40 – 45
1/4 inch	Copper	40 – 45
1/4 (Bat. Shunt Ground)	Copper	40 – 45
No. 10	Copper	15 – 20
No. 8	Copper	9 – 12
No. 6	Copper	Tighten to depress lockwasher

DC Generation – General Data

1. General

The DC generation system provides the aircraft with the primary source of 28 Volt DC power, and also provides the starting facility for starting the aircraft engines. (Refer to 80-00-00, Starting – General Data). The system consists of left and right starter-generator systems designed to either operate in parallel during normal operation or, if required, both systems can be operated as completely separate power systems as determined by a BUS TIE switch. A secondary source of DC power is provided by a 24 Volt main battery.

Each starter-generator system consists of a starter-generator driven by the engine auxiliary gearbox. With both generators on-line and operating as separate systems (i.e. BUS TIE switch set to OPEN) the left generator powers the left DC bus and the battery bus, and also charges the battery, and the right generator powers the right DC bus. With the BUS TIE switch set to NORMAL, the generators are connected in parallel, thus, should one generator fail, all the buses are supplied by the single on-line generator.

Generator protection from reverse current, undervoltage and overvoltage is provided by a reverse current relay, and overvoltage relay, and generator output is regulated by a voltage regulator. The overvoltage sensing is achieved by a circuit in the voltage regulator.

Generator status is annunciated on the crew alerting system (CAS) text message list. Voltage level and amperage demand are continually displayed on the electrical system window.

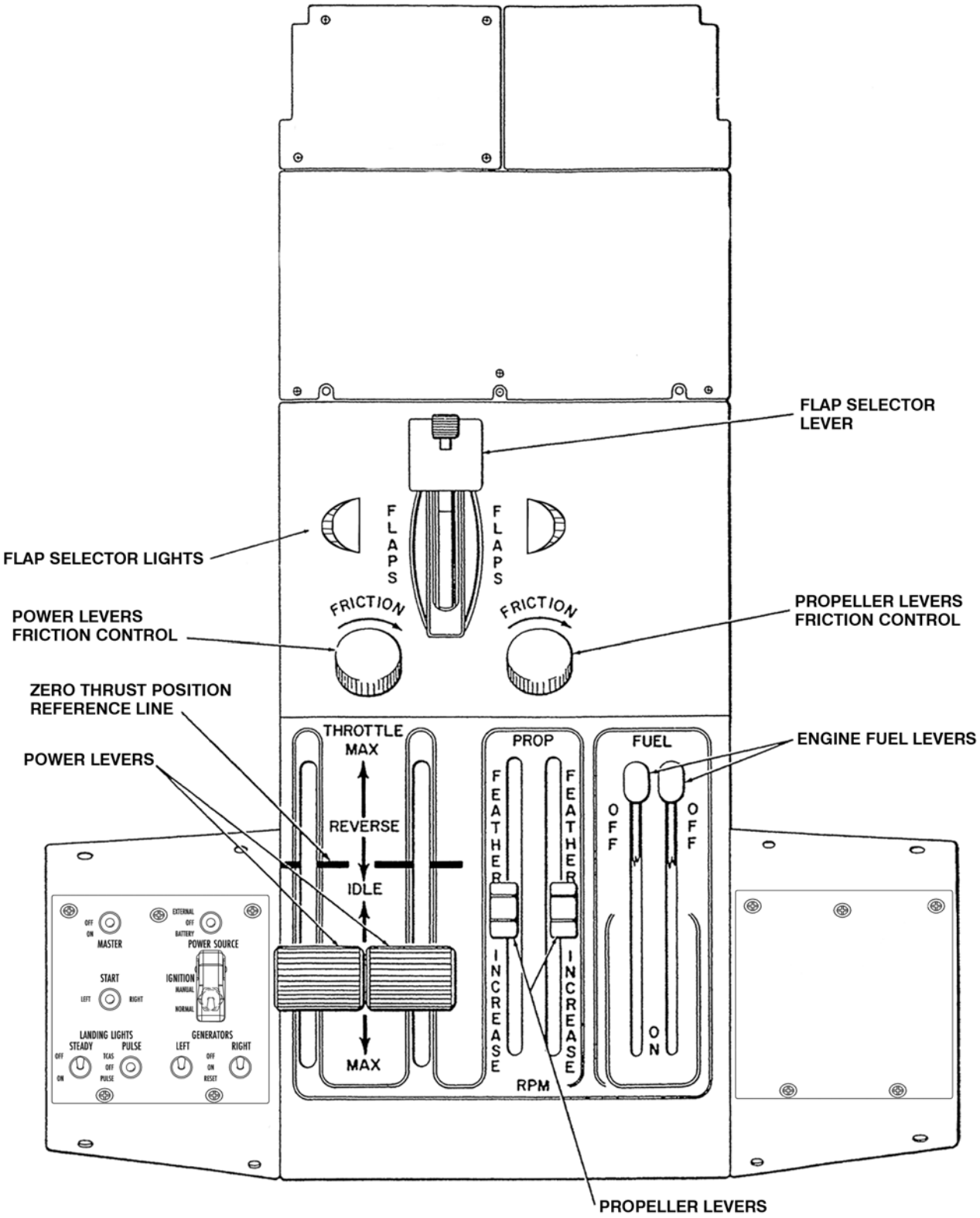
The left generator caution circuit is powered from the right DC bus and protected by a GENERATOR FAIL L circuit breaker. The right generator caution circuit is powered from the left DC bus and protected by a GENERATOR FAIL R circuit breaker. Both Circuit breakers are on the main circuit breaker panel.

Electrical system parameters are detailed in the electrical system window which is normally displayed on the lower Multi-Function Display (MFD). (Refer to 31-11-00, Display Units – General Data).

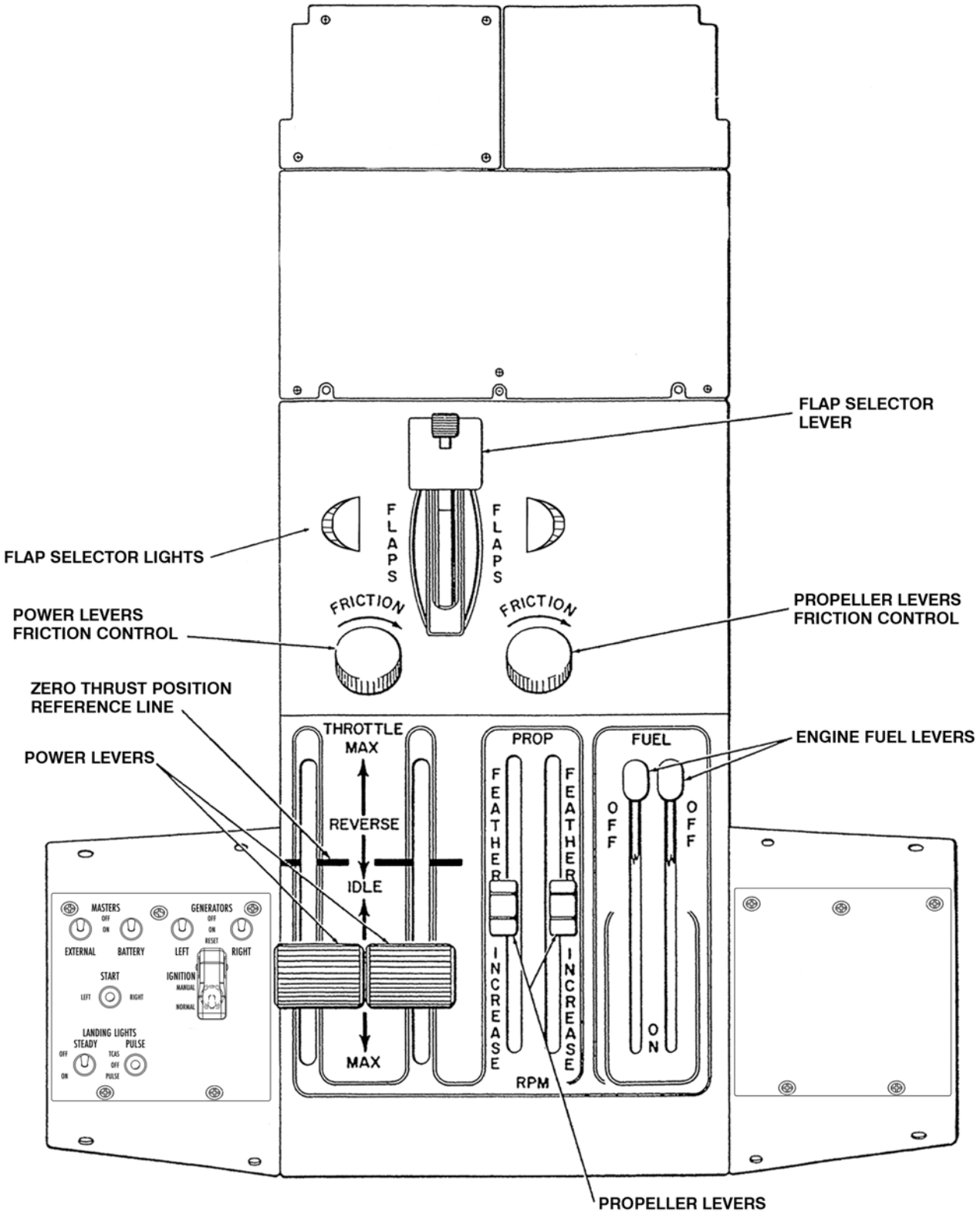
Note Mod 6/2314 (Independent External and Battery Power switches) and Mod 6/2315 (Introduction of Avionics Start Battery) affect the layout of the overhead switch panel. The differences are illustrated in Figure 1, Figure 2 and Figure 3.

Two switches labelled GENERATORS, LEFT and RIGHT, are located on the start control panel directly above the left pilot windshield (refer to Figure 1, Figure 2 or Figure 3). Each is a three position switch with positions labelled OFF, ON, and RESET. The ON position connects the applicable generator output to the electrical system through a reverse current relay and a voltage regulator. Undervoltage or overvoltage conditions de-energize the generator field relay to disconnect the generator from the system. Before reconnection can take place, the applicable generator switch must be held momentarily at RESET to restore the generator field circuit, and then released to ON.

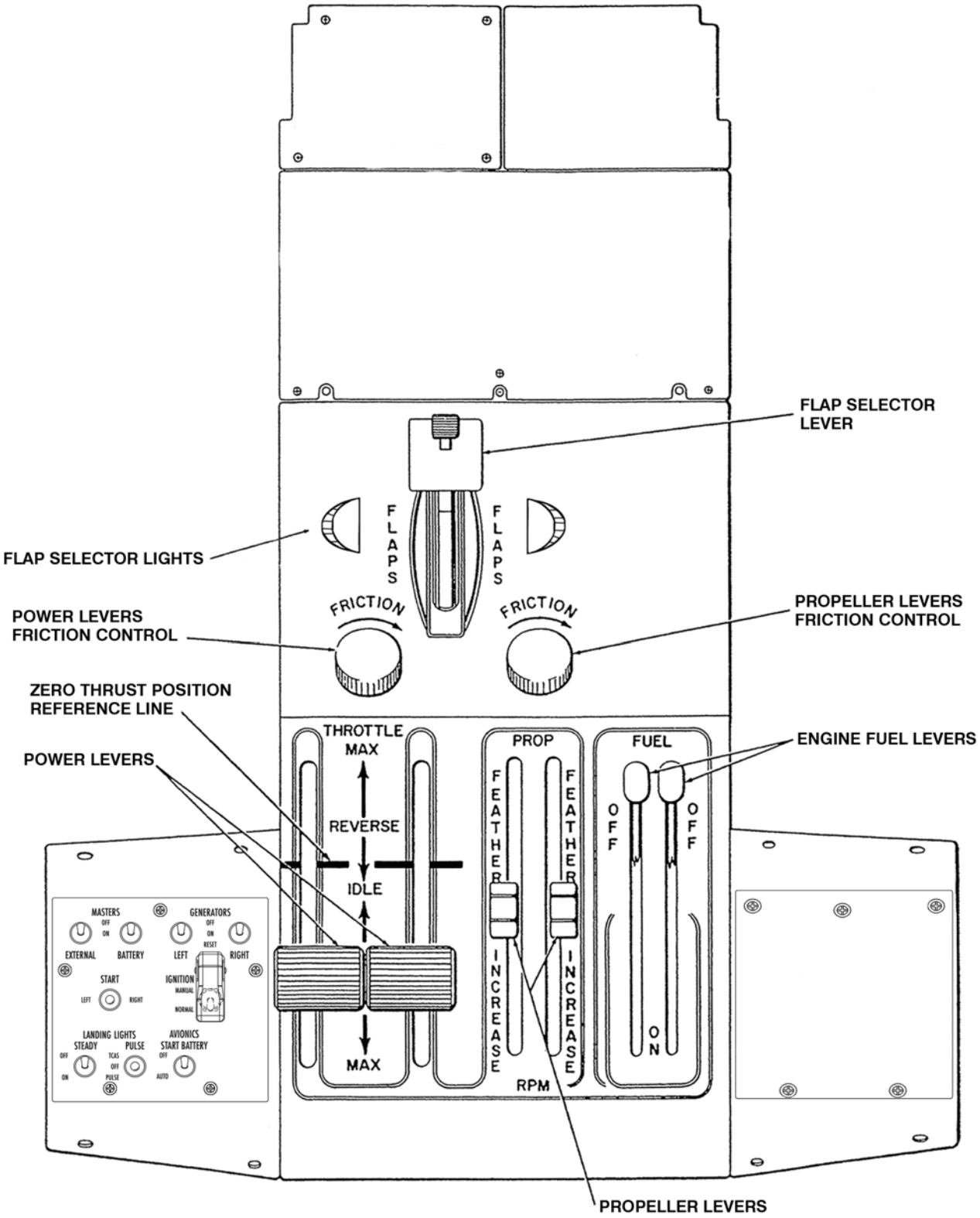
For specific details on the DC generation system refer to PSM 1-64-2W Chapter 24.



Overhead Console – Typical (Pre Mod 6/2314)
Figure 1



Overhead Console – Typical (Post Mod 6/2314) (Pre Mod 6/2315)
Figure 2



Overhead Console – Typical (Post Mod 6/2315)
Figure 3

2. Description

A. Starter-generator

A starter-generator is mounted on the accessory gearbox of each engine. These function as a direct drive starter during engine start, and as a DC generator driven by the engine once the engine is running. The starter-generators each have a nominal regulated output of 28.5 volts at 200 amperes. The full rated 200 amp capacity is available at 59% N_G and above. Following engine start a generator begins generating power when the generator switch is selected to RESET and set to ON. The field circuits are protected by GEN RESET circuit breakers, each of which is inside its respective engine nacelle top cowling.

B. Voltage Regulators

Voltage regulators in each generator circuit control the generator output at 28.5 volts nominal. At low N_G rpm (below 59%) maximum currents as depicted in the following table should not be exceeded.

N_G	Recommended Maximum Current per Generator
Idle (51%) to 57%	40 Amperes Maximum
57% to 59%	100 Amperes Maximum
Above 59% (65% recommended, refer to Note)	200 Amperes Maximum

Note Rated generator capacity is available at 59% N_G, however improved cooling is provided at higher rpm. 65% N_G is recommended for continuous loads in excess of 100 amperes.

The voltage regulators are located on the aft side of the baggage bulkhead (station 332.00). An equalizer circuit in each voltage regulator ensures equal loading (within 20 amperes) when the generators are operating in parallel (relays K5 and K6 energized, and the bus tie switch at NORMAL). The equalizer circuit is disconnected during engine starting (only relays K7 or K8 is energized). When both relays K7 and K8 are energized (IGNITION switch set to MANUAL) the equalizer circuit is connected.

The voltage regulators have an overvoltage sensing function to provide protection in the event of an overvoltage condition developing in the system during operation. An external adjustment screw and locknut is incorporated.

C. Reverse Current Relays

Reverse current relays, K1 and K2, connect starting power to the left and right starter-generators, and also connect generator output to the DC buses and batteries. The relays are mounted in the power distribution and generator control box in the RH cabin roof. Each relay unit comprises three relays. Relay 1 is energized (22 volts minimum pull-in and 18 volts drop-out) through terminal marked switch (SW). Relay 1 closes relay 2, which closes relay 3. Relay 3 senses differential voltage (0.35 to 0.65 volt for pull-in), and reverse current (9 to 25 amperes for drop-out) between terminals marked generator (GEN) and battery (BATT). When a starting circuit is applied to terminal marked APP, relays 1 and 2 are bypassed, and relay 3 is directly energized, connecting DC power from the BATT terminal to the GEN terminal for starting power input to the starter-generator. When the starting circuit is disconnected, the generator circuit is applied to the SW terminal, and normal relay operation becomes effective for generator power output through the GEN terminal, and relay 3, to the BATT terminal. With relay 3 energized, an output circuit on the terminal marked indicating (IND) is used for the generator fail relay circuit (PD-K5 or PD-K6).

In the event of an overvoltage condition developing, these relays are energized by the voltage regulator overvoltage sensing circuit to de-energize the associated generator field relay. Refer to PSM 1-64-2W.

D. Bus Feed Circuit Breakers

Circuit breakers are installed in the power distribution and generator control box in the cabin roof to protect the generation and distribution systems connected to the left and right DC buses. Additional circuit breakers are installed on the main and the avionics power distribution panels in the flight compartment for triplicate left and right DC bus feeds to the appropriate circuit breaker panels.

E. Main Battery

The main battery is a 24 volt, 48 ampere-hour, recombinant gas lead acid type, located below the floor in the rear baggage compartment. On aircraft incorporating S.O.O. 6371, the main battery has been relocated to the nose baggage compartment. The battery is provided with a cover and sealing gasket. The battery does not require provisions for overboard venting and drains.

F. Diodes

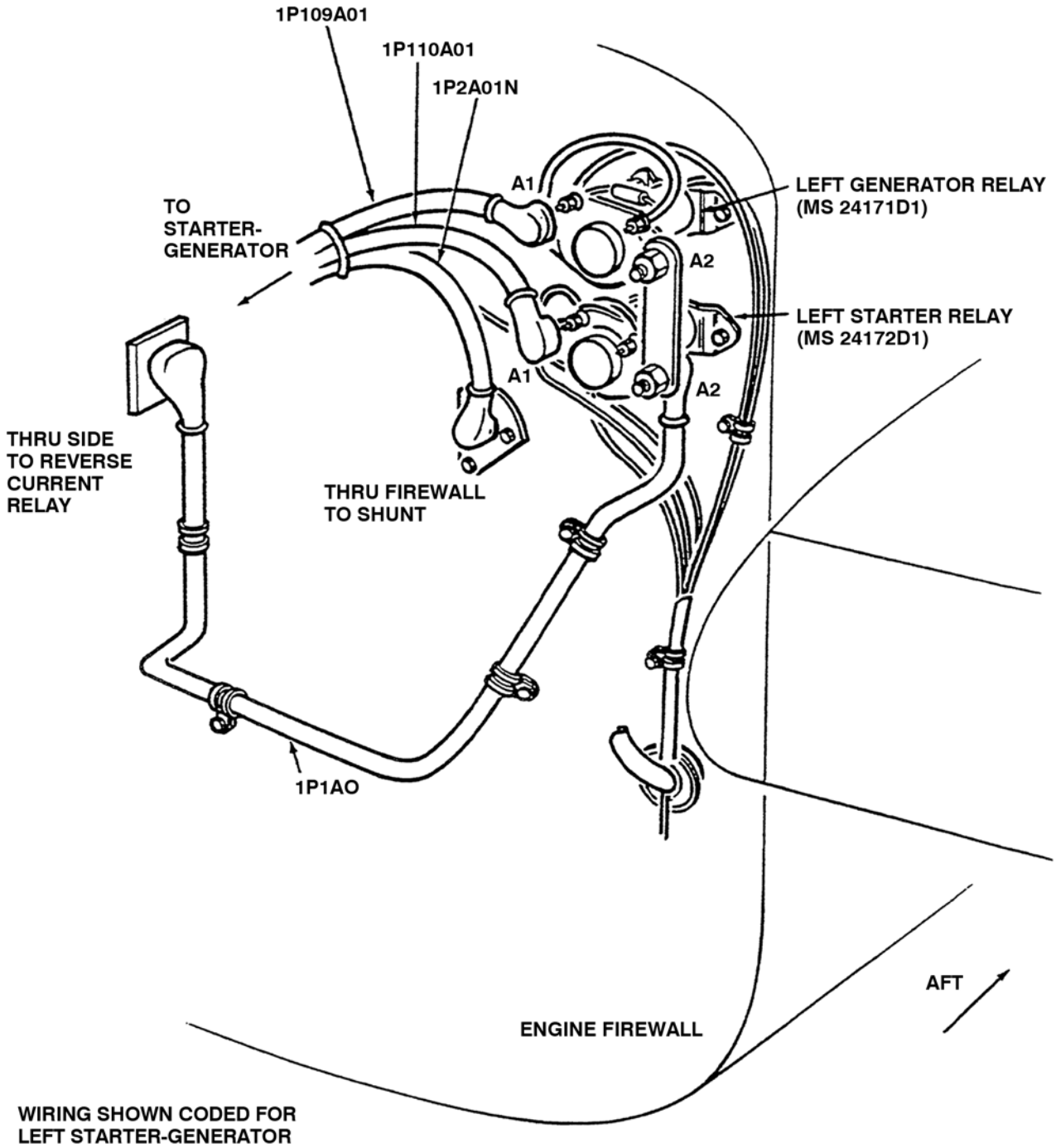
Diodes are incorporated for the following purposes:

1) To block a DC circuit feedback, installed in series with the load, in a forward biased position (anode towards the power source).

2) For DC voltage spike suppression (when a solenoid or relay coil is de-energized), installed in parallel with the coil, in a reverse biased position (cathode towards the power source).

G. Generator and Starter Relays

Left and right generator and starter relays are located on the forward face of each engine firewall (refer to Figure 4). The generator relay (MS24171–D1) is a SPST, 200 ampere continuous duty relay, the starter relay (MS24172D1) is a SPST 200 ampere intermittent duty relay. The relays are not interchangeable.



Location of Generator and Starter Relays
Figure 4

3. Operation

CAUTION **DO NOT CONNECT DC GENERATOR UNTIL ENGINE STABILIZES AT IDLE RPM (N_G) + 15% WITH BOTH ENGINES OPERATING, CONNECT GENERATORS SEPARATELY TO AVOID TRIPPING REVERSE CURRENT CIRCUIT BREAKER. AFTER A BATTERY START MAINTAIN POWER LEVER ABOVE IDLE RPM (N_G) + 15% UNTIL THE APPLICABLE GENERATOR AMPS DECLINE BELOW 100 AMPS. SWITCH OFF OPERATING GENERATOR BEFORE BATTERY START OF SECOND ENGINE.**

Refer to PSM 1-64-2W for the DC Electrical System Schematics.

Refer to Figure 1, Figure 2 and Figure 3 for details.

With the engine started (refer to 80-00-00, Starting – General Data), the engine speed stabilized at IDLE rpm (N_G) + 15%, and the starter switch released, the relevant GENERATORS control switch is momentarily selected to RESET (to excite the generator field) and then released to the ON position. Output from the generator energizes the generator relay, and through the GENERATOR CONTROL 10 ampere circuit breakers in series with a 15 ampere breaker and the GENERATORS control switch, it also energizes the generator field relay. The reverse current main relay coil is energized (via the GEN RESET circuit breaker, the GENERATORS control switch, the MASTER switch (Pre Mod 6/2314) or GENERATOR switch (Post Mod 6/2314), the generator field relay and terminal SW of the reverse current relay) to connect the generator output to the DC buses (BUS TIE switch set to NORMAL) providing the generator voltage is higher than the bus volts. The generator field relay de-energizes the external power relay (if external power is selected) to disconnect the external power circuit. When the generator is connected on-line (reverse current main relay contacts closed), the generator fail relay is energized (via the IND terminal of the reverse current relay) to disconnect the GENERATOR FAIL circuit breakers. A circuit from the left DC bus through the gen excitation breaker is connected through a resistor to terminals A of the generator and voltage regulator to maintain generator buildup and polarity of the field.

With the POWER SOURCE switch selected to BATTERY (Pre Mod 6/2314) or BATTERY switch to ON (Post Mod 6/2314), the main battery relay is energized connecting the main battery to the DC buses. With the generators on-line, the battery charge is maintained by the generators. With the generators off-line, the batteries or external power (refer to 12-00-00, Servicing – General Data), supply all the DC buses providing the BUS TIE switch is selected to NORMAL.

In the event of a generator going off-line, due to a generator overvoltage or line failure, selecting the appropriate GENERATORS control switch momentarily to RESET applies direct generator output through the GEN RESET circuit breaker to the generator field allowing the generator to build up. Moving (or setting) the control switch to ON energizes the reverse current relay to reconnect the generator on-line.

The appropriate GENERATORS control switch must first be selected OFF, to de-energize the latched generator field control relay, before selecting RESET to energize the generator field. Thereafter, releasing the control switch to ON energizes the reverse current relay as above.

CAUTION **LIMIT THE RESET OF A FAILED GENERATOR TO TWO ATTEMPTS ONLY. IF THE ASSOCIATED GENERATOR OFF MESSAGE IS NOT PRESENT BEFORE AND AFTER AN ENGINE START SELECTION (OR AFTER GENERATOR SWITCH-OFF PRIOR TO ENGINE SHUTDOWN), SET BUS TIE SWITCH TO OPEN. ALSO SET BATTERY/EXTERNAL SWITCH TO OFF WHEN LEFT GENERATOR OFF MESSAGE IS INOPERATIVE.**

4. Troubleshooting

Table 1 — DC Generation – Troubleshooting has been compiled to assist operators in the recognition of the causes, with the subsequent rectification action required, of the more commonly reported snags in the Twin Otter's main electrical system. It is hoped that this will help to prevent the unnecessary removal for investigation and repair of items which are actually serviceable or which require only minor adjustment.

It must be appreciated that the following table does not represent a complete troubleshooting guide but reflects the snags which experience has shown to be those most likely to cause the symptoms discussed.

It is assumed that the following table will be used in conjunction with the circuit schematics and diagrams.

Table 1 DC Generation – Troubleshooting

SYMPTOM	POSSIBLE CAUSE	CHECK THAT	REMEDIAL ACTION
1. (a) Low main battery output.	Poor connections of main battery ground line.	Connections of ground line to shunt and shunt to airframe are secure.	Torque nuts to 40 – 45 pound-inches. Ensure shunt contact surfaces are clean.
(b) No main battery relay output.	Open circuit diode in main battery relay control circuit	Power available at terminal A1 of main battery relay.	Replace defective diode.
2. (a) Starter will not operate when start switch selected.		Ignition switch is in "NORMAL" position.	Set correctly.
(b)	Defective relay K16 (left engine) or K17 (right engine).	No power available at terminal "B1" of relay.	Replace defective relay.
(c)	Open circuit diode CR5 – (left engine) CR6 – (right engine).	No power available at "APP" terminal of reverse current relay but power available at terminal "B1" of either K16 or K17 relay.	Replace defective diode. Check diode mounting holes are properly deburred and mica washers are undamaged and not providing short to ground.
(d)	Reverse current relay fails to close.	Power available at "APP" terminal but not at "GEN" terminal of reverse current relay.	Replace reverse current relay.
(e)	Defective starter relay.	Power available at "GEN" terminal of reverse current relay.	Check functioning of starter relay. Replace if confirmed defective.
(f)	Defective relay. K9 – (left engine) K10 – (right engine).	With relay energized terminals A2 and A3 are open from ground.	Replace defective relay.
3. (a) Generator will not come on line when GENERATORS switch is pressed to "RESET", then set to "ON".	Defective switch.	Continuity of switch satisfactory.	Replace switch if defective.
(b) Generator will not come on line when GENERATORS switch is selected "OFF", then to "RESET", then set to "ON" (Post Mod 6/1590).	Generator field control relay stuck in latched position, or normally closed contacts defective. K1A – (Left generator), K2A – (right generator).	There is continuity between terminals A2 and A3 of associated relay de-energized.	Replace defective relay.
(c) (Pre and Post Mod 6/1590 or 6/1636).	Open circuit diode CR3 (left engine) CR4 (right engine).	Power is at diode CR3 (left engine) CR4 (right engine) and continuity across diode.	Replace diode if defective.

Table 1 DC Generation – Troubleshooting (continued)

SYMPTOM	POSSIBLE CAUSE	CHECK THAT	REMEDIAL ACTION
(d)	Defective relay K9 (left engine) K10 (right engine).	With relay de-energized contacts A2 and A3 read to ground.	Replace defective relay.
(e)	Inoperative field exciter circuit.	Check voltmeter circuit breaker is made. Check continuity across resistors R1 (left engine) or R3 (right engine).	Replace defective resistor (100 ohm) if necessary.
4. (a) Generator will not stay on line.	Incorrect setting of voltage regulator, permitting overvoltage relay to operate under transient overvoltage conditions.	Check voltage regulator base contacts. Check relays as at 3.(b).	Carry out voltage regulator adjustment procedure. Replace defective relay as required.
(b)	Excessive load unbalance.		Refer to item 5.(a).
5. (a) Generator fails to load share.	Improperly adjusted voltage regulator.		Adjust voltage regulator. Difference in regulator settings should be less than 0.2 volts.
(b)	Poor connection of generator ground line in rear of nacelle.	Security of ground line to shunt and of shunt to airframe.	Torque nuts to 110 – 124 pound-inches. Ensure shunt contact surfaces are clean.
6.			Reset (always set generators to “ON” separately). Check battery charge condition.
6. Damage to external power relay, battery relay, battery shunt and associated bus bars.	Reverse polarity of ground power unit (Pre Mod 6/1293).		Replace damaged components. As an insurance against occurrence install Mod 6/1293.
7. No external power input	Open circuit diode – external power control input circuit (Post Mod 6/1293).	Power available at terminal X1 of external power relay.	Replace diode if defective.

5. Adjustment/Test

A. Voltage Regulator Adjustment

Note The following adjustment procedure must only be done if a new or overhauled voltage regulator is fitted, or if the load sharing is out of limits (between 10 – 20 amps).

Voltage regulators are normally pre-set to 28.5 volts. When operating in extremely high or low temperatures, slight variation in voltage settings as shown in Table 2 — Temperature/Voltage Settings, will assist in preserving battery condition.

Table 2 Temperature/Voltage Settings

Temperature	Voltage Settings
32°F and below	29.0 volts
33°F to 80°F	28.5 volts
81°F and above	28.0 volts

1) Connect generators on-line (refer to Para 5.B – Function Test DC Generation System).

2) Switch off all loads.

3) Set BUS TIE switch to OPEN.

4) Remove plug button from regulation adjustment control access hole.

5) Loosen locknut on adjustment control using a 5/16 inch hexagon nut drive with a hollow shaft.

6) Adjust voltage in accordance with Table 2 — Temperature/Voltage Settings.

7) Select appropriate power lever to IDLE RPM (N_G) and back to idle (N_G) + 15% to ensure voltage is stable (± 0.2 volts).

CAUTION IT IS IMPORTANT THAT THE RELEVANT GENERATOR LOAD IS LESS THAN 0.5 WHEN THE APPROPRIATE POWER LEVER IS SET TO IDLE RPM.

8) Repeat adjustment procedure Para 5.A.4) to Para 5.A.7) for second generator voltage regulator.

9) Parallel generators by setting BUS TIE switch to NORMAL.

10) Ensure appropriate power levers are set to IDLE (N_G) + 15%.

CAUTION IT IS IMPORTANT THAT THE ENGINES ARE MAINTAINED AT THIS SPEED, AND NOT REDUCED, DURING THE FOLLOWING ADJUSTMENT PROCEDURE.

11) Connect as much load as possible.

12) Verify that each generator is load sharing between 10 – 20 amps (= 20 amps).

13) If load sharing is not within limits, individually and alternately adjust (in small increments) the regulation controls by balancing the high load generator (decrease) against the low load generator (increase) until the limits in Para 5.A.12) are achieved.

Note If a generator comes off-line, repeat Para 5.A.6) and Para 5.A.7) and then repeat balancing procedure but in smaller increments.

14) On completion of adjustment, carefully tighten the regulator adjustment locknut, ensuring the setting is not altered.

CAUTION DO NOT OVERTIGHTEN CONTROL LOCKNUT.

15) Recheck load sharing.

16) Disconnect generators (refer to Para 5.B) and shutdown engines (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

B. Function Test DC Generation System

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Start left engine (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data). Verify no Left Generator Off CAS message is displayed while START switch is held at ON.

3) Release START switch, verify the advisory cyan Left Generator Off advisory CAS message is displayed, and advance power lever to IDLE (N_G) + 15%.

Note After each engine has been running for 90 seconds the advisory CAS message is replaced by a caution CAS message.

Note While either generator is selected off note that the Generators Off indication is shown in the electrical panel.

4) When the engine speed has stabilized, momentarily select the LEFT GENERATOR control switch to RESET and then release to the ON position.

5) Verify no Left Generator Off CAS message is displayed and the Left Generator Amps displays a positive load.

6) If the battery was used to start the left engine and there is any reason to doubt the capability of the battery for starting the right engine, maintain the engine at IDLE + 15% to charge the battery. When the battery charge reading is less than 40 amperes on the Main Battery Amps display, select the LEFT GENERATOR control switch to OFF and verify the amber Left Generator Off caution CAS message is displayed.

7) Start right engine (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data). Verify no Right Generator Off CAS message is displayed while START switch is held at ON.

8) Release start switch, verify the cyan Right Generator Off advisory CAS message is displayed, and advance power lever to IDLE (N_G) + 15%.

9) When the engine speed has stabilized, momentarily select the RIGHT GENERATOR control switch to RESET and then release to the ON position.

10) Verify no Right Generator Off CAS message is illuminated and the Right Generator Amps displays a positive load.

11) Momentarily select LEFT GENERATOR control switch to RESET and then release to the ON position. Verify no Left Generator Off CAS message is illuminated and the Left Generator Amps displays a positive load. Verify the Generators Off is not illuminated.

12) With aircraft loads of approximately 50 amps per bus selected, verify that loads are approximately evenly distributed (within 10 – 20 amps) between the two generators.

13) If external power has been used to start engines, verify the POWER SOURCE switch is set to BATTERY (Pre Mod 6/2314), verify the BATTERY switch is set to ON then set EXTERNAL switch to OFF (Post Mod 6/2314). Verify Main Battery Amps indicates a battery charging condition indicated by a positive reading.

14) Select LEFT GENERATOR control switch to OFF and verify the amber Left Generator Off caution CAS message is displayed and the Right Generator Amps displays as increased reading (i.e., all loads supplied by right generator).

15) Momentarily select LEFT GENERATOR switch to RESET and then release to the ON position. Verify no Left Generator Off CAS message is illuminated and the Left Generator Amps displays a positive load and is balanced with Right Generator Amps.

16) Select RIGHT GENERATOR switch to OFF and verify the amber Right Generator Off caution CAS message is displayed and the Left Generator Amps displays an increased reading (i.e., all loads supplied by right generator).

17) Select BUS TIE switch to OPEN and verify loss of right DC bus. Verify the Bus Tie Open indication is displayed.

18) Reset BUS TIE switch to NORMAL. Verify Right DC Bus is restored and Bus Tie Open indication is not displayed.

19) Select LEFT GENERATOR control switch to OFF and verify the amber Left Generator Off caution CAS message is displayed and the Main Battery Amps displays discharge indicated by a negative reading (i.e. all loads supplied by battery).

20) Shutdown engines (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

21) Select MASTER switch and POWER SOURCE switch to OFF (Pre Mod 6/2314); or select EXTERNAL switch to ON (Post Mod 6/2314).

22) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Bench Test Reverse Current Relay

Refer to Figure 5 for details.

Equipment required:

- Direct current generator, 0 – 28 volts 300 amperes
- Voltage regulator
- SPDT switch No. 3
- Generator field rheostat
- SPDT switch No. 5
- SPDT switch No. 6
- 0 – 30 Volt DC voltmeter
- SPST switch No. 8
- SPDT switch No. 9
- 0 – 5 Volt DC voltmeter
- 300 – 0 – 30 ampere DC ammeter
- 300 ampere at 28 volts load bank
- 0 – 100 millivolts DC millivoltmeter
- 24 Volt DC battery

1) Differential Voltage and Reverse Current Test

a) Connect relay.

b) Set all switches to OFF position.

c) Operate generator at a speed between 3000 and 4500 revolutions per minute.

d) With switch (3) in position 1, adjust generator voltage by means of rheostat (4) so that voltage is approximately one volt less than battery voltage.

Note The voltages of battery (14) and generator (1) can be read and readily compared by means of placing switch (6) alternately in positions 1 and 2 and noting reading on voltmeter (7).

e) With switch (5) in position 1 and switch (8) closed, place switch (9) in position 1 and slowly increase generator voltage by means of the rheostat (4). When cutout operates, voltage indicated on voltmeter (10) will go to zero. Differential voltage is indicated just before voltmeter reading goes to zero and should be between 0.35 and 0.65 volts.

CAUTION SWITCH (9) MUST BE OPENED BEFORE NEXT TEST.

f) Measure reverse current opening by slowly decreasing generator voltage by means of rheostat (4) until cutout opens. Reverse current is read on ammeter (11) just before reading goes to zero. Reverse current opening should be between 9 and 25 amperes when generator voltage is at 26 volts. If generator voltage read on voltmeter (7) with switch (6) in position 2 is not 26 volts, use Table 3 — Reverse Current Opening to obtain the corresponding reverse current opening for various voltages.

Table 3 Reverse Current Opening

GENERATOR VOLTAGE (volts)	REVERSE CURRENT (amperes)	
	Min	Max
28	11	27
26	9	25
24	7	23
22	5	21

g) Test for opening of differential relay due to removal of power to “SW” terminal, first repeat procedure Para 5.C.1).c) through Para 5.C.1).e).

h) Decrease generator voltage by means of rheostat (4) until current reading on ammeter (11) has decreased to approximately zero, then open switch (18).

i) Again decrease generator voltage, dropping out contactor, then continue to decrease voltage to lowest value possible.

j) Open switch (5). Differential contacts now should have opened. To check for open contacts, place ohmmeter across terminals “SW” and “APP”, and note resistance reading. Ohmmeter should read approximately 200 ohms. If contacts remained closed, ohmmeter will read zero resistance.

2) Voltage Relay Tests

a) Set all switches to OFF position.

b) Operate generator at speed between 3000 and 4500 rpm.

c) Set switches (3 and 5) in position 1, and switch (6) in position 2. Slowly vary voltage by means of rheostat (4). Closing voltage should be between 20 and 24 volts and is read on voltmeter (7) when relay operates. Opening voltage should be above 18 volts.

d) Connect ohmmeter from “GEN” terminal to “T” test terminal to indicate when voltage relay contacts have closed. Remove name plate and make a connection to terminal “T” by means of 6-32 screw, 1/2–inch or longer, screwed into tapped hole. This hole is located approximately midway between

terminals "APP" and "SW". File point on screw to be sure of a positive connection. After testing, remove screw and replace name plate.

3) Contactor Test

a) Set all switches to OFF position.

Note This test should be made before contactor coil has become heated, and with relay placed such that four mounting holes are flat on table. Position of relay will have some effect on this operating voltage, even though relay may be mounted in any position in aircraft.

b) Operate generator at speed between 3000 and 4500 rpm.

c) Place switch (3) in position 1 switches (5 and 6) in position 2, then vary voltage by means of rheostat (4) to close and open contactor. Closing voltage, read on voltmeter (7), should be between 14 and 15.5 volts. Opening voltage should be 5 volts or below.

4) Millivolt Drop Test

a) Set all switches to OFF position.

b) Operate generator at speed between 4500 and 6000 rpm.

c) Place switch (3) in position 2, and adjust regulator to 28.5 volts.

d) Place switch (5) in position 2, then close load bank switches until current of 300 amperes is read on ammeter (11).

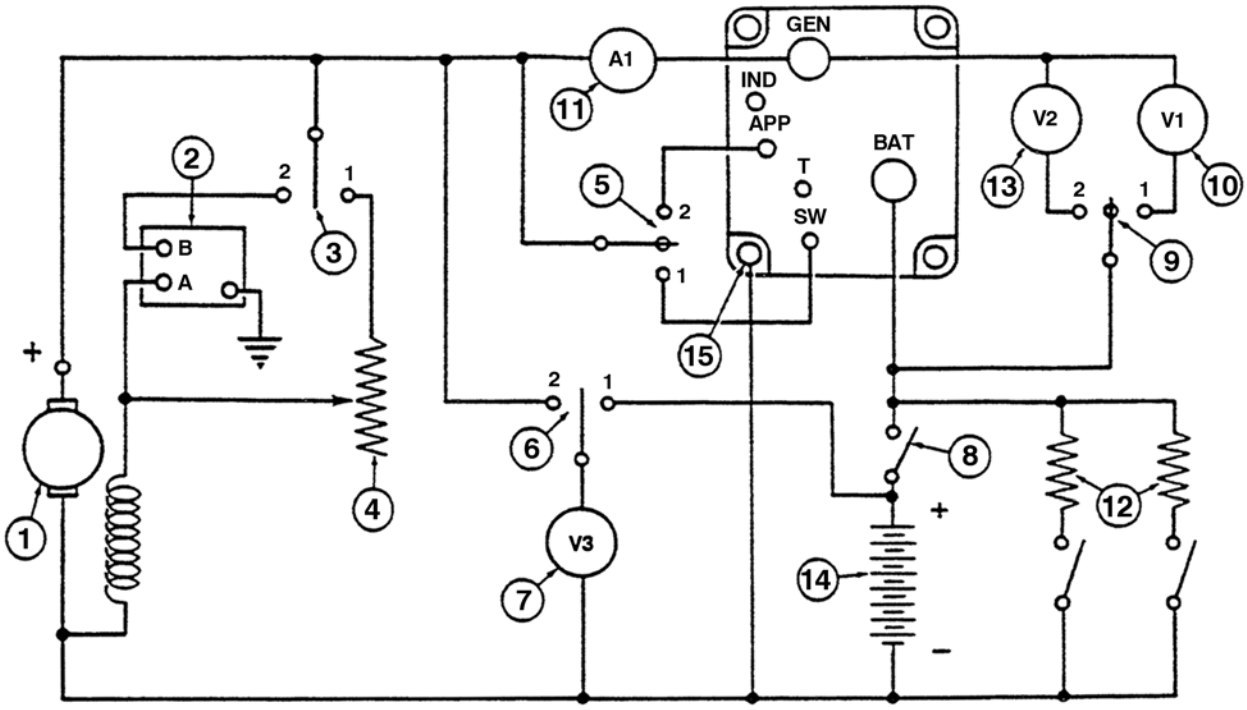
e) Place switch (9) in position 2 and read millivolt drop on millivoltmeter (13). Drop should not exceed 100 millivolts.

CAUTION PLACE SWITCH (9) IN OPEN POSITION BEFORE PERMITTING RELAY TO OPEN. DIFFERENTIAL RELAY CONTACTS SHOULD BE LEFT IN OPEN POSITION TO PREVENT DAMAGE FROM A REVERSE POLARITY GENERATOR.

5) Dielectric Test

a) Using suitable dielectric tester, apply voltage of 1250 volts rms, commercial frequency, for period of one second between following points: each terminal to cover; "BAT" terminal to terminals "IND", "APP" and "SW"; "GEN" terminal to terminals "IND", "APP" and "SW". No breakdown shall occur as a result of this test.

b) Apply 500 volt rms, commercial frequency, for a period of one second between terminals "GEN" and "BAT". No breakdown shall occur.



1. 28V DC 300 AMPERE GENERATOR
2. VOLTAGE REGULATOR
3. SPDT SWITCH
4. GENERATOR FIELD RHEOSTAT
5. SPDT SWITCH
6. SPDT SWITCH
7. 0-30V DC VOLTMETER
8. SPST SWITCH
9. SPDT SWITCH
10. 0-5V DC VOLTMETER
11. 300-0-30 AMPERE DC AMMETER
12. 300 AMP AT 28 VOLTS LOAD BANK
13. 0-100 MILLIVOLTS DC MILLIVOLTMETER
14. 24V DC BATTERY
15. NEGATIVE COVER CONNECTION

Reverse Current Relay Test Circuit
Figure 5

D. Function Test Battery Reserve Polarity Protection Diode (CR109)

Refer to PSM 1-64-2W for electrical schematic.

The following procedure checks the functionality of the battery reserve polarity protection diode (CR109).

Note Diodes that do not pass this test are considered defective and must be replaced.

Equipment Required:

- Digital multimeter, Fluke 87V or similar

1) Aircraft with S.O.O. 6371 (Main Battery installed in Nose Compartment)

a) Ensure aircraft power is OFF and ground power source is disconnected.

b) Disconnect the main battery.

c) Set the MASTER switch to ON and POWER SOURCE switch to BATTERY (Pre Mod 6/2314); or set BATTERY switch to ON (Post Mod 6/2314). This step connects the cathode side of the diode to airframe ground.

d) Gain access to the electrical panel inside the nose compartment at station –6.0.

e) Remove the protective cover to gain access to the main battery relay (K106).

f) Using a digital multimeter in diode test mode, connect the positive test lead to K106 terminal X2 and the negative test lead to airframe ground. This step puts the diode in a forward biased condition.

g) Confirm the forward-bias reading is between 0.5 and 0.8V.

h) Reverse the test leads by connecting the positive test lead to airframe ground and the negative test lead to K106 terminal X2. This step puts the diode in a reverse biased condition.

i) Confirm the reverse bias reading is an open circuit.

j) Disconnect the multimeter and re-install the protective cover at station –6.0.

k) Set the MASTER switch and POWER SOURCE switch to OFF (Pre Mod 6/2314); or set the BATTERY Switch to OFF (Post Mod 6/2314).

l) Reconnect the main battery.

2) Aircraft without S.O.O. 6371 (Main Battery installed underneath floor of rear baggage compartment)

a) Ensure aircraft power is OFF and ground power source is disconnected.

b) Remove the battery compartment floor panel to gain access to the main battery and main battery relay (K106) (refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram).

c) Disconnect the main battery.

d) Set the MASTER switch to ON and POWER SOURCE switch to BATTERY (Pre Mod 6/2314); or set BATTERY switch to ON (Post Mod 6/2314). This connects the cathode side of the diode to airframe ground.

e) Using a digital multimeter in diode test mode, connect the positive test lead to K106 terminal X2 and the negative test lead to airframe ground. This puts the diode in a forward biased condition.

f) Confirm the forward-bias reading is between 0.5 and 0.8V.

g) Reverse the test leads by connecting the positive test lead to airframe ground and the negative test lead to K106 terminal X2. This puts the diode in a reverse biased condition.

h) Confirm the reverse bias reading is an open circuit.

i) Disconnect the multimeter.

j) Set the MASTER switch and POWER SOURCE switch to OFF (Pre Mod 6/2314); or set the BATTERY Switch to OFF (Post Mod 6/2314).

k) Reconnect the main battery and re-install the floor panel.

E. Function Test Ground Power Source Reserve Polarity Protection Diode (CR108)

Refer to PSM 1-64-2W for electrical schematic.

The following procedure checks the functionality of the battery reserve polarity protection diode (CR108).

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR24-17**

Insert in 24-30-00 after Page 18
DC GENERATION

REASON FOR ISSUE:

To add Notes for S.O.O. 6431, Introduction of Station 15.45 Bulkhead – Nose Luggage Compartment.

ACTION:

In Section 24-30-00, ignore existing Para 5.E.1) and insert revised Para, as shown on Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

1) Aircraft with S.O.O. 6371 (Main Battery installed in Nose Compartment)

a) Ensure aircraft power is OFF and ground power source is disconnected.

Note If S.O.O. 6431 is incorporated, remove upper and lower bulkhead panels, removing the bolts which attach the bulkhead panels to the securing brackets.

b) Disconnect the main battery.

c) Set the MASTER switch to ON and POWER SOURCE switch to BATTERY (Pre Mod 6/2314); or set BATTERY switch to ON (Post Mod 6/2314). This step connects the cathode side of the diode to airframe ground.

d) Gain access to the electrical panel inside the nose compartment at station –6.0.

e) Remove the protective cover to gain access to the main battery relay (K106).

f) Using a digital multimeter in diode test mode, connect the positive test lead to K106 terminal X2 and the negative test lead to airframe ground. This step puts the diode in a forward biased condition.

g) Confirm the forward-bias reading is between 0.5 and 0.8V.

h) Reverse the test leads by connecting the positive test lead to airframe ground and the negative test lead to K106 terminal X2. This step puts the diode in a reverse biased condition.

i) Confirm the reverse bias reading is an open circuit.

j) Disconnect the multimeter and re-install the protective cover at station –6.0.

k) Set the MASTER switch and POWER SOURCE switch to OFF (Pre Mod 6/2314); or set the BATTERY Switch to OFF (Post Mod 6/2314).

l) Reconnect the main battery.

Note If S.O.O. 6431 is incorporated, re-install upper and lower bulkhead panels.

NOTE: Please see the TEMPORARY REVISION that revises this page.

Note Diodes that do not pass this test are considered defective and must be replaced.

Equipment Required:

- Digital multimeter, Fluke 87V or similar

1) Aircraft with S.O.O. 6371 (Main Battery installed in Nose Compartment)

   a) Ensure aircraft power is OFF and ground power source is disconnected.

   b) Disconnect the main battery.

   c) Gain access to the external ground power receptacle inside the forward avionics compartment by removing the HIRF protective covers.

   d) Remove the protective cover next to the external ground power receptacle to gain access to the external power relay (K105).

   e) Using a digital multimeter in diode test mode, connect the positive test lead to the control pin terminal on the external ground power receptacle and the negative test lead to K105 terminal X1. This puts the diode in a forward biased condition.

   f) Confirm the forward-bias reading is between 0.5 and 0.8V.

   g) Reverse the test leads by connecting the positive test lead to K105 terminal X1 and the negative test lead to the control pin terminal on the external ground power receptacle. This puts the diode in a reverse biased condition.

   h) Confirm the reverse bias reading is an open circuit.

   i) Disconnect the multimeter and re-install the protective covers.

   j) Reconnect the main battery.

2) Aircraft without S.O.O. 6371 (Main Battery installed in AFT Battery Compartment)

   a) Ensure aircraft power is OFF and ground power source is disconnected.

   b) Remove the floor panel to gain access to the main battery and external power relay (K105) (refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram).

   c) Disconnect the main battery.

   d) Using a digital multimeter in diode test mode, connect the positive test lead to the control pin terminal on the external ground power receptacle and the negative test lead to K105 terminal X1. This puts the diode in a forward biased condition.

   e) Confirm the forward-bias reading is between 0.5 and 0.8V.

   f) Reverse the test leads by connecting the positive test lead to K105 terminal X1 and the negative test lead to the control pin terminal on the external ground power receptacle. This puts the diode in a reverse biased condition.

   g) Confirm the reverse bias reading is an open circuit.

   h) Disconnect the multimeter.

   i) Reconnect the main battery and re-install the floor panel.

F. Power Feeder Cable Inspection at Station 111 Cabin Floor Cut-Out

The following inspection is only applicable to aircraft with S.O.O. 6371 (Forward Relocation of Main Battery and GPU Receptacle).

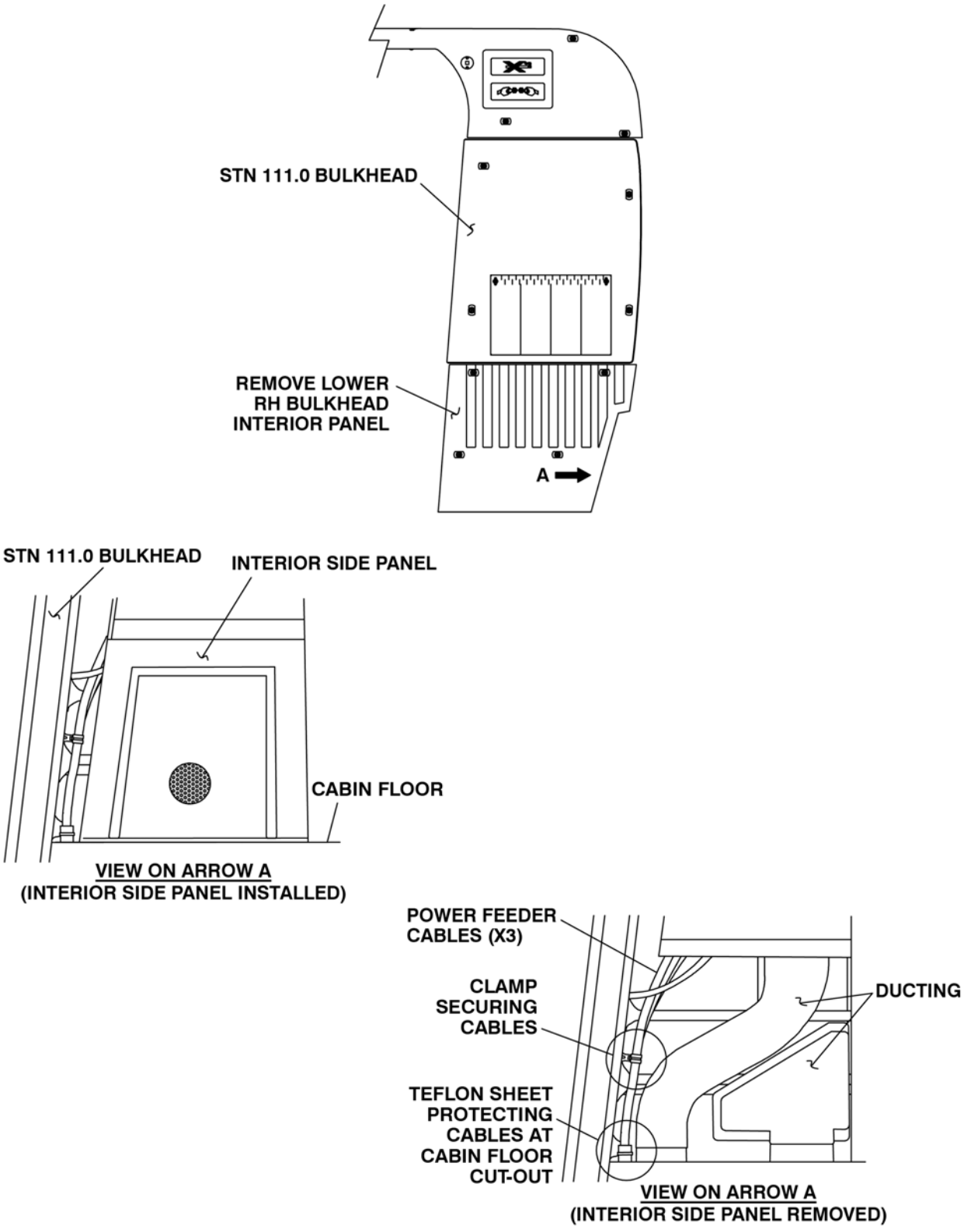
1) Remove the station 111 bottom RH bulkhead interior panel and side panel as shown in Figure 6.

2) Inspect the three power feeder cables near the cabin floor cut-out to ensure there are no signs of mechanical impact, chafing, or overheating.

3) Verify the teflon sheet is securely installed around the cable bundle as shown in Figure 6.

4) Verify the clamp above the cabin floor cut-out is securely installed around the cable bundle as shown in Figure 6.

5) Reinstall the interior panels.



Power Feeder Cable Inspection at Station 111 Cabin Floor Cut-Out
Figure 6

6. Inspection/Check

Refer to Figure 7 for details.

A. Check Installation and Back-to-Back Ratio of Diodes

1) Ensure that electrical power is disconnected, and check for correct polarity of ohmmeter leads on meter to be used for checking.

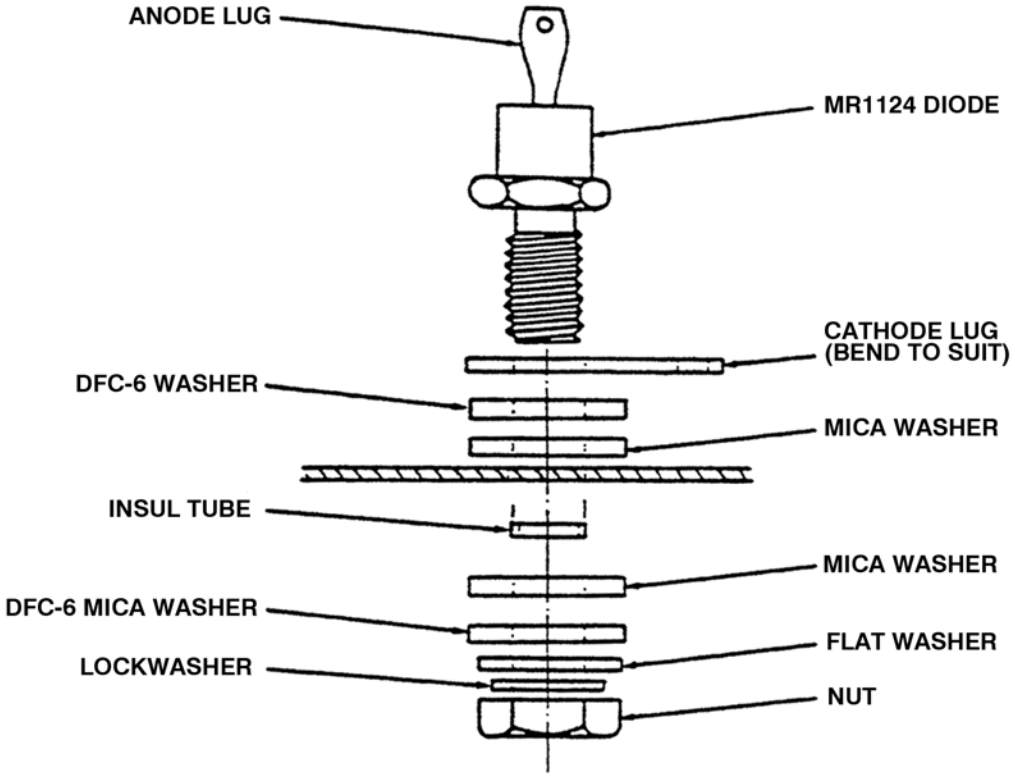
2) Check that blocking diode is in series with load (i.e., anode towards power source).

3) Check that voltage spike suppression diode is connected in parallel with solenoid or relay coil, (i.e., cathode towards power source).

4) When diode is connected with parallel circuit (e.g. soldered across relay coil) apply meter positive lead to anode and negative lead to cathode, and check for low resistance. Then apply negative lead to anode and positive lead to cathode and check that resistance measured is that of the parallel circuit.

5) If possible, disconnect diode from any parallel circuitry, apply meter positive lead to anode and negative lead to cathode of diode, and check for low resistance. Then apply negative lead to anode and positive lead to cathode and check for infinity.

6) When installing MR1124 diodes identified CR5 and CR6 in the main distribution box, ensure that diode mounting holes are properly deburred and install diodes in accordance with Figure 7 and the wrench torque values given in Table 4 — Wrench Torque Values for Terminal Nuts for MR1124 Diodes.



CR5 and CR6 (MR1124) Diode Installation
Figure 7

Table 4 Wrench Torque Values for Terminal Nuts for MR1124 Diodes

STUD SIZE	TORQUE (POUND–INCHES)	
	MINIMUM	MAXIMUM
10/32 inch	12	15
1/4 inch	25	30

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TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR24-18**

Insert in 24-30-31 before Page 1
MAIN BATTERY SYSTEM

REASON FOR ISSUE:

To add Notes for S.O.O. 6431, Introduction of Station 15.45 Bulkhead – Nose Luggage Compartment.

ACTION:

In Chapter 24, Section 24-30-31, ignore existing Para 1. and insert revised Para, as shown on Pages 3 to 4 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

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Main Battery System – Maintenance Practices

1. Removal/Installation

A. Remove Main Battery

CAUTION **BEFORE DISCONNECTING THE BATTERY, ENSURE THAT THE MASTER SWITCH (PRE MOD 6/2314) OR BATTERY SWITCH (POST MOD 6/2314) IS OFF, AND PULL COMP LTS CIRCUIT BREAKER. ANY SUBSTANTIAL LOAD ON THE BATTERY COULD CAUSE DAMAGE TO CONNECTOR PINS AND SOCKETS WHEN BATTERY IS BEING DISCONNECTED.**

Before removing the battery, refer to safety precautions in 24-00-00, Electrical Power – General Data. Refer to Figure 2 for details.

1) Set the MASTER switch to OFF (Pre Mod 6/2314); or set BATTERY and EXTERNAL switches to OFF (Post Mod 6/2314).

2) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Remove the access panel in the rear of the baggage compartment floor. For aircraft incorporating S.O.O. 6371, gain access to the battery through the nose compartment door. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram)

Note If S.O.O. 6431 is incorporated, remove upper and lower bulkhead panels, removing the bolts which attach the bulkhead panels to the securing brackets.

4) Disconnect the battery quick disconnect cable.

5) Remove battery hold down bars.

6) Disengage the battery.

WARNING **BATTERIES ARE VERY HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.**

7) Remove the battery from the tray.

B. Install Main Battery

1) Inspect the battery for damage. Cracks in metal or plastic containers are not permitted. Dents in the metal containers that impinge on the interior plastic container are not acceptable.

2) Set the MASTER switch to OFF (Pre Mod 6/2314); or set BATTERY and EXTERNAL switches to OFF (Post Mod 6/2314).

3) Disconnect DC Power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Remove the access panel in the rear of the baggage compartment floor. For aircraft incorporating S.O.O. 6371, gain access to the battery through the nose compartment door. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram)

Note If S.O.O. 6431 is incorporated, remove upper and lower bulkhead panels, removing the bolts which attach the bulkhead panels to the securing brackets.

5) Ensure the battery tray is clean and dry.

WARNING **BATTERIES ARE VERY HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.**

6) Install the battery in the container.

CAUTION **USE AN OPEN END WRENCH ON THE FLATS OF THE BATTERY TERMINAL, WHERE AVAILABLE, WHILE TORQUING TERMINAL BOLTS. FAILURE TO DO SO MAY RESULT IN RUPTURE OF THE BATTERY SEAL AT THE TERMINAL AND THE PREMATURE FAILURE OF THE BATTERY.**

CAUTION **USE ONLY THE HARDWARE PROVIDED WITH THE BATTERY. DO NOT USE STAINLESS STEEL OR STEEL WASHERS BETWEEN THE RING TERMINAL AND THE BATTERY TERMINAL.**

7) Engage battery hold down hardware, torque and lock wire battery connector.

8) Connect the battery quick connect.

9) Replace the access panel in the rear baggage compartment floor.

NOTE: Please see the TEMPORARY REVISION that revises this page.

Main Battery System – Maintenance Practices

1. Removal/Installation

A. Remove Main Battery

CAUTION **BEFORE DISCONNECTING THE BATTERY, ENSURE THAT THE MASTER SWITCH (PRE MOD 6/2314) OR BATTERY SWITCH (POST MOD 6/2314) IS OFF, AND PULL COMP LTS CIRCUIT BREAKER. ANY SUBSTANTIAL LOAD ON THE BATTERY COULD CAUSE DAMAGE TO CONNECTOR PINS AND SOCKETS WHEN BATTERY IS BEING DISCONNECTED.**

Before removing the battery, refer to safety precautions in 24-00-00, Electrical Power – General Data. Refer to Figure 2 for details.

1) Set the MASTER switch to OFF (Pre Mod 6/2314); or set BATTERY and EXTERNAL switches to OFF (Post Mod 6/2314).

2) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Remove the access panel in the rear of the baggage compartment floor. For aircraft incorporating S.O.O. 6371, gain access to the battery through the nose compartment door. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram)

4) Disconnect the battery quick disconnect cable.

5) Remove battery hold down bars.

6) Disengage the battery.

WARNING **BATTERIES ARE VERY HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.**

7) Remove the battery from the tray.

B. Install Main Battery

1) Inspect the battery for damage. Cracks in metal or plastic containers are not permitted. Dents in the metal containers that impinge on the interior plastic container are not acceptable.

2) Set the MASTER switch to OFF (Pre Mod 6/2314); or set BATTERY and EXTERNAL switches to OFF (Post Mod 6/2314).

3) Disconnect DC Power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Remove the access panel in the rear of the baggage compartment floor. For aircraft incorporating S.O.O. 6371, gain access to the battery through the nose compartment door. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram)

5) Ensure the battery tray is clean and dry.

WARNING **BATTERIES ARE VERY HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.**

6) Install the battery in the container.

NOTE: Please see the TEMPORARY REVISION that revises this page.

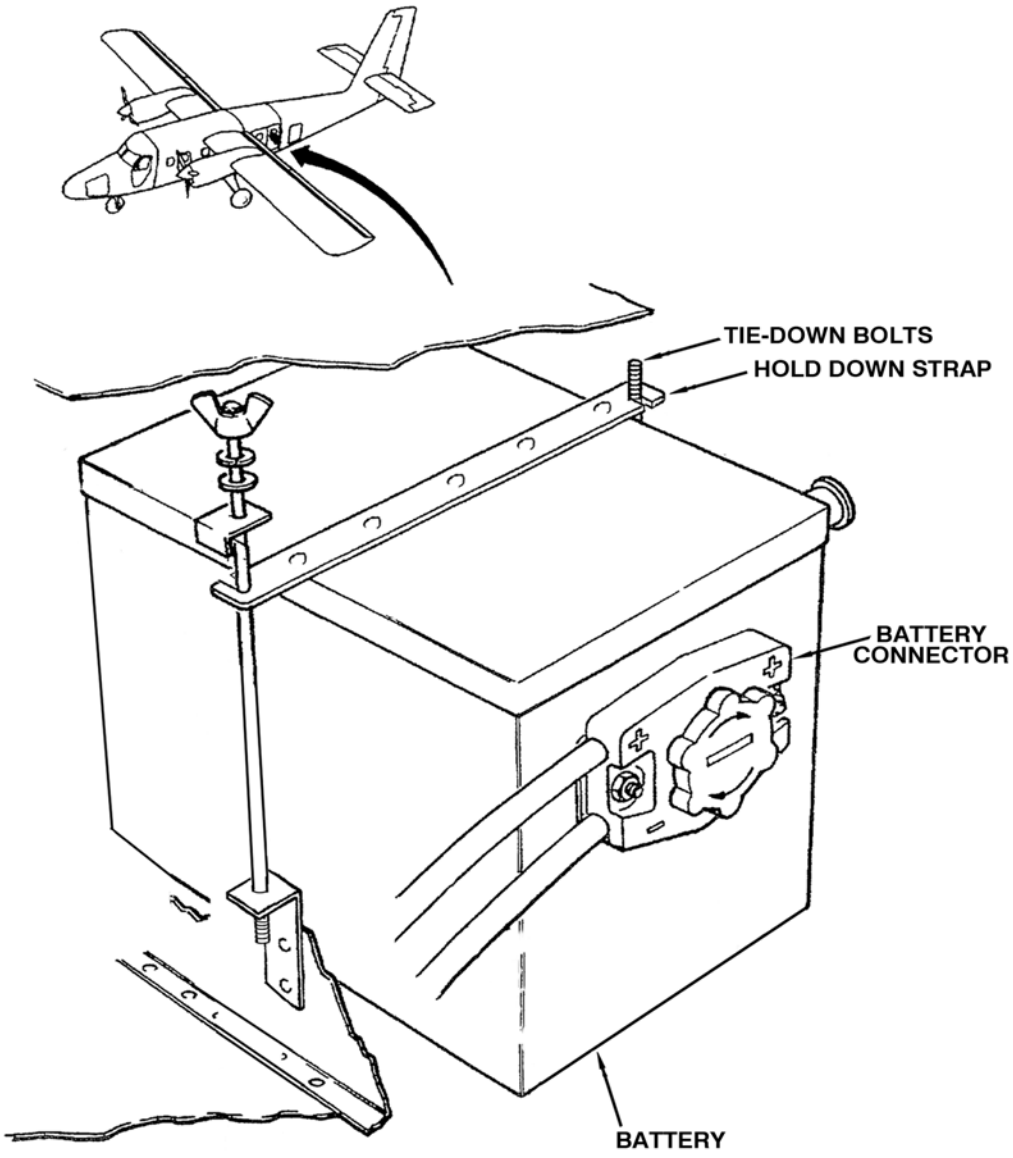
CAUTION **USE AN OPEN END WRENCH ON THE FLATS OF THE BATTERY TERMINAL, WHERE AVAILABLE, WHILE TORQUING TERMINAL BOLTS. FAILURE TO DO SO MAY RESULT IN RUPTURE OF THE BATTERY SEAL AT THE TERMINAL AND THE PREMATURE FAILURE OF THE BATTERY.**

CAUTION **USE ONLY THE HARDWARE PROVIDED WITH THE BATTERY. DO NOT USE STAINLESS STEEL OR STEEL WASHERS BETWEEN THE RING TERMINAL AND THE BATTERY TERMINAL.**

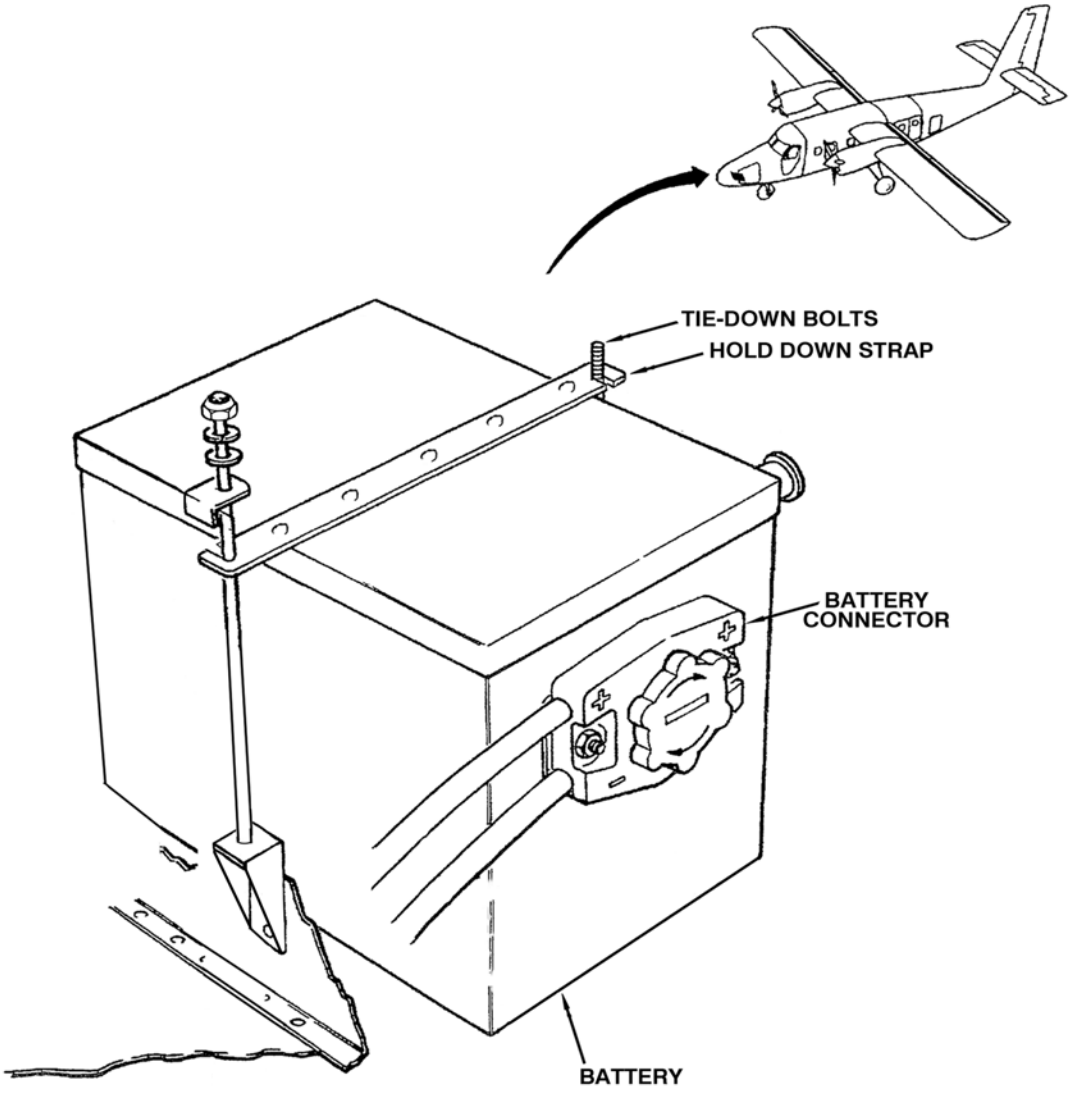
7) Engage battery hold down hardware, torque and lock wire battery connector.

8) Connect the battery quick connect.

9) Replace the access panel in the rear baggage compartment floor.



Main Battery Installation
Figure 1



Main Battery Installation (S.O.O. 6371)
Figure 2

2. Adjustment/Test

A. Constant Voltage Charging

In the DHC-6 Series 400 aircraft, valve regulated lead acid batteries are used, and the constant voltage charging method is employed, using the voltage-regulated generator charging circuit. The graph shown in PSM 1-6-6A, Manufacturer's Information, indicates what happens essentially during this type of charge.

A voltage produced by the generator permits current to flow to the battery. In a discharged battery, the maximum surge of current will be approximately 10 times the rated capacity. The high surge currents are due to the low internal resistance of the battery. As the battery approaches a charged state, the current falls off rapidly and the voltage rises. The recommended voltage setting for the regulator, is computed by multiplying the number of cells in the battery by a factor of 1.5 to 1.55.

B. Apparent Loss of Capacity

When apparent loss or temporary loss of capacity occurs, the battery capacity will be lower than the rated capacity. The lower capacity is normally indicated by an abnormal slow engine starting performance or if abnormally high charging current is required to maintain the battery at bus voltage.

C. Capacity Test

1) Special Tools

a) Advanced Power Products Beta D–50 Aircraft Battery Analyzer, P/N 4146 or 4159 or equal.

b) Advanced Power Products Alpha C–25 Battery Charger Part Number 4142 or 4159 or equal.

WARNING **CONTACT CONCORDE FOR DETERMINATION OF EQUAL TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY BATTERY.**

2) Procedure

a) Remove battery from aircraft as per Para 1.A.

b) Stabilize the battery at 15°C or higher. The battery must be at this temperature for at least 24 hours.

c) If battery is not fully charged at this point, charge it as per Para 3.B.

d) Discharge the battery at the C1 rate on the label to the end point voltage (EPV) of 20 volts.

e) Record the time to the end point voltage (EPV).

f) The battery is acceptable for continued use if the ampere hour capacity (actual hours if discharge X ampere rate of discharge) is greater than 85% of the normal capacity (C1) shown on the battery label (i.e. 51 minutes or more).

g) If the battery passes first capacity test, charge the battery as per Para 3.B and return to service.

h) If the battery failed the capacity test for the first time, follow the battery conditioning procedure as per Para 3.C.

i) If the battery failed the capacity test for a second time, replace the battery.

j) After battery has been conditioned, repeat the capacity test.

Note If the battery gets very hot (external case temperature greater than 55°C (130°F)) during charge, it should be replaced.

- If battery passes, refer to Para 2.C.2).g).
- If the battery fails, replace it.

k) Install battery as per Para 1.B.

3. Inspection/Check

A. Inspect Main Battery

A visual inspection in the aircraft should be made of the battery, and battery connector, receptacle and vent lines should be checked for contact failure or damage and the allen head screws or nuts should be torqued, as per manufacturer's data.

B. Charge the Battery

1) Special Tools

a) Advanced Power Products Beta D–50 Aircraft Battery Analyzer, P/N 4126, or equal.

b) Advanced Power Products Alpha C–25 Battery Charger, P/N 4142, or equal.

c) Advanced Power Products CA15–50 Charger/analyzer P/N 4159, or equal.

WARNING CONTACT CONCORDE FOR DETERMINATION OF EQUAL TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY THE BATTERY.

2) Depending on the type of charger available, charge the battery Constant Potential (CP). Charge at 28.2 volts for 24 volt batteries until the charge current stabilizes for 1 hour.

WARNING ONLY CONSTANT POTENTIAL CHARGING MAY BE DONE ON THE AIRCRAFT. DO NOT CONSTANT CURRENT CHARGE A BATTERY ON THE AIRCRAFT. THERE MAY BE A SERIOUS RISK OF INJURY TO PERSONNEL AND/OR DAMAGE TO THE AIRCRAFT OR AIRCRAFT SYSTEMS.

C. Battery Conditioning Procedure

WARNING REMOVE BATTERY FROM AIRCRAFT PRIOR TO DOING A BATTERY CONDITIONING PROCEDURE.

1) Special Tools

a) Advanced Power Products Beta D–50 Aircraft Battery Analyzer, P/N 4146 or 4159 or equal.

b) Advanced Power Products Alpha C–25 Battery Charger P/N 4142 or 4159 or equal.

WARNING CONTACT CONCORDE FOR DETERMINATION OF EQUAL TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY BATTERY.

2) Procedure

a) Remove battery from aircraft as per Para 1.A.

b) Discharge battery at C1 rate, noted on the battery label, to end point voltage of 18 volts.

c) Constant current charge at the (C1/10) rate for 16 hours or until the battery voltage on charge is 31 volts or higher for 4 hours.

Note C1 as shown on battery label.

WARNING BATTERY CAN BE DAMAGED IF CONDITION PROCEDURE DONE TOO MANY TIMES.

d) Allow battery to cool down. (Approximately 8 hours).

e) After battery has been conditioned, repeat capacity test as per Para 2.C.

4. Battery Top Charging using Ground Power (Mod 6/2314)

WARNING **ONLY CONSTANT POTENTIAL CHARGING MAY BE DONE ON THE AIRCRAFT. DO NOT CONSTANT CURRENT CHARGE A BATTERY ON THE AIRCRAFT, THERE MAY BE A SEROUS RISK OF INJURY TO PERSONNEL AND/OR DAMAGE TO THE AIRCRAFT OR AIRCRAFT SYSTEM.**

CAUTION **THIS PROCEDURE IS ACCEPTABLE ONLY WHEN A KNOWN GOOD BATTERY HAS BEEN DISCHARGED THROUGH EXTENDED GROUND USE AND THERE IS CONCERN ABOUT ITS CAPABILITY TO POWER THE AIRCRAFT AND AVIONICS DURING AN ENGINE START.**

CAUTION **THIS PROCEDURE REQUIRES A KNOWN GOOD EXTERNAL GROUND POWER SOURCE THAT IS CAPABLE OF PROVIDING CLEAN AND CONSTANT DC VOLTAGE AND HAS AN AMPERE RATING CAPABLE OF POWER THE BASIC AIRCRAFT (APPROXIMATELY 35 AMPERES) PLUS BATTERY CHARGING CURRENT WHICH CAN EXCEED 100 AMPERES (135 AMPERES TOTAL).**

When Mod 6/2314 is embodied, it is possible to turn on EXTERNAL power and BATTERY power at the same time. When the BATTERY and EXTERNAL switches are both ON it is possible to top charge the battery using ground power.

Note This procedure does not replace the battery conditioning procedure or the battery charge procedure.

A. Check the battery voltage:

1) BATTERY switch ON, EXTERNAL switch OFF.

2) When the cockpit displays illuminate, check BATTERY VOLTS and AMPS.

3) If the BATTERY VOLTS indicates 20 Volts or lower, do not use this procedure. Refer to Para 3.B.

4) Battery discharge current is displayed as a negative number, typically –35 A, depending on aircraft electrical loads.

B. Connect ground power:

1) With the EXTERNAL switch OFF, connect 28 Volt DC power source to external power receptacle.

2) Check the GND PWR voltage displayed on the avionics display.

3) Set the ground power voltage as shown in the following table. The voltage is displayed on the cockpit display.

CAUTION **BATTERY CAN BE OVERCHARGED BY SETTING THE GROUND POWER VOLTAGE TOO HIGH. HIGH CHARGING VOLTAGE WILL SHORTEN THE BATTERY LIFE SIGNIFICANTLY.**

Battery Temperature (use OAT reading)	Ground Power Voltage Setting (Volts DC)
Below 0°C (32°F)	29.0 – 29.5
0 to 15°C (32 to 59°F)	28.5 – 29.0
16 to 30°C (60 to 86°F)	28.0 – 28.5
31 to 45°C (87 to 113°F)	27.5 – 28.0
Above 45°C (113°F)	27.0 – 27.5

4) While observing the cockpit electrical display, switch the EXTERNAL switch ON. GND PWR turns green. BATTERY V and GND PWR V displays the same value (within 0.1 volt). A slight drop of GND PWR voltage is acceptable depending on the capability of the external DC power source.

5) Immediately check the battery current. The value on the cockpit display is positive indicating battery charge. (The cockpit dipslay is capable of presenting values between –100 A and + 100 A. A blank value indicates out of range, i.e. > 100 Amps).

WARNING **IF THE READING EXCEEDS THE CAPABILITY OF THE EXTERNAL DC POWER SOURCE, OR IF THE VOLTAGE IS FLUCTUATING, SET THE EXTERNAL SWITCH TO OFF IMMEDIATELY AND INVESTIGATE.**

Note Normal top charge is indicated by slowly decreasing battery charge current. Typical values start at below 70 Amps decreasing to 5 Amps, the top charge is complete.

6) When the charge current has dropped to less than 5 Amps, the top charge is complete.

7) Set BATTERY and EXTERNAL switches as required.

Avionics Start Battery (Mod 6/2315) – General Data

1. General

Mod 6/2315 introduces a 24 volt, 3.3 amp-hour lead-acid avionics start battery. The battery supplies power to the electrical systems required for monitoring engine indications during an engine start. When an engine start is initiated, larger electrical loads are placed on the main battery. This can cause the voltage on the main battery to drop below the minimum voltage required to operate the avionics systems. The avionics start battery is used to maintain the voltage to certain avionics systems so that a start can be accomplished without the loss of engine indications and displays. These systems include MAU1B, MAU2B, DAU1A, DAU1B, DAU2A, DAU2B, AURAL ALERT 1, AURAL ALERT 2, CAUTION LIGHTS, and all four displays. The circuit breakers for these systems are identified with a white circle around the circuit breaker cutouts on the avionics circuit breaker panel.

A schematic of the avionics start battery circuit is shown in Figure 1.

2. Description

A. Avionics Start Battery

The avionics start battery is a 24 volt, 3.3 ampere-hour recombinant gas lead acid type. It is located on the upper of the three avionics racks between the nose baggage compartment and the flight compartment at approximately STA 50.00, beside the MAU. The battery does not require provisions for overboard venting and drains.

B. AVIONICS START BATTERY Switch

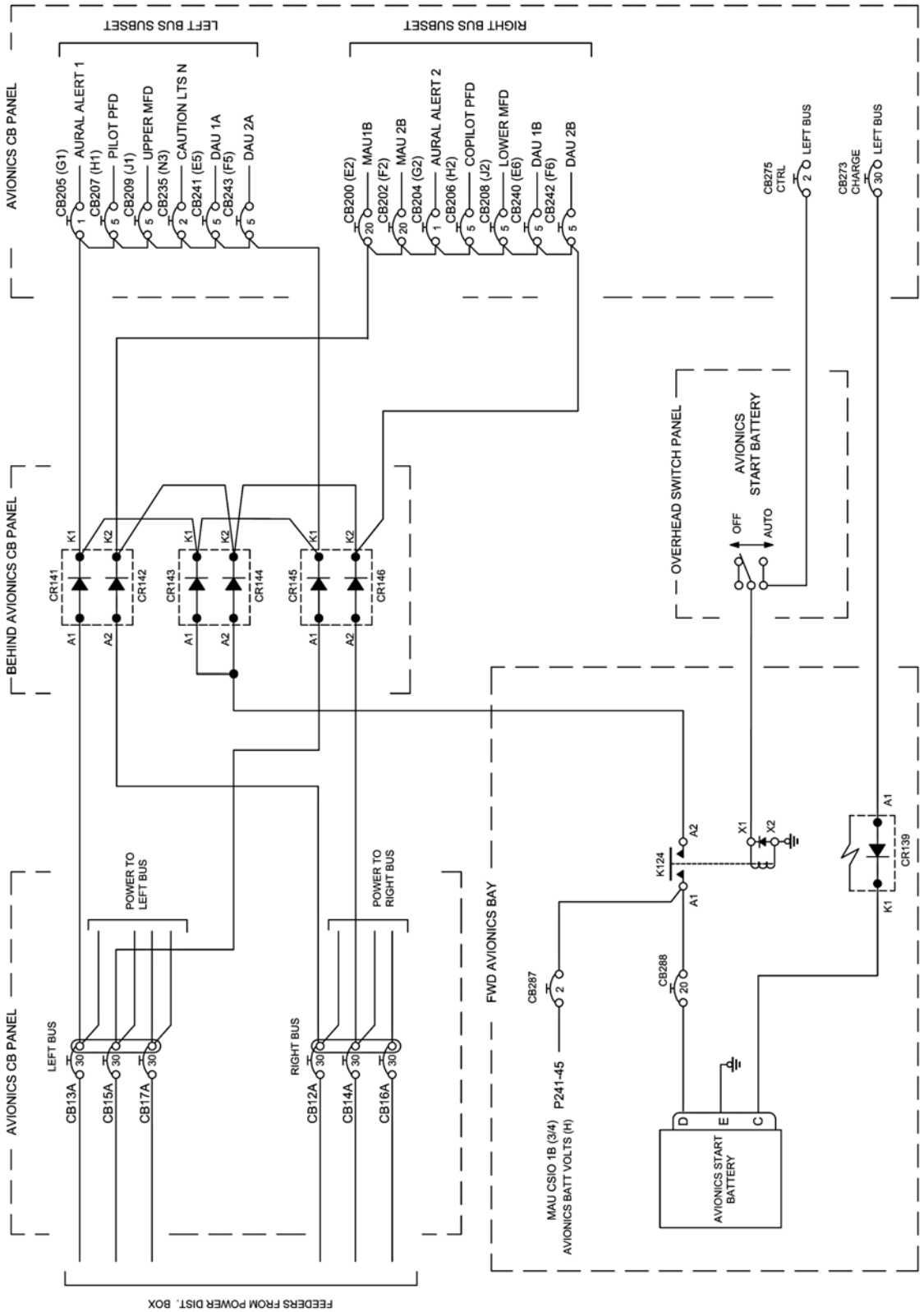
The AVIONICS START BATTERY switch is mounted on the overhead console and is marked, OFF and AUTO. When the switch is set to OFF the avionics start battery output is disconnected. When the switch is set to AUTO, the avionics start battery will maintain voltage to the equipment during the low voltage period, so that engine instruments and warnings continue to function for the engine start.

C. Avionics Start Battery Circuit Breakers

The avionics start battery circuit includes four circuit breakers.

Two of the circuit breakers are located on the top left shelf in the forward avionics bay and are labeled CB287 and CB288. CB287 (2 amps) protects the voltage indication circuit connecting to the MAU. CB288 (20 amps) protects the power output circuit from the battery.

The other two circuit breakers are installed on the avionics circuit breaker panel and are labeled AVIONICS STRT BATT CHARGE (M7) and AVIONICS STRT BATT CTRL (N7). The CHARGE circuit breaker (30 amps) protects the circuit connecting to the battery charge input. The CTRL circuit breaker (2 amps) protects the circuit connecting to the coil of the battery control relay (K124).



Avionics Start Battery Circuit – Electrical Schematic
Figure 1

D. Avionics Start Battery Circuit Diodes

The avionics start battery circuit includes seven Schottky power diodes. The diodes are packaged in modules containing two diodes per module.

1) Charge Circuit Diode (CR139)

The charge circuit diode is located on the top left shelf in the forward avionics bay and is labeled CR139. The diode is connected to avionics start battery charge input and prevents the avionics start battery from discharging through the charge circuit.

2) Power Isolation Diodes (CR141 thru CR146)

The power isolation diodes are located in the center pedestal forward of the avionics circuit breaker panel and are labeled CR141, CR142, CR143, CR144, CR145, and CR146. The diodes are used to isolate the systems that are not powered by the avionics start battery. The diodes are also used to maintain isolation between the left and right bus.

E. Avionics Start Battery Control Relay

The avionics start battery control relay is located on the top left shelf in the forward avionics bay and is labeled K124. The relay is a SPST type with a continuous duty rating of 50 amps.

3. Operation

The battery is controlled by the AVIONICS START BATTERY switch located on the overhead switch panel. When aircraft power is on and the AVIONICS START BATTERY switch is in the AUTO position, the battery control relay closes and connects the battery in parallel with the main battery. Diodes are used to isolate the battery from the systems that are not required for monitoring engine indications during the start. These diodes are also used to maintain proper isolation between the left and right bus.

The output voltage of avionics start battery is monitored by the AVIONICS BATTERY LOW advisory level CAS message. The CAS message is displayed when the avionics start battery voltage is below 19 volts.

A schematic of the avionics start battery circuit is shown in Figure 1.

4. Removal/Installation

A. Removal of Avionics Start Battery

CAUTION **BEFORE DISCONNECTING THE BATTERY, ENSURE THAT THE FOLLOWING SWITCHES ARE OFF:**

- **AVIONICS START BATTERY SWITCH**
- **MASTER SWITCH (PRE MOD 6/2314)**
- **BATTERY AND EXTERNAL SWITCH (POST MOD 6/2314)**

SUBSTANTIAL LOAD ON THE BATTERY COULD CAUSE DAMAGE TO THE CONNECTOR PINS AND SOCKETS WHEN THE BATTERY IS BEING DISCONNECTED.

Before removing the battery, refer to safety precautions in 24-00-00, Electrical Power – General Data.

1) Set the MASTER switch to OFF (Pre Mod 6/2314); or set BATTERY and EXTERNAL switches to OFF (Post Mod 6/2314).

2) Set the AVIONICS START BATTERY switch to OFF.

3) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Ensure the aircraft power is OFF.

5) Gain access to the forward avionic shelves through the forward baggage compartment.

6) Remove the top cover attached to bulkhead station 37.21.

7) Pull CB273 and CB288 and fit with safety clips.

Note CB273 is located on the avionics circuit breaker panel and is labeled AVIONCIS STRT BATT CHARGE (M7). CB288 is located on the bracket behind the avionics start battery.

8) Disconnect the battery connector (P474).

9) Remove battery mounting screws.

WARNING BATTERIES ARE VERY HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT.

10) Carefully remove battery from aircraft.

B. Installation of Avionics Start Battery

1) Inspect the battery for damage. Cracks in metal or plastic containers are not permitted. Dents in the metal containers that impinge on the interior plastic container are not acceptable.

2) Set the MASTER switch to OFF (Pre Mod 6/2314); or set BATTERY and EXTERNAL switches to OFF (Post Mod 6/2314) and the AVIONICS START BATTERY switch to OFF.

3) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Ensure aircraft power is OFF.

5) Gain access to the forward avionic shelves through the forward baggage compartment.

6) Remove the top cover attached to bulkhead station 37.21.

7) Ensure that avionics shelf is clean and dry.

8) Verify CB273 and CB288 are pulled.

Note CB273 is located on the avionics circuit breaker panel and is labeled AVIONCIS STRT BATT CHARGE (M7). CB288 is located on the bracket behind the avionics start battery.

9) Install battery mounting hardware. Ensure battery is secure on shelf.

10) Connect the battery connector (P474)

11) Reset circuit breakers CB273 and CB288.

12) Re-install the top cover attached to bulkhead station 37.21.

5. Adjustment/Test

A. Capacity Test

Equipment Required:

- Electronic Load, capable of constant current at C1 ADC and 20-28 VDC with EPV termination at 20 VDC
- Mating connector MS27467T15B5P
- Digital Multimeter capable of measuring 18-34 VDC with accuracy of 1% or better

1) Stabilizer the battery at 20°C or higher. The battery must be at this temperature for at least 24 hours.

2) Connect the battery to a digital multimeter using mating connector MS27467T15B5P and measure the open circuit voltage (OCV):

a) If battery is at 20 volts or above, charge the battery following the constant potential charging procedure in Para 6.B.

b) If battery is below 20 volts, remove the battery from the aircraft and do the deep discharge recovery procedure in Para 6.C.

3) Discharge the battery at the C1 rate on the label to the end point voltage (EPV) of 20 volts.

4) Record the time to the end point voltage (EPV).

5) The battery is acceptable for continued use if the ampere hour capacity (actual hours of discharge X ampere rate of discharge) is greater than 85% of the normal capacity (C1) shown on the battery label (i.e. 51 minutes or more).

6) If the battery passes first capacity test, charge the battery as per Para 6.B and return to service.

7) If the battery failed the capacity test for the first time, follow the battery conditioning procedure as per Para 6.D. After battery has been conditioned, repeat the capacity test.

8) If the battery failed the capacity test for a second time, follow the battery conditioning procedure as per Para 6.D. After battery has been conditioned, repeat the capacity test.

9) If the battery failed the capacity test for a third time, or gets very hot during constant charging, replace the battery.

6. Inspection/Check

A. Inspect Avionics Start Battery

A visual inspection in the aircraft should be made of the battery, battery connector and receptacle should be checked for corrosion, contact failure or damage and loose or missing fasteners. Ensure that the identification and information labels are securely attached and legible.

B. Constant Potential Charging Procedure

Equipment Required:

- Constant voltage power supply capable of 28.0 - 28.5 VDC and at least 0.2xC1 ADC output
- Mating connector MS27467T15B5P

Note Constant potential charging is the preferred method of charging the battery. If the battery does not pass the capacity test, the conditioning charge procedure should then be used.

1) Connect the battery to the constant potential charging equipment, using mating connector MS27467T15B5P.

2) Apply a constant potential of 28.25 ± 0.25 volts with a current capacity of at least 0.2xC1 amperes.

Note The output current of the charging equipment should be set as high as possible. RG® series batteries can accept charging rates up to 8xC1.

3) Charge until the charge current falls below 5% of the battery C1 rating (i.e., 0.17 amperes for a 3.3 AH battery) and then continue charging at the same constant potential for a minimum of 4 more hours.

Note Terminate charging if the charge current does not fall below 5% of the battery C1 rating within 8 hours.

C. Deep Discharge Recovery Procedure

WARNING REMOVE BATTERY FROM AIRCRAFT PRIOR TO DOING A DEEP CYCLE CHARGE PROCEDURE. THE PROCEDURE SHOULD BE DONE IN A WELL VENTILATED AREA BECAUSE A SIGNIFICANT AMOUNT OF HYDROGEN GAS MAY BE RELEASED FORM THE BATTERY.

CAUTION IF THE BATTERY BECOMES HOT (ABOVE 55°C/130°F) DURING THE CHARGE, STOP THE CURRENT AND ALLOW THE BATTERY TO COOL TO ROOM TEMPERATURE BEFORE CONTINUING THE CHARGE.

Equipment Required:

- Constant current power supply, capable of C1/10 ADC and at least 34 VDC output
- Mating connector MS27467T15B5P

1) Connect the battery to the constant current charging equipment using mating connector MS27467T15B5P.

2) Charge at a constant current rate of C1/10 (i.e., 0.33 amps for a battery with a C1 rating of 3.3 AH) until the battery voltage reaches 31.0 volts, then continue charging at the same constant current rate for an additional 4 hours. The charge cycle should be continuous when using this method, i.e., no pauses should be included. However, if there is an interruption (e.g., due to a local power outage), continue from where the profile stopped and run to completion.

Note If the battery voltage exceeds 31 volts at the beginning of charge and then drops below 31 volts within 2 hours, continue charging at constant current until the voltage reaches 31 volts a second time. Then continue charging for an additional 4 hours as specified above.

Note If the battery voltage does not reach 31 volts within 24 hours, the charge should be terminated.

3) After the charge is complete, allow the battery to cool down for at least 8 hours or until the battery temperature is within 10°C/18°F of the ambient temperature, then proceed with the capacity test.

D. Battery Conditioning Procedure

WARNING REMOVE BATTERY FROM AIRCRAFT PRIOR TO DOING A BATTERY CONDITIONING PROCEDURE. THE PROCEDURE SHOULD BE DONE IN A WELL VENTILATED AREA BECAUSE A SIGNIFICANT AMOUNT OF HYDROGEN GAS MAY BE RELEASED FORM THE BATTERY.

CAUTION A CONDITIONING CHARGE IS ONLY NECESSARY IF THE BATTERY FAILS THE CAPACITY TEST. THIS PROCEDURE MAY SHORTEN THE BATTERY'S LIFE IS DONE ON A REPETITIVE BASIS.

CAUTION IF THE BATTERY BECOMES HOT (ABOVE 55°C/130°F) DURING THE CONDITIONING CHARGE, STOP THE CURRENT AND ALLOW THE BATTERY TO COOL TO ROOM TEMPERATURE BEFORE CONTINUING THE CHARGE.

Equipment Required:

- Electronic Load, capable of constant current at C1 ADC and 20-28 VDC with EPV termination at 20 VDC
- Constant Current Power Supply, capable of C1/10 ADC and at least 34 VDC output
- Mating connector MS27467T15B5P

1) Remove battery from aircraft as per Para 4.A.

2) Connect battery to electronic load using mating connector MS27467T15B5P.

3) Discharge battery at C1 rate, noted on the battery label, to end point voltage of 18 volts.

4) Charge at a constant current rate of C1/10 for 16 hours and then discontinue charging. The conditioning charge does not need to be continuous, i.e., pauses may be included. For example, an 8 hour charge on the first day followed by an 8 hour charge on the second day is acceptable.

5) Allow the battery to cool down for at least 8 hours or until the battery temperature is within 10°C/18°F of the ambient temperature, then proceed with the capacity test as per Para 5.A.

E. Check Avionics Battery Diodes

Note Refer to the PSM 1-64-2W Wiring Diagrams manual to familiarize with the circuitry.

Equipment Required:

- Digital multimeter
- Torque driver

Note Diodes that do not pass this check must be replaced.

1) Charge Circuit Diode (CR139)

a) Ensure aircraft power is OFF.

b) Gain access to the forward avionic shelves through the forward baggage compartment.

c) Remove the top cover attached to bulkhead station 37.21.

d) Pull CB273 and CB288 and fit with safety clips.

Note CB273 is located on the avionics circuit breaker panel and is labeled AVIONCIS STRT BATT CHARGE (M7). CB288 is located on the bracket behind the avionics start battery.

e) Disconnect battery connector (P474).

f) Using a digital multimeter in diode test mode, measure CR139 and confirm forward-bias reading is approximately 0.25V and reverse-bias reading is open.

g) Reconnect battery connector (P474).

h) Reset CB273 and CB288 circuit breakers.

i) Reinstall the top cover attached to bulkhead station 37.21.

2) Power Isolation Diodes (CR141 thru CR146)

Refer to Figure 2 for terminal arrangement of diodes.

a) Ensure aircraft power is OFF.

b) Remove center pedestal cover to gain access to diodes behind avionics circuit breaker panel.

c) Disconnect wire 1P128A14 from CR141 terminal A1.

d) Using a digital multimeter in diode test mode, measure CR141 and CR145 and confirm forward-bias reading is approximately 0.25V and reverse-bias reading is open.

e) Reconnect wire 1P128A14 to CR141 termianl A1. Torque terminal screw to 11.5 - 12.5 in-lbs.

CAUTION **DO NOT OVER TIGHTEN SCREWS. THIS CAN DO DAMAGE TO THE DIODE MODULE.**

f) Disconnect wire 2P128A14 from CR142 terminal A1.

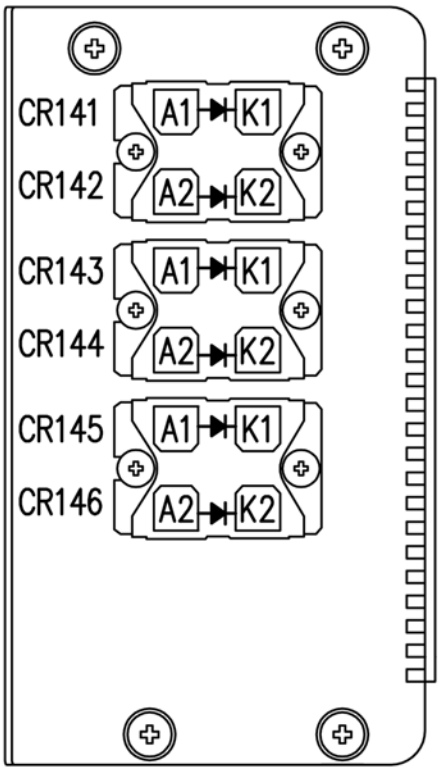
g) Using a digital multimeter in diode test mode, measure CR142 and CR146 and confirm forward-bias reading is approximately 0.25V and reverse-bias reading is open.

h) Reconnect wire 2P128A14 to CR142 terminal A1. Torque terminal screw to 11.5 - 12.5 in-lbs.

CAUTION **DO NOT OVER TIGHTEN SCREWS. THIS CAN DO DAMAGE TO THE DIODE MODULE.**

i) Using a digital multimeter in diode test mode, measure CR143 and CR144 and confirm forward-bias reading is approximately 0.25V and reverse-bias reading is open.

j) Re-install center pedestal cover.



CR141 thru CR146 Diode Terminal Arrangement
Figure 2

7. Troubleshooting

SYMPTOM	POSSIBLE CAUSE	CHECK THAT	REMEDIAL ACTION
1) Battery does not hold charge.	Battery beyond serviceable life.		Replace battery.
2) Battery capacity less than 85% of rating.	Battery beyond serviceable life.		Replace battery.
3) Battery gets hot during constant potential charging.	Battery beyond serviceable life.		Replace battery.
4) Battery not charging in aircraft.	Faulty charge circuit diode.	Check CR139 in accordance with Para 6.E.1).	Replace faulty diode.
	Battery beyond serviceable life.	Do capacity test in accordance with Para 5.A.	Replace battery.
5) Power interruption to displays during start with fully charged avionics start battery.	Battery output circuit breaker (CB288) tripped.		Reset circuit breaker.
	Faulty power isolation diodes.	Check CR141 thru CR146 in accordance with Para 6.E.2).	Replace faulty diodes.
	Faulty battery control relay.	Relay does not close when AVIONICS START BATTERY switch is in AUTO position.	Replace faulty relay.

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External Power – General Data

1. General

External 28 Volt DC power is connected to the aircraft through an external power receptacle located on the left side of the rear fuselage, just aft of the cargo door. On aircraft incorporating S.O.O. 6371, the external ground power receptacle has been relocated to the outer metallic structure at approximately station 55.

EXTERNAL power allows ground testing and engine starting procedures to be done without using the aircraft main batteries. EXTERNAL power is available with external power connected to the aircraft, the MASTER switch selected to the ON position, the BUS TIE switch to NORMAL and the POWER SOURCE switch selected to EXTERNAL (Pre Mod 6/2314) or EXTERNAL switch selected to ON (Post Mod 6/2314) (refer to PSM 1-64-2W for DC Generation Electrical Schematics for details). 28 Volt DC from the ground source energizes the external power relay which connects the external power to the aircraft DC buses; the normally open contacts of the main battery relay isolate the battery from the external power (Pre Mod 6/2314). For Post Mod 6/1293 a diode in the external power control circuit protects the aircraft system from an input of reversed polarity.

When either generator is selected on-line, the generator field relay (K5 or K6) is energized disconnecting the external power relay circuit which, in turn, disconnects the external power source from the buses. (During initial start-up, the generator field circuit must be energized, therefore the starting circuit bypasses the appropriate contacts of relay K5 or K6 to maintain external power input).

2. Servicing

A. Application of External Power (Pre Mod 6/2314)

To apply power to the aircraft buses from a ground power source, proceed as follows:

1) Connect 28 Volt DC power source to external power receptacle.

2) Set MASTER switch to ON.

3) Set POWER SOURCE switch to EXTERNAL.

4) Set BUS TIE switch to NORMAL.

5) Check GND PWR voltmeter for 28 Volt DC reading.

B. Removal of External Power (Pre Mod 6/2314)

1) Set MASTER switch to OFF.

2) Set POWER SOURCE switch to OFF.

3) Disconnect ground power source from external power receptacle.

C. Application of External Power (Post Mod 6/2314)

To apply power to the aircraft buses from a ground power source, proceed as follows:

1) Connect 28 Volt DC power source to external power receptacle.

2) Set EXTERNAL switch to ON.

3) Set BUS TIE switch to NORMAL.

4) Check GND PWR voltmeter for 28 Volt DC reading.

D. Removal of External Power (Post Mod 6/2314)

1) Set EXTERNAL switch to OFF.

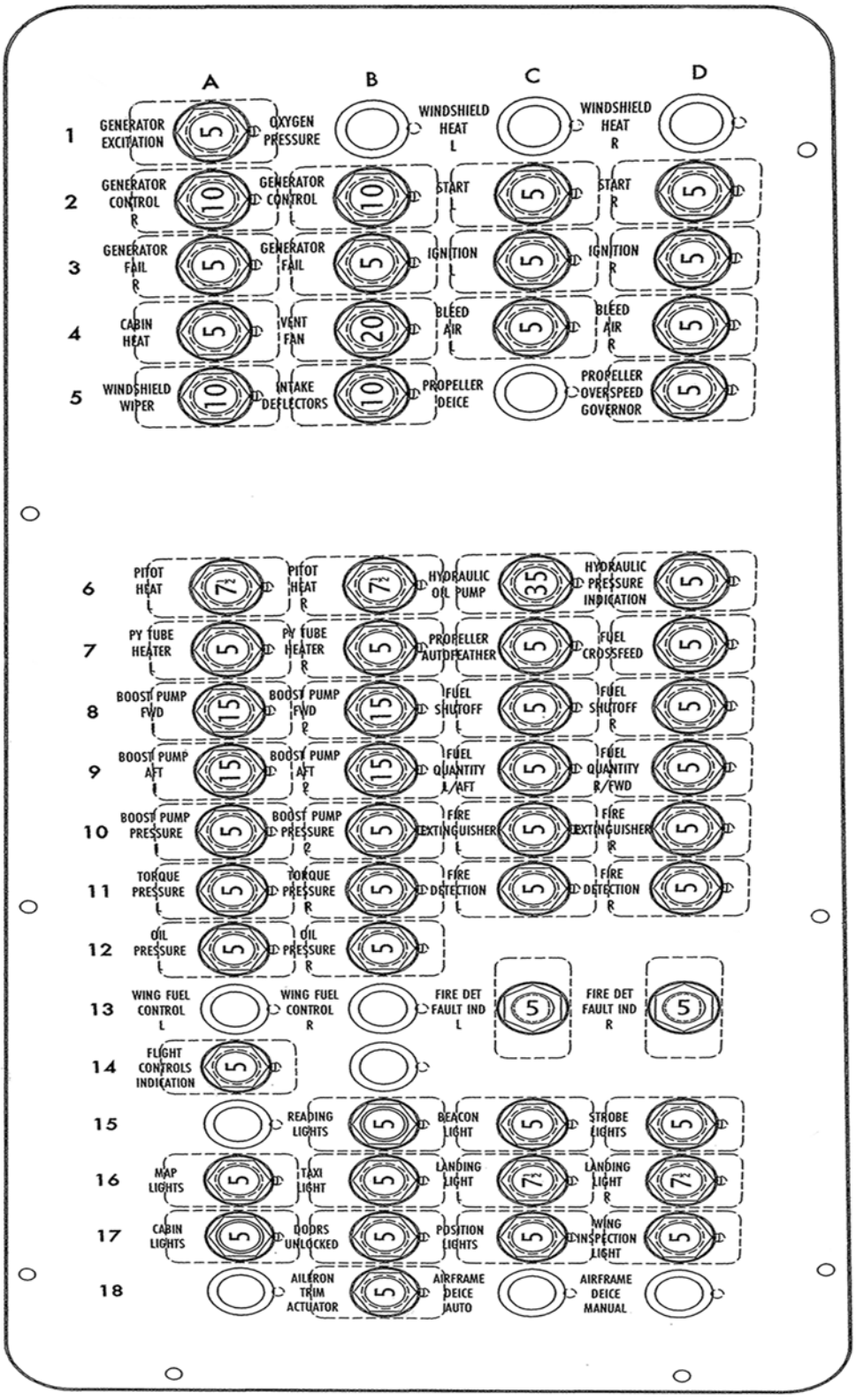
2) Disconnect ground power source from external power receptacle.

DC Electrical Load Distribution – General Data

1. General

DC power distribution is by means of a multiple bus system consisting of left and right 28 Volt DC buses, and main battery, battery/external power. The left generator is connected to the left DC bus and the right generator to the right DC bus by LEFT and RIGHT GENERATOR switches marked OFF–ON–RESET, together with a POWER SOURCE switch and MASTER switch. When Mod 6/2314 is installed, the POWER SOURCE and MASTER switches are deleted, and replaced with BATTERY and EXTERNAL switches which independently select DC voltages from the BATTERY or EXTERNAL sources. All buses can be supplied from one generator or the main battery, through a bus-tie relay operated when a BUS TIE switch is set to NORMAL. The switches are located on the overhead console panels. The main battery bus, connected directly to the main battery positive terminal, is connected through a battery relay to the battery/external power bus, then through a reverse current circuit breaker to the left DC bus. If the BUS TIE switch is set to OPEN, the right DC bus is isolated and is only powered from the right generator. Power is distributed to the left and right DC buses, through bus feed circuit breakers in the power distribution and generator control box and through circuit breakers on the main overhead and avionics distribution panels added for Mod 6/1274, to supply electrical circuits through individual system circuit breakers (refer to Figure 1), mounted on panels in the flight compartment, the main panels on the bulkhead to the left of the pilot, and avionics panels on the center pylon (refer to Figure 1). Circuits from the main battery bus operate cabin entrance and baggage compartment lights. (Refer to 33-10-00, Flight Compartment Lights (Mod 6/2028) – General Data).

Six bus feed, 50 ampere circuit breakers (left DC CB5, CB7, CB9 and right DC CB4, CB6, CB8) are installed in the power distribution and generator control box (24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data, Figure 1) to supply six 50-ampere circuit breakers (CB5A, CB7A, CB9A, CB4A, CB6A and CB8A) on the main power distribution circuit breaker panel (refer to Figure 3). Six bus feed, 30-ampere circuit breakers (left DC CB13, CB15, CB17 and right DC CB12, CB14, CB16) are installed in the power distribution and generator control box to six 30-ampere circuit breakers (CB13A. CB15A, CB17A, CB12A. CB14A and CB16A) on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).



Circuit Breaker Panel (Typical)
Figure 1

Table 1 Main Circuit Breaker Panel CBP-A

Left Bus		Amperage	SOO
GENERATOR EXCITATION	CB101	5	
GENERATOR CONTROL R	CB103	10	
GENERATOR FAIL R	CB105	5	
CABIN HEAT	CB107	5	
WINDSHIELD WIPER	CB109	10	
PITOT HEAT	CB111	7.5	
PY TUBE HEATER L	CB113	5	
BOOST PUMP FWD 1	CB115	15	PRE MOD 6/2299
		10	POST MOD 6/2299
BOOST PUMP AFT 1	CB117	15	PRE MOD 6/2299
		10	POST MOD 6/2299
BOOST PUMP PRESSURE 1	CB119	5	
TORQUE PRESSURE L	CB121	5	
OIL PRESSURE L	CB123	5	
WING FUEL CONTROL L	CB125	7.5	6247
FLIGHT CONTROL INDICATION	CB127	5	
RESERVED	CB129		
MAP LIGHTS	CB131	5	
CABIN LIGHTS	CB133	5	
RESERVED	CB135		
WINDSHIELD HEAT L	CB137	5	6187
START L	CB139	5	
IGNITION L	CB141	5	
BLEED AIR L	CB143	5	
PROPELLER DEICE	CB145	20	6202
HYDRAULIC OIL PUMP	CB147	35	
PROPELLER AUTOFEATHER	CB149	5	
FUEL SHUTOFF L	CB151	5	
FUEL QUANTITY L/AFT	CB153	5	
FIRE EXTINGUISHER L	CB155	5	
FIRE DETECTION L	CB157	5	
FIRE DET FAULT IND L	CB161	5	
BEACON LIGHT	CB165	5	

Table 1 Main Circuit Breaker Panel CBP-A (continued)

Left Bus		Amperage	SOO
LANDING LIGHT L	CB167	7.5	PRE MOD 6/2285
		5	POST MOD 6/2285
POSITION LIGHTS	CB169	5	
AIRFRAME DEICE AUTO	CB171	5	6004

Right Bus		Amperage	SOO
RESERVED	CB100		
GENERATOR CONTROL L	CB102	10	
GENERATOR FAIL L	CB104	5	
VENT FAN	CB106	20	
INTAKE DEFLECTORS	CB108	10	
PITOT HEAT R	CB110	7.5	
PY TUBE HEATER R	CB112	5	
BOOST PUMP FWD 2	CB114	15	PRE MOD 6/2299
		10	POST MOD 6/2299
BOOST PUMP AFT 2	CB116	15	PRE MOD 6/2299
		10	POST MOD 6/2299
BOOST PUMP PRESSURE 2	CB118	5	
TORQUE PRESSURE R	CB120	5	
OIL PRESSURE R	CB122	5	6247
WING FUEL CONTROL R	CB124	7.5	
RESERVED	CB126		
READING LIGHTS	CB128	5	
TAXI LIGHT	CB130	5	
DOORS UNLOCKED	CB132	5	
AILERON TRIM ACTUATOR	CB134	5	
WINDSHIELD HEAT R	CB136	5	6187
START R	CB138	5	
IGNITION R	CB140	5	
BLEED AIR R	CB142	5	
PROPELLER OVERSPEED GOVERNOR	CB144	5	
HYDRAULIC PRESSURE INDICATION	CB146	5	
FUEL CROSSFEED	CB148	5	
FUEL SHUTOFF R	CB150	5	
FUEL QUANTITY R/FWD	CB152	5	

Right Bus		Amperage	SOO
FIRE EXTINGUISHER R	CB154	5	
FIRE DETECTION R	CB156	5	
FIRE DET FAULT IND R	CB160	5	
STROBE LIGHTS	CB164	5	
LANDING LIGHT R	CB166	7.5	PRE MOD 6/2285
		5	POST MOD 6/2285
WING INSPECTION LIGHT	CB168	5	
AIRFRAME DEICE MANUAL	CB170	7.5	6004
CABIN FANS	CB173	15	6378 6379

Table 2 Avionics Circuit Breaker Panel CBP-B

Left Bus		Amperage	SOO
MAU 1A	CB201	20	
MAU 2A	CB203	20	
AURAL ALERT 1	CB205	1	
PILOT PFD	CB207	5	
UPPER MFD	CB209	5	
ADAHRS 1A	CB211	3	
FLT CTRL 1A	CB213	3	
PITCH TRIM	CB215	5	6221
AP SERVOS	CB217	7.5	6221
PILOT PFD CTRL	CB219	3	
PILOT AUDIO	CB221	5	
MMDR 1	CB223	7.5	
XPDR 1	CB225	5	
DME 1	CB227	3	
GPS 1	CB229	3	
CVR	CB231	5	
FDR	CB233	5	
CAUTION LTS NORMAL	CB235	2	
WX RADAR	CB237	5	
MFD CTRL	CB239	3	
DAU 1A	CB241	5	
DAU 2A	CB243	5	
TAWS	CB245	3	

Table 2 Avionics Circuit Breaker Panel CBP-B (continued)

Left Bus		Amperage	SOO
TCAS	CB247	5	
RAD ALT 1	CB249	3	
ELT NAV	CB251	1	
CABIN PA	CB253	5	
ESIS BATTERY	CB255	15	
ESIS INST	CB257	3	
ESIS MAG	CB259	1	
AVIONICS FAN	CB261	1	
RESERVED	CB263		
RESERVED	CB265		
GEAR IND	CB267	5	6001
RESERVED	CB269		
RESERVED	CB271		
AVIONICS STRT BATT CHARGE	CB273	30	POST MOD 6/2315
AVIONICS STRT BATT CTRL	CB275	2	POST MOD 6/2315
PANEL LIGHTS L	CB277	5	
FLOOD LIGHTS	CB279	5	

Right Bus		Amperage	SOO
MAU 1B	CB200	20	
MAU 2B	CB202	20	
AURAL ALERT 2	CB204	1	
COPILOT PFD	CB206	5	
LOWER MFD	CB208	5	
ADAHRS 1B	CB210	3	
FLT CTRL 1B	CB212	3	
CAUTION LTS BYPASS	CB214	2	
RESERVED	CB216		
COPILOT PFD CTRL	CB218	3	
COPILOT AUDIO	CB220	5	
MMDR 2	CB222	7.5	
XPDR 2	CB224	5	
DME 2	CB226	3	
GPS 2	CB228	3	
SATELLITE WX	CB230	5	6211

Right Bus		Amperage	SOO
STORM SCOPE	CB232	3	6215
SAT COM	CB234	3	6298
HF CTRL	CB236	2	6215
ECTM/FOQA	CB238	3	6288
DAU 1B	CB240	5	
DAU 2B	CB242	5	
HEADSETS	CB244	2	
RESERVED	CB246		
RAD ALT 2	CB248	3	6218
RESERVED	CB250		
28V INPUT	CB252	5	
		20	6282
14V INPUT	CB254	7.5	
RESERVED	CB256		
VHF-FM	CB258	3	6296

Table 3 CBP-C Main Circuit Distribution Panel

Left Bus	Amperage	SOO
CB5A	50	
CB7A	50	
CB9A	50	

Right Bus	Amperage	SOO
CB4A	50	
CB6A	50	
CB8A	50	

Table 4 FWD Converter Circuit Breaker Panel

Right Bus		Amperage	SOO
STA 128 14VDC	CB282	7.5	6282
STA 176L 14VDC	CB281	7.5	6282
STA 176R 14VDC	CB280	7.5	6282

Table 5 AFT Converter Circuit Breaker Panel

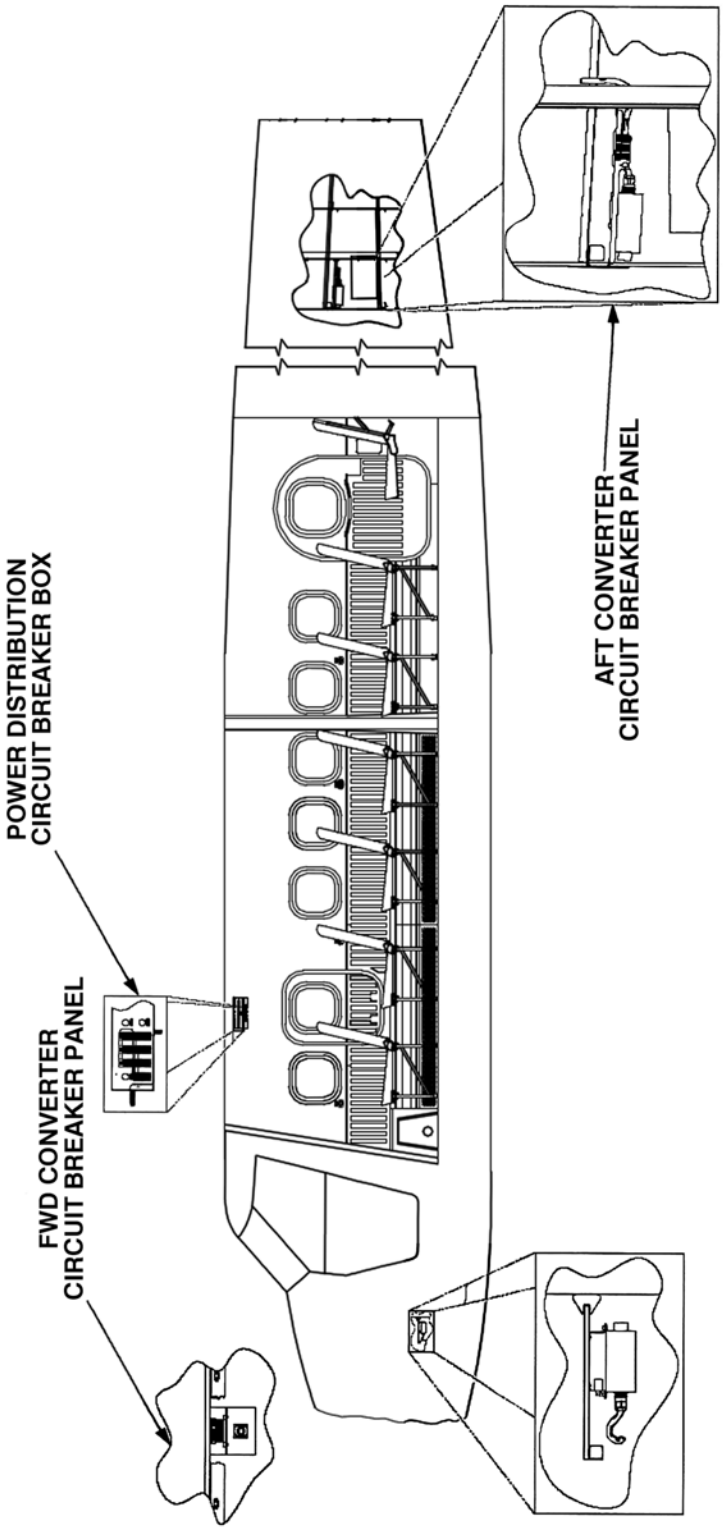
Right Bus		Amperage	SOO
STA 223	CB286	7.5	6281
STA 258	CB285	7.5	6281

Table 5 AFT Converter Circuit Breaker Panel (continued)

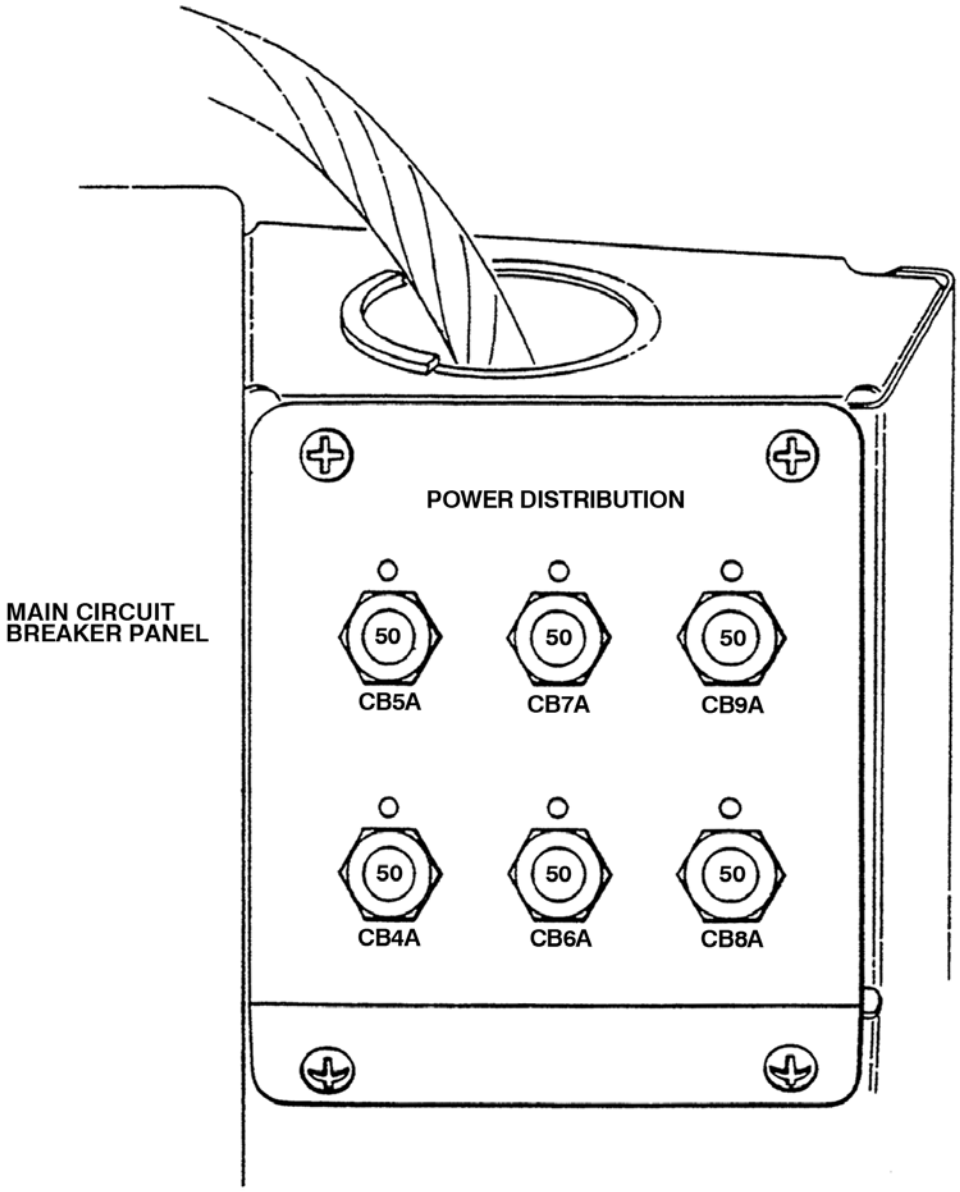
Right Bus		Amperage	SOO
STA 332	CB284	7.5	6281
		7.5	6272

Table 6 Power Distribution Circuit Breaker Box

Right Bus		Amperage	SOO
CBP-E	CB18	5	6272
CBP-E	CB18	15	6281
CBP-E	CB18	5	6283



Cabin Power Converter and Power Distribution Box Locations (S.O.O. 6272, 6281, 6282, 6283)
Figure 2



Main Circuit Breaker Panel
Figure 3

Electrical Load Distribution (Mod 6/2028) – Maintenance Practices

1. Adjustment/Test

A. Operational Test DC Bus Feed Circuits

Refer to PSM 1-64-2W for electrical schematic.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Set BUS TIE switch to NORMAL.

3) If required, remove upholstery beneath circuit breaker brackets on power distribution and generator control box to gain access to bus feed circuit breakers.

4) Check the left bus feed circuits (CB5A, CB7A and CB9A) by observing that the Hydraulic Power Fail CAS message does not come ON and remains OFF for the duration of the following tests and the cabin lights; select CABIN GENERAL switch BRIGHT, remain ON (make sure the HYDRAULIC OIL PUMP and HYDRAULIC PRESSURE INDICATION circuit breakers and CABIN LIGHTS circuit breaker are engaged on the main circuit breaker panel:

a) Pull circuit breakers CB5 and CB7 together, then reset CB5.

b) Pull circuit breaker CB9, then reset CB7.

c) Pull circuit breaker CB5, then reset CB9 and CB5.

5) Check the right bus feed circuits (CB4A, CB6A and CB8A) by observing that the Fuel Cross Feed Fail CAS message does not come ON and remains OFF for the duration of the following tests and the CABIN reading lights remain ON (make sure the FUEL CROSSFEED and READING LIGHTS circuit breaker is engaged on the main circuit breaker panel:

a) Pull circuit breakers CB4 and CB6 together, then reset CB4.

b) Pull circuit breaker CB5, then reset CB6.

c) Pull circuit breaker CB4, then reset CB8 and CB4.

6) To check the left bus feed circuits (CB13, CB15 and CB17), verify that the pilot's PFD and upper MFD remain ON for the following tests which is fed from the left DC bus, and proceed as follows:

a) Pull circuit breakers CB13 and CB15 together, then reset CB13.

b) Pull circuit breaker CB17, then reset CB15.

c) Pull circuit breaker CB13, then reset CB17 and CB13.

7) To check the right bus feed circuits (CB12, CB14 and CB16), verify that the co-pilot's PFD and lower MFD remain ON for the following tests which are fed from the right DC bus, and proceed as follows:

a) Pull circuit breakers CB12 and CB14 together, then reset CB12.

b) Pull circuit breaker CB16, then reset CB14.

c) Pull circuit breaker CB12, then reset CB16 and CB12.

8) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

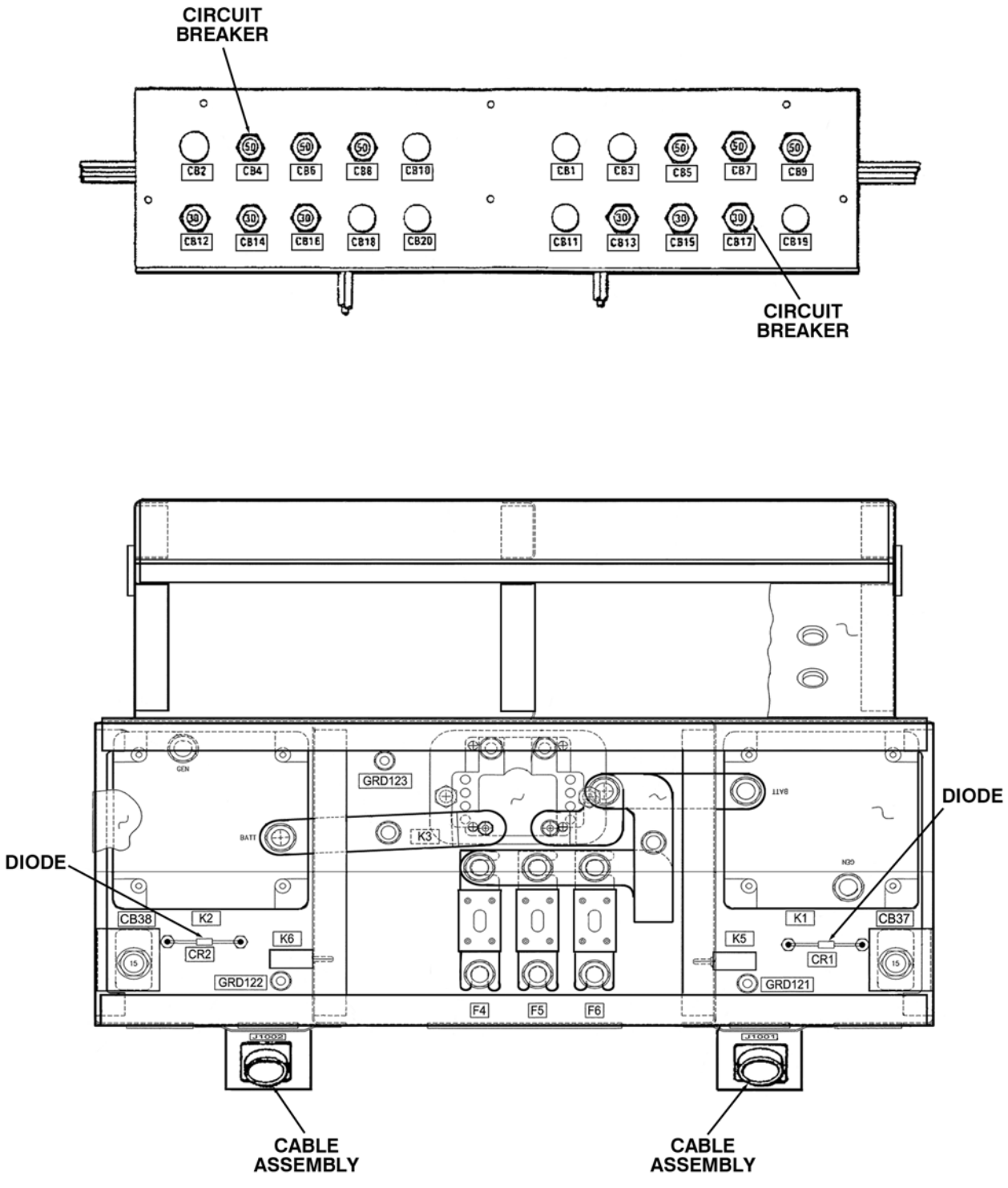
9) Reinstall upholstery removed in Para 1.A.3).

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Power Distribution and Generator Control Box (Mod 6/2028) – General Data

1. Description

The power and distribution and generator control box (refer to Figure 1), located in the cabin roof on the right side between stations 162.35 and 180.75, consists of a rectangular aluminum alloy box and cover. The cover is attached to the box by eight quick-release fasteners. Installed in the box are the left and right generator reverse current relays (K1 and K2), the bus tie relay (K3), two generator fail relays (K5 and K6), four blocking diodes (CR1, CR2, CR3 and CR4), a voltage spike suppression diode and the associated electrical wiring. The aircraft electrical power system wiring is routed into the box by two electrical connectors (J1001 and J1002) and through two holes fitted with grommets. A circuit breaker panel mounted on the outside of the box contains six 50-ampere and six 30-ampere bus feed circuit breakers.



Power Distribution and Generator Control Box (Mod 6/2028)
Figure 1

2. Inspection/Check

A. General Visual Inspection and Terminal Torque

Remove the control box cover assembly and inspect the inside components for:

1) General cleanliness. Contamination by metal shavings, dust or fluids. Remove accumulated contamination.

2) Mechanical security.

3) Bus bars and terminal lugs for signs of overheat, burning or arcing.

4) RCR and adjacent components for signs of overheat.

5) Deterioration of rubber grommets and extruded rubber pad.

6) Wiring for:

- General condition
- Routing and security
- Sagging
- Chafing
- Signs of overheat, burning or arcing, discoloration of insulation, deterioration of insulation
- Lacing tape or cable ties missing or incorrectly installed
- Deterioration of previous repairs (e.g. splices)
- Inappropriate repairs

7) Wire terminations and bus bars for:

- Security of terminations, crimped connections and terminal lugs.
- Torque: with aircraft power removed, verify that the nuts on the large RCR and Bus Tie Relay contact posts are torqued within specification (refer to 24-00-00, Electrical Power – General Data, Table 1).

8) Function test DC Generation System (refer to 24-30-00, DC Generation – General Data, Para 5.B.).

B. Reverse Current Relay Contacts – Visual Inspection

Inspect reverse current relay contactors as follows:

1) Disconnect all electrical cables from relay terminals. Refer to Figure 1.

Note Refer to Hartman Overhaul Instructions contained in PSM 1-6-6A for disassembly, repair or replacement procedures.

2) Remove and retain four screws securing each relay and remove both relays from control box.

3) Disassemble to extent required and visually inspect the four contact faces in each relay. If contact faces are free from pitting and only lightly discoloured, relay may be reassembled.

Note Any evidence of pitting of the contacts is unacceptable. Slight discoloration is permitted.

4) If one or more contact faces in either relay are burned or pitted, replace the RCR, or send the RCR to a qualified repair facility for repair and recertification.

5) Reinstall both relays in control box using screws retained in Para 2.B.2) and reconnect electrical cables (refer to PSM 1-64-2W).

6) Function test DC Generation System (refer to 24-30-00, DC Generation – General Data, Para 5.B.).

3. Approved Repairs

WARNING - ENSURE ADHESIVE AND SOLVENTS ARE KEPT CLEAR OF FIRE AND OTHER SOURCES OF IGNITION.
- ENSURE SUFFICIENT VENTILATION IS PROVIDED WHEN USING SOLVENTS AND ADHESIVE IN CONFINED AREAS.
- ALWAYS WEAR PROTECTIVE CLOTHING AND GLOVES TO AVOID SKIN CONTACT WITH SOLVENTS AND ADHESIVE.

A. Replace Extruded Rubber Pad

Materials and Equipment required:

- Bostik 1142 (B. B. Chemical Co.)
- Toluol (or Methyl Ethyl Ketone) cleaning solution
- Napthol spirits
- 180 grit paper
- Suitable scraper
- Small brush
- Clean lint free cloths

1) Using a suitable scraper and Toluol cleaning solution, remove all traces of old rubber pad and adhesive from box structure.

2) Lightly scuff surface of new rubber pad with 180 grit paper and clean scuffed area with Napthol spirits and wipe dry with a clean cloth before the solvent evaporates.

3) Clean bond area on box structure with Napthol spirits and wipe dry before solvent evaporates.

Note After cleaning bond areas, the areas must not be touched with bare hands. If handling is necessary, wear clean cotton gloves.

4) Spread an even coat of Bostik 1142 adhesive to surface of bond areas.

Note Two coats of adhesive may be required for the rubber pad.

5) Allow the adhesive to dry (i.e. when lightly touched, adhesive will not come away on finger).

6) Mate the rubber pad to the box structure and press down with fingers to ensure overall contact.

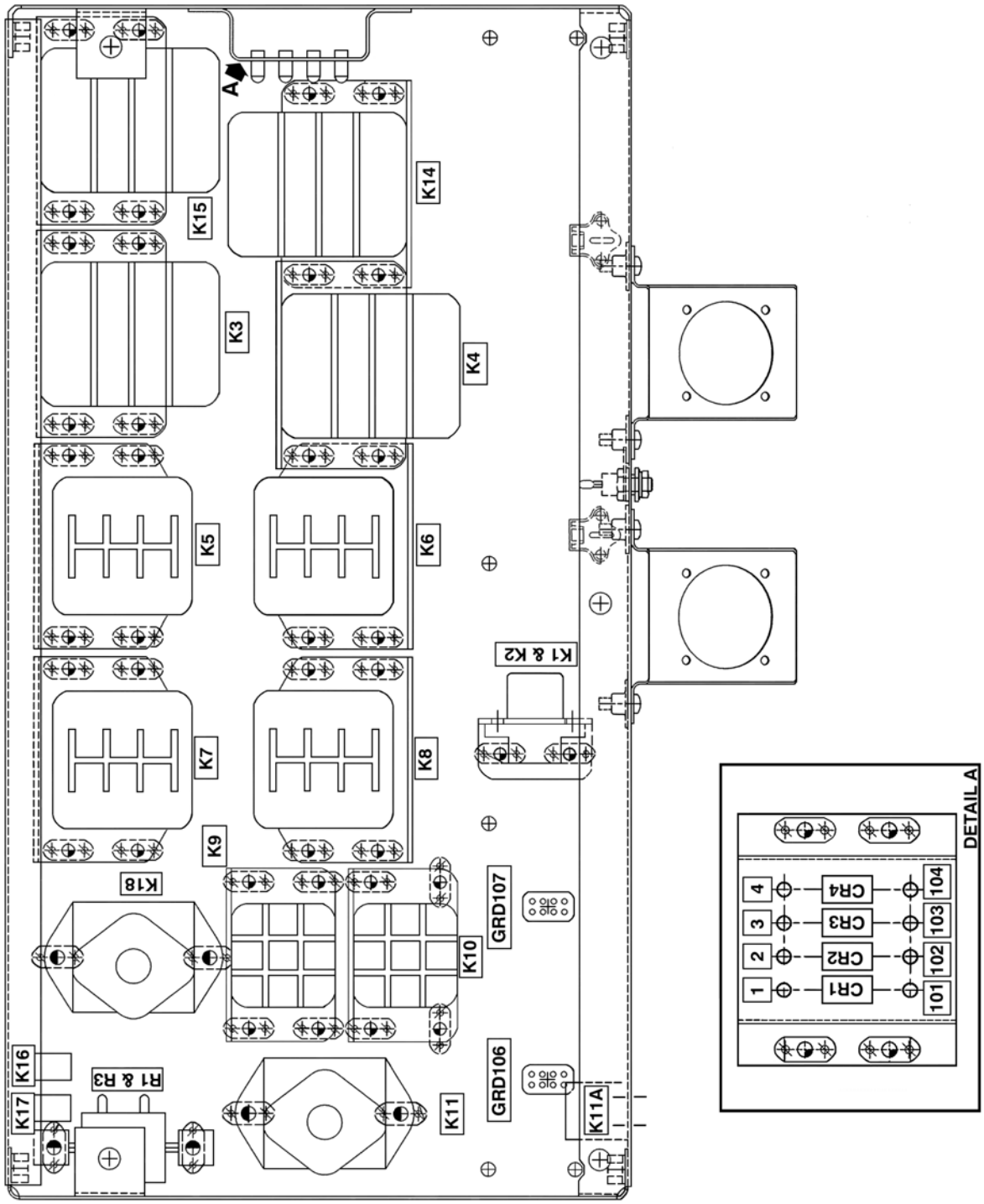
7) Allow joint to dry for 2 – 4 hours.

8) Remove excess adhesive with Toluol.

Main Distribution Box – General Data

1. Description

The Viking Air Limited Main Distribution Box, P/N C6NF1400–1 (refer to Figure 1) forms part of the aircraft electrical power system and consists of a rectangular aluminum alloy box closed by a cover secured by quick release fasteners. Mounted within the box are sixteen relays (K1 to K11 and K14 to K18), a bank of resistors and several diodes; and when Mod 6/1605 is incorporated, another relay (K11A) is installed, making a total of seventeen relays housed in the box. Fed through J100 to J101 large gauge wires still go through original opening to appropriate relays.



Main Distribution Box
Figure 1

CHAPTER 25

EQUIPMENT /FURNISHING

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CHAPTER 25 – EQUIPMENT/FURNISHING

LIST OF EFFECTIVE PAGES

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Equipment/Furnishing – General Data

1. General

The equipment/furnishings consist of flight compartment seats, cabin compartment seats, cargo tie-down fittings, and emergency equipment. The basic aircraft is fully equipped to accommodate up to nineteen passengers within a “Commuter” styled interior.

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Flight Compartment – General Data

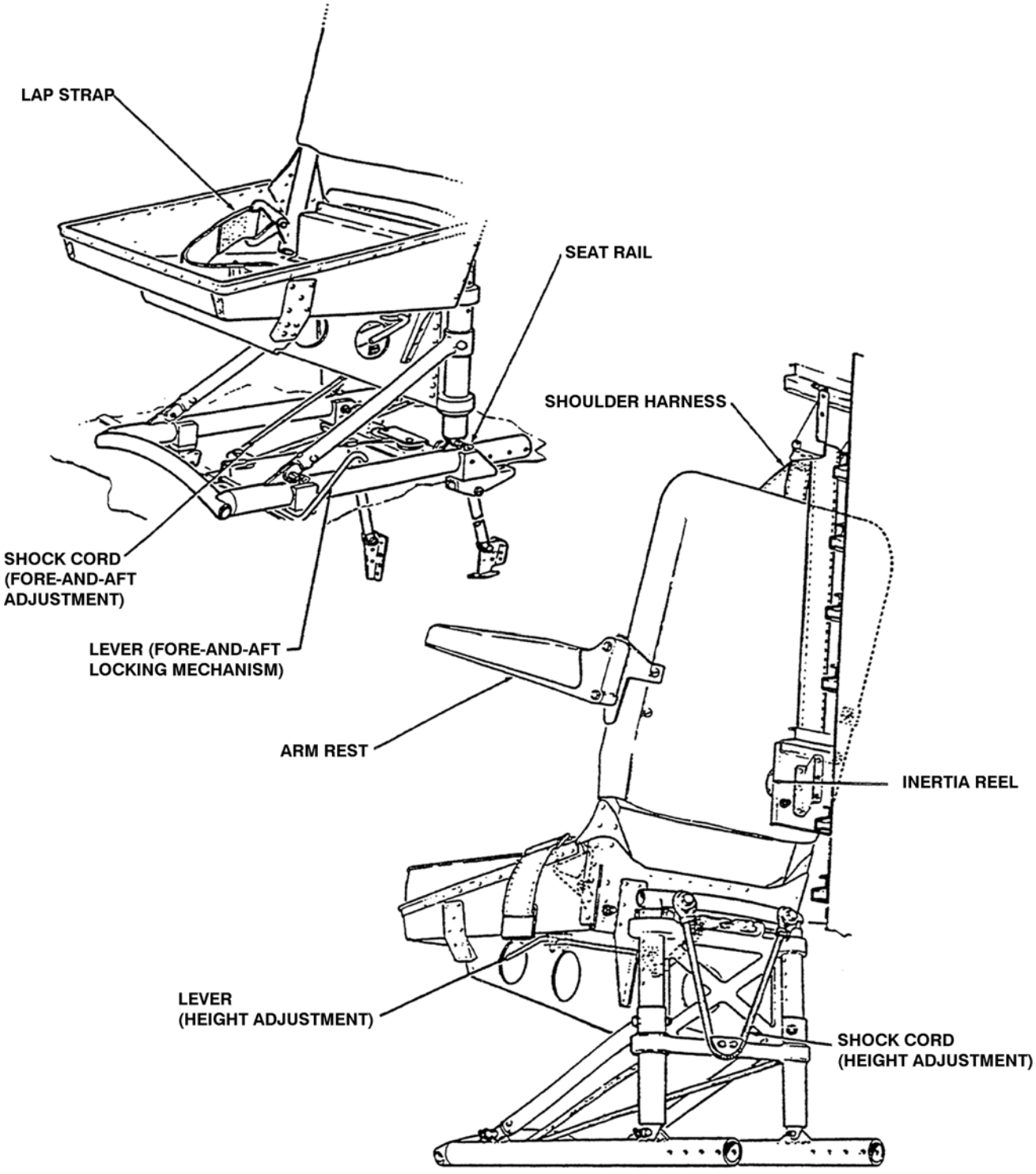
1. General

   The flight compartment equipment furnishings consist of the pilot's and co-pilot's seats, which are both of the same type.

2. Flight Compartment Seats

   Each seat has independent adjustment for height, and fore-and-aft movement. Elastic shock cords are used to take part of the weight of the seat and occupant and make adjustment easier. Both seats have lap-type safety belts and swing back arm rests fitted. Shoulder straps and an inertia reel (Mod 6/1601) are provided for pilot's and co-pilot's seats. Refer to Figure 1 for details.

   As a customer standard order option, S.O.O. 6268 – Sheepskin Covers, may be installed for pilot and co-pilot seats. Refer to 12-20-05, Cleaning – General Data for cleaning instructions.



Pilot's and Co-pilot's Seats
Figure 1

3. Removal/Installation

A. Remove Pilot's and Co-pilot's Seats

1) Move seat to its fully-forward position and release shock cord from block on floor.

2) Withdraw four seat attachment pins and lift out seat.

B. Install Pilot's and Co-pilot's Seats

1) Position seat and insert four seat attachment pins.

2) Move seat to fully forward position and engage shock cord with block on floor.

3) Operate seat height adjustment lever and fore-and-aft adjustment lever and check mechanism for release and correct locking.

4) If portable ELT mounting provisions are installed on co-pilot's seat, ensure that safety wire from No. 40 hole in case to ring on pull pin is intact. If safety wire is broken or improperly installed, install safety wire using 0.020–inch diameter copper locking wire. Using single method of safety wiring to ensure that it can easily be broken in an emergency.

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14 Volt Power Converter System – General Data

1. General

There are two 14 volt power outlets installed in the cockpit, on the pilot and co-pilot's side of center pedestal, to provide power to peripherals; cell phone, laptop, etc.

2. Description

Two 14 volt convenience outlets are provided in the flight compartment. They are located on either side of the instrument panel center pedestal, just below the base of the instrument panel. These two outlets have maximum combined load of 7.5 amps. The 28 Volt DC to 14 Volt DC power converter is located in the forward avionics compartment, and is mounted to the underside of the middle avionics shelf.

28 Volt DC power is provided from the right bus through a 5 amp PWR CONV 28V INPUT (M6) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data). 14 Volt DC power is provided to the outlets through a 7.5 amp PWR CONV 14V OUTPUT (N6) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

On aircraft with Forward Cabin 14V convenience outlets (Mod S.O.O. 6282) incorporated (refer to 25-22-00, 14 Volt Power System (Mod S.O.O. 6272, 6281 and 6282) – General data), 28 Volt DC power is provided through the right bus through a 20 amp PWR CONV 28V INPUT (M6) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data). 14 Volt DC power is provided to the outlets through a 7.5 amp PWR CONV 14V OUTPUT (N6) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

3. Adjustment/Test

A. Function Test of 14 Volt Power Converter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +13.8V (± 0.5V).

3) Plug an accessory into each outlet, as required, and ensure the accessory is being powered by the outlet.

4) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Adjustment Test of 14 Volt Power Converter (Post Mod 6/2406)

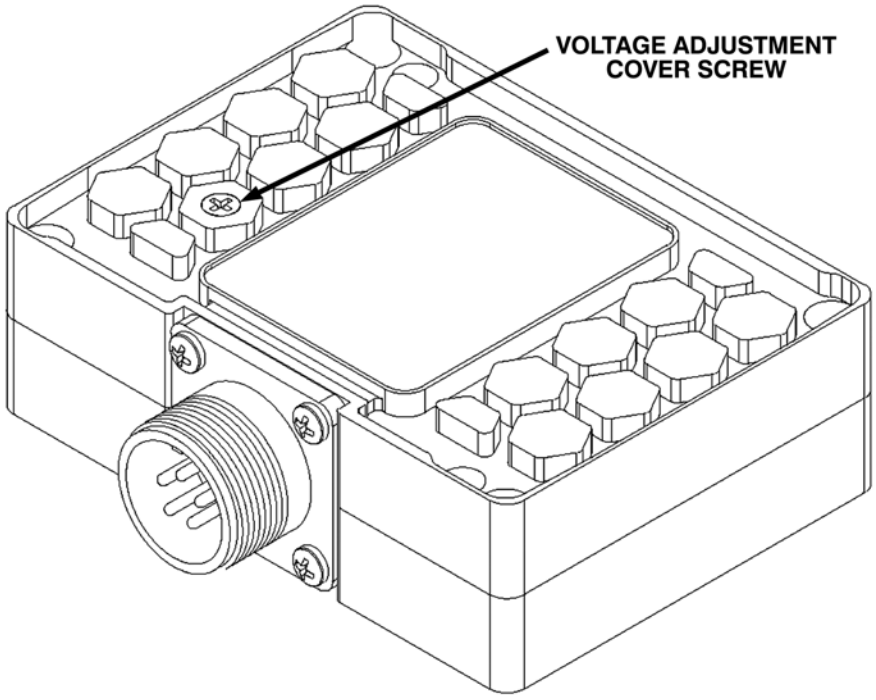
1) Disconnect any electrical load from outlet plug.

2) Remove voltage adjustment cover screw. (Refer to Figure 1).

3) Using a small flat screwdriver, adjust voltage trimmer clockwise to increase voltage or counterclockwise to decrease voltage.

4) Function test in accordance with Para 3.A.

5) Replace voltage adjustment cover screw after voltage adjustment is complete.



14V Power Converter (Post Mod 6/2406)
Figure 1

4. Removal/Installation

A. Removal of 14 Volt Power Converter

1) Ensure the aircraft electrical system is OFF.

2) Open the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel and fit with safety clips.

3) Gain access to the forward avionics shelves through the forward baggage compartment.

4) Remove the middle and bottom covers attached to bulkhead at station 44.00 as required.

5) Disconnect electrical connector.

6) While supporting the 14 volt power converter, remove the four mounting screws.

7) Remove unit from shelf.

Note Stow connectors as required.

B. Installation of 14 Volt Power Converter

1) Ensure the aircraft electrical system is OFF.

2) Ensure the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel is opened and fitted with safety clips.

3) Set unit in position and install four mounting screws.

4) Reconnect electrical connector.

5) Install the middle and bottom covers attached to bulkhead at station 44.00. Ensure the covers are mounted securely.

6) Remove the safety clips and close the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel.

7) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +13.8V (± 0.5V).

8) Do a function test, refer to Para 3.A.

C. 14 Volt Power Outlet

1) Removal of 14 Volt Power Outlet

a) Ensure the aircraft electrical system is OFF.

b) Open the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Remove side panel(s) of center pedestal as required to access the rear of the 14 volt outlet.

d) Gently remove electrical connectors from the rear terminals of the outlet. Note which connector came off the center terminal and which came off the outer terminal.

e) Unthread the two metallic locknuts from the back of the outlet.

f) Remove the outlet by pulling it out the front of the mounting hole.

Note Stow connectors as required.

2) Installation of 14 Volt Power Outlet

a) Ensure the aircraft electrical system is OFF.

b) Ensure the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel is opened and fitted with safety clips.

c) Install the outlet in the mounting hole from the front of the panel.

d) Thread the two metallic locknuts onto the back of the outlet and tighten. Ensure outlet is secured in the panel.

e) Install electrical disconnects onto the appropriate leads on the rear of outlet.

f) Install side panels as required.

g) Remove the safety clips and close the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel.

h) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +13.8V (± 0.5V).

i) Do a function test, refer to Para 3.A.

5. Troubleshooting

A. 14 Volt Power Outlet

1) Ensure PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers are closed.

2) Visually inspect the outlet connectors for damaged wiring or contacts. Ensure electrical connectors on the outlets are fully seated and secure.

3) Visually inspect the converter connector for damaged wiring or contacts. Ensure electrical connector on the converter is fully seated and secure.

4) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

5) Measure 14V output voltage at PWR CONV.

6) If repair or service is required, send to appropriately rated avionics repair facility.

Cabin Compartment – Commuter Version (Mod 6/2050) – General Data

1. General

The cabin compartment equipment/furnishings consist of passenger seats, 14V or 28V outlets, cabin upholstery and draftproof bulkhead.

2. Seating Arrangement

Refer to 25-20-01, Cabin Compartment – Seating Arrangement – General Data.

The basic interior layout of the cabin compartment provides for 19 seats, fitted as standard configuration. The 6th row consists of only one single seat, providing a larger entry area at the airstair door.

Two configurations of commuter seating arrangements are available:

A. Commuter – Nineteen Seats (Basic)

The nineteen seat configuration comprises of five double seats and one single seat on the right side of the cabin, five single seats on the left side and three single tip-up type seats mounted across the rear bulkhead, all of which face forward.

B. Commuter – Fourteen Seats (S.O.O. 6286)

The customer option fourteen passenger seat configuration comprises of six single seats on the right side of the cabin, five single seats on the left side and three single tip-up type seats mounted across the rear bulkhead, all of which face forward.

3. 28 Volt Power System (Mod S.O.O. 6283)

Refer to 25-21-00, 28 Volt Power System (Mod S.O.O. 6283) – General Data.

As a customer option, a 28 volt power outlet may be installed on the forward right hand side of STA 332 bulkhead in the cabin.

4. 14 Volt Power System (Mod S.O.O. 6272, 6281 and 6282)

Refer to 25-22-00, 14 Volt Power Converter System (Mod S.O.O. 6272, 6281 and 6282) – General Data.

Three 14 volt power outlet configurations are available to provide power to peripherals: cell phone, laptops, etc.

A. STA 332 14V Convenience Outlet (Mod S.O.O. 6272)

As a customers option, a single 14 volt power outlet may be installed on the forward right hand side of STA 332 bulkhead in the cabin.

B. Aft Cabin 14V Convenience Outlet (Mod S.O.O. 6281)

As a customer option, three 14 volt power outlets may be installed between STA 220 and 332 bulkhead in the cabin.

C. Forward Cabin 14V Convenience Outlet (Mod S.O.O. 6282)

As a customer option, three 14 volt power outlets may be installed between STA 111 and 186 in the cabin.

5. Cabin Upholstery (Mod 6/2013)

Refer to 25-20-02, Cabin Compartment – Cabin Upholstery (Mod 6/2013) – General Data for Cabin Upholstery specifications.

6. Draftproof Bulkhead

The draftproof bulkhead has a foam sandwich panel with a thermoplastic fascia bonded to both sides and an aluminum channel edging the aisle side. This bulkhead houses the entrance light switch.

7. Cleaning

For cleaning of the cabin floor, interior and upholstery, refer to the procedures in 12-20-05, Cleaning – General Data, Para 4.

Cabin Compartment – Seating Arrangement – General data

1. General

The commuter style seats comprise of a metal frame covered with padding and upholstery material, fitted with lap type safety belts. Increased seat padding and new seat covering material is introduced, the seat covering being easily removed for cleaning (Mod 6/2054 and Mod 6/2281). Single and double seats are provided, and are installed to a side track rail on the cabin wall and to a seat rail and track assembly on the cabin floor. Seats may be removed at the operator's discretion. However, any seating plan or carriage of cargo that deviates from Figure 1 through Figure 4 must be approved by Viking Air Limited, or the operators local airworthiness authority.

2. Configuration

Two configurations of commuter seating arrangements are available:

A. Commuter – Nineteen Seats (Basic)

The nineteen seat configuration comprises five double seats and one single seat on the right side of the cabin, five single seats on the left side and three single tip-up type seats mounted across the rear bulkhead, all of which face forward. Refer to Figure 1 and Figure 3 for details.

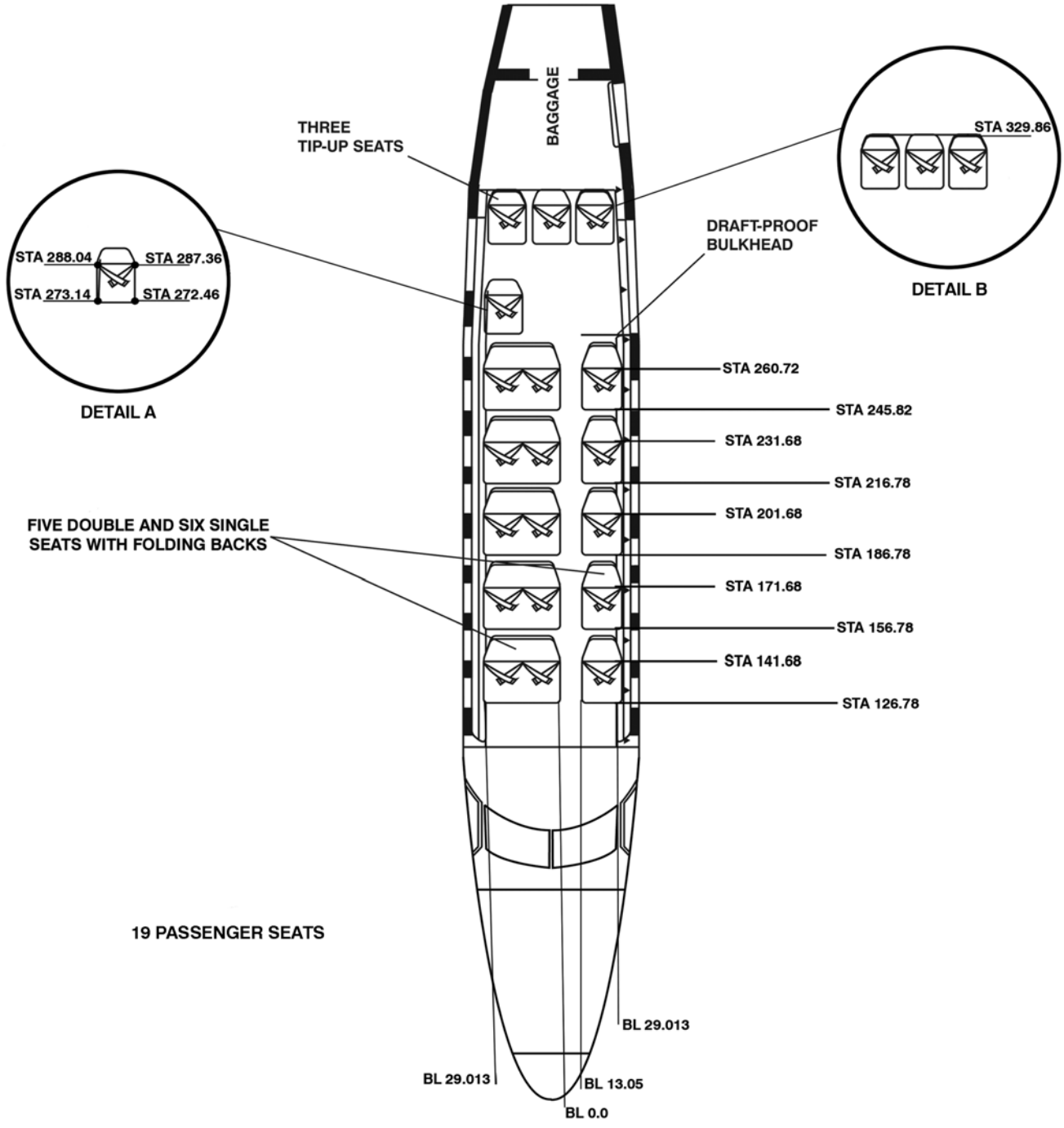
B. Commuter – Fourteen Seats (S.O.O. 6286)

The customer option fourteen passenger seat configuration comprises of six single seats on the right side of the cabin, five single seats on the left side and three single tip-up type seats mounted across the rear bulkhead, all of which face forward. Refer to Figure 2 and Figure 4 for details.

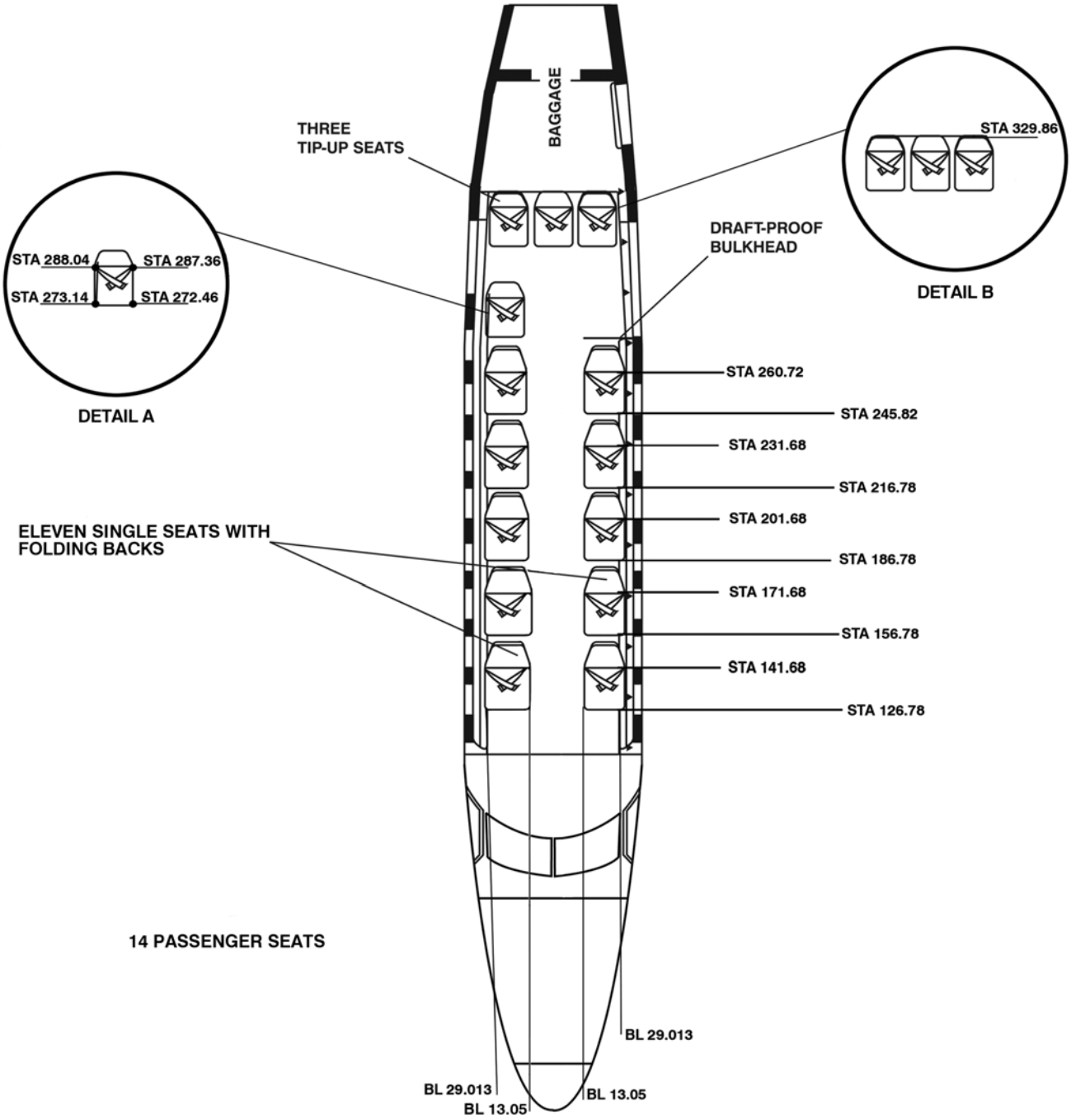
C. Commuter – Nineteen Seats (S.O.O. 6407)

The customer option nineteen passenger seat configuration comprises of six double seats on the right side of the cabin, five single seats on the left side and two single tip-up type seats on STA 332 bulkhead, all of which face forward. Refer to Figure 5.

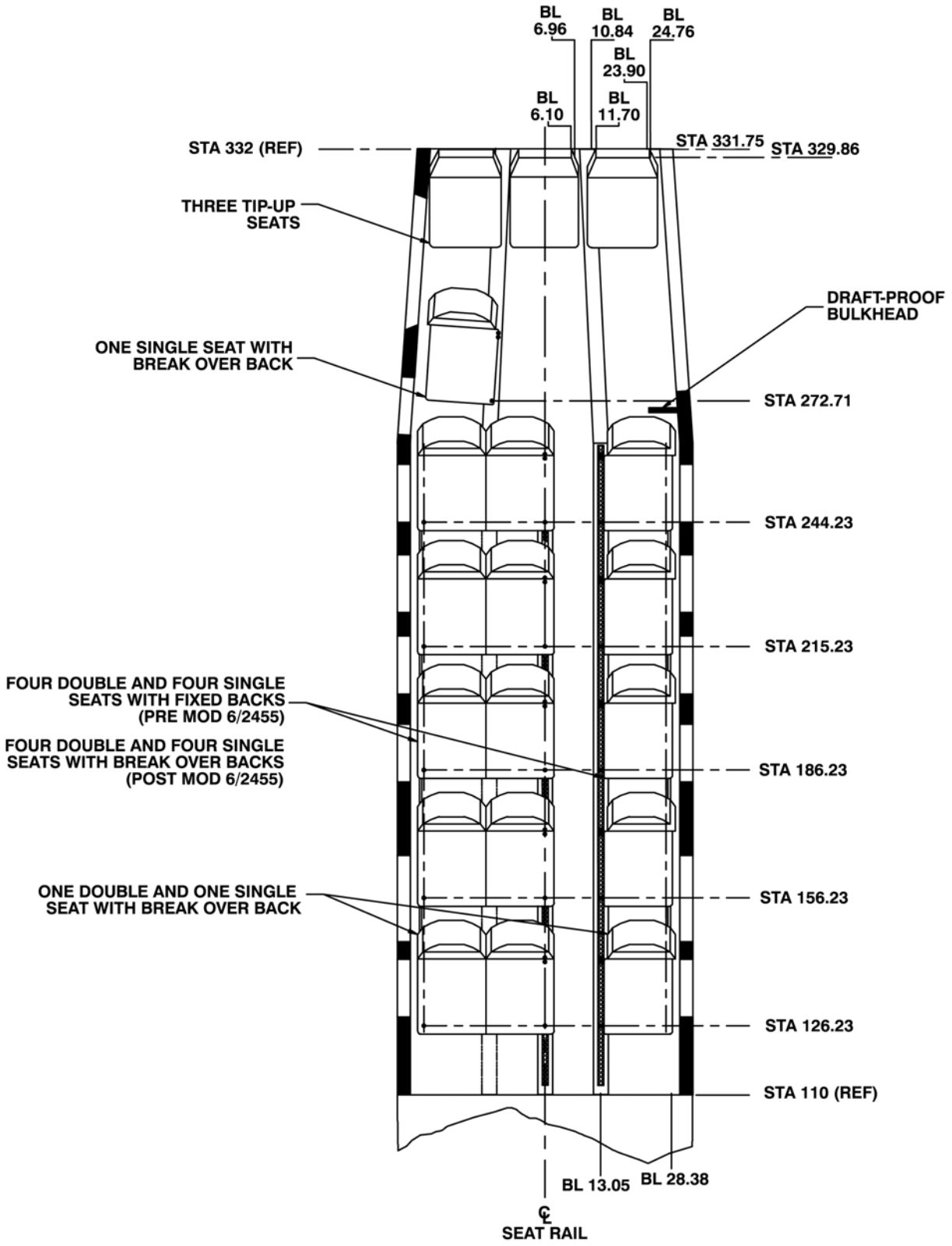
Note S.O.O. 6389 (STA 332 Bulkhead Modification and Cargo Net Installation) and S.O.O. 6377 (Structural Provisions, 6th Row Double Seat RHS) are prerequisites of this option.



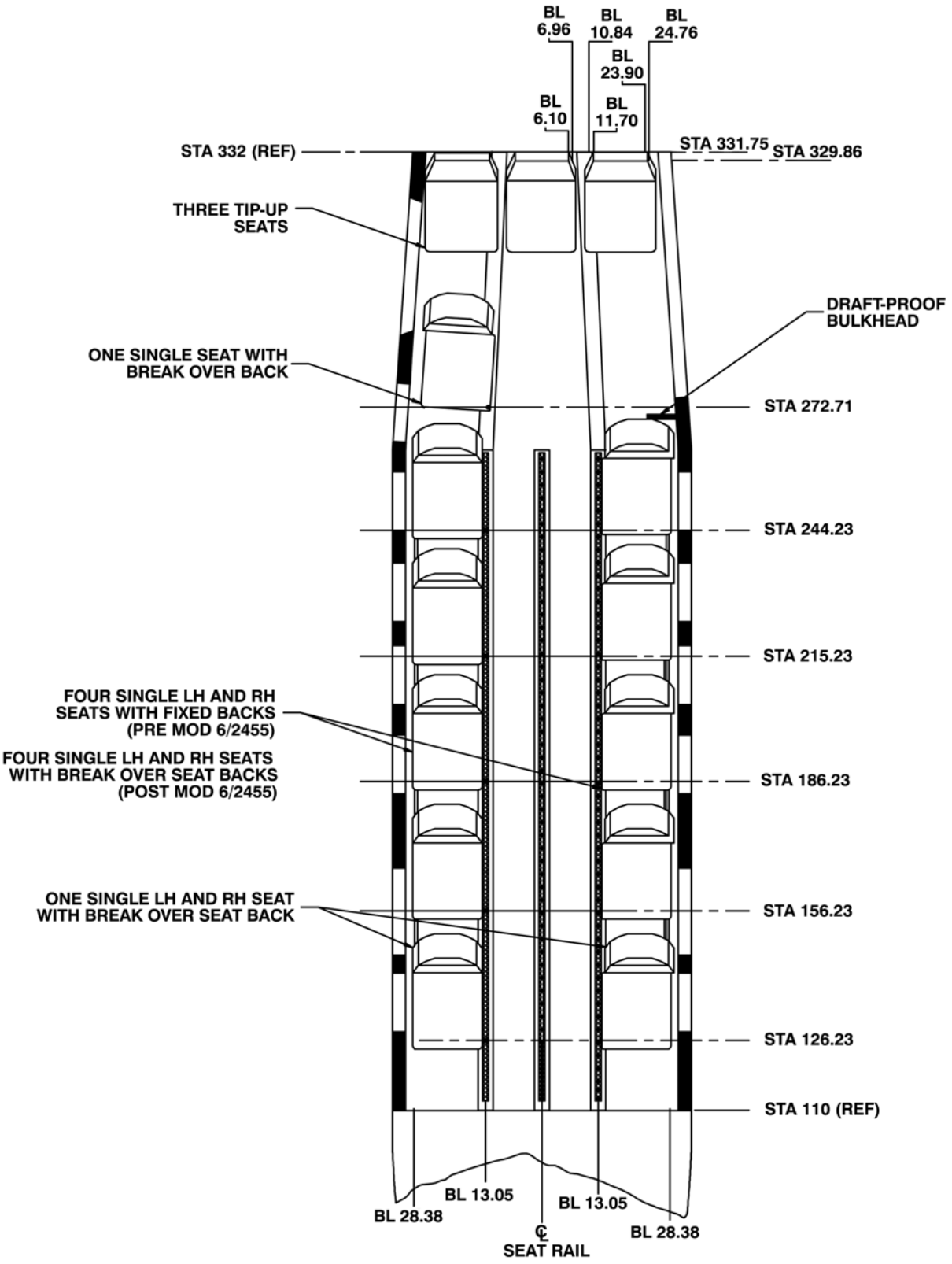
Seat Arrangement – Commuter – 19 Passenger (Pre Mod 6/2281)
Figure 1



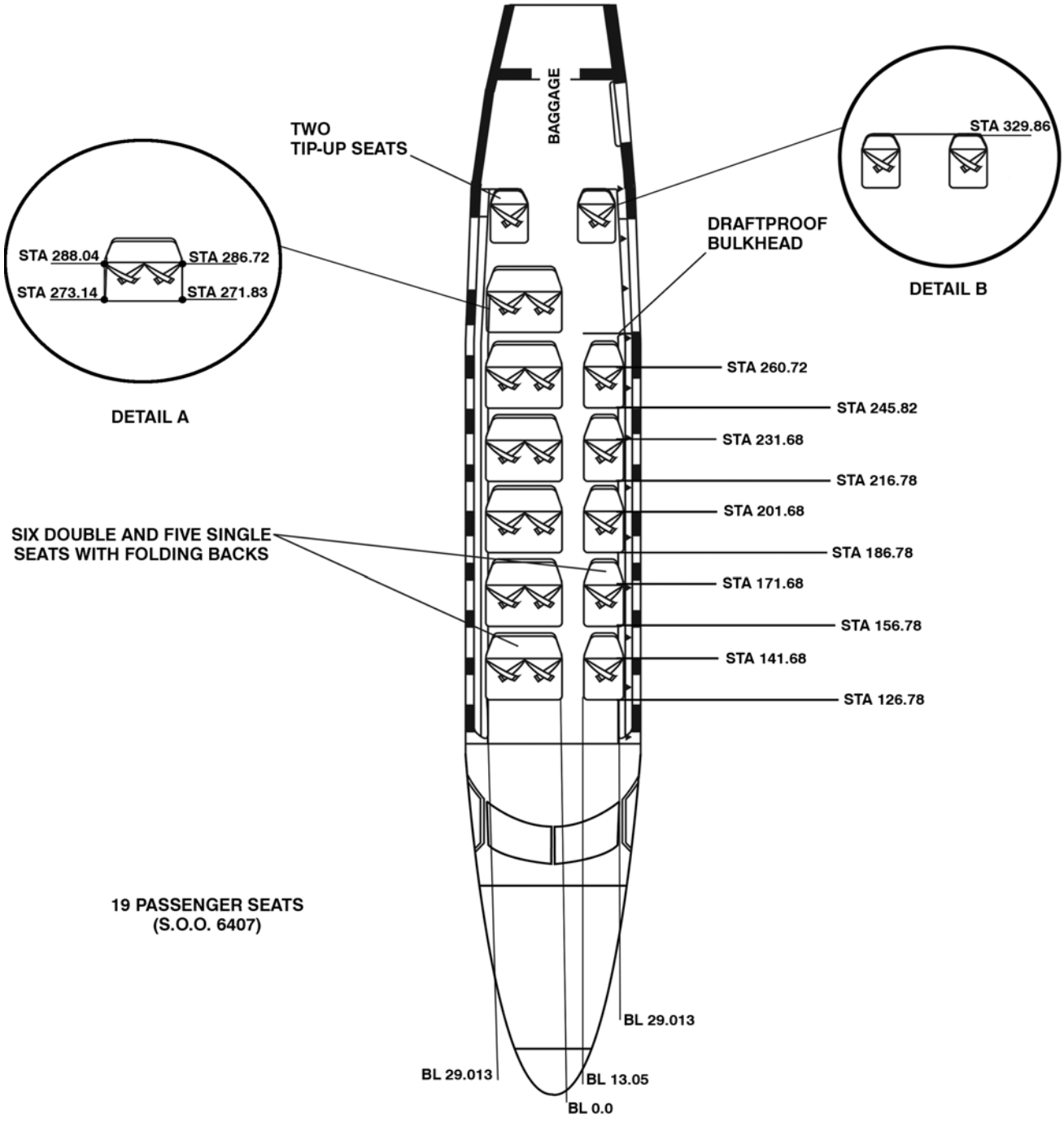
Seat Arrangement – Commuter – 14 Passenger (S.O.O. 6286) (Pre Mod 6/2281)
Figure 2



Seat Arrangement – Commuter – 19 Passenger (Post Mod 6/2281)
Figure 3



Seat Arrangement – Commuter – 14 Passenger (Post Mod 6/2281)
Figure 4



Seat Arrangement – Commuter – 19 Passenger (S.O.O. 6407)
Figure 5

3. Removal/Installation

A. Removal of Single or Double Seat (Pre Mod 6/2281)

1) Pull and rotate the spring loaded “lock and ring” on the forward outboard side of the seat and rotate until the roll pin protrudes from the seat lock.

2) Pull up and rotate the forward and aft spring loaded “lock collar” and slide the forward and aft seat legs clear of the seat track.

3) Slide the seat forward or aft until side seat support and lower seat leg supports are clear of the seat retaining tracks.

B. Installation of Single or Double Seat (Pre Mod 6/2281)

1) Pull and rotate the spring loaded “lock and ring” on the forward outboard side of the seat and rotate until the roll pin protrudes from seat lock.

2) Pull up on the forward and aft spring loaded seat leg “lock collars”.

3) Slide the seat assembly into the side and floor seat tracks at the correct locations. (Refer to Figure 1 and Figure 2 for locations).

4) Turn the forward outboard side “lock ring” to engage the forward seat lock.

5) Rotate the lower seat leg collar to engage the seat leg support in the floor track.

C. Removal of JB6 Series Seat Assemblies – Single or Double (Post Mod 6/2281)

Note Seat assemblies are secured to the aircraft seat tracks through the use of a stud and anti-rattle track lock.

1) Use an anti-rattle nut wrench to loosen the anti-rattle nuts located under the forward leg and sidewall attachment points.

2) Back out the threaded plunger on top of the aft track floor fitting and sidewall fitting.

3) Slide the seat assembly forward or aft one-half inch and lift aircraft seat off of tracks.

D. Installation of JB6 Series Seat Assemblies – Single or Double (Post Mod 6/2281)

Note Centered seat track (Mod 6/2280), is a prerequisite for installation of JB6 series seats.

1) Ensure the shear plungers are fully retracted on the aft sidewall fitting and floor track fitting.

2) Ensure anti-rattle nuts are fully backed off and against the seat frame.

3) Slide the seat assembly into the side and floor seat tracks at the correct locations. (Refer to Figure 3 and Figure 4 for locations).

4) Ensure break over seats are located at the first row (forward of emergency exit) and sixth row on the right hand side of the fuselage (refer to Figure 3 and Figure 4).

5) Thread the shear plungers into the wall and floor seat track mounts.

Note

- For floor track fitting torque to 36 to 40 in–lbs.
- For sidewall fitting torque to 28 to 32 in–lbs.

6) Using an anti-rattle nut wrench, tighten the anti-rattle nuts against the seat track. Hand tighten, plus 1/4 turn, the anti-rattle nuts against seat track using an anti-rattle nut wrench.

E. Removal of Station 332 Seats

1) Remove and retain two (2) AN5–4A bolts attaching the lower seat support to Station 332 bulkhead.

2) While supporting the seat, remove and retain two (2) AN5–7A bolts attaching the upper seat support to Station 332 bulkhead. Remove seat from aircraft and move to a suitable work area.

3) Repeat Para 3.E.1) and Para 3.E.2) on remaining 2 seats.

F. Installation of Station 332 Seats

1) Line up the seat mounting pads with bulkhead seat mounting locations.

2) Install upper support AN5–7A bolts prior to installing lower support AN5–4A bolts.

3) Repeat Para 3.F.1) and Para 3.F.2) on the remaining seats.

G. Removal of Dress Covers and Cushions – Single and Double

1) Carefully pull up on the seat bottom cushion to separate the hook and loop fasteners.

2) Remove dress cover upholstery from the seat bottom cushion.

3) Carefully pull the seat back cushion to separate hook and loop fasteners.

4) Remove dress cover upholstery from the seat back cushion.

H. Installation of Dress Covers and Cushions – Single and Double

1) Replace dress cover upholstery on the seat back cushion.

2) Carefully reconnect the seat back cushion hook and loop fasteners.

3) Replace dress cover upholstery to the seat bottom cushion.

4) Carefully reconnect the seat bottom cushion hook and loop fasteners.

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR25-19**

Insert in 25-20-02 before Page 1

CABIN COMPARTMENT – CABIN UPHOLSTERY (MOD 6/2013)

REASON FOR ISSUE:

To introduce Mod 6/2414, Cabin Fresh Air and Distribution – O/H Ducting Sealing Improvements.

ACTION:

In Chapter 25, ignore existing Section 25-20-02 and insert revised section, as shown on Pages 3 to 8 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

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Cabin Compartment – Cabin Upholstery (Mod 6/2013) – General data

1. General

The cabin compartment upholstery (Mod 6/2013) consists of ceiling panels, sidewall panel, station 111 bulkhead padding (Mod 6/2283) and with S.O.O. 6310 installed, custom logo bulkhead padding, a reading light and cold air outlet.

2. Description

A. Ceiling

The ceiling has a centre section made from a high impact abrasion resistant Acrylic PVC thermoplastic sheet and includes lens covers for the main cabin lights and speaker grills for the PA system. These panels are retained by a “H” section aluminum extrusion.

The main ceiling panels, made from the same thermoplastic material, comprises of 10 individual panels retained in place by “H” section aluminum extrusions and High Impact Poly Styrene (HIPS) panel joiners.

B. Cabin Interior Panels

The cabin interior panels, divided into upper and lower panels by an aluminum extrusion, are manufactured from a high impact abrasion resistant Acrylic PVC thermoplastic sheet. The cabin sidewalls and rear bulkhead have a sound deadening foam bonded to the rear surface.

C. Bulkhead (Mod 6/2283)

The lightweight forward bulkhead is a foam sandwich panel, with a moulded thermoplastic fascia bonded to the cabin side. Included on this forward bulkhead is a left and right Padding Assy (Mod 6/2099 and Mod 6/2123) with a magazine pocket on the fascia. An LED type passenger service sign displaying No Smoking and Fasten Seat Belt symbols, is located on the upper right bulkhead. An aluminum channel edges the doorway.

For aircraft incorporating customer standard option S.O.O. 6310, the station 111 bulkhead padding will incorporate custom embroidered logos on the padding cover.

D. Reading Light and Cold Air Outlet

A reading light and cold air outlet, located on the cold air ducts on both sides of the cabin, are provided for each passenger seat. The lights and cold air outlets can be individually controlled by the passengers. (Refer to 33-20-00, Passenger Compartment Lights (Mod 6/2028) – General Data).

E. Light and Switch Cover (Mod 6/2414)

The light and switch cover improves air distribution at cabin gasper vents. The box section ducts house the cabin gaspers and light/switch covers.

3. Removal/Installation

A. Remove Lower Sidewall Panels

1) Lift the lower aft edge of the panel with a flat bladed screwdriver.

2) With the screwdriver sliding over the side mounted seat track, push the lower aft edge of the panel forward.

B. Remove Sidewall Panels – Rear RH Door and Emergency Exits

1) Without removing the door surrounds, carefully remove the sidewall panel from the aluminum extrusions.

2) Gently pull the panel out from under the lip/edge of the door surround.

C. Install Sidewall Panels – Rear RH Door and Emergency Exits

1) Without removing the door surrounds, carefully insert the sidewall panel under the aluminum extrusions.

2) Gently push the panel behind the lip/edge of the door surround.

D. Remove Centre Ceiling Panels

1) Remove screws, nuts, washers and the extrusions on either side of the centre ceiling panel.

2) Remove the aluminum extrusions from each side of the ceiling panel and move the centre ceiling panel.

E. Install Centre Ceiling Panels

1) Position the aluminum extrusions on either side of the centre ceiling panel.

2) Join the aluminum extrusions using the extrusions joiners.

3) Fasten the aluminum extrusions with screws, nuts and washers.

F. Remove Cabin Ceiling

CAUTION **DON'T USE SCREWDRIVERS OR OTHER SHARP OBJECTS AS THIS WILL DAMAGE THE PANELS AND JOINERS.**

Refer to Figure 1 for details.

The easiest method to remove the ceiling panels for maintenance is to remove the left and right #2 panels first. These are located second from the forward bulkhead.

1) Push with the palm of the hand against the outboard radius of the #2 panel.

Note This will pull the panel away from the centre section extrusion and allow removal of the inboard edge with the other hand.

2) Pull down and remove the panel.

Note Once the #2 panel is removed, removal of the remaining panels is easier.

G. Install Cabin Ceiling

Refer to Figure 1 for details.

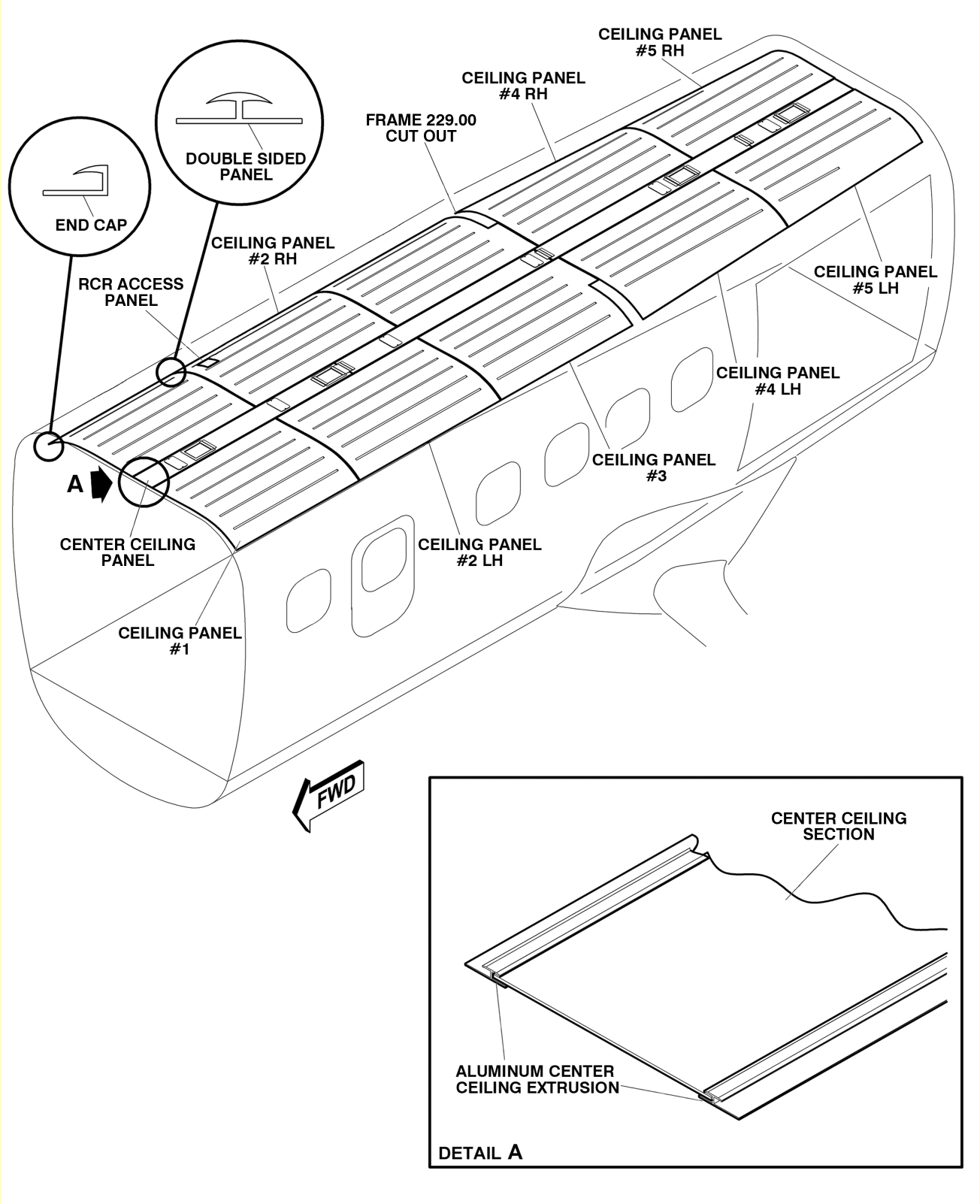
Install the forward #1 panel first. If the rear left and right #4 and #5 panels were removed, install #4 and #5 left and right then the #3 panel.

Panel installation is best achieved by fitting the outboard edge into the air duct first, then fit the inboard edge to the centre section extrusion.

1) At this stage the ceiling panel should be curved downwards. With the palm of the hand placed in the centre of the panel, push the panel upwards allowing it to spring upwards into position.

2) Install the plastic joining strips to #3 and #4 panels.

3) Install the #2 panels left and right.



Ceiling Panel Layout
Figure 1

4) Once #2 panel has been installed, install the joining strips as follows:

   a) Carefully press the outboard curve over the joint of the two panels.

      Note Gently ease the two corners down from the centre section extrusion, not the entire edge. This will allow the joiner to slide down between the two panels.

   b) Once the joiner is in place, reapply pressure with the palm of the hand to the outboard radius of the panels and carefully push the inboard corners inwards to return the inboard edge to the groove of the extrusion.

H. Removal of Bulkhead Padding Assembly (STA 111) (Pre Mod 6/2283)

1) Stand aft of the STA 111 bulkhead.

2) Hold the padding assembly near the top and bottom along the inboard edge and carefully pull assembly from sidewall of STA 111 bulkhead.

3) Repeat Para 3.H.2) for other side.

I. Installation of Bulkhead Padding Assembly (STA 111) (Pre Mod 6/2283)

1) Inspect padding assembly for condition and replace if necessary.

2) Align top and bottom edges of assembly with the sidewall of the STA 111 bulkhead.

3) Securely push assembly in place.

4) Repeat Para 3.I.2) and Para 3.I.3) for other side.

J. Removal of Bulkhead Padding Assembly (STA 111) (Post Mod 6/2283)

1) Remove the cockpit soft panels by holding the corner of the panels and carefully pull to separate the hook and loop fasteners on the STA 111 bulkhead, to gain access to the padding assembly retaining fasteners.

2) Remove the four (4) screws and washers that secure the assembly.

3) Remove the assembly.

K. Installation of Bulkhead Padding Assembly (STA 111) (Post Mod 6/2283)

1) Inspect padding assembly for condition and replace if necessary.

2) Position assembly aft of sidewall of STA 111 bulkhead.

3) Locate one mounting screw on the forward side of the STA 111 bulkhead and loosely secure into the standoff of the padding assembly.

4) Locate and install remaining three (3) screws and washers.

   CAUTION DO NOT OVER TIGHTEN SCREWS, THIS COULD DAMAGE THE BULKHEAD.

5) Tighten screws to snug to secure.

6) Install the cockpit soft panels by pressing firmly on the edges to secure the hook and loop fastening tape. Ensure panels are secure.

7) Repeat Para 3.K.2) through Para 3.K.6) for other side.

L. Removal of Light/Switch Cover (Mod 6/2414)

1) Carefully remove inboard half duct of the box section ducts.

2) Remove fillet seal around light/switch cover.

3) Disconnect wire in the nearest terminal lug to remove cover.

4) Replace light unit, if required.

M. Installation of Light/Switch Cover (Mod 6/2414)

1) Reconnect wire disconnected in Para 3.L.3).

2) Install cover by re-sealing with DAPCO 2200 sealant.

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NOTE: Please see the TEMPORARY REVISION that revises this page.

Cabin Compartment – Cabin Upholstery (Mod 6/2013) – General data

1. General

The cabin compartment upholstery (Mod 6/2013) consists of ceiling panels, sidewall panel, station 111 bulkhead padding (Mod 6/2283) and with S.O.O. 6310 installed, custom logo bulkhead padding, a reading light and cold air outlet.

2. Description

A. Ceiling

The ceiling has a centre section made from a high impact abrasion resistant Acrylic PVC thermoplastic sheet and includes lens covers for the main cabin lights and speaker grills for the PA system. These panels are retained by a “H” section aluminum extrusion.

The main ceiling panels, made from the same thermoplastic material, comprises of 10 individual panels retained in place by “H” section aluminum extrusions and High Impact Poly Styrene (HIPS) panel joiners.

B. Cabin Interior Panels

The cabin interior panels, divided into upper and lower panels by an aluminum extrusion, are manufactured from a high impact abrasion resistant Acrylic PVC thermoplastic sheet. The cabin sidewalls and rear bulkhead have a sound deadening foam bonded to the rear surface.

C. Bulkhead (Mod 6/2283)

The lightweight forward bulkhead is a foam sandwich panel, with a moulded thermoplastic fascia bonded to the cabin side. Included on this forward bulkhead is a left and right Padding Assy (Mod 6/2099 and Mod 6/2123) with a magazine pocket on the fascia. An LED type passenger service sign displaying No Smoking and Fasten Seat Belt symbols, is located on the upper right bulkhead. An aluminum channel edges the doorway.

For aircraft incorporating customer standard option S.O.O. 6310, the station 111 bulkhead padding will incorporate custom embroidered logos on the padding cover.

D. Reading Light and Cold Air Outlet

A reading light and cold air outlet, located on the cold air ducts on both sides of the cabin, are provided for each passenger seat. The lights and cold air outlets can be individually controlled by the passengers. (Refer to 33-20-00, Passenger Compartment Lights (Mod 6/2028) – General Data).

3. Removal/Installation

A. Remove Lower Sidewall Panels

1) Lift the lower aft edge of the panel with a flat bladed screwdriver.

2) With the screwdriver sliding over the side mounted seat track, push the lower aft edge of the panel forward.

B. Remove Sidewall Panels – Rear RH Door and Emergency Exits

1) Without removing the door surrounds, carefully remove the sidewall panel from the aluminum extrusions.

2) Gently pull the panel out from under the lip/edge of the door surround.

C. Install Sidewall Panels – Rear RH Door and Emergency Exits

1) Without removing the door surrounds, carefully insert the sidewall panel under the aluminum extrusions.

2) Gently push the panel behind the lip/edge of the door surround.

NOTE: Please see the TEMPORARY REVISION that revises this page.

D. Remove Centre Ceiling Panels

1) Remove screws, nuts, washers and the extrusions on either side of the centre ceiling panel.

2) Remove the aluminum extrusions from each side of the ceiling panel and move the centre ceiling panel.

E. Install Centre Ceiling Panels

1) Position the aluminum extrusions on either side of the centre ceiling panel.

2) Join the aluminum extrusions using the extrusions joiners.

3) Fasten the aluminum extrusions with screws, nuts and washers.

F. Remove Cabin Ceiling

CAUTION **DON'T USE SCREWDRIVERS OR OTHER SHARP OBJECTS AS THIS WILL DAMAGE THE PANELS AND JOINERS.**

Refer to Figure 1 for details.

The easiest method to remove the ceiling panels for maintenance is to remove the left and right #2 panels first. These are located second from the forward bulkhead.

1) Push with the palm of the hand against the outboard radius of the #2 panel.

Note This will pull the panel away from the centre section extrusion and allow removal of the inboard edge with the other hand.

2) Pull down and remove the panel.

Note Once the #2 panel is removed, removal of the remaining panels is easier.

G. Install Cabin Ceiling

Refer to Figure 1 for details.

Install the forward #1 panel first. If the rear left and right #4 and #5 panels were removed, install #4 and #5 left and right then the #3 panel.

Panel installation is best achieved by fitting the outboard edge into the air duct first, then fit the inboard edge to the centre section extrusion.

1) At this stage the ceiling panel should be curved downwards. With the palm of the hand placed in the centre of the panel, push the panel upwards allowing it to spring upwards into position.

2) Install the plastic joining strips to #3 and #4 panels.

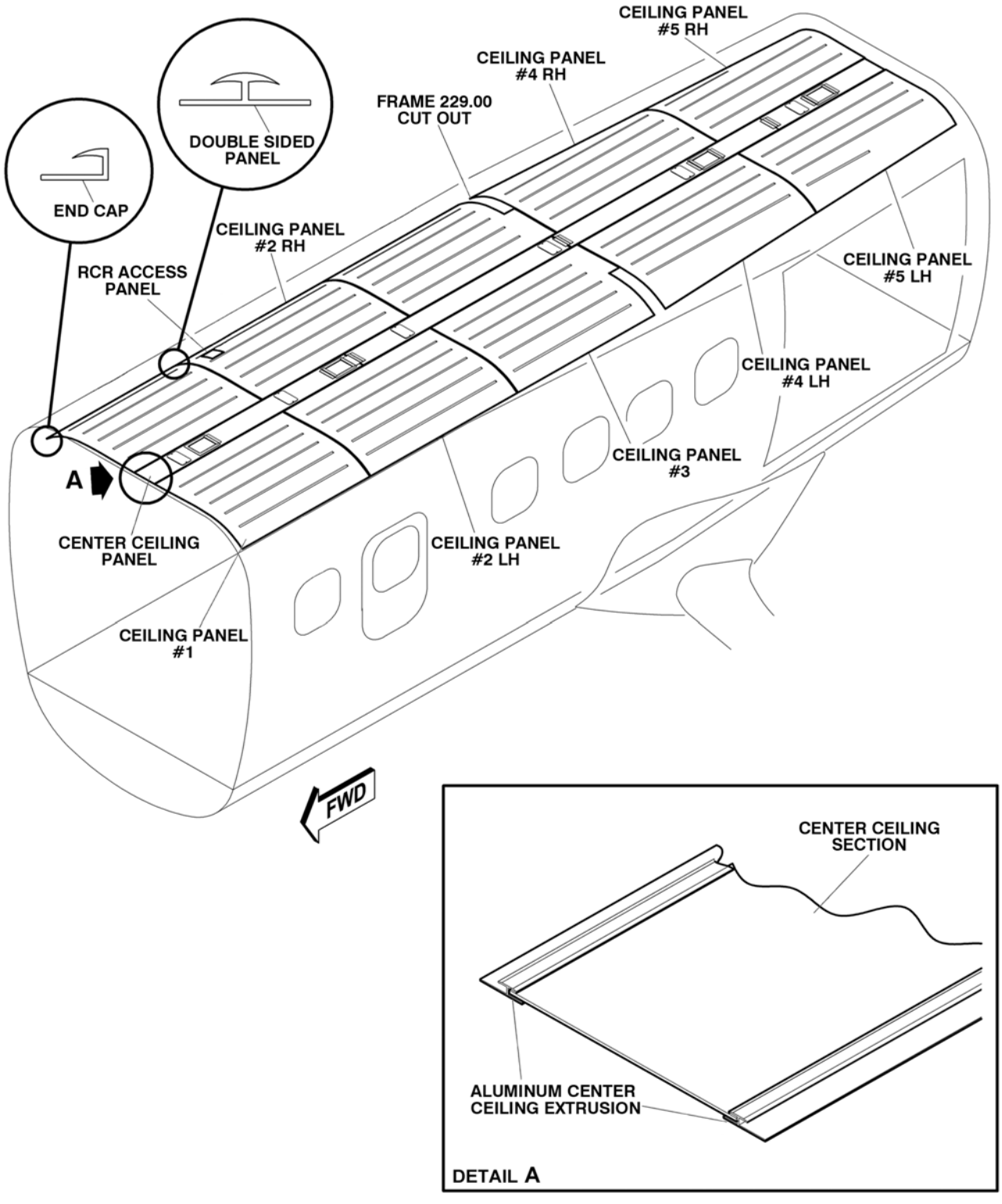
3) Install the #2 panels left and right.

4) Once #2 panel has been installed, install the joining strips as follows:

a) Carefully press the outboard curve over the joint of the two panels.

Note Gently ease the two corners down from the centre section extrusion, not the entire edge. This will allow the joiner to slide down between the two panels.

b) Once the joiner is in place, reapply pressure with the palm of the hand to the outboard radius of the panels and carefully push the inboard corners inwards to return the inboard edge to the groove of the extrusion.



Ceiling Panel Layout
Figure 1

NOTE: Please see the TEMPORARY REVISION that revises this page.

H. Removal of Bulkhead Padding Assembly (STA 111) (Pre Mod 6/2283)

1) Stand aft of the STA 111 bulkhead.

2) Hold the padding assembly near the top and bottom along the inboard edge and carefully pull assembly from sidewall of STA 111 bulkhead.

3) Repeat Para 3.H.2) for other side.

I. Installation of Bulkhead Padding Assembly (STA 111) (Pre Mod 6/2283)

1) Inspect padding assembly for condition and replace if necessary.

2) Align top and bottom edges of assembly with the sidewall of the STA 111 bulkhead.

3) Securely push assembly in place.

4) Repeat Para 3.I.2) and Para 3.I.3) for other side.

J. Removal of Bulkhead Padding Assembly (STA 111) (Post Mod 6/2283)

1) Remove the cockpit soft panels by holding the corner of the panels and carefully pull to separate the hook and loop fasteners on the STA 111 bulkhead, to gain access to the padding assembly retaining fasteners.

2) Remove the four (4) screws and washers that secure the assembly.

3) Remove the assembly.

K. Installation of Bulkhead Padding Assembly (STA 111) (Post Mod 6/2283)

1) Inspect padding assembly for condition and replace if necessary.

2) Position assembly aft of sidewall of STA 111 bulkhead.

3) Locate one mounting screw on the forward side of the STA 111 bulkhead and loosely secure into the standoff of the padding assembly.

4) Locate and install remaining three (3) screws and washers.

CAUTION DO NOT OVER TIGHTEN SCREWS, THIS COULD DAMAGE THE BULKHEAD.

5) Tighten screws to snug to secure.

6) Install the cockpit soft panels by pressing firmly on the edges to secure the hook and loop fastening tape. Ensure panels are secure.

7) Repeat Para 3.K.2) through Para 3.K.6) for other side.

28 Volt Power System (Mod S.O.O. 6283) – General data

1. General

There is a 28 volt power outlet configuration available in the cabin to provide power to peripherals: cell phone, laptops, etc.

2. Description

There is a single 28 volt power outlet that may be installed on the forward right hand side of the STA 332 bulkhead in the cabin. The outlet has a maximum load of 5 amps.

28 volt DC Power is provided from the right bus, through a 5 amp circuit breaker (CB18) located on the power distribution circuit box.

3. Removal/Installation

A. Removal of 28V Power Outlet

1) Ensure the aircraft electrical system is OFF.

2) Open the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

3) Remove the countersunk screws from the outlet mounting plate.

4) Remove the outlet assembly from the interior paneling.

5) Gently remove electrical connectors from the rear terminals of the outlet. Note which connector came off the center terminal and which came off the outer terminal.

6) Unthread the two metallic locknuts from the back of the outlet.

7) Remove outlet by pulling it out of the front of the mounting hole.

Note Stow the connectors as required.

B. Installation of 28V Power Outlet

1) Ensure the aircraft electrical system is OFF.

2) Ensure the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

3) Install the outlet in mounting panel from the front of the panel.

4) Thread the two metallic locknuts onto the back of the outlet and tighten. Ensure the outlet is secured in the panel.

5) Install the electrical connectors onto the appropriate terminals on the rear of the outlet.

6) Install outlet assembly from the interior paneling.

7) Install the countersunk screws from the outlet mounting plate.

8) Remove the safety clips and close the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

9) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +28V (± 1.0V).

4. Troubleshooting

A. Ensure CB18 circuit breakers located on the power distribution circuit box are closed.

B. Visually inspect the outlet connectors for damaged wiring or contacts. Ensure electrical connectors on outlet are fully seated and secure.

C. If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in PSM 1-64-2W.

14 Volt Power System (Mod S.O.O. 6272, 6281 and 6282) – General data

1. General

There are three 14 volt power outlets configuration are available in the cabin to provide power to peripherals: cell phone, laptops, etc.

2. Description

A. STA 332 14V Convenience Outlet – (Mod S.O.O. 6272)

There is a single 14 volt power outlet may be installed on the forward right hand side of STA 332 bulkhead in the cabin. A 14 volt convenience outlet on the forward, right hand side of STA 332 bulkhead. The outlet has a maximum load 7.5 amps. The 28 Volt DC to 14 Volt DC power converter is located on the aft avionics racks, mounted underside of the top shelf below the CVR.

28 Volt DC power is provided from the right bus, through a 5 amp circuit (CB18) located on the power distribution circuit box. 14 Volt DC power is provided to the outlet and is protected by a 7.5 amp STA 332 circuit breaker on the aft converter circuit breaker panel located near the power converter.

B. Aft Cabin 14V Convenience Outlet – (Mod S.O.O. 6281)

There are three 14 volt power outlets may be installed between STA 220 and 332 bulkhead in the cabin. Each outlet has maximum load 7.5 amps. The 28 Volt DC to 14 Volt DC power converter is located on the aft avionic racks, mounted underside of the top shelf below the CVR.

28 Volt DC power is provided from the right bus, through a 15 amp circuit (CB18) located on the power distribution circuit box. 14 Volt DC power is provided to each of the three outlets and each outlet is protected by a 7.5 amp STA 223, STA 258, STA 332 circuit breakers located on the aft converter circuit breaker panel located near the power converter.

C. Fwd Cabin 14V Convenience Outlet – (Mod S.O.O. 6282)

There are three 14 volt power outlets that may be installed between STA 111 and 186 in the cabin. Each outlet has maximum load 7.5 amps. The 28 Volt DC to 14 Volt DC power converter is located on the forward avionics compartment, mounted in under the middle shelf. In addition to powering the three cabin outlets the power converter also powers the existing two 14 volt outlets located in the cockpit, on the pilot and co-pilot's side of the center pedestal, refer to 25-11-00, 14 Volt Power Converter System – General Data.

The 28 Volt DC power is provided from the right bus through a 15 amp PWR CONVR 28V INPUT (M6) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data). 14 Volt DC power is provided to the existing cockpit outlets through a 7.5 amps PWR CONVR 14V OUTPUT (N6) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data), and to the cabin outlets through a 7.5 amp STA 128 14 VDC, STA 176L 14 VDC, STA 176R VDC circuit breaker located on the forward converter circuit breaker panel located near the power converter.

3. Adjustment/Test

A. Function Test of 14 Volt Power Converter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +13.8V (± 0.5V).

3) Plug an accessory into each outlet, as required, and ensure the accessory is being powered by the outlet.

4) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal/Installation

A. Removal of 14 Volt Power Converter for Mod S.O.O. 6282

1) Ensure the aircraft electrical system is OFF.

2) Open the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel, and the CB18 circuit breakers located on the power distribution circuit box and fit safety clips.

3) Gain access to the forward avionics shelves through the forward baggage compartment.

4) Remove the middle and bottom covers attached to bulkhead 44.00 as required.

5) Disconnect the electrical connector.

6) While supporting the 14V Power Converter, remove the four mounting screws.

7) Remove unit from shelf.

Note Stow connectors as required.

B. Removal of 14 Volt Power Converter for Mod S.O.O. 6272 and 6281

1) Ensure the aircraft electrical system is OFF.

2) Open the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

3) Gain access to the 14 Volt Power Converter, which is accomplished through the aft baggage compartment into the rear compartment.

4) Disconnect the electrical connector.

5) While supporting the 14V Power Converter, remove the four mounting screws.

6) Remove unit from shelf.

Note Stow connectors as required.

C. Installation of 14 Volt Power Converter for Mod S.O.O. 6282

1) Ensure the aircraft electrical system is OFF.

2) Ensure the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel.

3) Set unit in position and install four mounting screws.

4) Reconnect electrical connector.

5) Install the middle and bottom covers attached to bulkhead 44.00. Ensure the covers are mounted securely.

6) Remove the safety clips and close the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel.

7) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +13.8V (± 0.5V).

8) Do a function test as per Para 3.A.

D. Installation of 14 Volt Power Converter for Mod S.O.O. 6272 and 6281

1) Ensure the aircraft electrical system is OFF.

2) Ensure the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

3) Set unit in position and install four mounting screws.

4) Reconnect electrical connector.

5) Remove the safety clips and close the CB18 circuit breakers located on the power distribution circuit box.

6) Do a function test as per Para 3.A.

E. Removal of 14V Power Outlets for Mod S.O.O. 6282

1) Ensure the aircraft electrical system is OFF.

2) Open the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel and fit safety clips.

3) Remove the countersunk screws from the outlet mounting plate.

4) Remove outlet assembly from the interior paneling.

5) Gently separate the outlet electrical disconnect. Note which disconnect came off the negative terminal and which disconnect came off the positive terminal, indicated on back side of plug.

6) Unthread the metallic locknut from the back of the outlet.

7) Remove outlet by pulling it out the front of the mounting hole.

Note Stow the connectors as required.

F. Removal of 14V Power Outlets for Mod S.O.O. 6272 and 6281

1) Ensure the aircraft electrical system is OFF.

2) Open the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

3) Remove the countersunk screws from the outlet mounting plate.

4) Remove outlet assembly from the interior paneling.

5) Gently separate the outlet electrical disconnect. Note which disconnect came off the negative terminal and which disconnect came off the positive terminal, indicated on back side of plug.

6) Unthread the metallic locknut from the back of the outlet.

7) Remove outlet by pulling it out from the front of the mounting hole.

Note Stow the connectors as required.

G. Installation of 14V Power Outlet for Mod S.O.O. 6282

1) Ensure the aircraft electrical system is OFF.

2) Ensure the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel, and the CB18 circuit breakers located on the power distribution circuit box and fit safety clips.

3) Install the outlet in mounting panel from the front of the panel.

4) Thread the metallic locknut onto the back of the outlet and tighten. Ensure the outlet is secured in the panel.

5) Install the electrical connectors onto the appropriate terminals on the rear of the outlet.

6) Install outlet assembly from the interior paneling.

7) Install the countersunk screws from the outlet mounting plate.

8) Remove the safety clips and close the PWR CONV 28V INPUT (M6) and PWR CONV 14V OUTPUT (N6) circuit breakers on the avionics circuit breaker panel, and the CB18 circuit breakers located on the power distribution circuit box.

9) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +13.8V (± 0.5V).

H. Installation of 14V Power Outlet for Mod S.O.O. 6272 and 6281

1) Ensure the aircraft electrical system is OFF.

2) Ensure the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

3) Install the outlet in mounting panel from the front of the panel.

4) Thread the metallic locknut onto the back of the outlet and tighten. Ensure the outlet is secured in the panel.

5) Install the electrical connectors onto the appropriate terminals on the rear of the outlet.

6) Install the countersunk screws from the outlet mounting plate.

7) Remove the safety clips and close the CB18 circuit breakers located on the power distribution circuit box and fit safety clips. (Refer to 24-60-11, Power Distribution and Generator Control Box (Mod 6/2028) – General Data).

8) Using a voltmeter, measure the voltage between the center pin and the outer ring of the outlet. Ensure the 'com' lead is connected to the outer ring. Confirm the voltage is +13.8V (± 0.5V).

5. Troubleshooting

A. Ensure associated system of the circuit breakers are closed.

B. Visually inspect the outlet connectors for damaged wiring or contacts. Ensure electrical connectors on outlet are fully seated and secure.

C. If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in PSM 1-64-2W.

D. Measure 14V output voltage at PWR CONV.

Parachute Anchor Line (S.O.O. 6291) – General Data

1. General

Provisions have been made for the installation of a Parachute Anchor Line from station 112.73 to 321.98 in the passenger compartment. This installation requires that the third seat rail standard customer option, Mod S.O.O. 6243, be installed on aircraft.

2. Description

The Parachute Anchor Line installation consists of a cable assembly that runs between two anchor plates; a forward anchor plate at station 112.73 and an aft anchor plate at station 321.98, secured on the right hand and center seat rails, and a tie down for an Emergency Static Line hook-up at station 301.19, located on the left hand seat rail.

The cable assembly is made from a 3/8 inch diameter 7x19 steel cable, with a turnbarrel for adjustments, and the required attachment fittings and hardware. The cable assembly runs down the right hand side of the aircraft between station 112.73 and 321.98.

3. Removal/Installation

A. Removal of Parachute Anchor Line

1) Remove the lockwire from the turnbarrel in the cable assembly.

2) Adjust the turnbarrel to release the tension in the cable assembly.

3) Remove the nuts, washers and bolts that secure the cable to the anchor bracket and remove the cable assembly.

4) Remove the nuts, washers and bolts that secure the forward anchoring plate to the right hand and center seat rail at station 112.73, and remove the anchoring plate.

5) Remove the nuts, washers and bolts that secure the aft anchoring plate to the right hand and center rail at station 321.98, and remove the anchoring plate.

6) Remove the tie down for the Emergency Static Line hook-up from the left hand seat rail at station 301.19.

7) Install the seat track end piece at the front of both the center and right hand seat rails, and secure with screws.

B. Installation of Parachute Anchor Line

1) Remove the screws securing the seat track end piece, and remove the seat track end piece at the front of both the right hand and center seat rails.

2) Install the bolts, washers and nuts to secure the anchoring plate to the right hand and center seat rail at station 112.73.

3) Install the bolts, washers and nuts to secure the anchoring plate to the right hand and center rail at station 321.98.

4) Install the cable assembly, and install the bolts, washers and nuts securing the cable assembly to the anchor brackets.

5) Adjust the turnbarrel to the tension in the cable assembly.

Note Do not over-tighten the turnbarrel. Adjust sufficiently to draw the cable taut.

6) Safety the turnbarrel with lock wire.

7) Install the tie down for the Emergency Static Line hook-up from the left hand seat rail at station 301.19.

Cargo Compartments – General Data

1. General

   Provision is made for carrying cargo in the front, rear and cabin compartments. Refer to Figure 1 for details.

2. Front Baggage Compartment

   Four permanently attached tie-down rings are provided in the front baggage compartment.

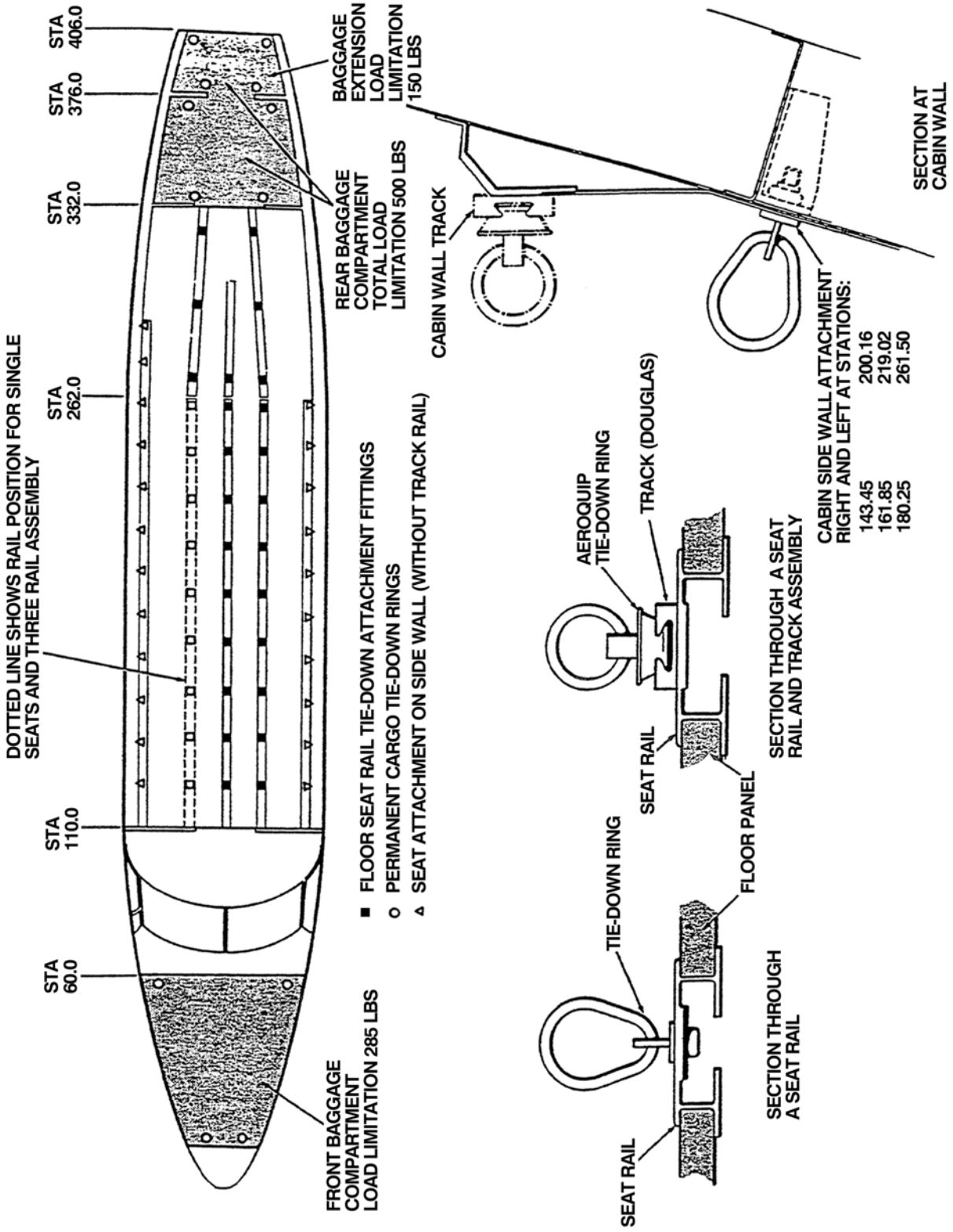
3. Rear Baggage Compartment

   Eight permanently attached tie-down rings are provided, four in the baggage compartment and four in the baggage compartment extension. A baggage restraint webbing is fitted at the semi-bulkhead between the baggage compartment and the extension.

4. Cabin Compartment

   The main cabin floor is stressed to carry loads of 200 pounds per square foot and the forward and rear baggage compartments are stressed to carry 100 pounds per square foot.

   With passenger seats removed, cargo tie-down rings can be attached to anchor nuts in the floor rails, seat side rails and the beam on the cabin rear bulkhead may be used for attaching tie-down rings when cargo is to be carried in the cabin. Douglas track is installed on the existing floor and side rails, and commercially-supplied tie-down rings designed for Douglas track may also be used.



Cargo Tie-Down Point Locations
Figure 1

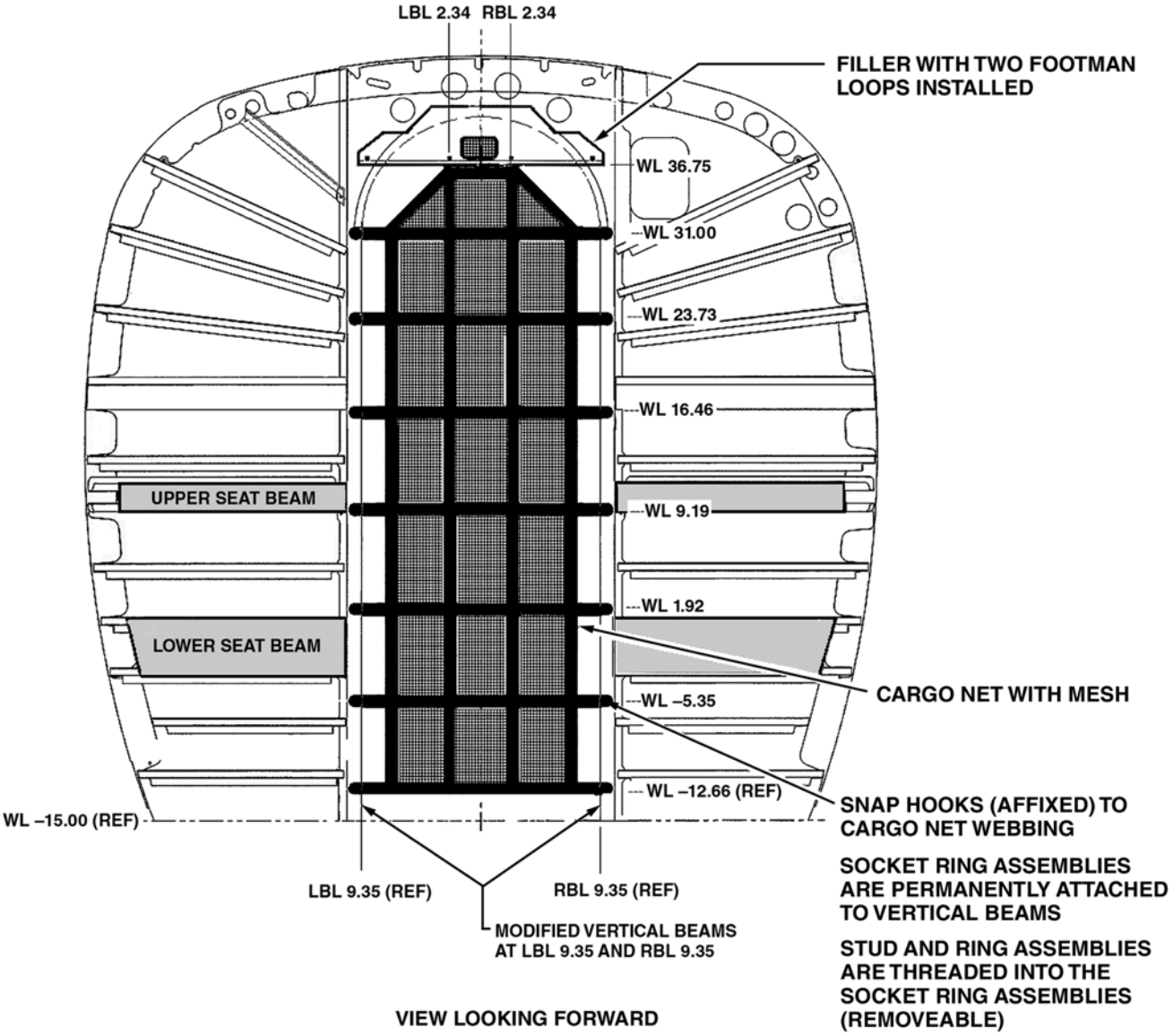
STA 332 Modified Bulkhead and Cargo Net Installation – General Data

1. General

The STA 332 bulkhead is modified to provide an opening into the rear baggage compartment as shown in Figure 1.

A removable lightweight cargo net is installed in the opening of the STA 332 modified bulkhead to separate the passenger cabin from the rear baggage compartment, and allows access to the rear baggage compartment from inside the aircraft.

The cargo net can be safely stowed in a Filler at the top of STA 332 bulkhead.



STA 332 Bulkhead Modification and Cargo Net Installation
Figure 1

2. Description

A. STA 332 Modified Bulkhead

The STA 332 bulkhead is modified by removing the center portion of the upper and lower seat beams and removing the center single folding seat. The two vertical beams are re-inforced to provide additional strength, and have stud and ring hardware installed to accommodate installation/removal of the cargo net.

B. Cargo Net

The cargo net is a light-weight web and mesh design which is secured in place by using fourteen snap hooks (seven per side), which are permanently sewn to the webbing. The cargo net snap hooks are “snapped” directly to the removable stud and ring assemblies on the vertical beams.

The top of the cargo net is secured using two Footman Loops attached to the filler installed at the top of the STA 332 bulkhead. The filler provides secure stowage when the cargo net is not in use.

C. Socket Ring Assemblies

The socket ring assemblies are permanently mounted to the modified vertical beams (7 socket ring assemblies per beam) and are threaded with a helicoil insert to accept the stud and ring assemblies.

D. Stud and Ring Assemblies

The stud and ring assemblies are removable and consist of a ring, which loops through a threaded stud. The cargo net snap hooks are “snapped” directly to the stud and ring assemblies.

3. Removal/Installation

A. Cargo Net

1) Removal

a) Open the clasp of each snap hook and disengage the snap hook from the stud and ring assembly down each side of the vertical beams (14 in total).

b) Prepare the cargo net for stowage inside the aircraft as per Para 3.A.3).

2) Installation

a) Prior to installing the cargo net, carry out the following:

1 Inspect the cargo net webbing for condition (refer to Para 4.). If damaged, repair or replace (refer to Para 5.).

2 Inspect the stud and ring assemblies on the aft side of the vertical beams for condition and corrosion. If damaged or missing, replace (refer to Para 5.).

3 Ensure that the identification tag of the cargo net faces aft.

b) Secure the top of the cargo net using two Footman Loops attached to the filler, located at the top of the STA 332 bulkhead.

c) Secure the cargo net in the opening of the STA 332 bulkhead by opening the clasp of each snap hook (14 in total) and engaging the snap hook into the stud and ring assemblies down each side of the vertical beams.

3) Stowage

a) Open the clasp of each snap hook (14 in total) and disengage the snap hook from the stud and ring assembly down each side of the bulkhead.

b) Roll the cargo net forward and upwards starting from the bottom.

c) Secure the rolled up cargo net in place using the two straps (Velcro) attached to the upper portion of the cargo net.

d) Ensure the cargo net is securely stowed.

B. Stud and Ring Assemblies

1) Removal

a) Remove the threaded stud and ring assemblies by turning the threaded assembly counterclockwise until it disengages from the socket ring assembly.

2) Installation

a) The stud and ring assemblies shall be oriented in the horizontal position for proper installation of the cargo net. The stud may be shimmed by adding AN960JD416 washers (or equivalent), as required to obtain the necessary horizontal orientation.

Note Torque for installing the stud and ring assembly (1/4 x 28 UNF thread): Hand tight.

C. Socket Ring Assemblies

If removal of the socket ring assemblies is required from inside the vertical beams, contact Viking Air Limited.

4. Inspection

A. Cargo Net Mesh

1) Mesh Damage Limitations are as follows:

a) Damage consisting of tears and/or wear that exceed 25% of the total number of meshed panels, is considered to be non-serviceable and the net shall be replaced.

b) No more than two (2) patches per section is permitted for field repairs.

c) Complete panel replacement shall be done by Cargo Systems Inc. only.

d) Material used for all patches is to be provided by Cargo Systems Inc. (P/N A4828–0705), and shall be made in accordance with procedures and specifications prescribed by Cargo Systems Inc.

e) For tears that are close to the edge of the webbing, patches may overlap webbing.

f) All patches are to be installed on the side OPPOSITE (backside) of the webbing.

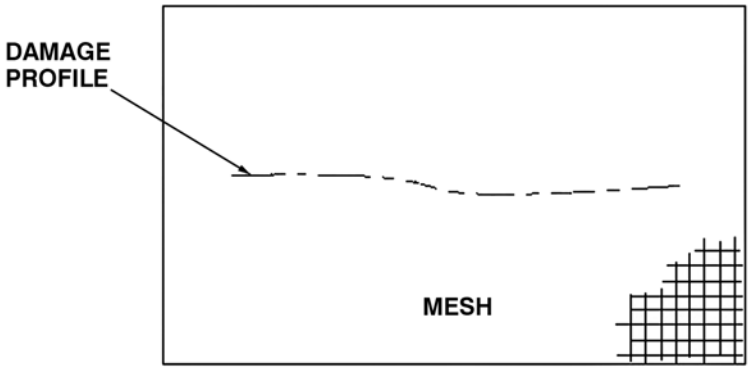
2) Mesh with Linear (Straight) Tears

Refer to Figure 2 for details.

a) Linear (straight) tears less than or equal to 2.0” long are acceptable. Repair NOT necessary at this time.

b) Field patch repair is acceptable for linear (straight) tears greater than 2.0” long, but less than 5.0” long. Refer to Figure 4.

c) Field patch repairs are not authorized for linear (straight) tears greater than 5.0” long. Return the cargo net to Cargo Systems Inc. for repair/replacement.



Mesh with Linear (Straight) Tears
Figure 2

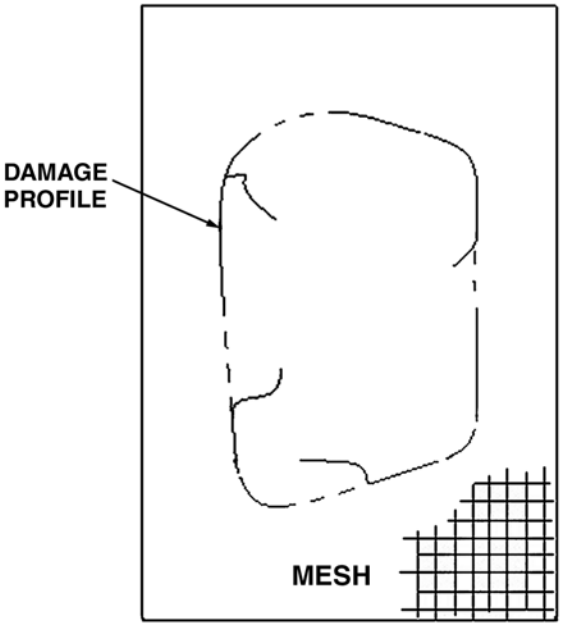
3) Mesh with Non-Linear (Uneven) Tears

Refer to Figure 3 for details.

a) Holes less than 1.0” diameter are acceptable. Repair is NOT necessary at this time.

b) Field patch repair is acceptable for holes greater than 1.0” diameter, but less than 3.0” diameter. Refer to Figure 5.

c) Field patch repairs are not authorized for tears greater than 3.0” diameter. Return the cargo net to Cargo Systems Inc. for repair/replacement.



Mesh with Non-Linear (Uneven) Tears
Figure 3

B. Cargo Net Webbing

1) Webbing Damage Limitations

a) Cargo net webbing that has tears, cuts and/or frayed edges (where the thread yarns have broken through) is not airworthy and must be repaired or returned to Cargo Systems Inc.

b) Light fraying (or fuzzing) of the webbing edges (where the thread yarns are still intact) is acceptable, provided that no tears or cuts are evident in the webbing yarns.

c) Replace the cargo net if any of the following conditions exits:

1 The length of an edge cut to the webbing exceeds the webbing thickness.

2 The penetration of the abrasion exceeds 15% of the webbing thickness.

3 If abrasion occurs on both sides of the webbing and the sum of the abrasion on both sides of the webbing exceeds 15% of the total webbing thickness.

4 Broken or worn stitching.

5 The identification tag is unreadable or missing.

C. Hardware

1) Snap Hooks

a) Inspect the snap hooks for security and condition. If corrosion is evident and cannot be removed using superfine steel wool or metal polish, the net should be removed from service and returned to Cargo Systems Inc. for repair.

b) Inspect the spring-loaded clasp to ensure that it is functional and operating smoothly.

c) Replace the snap hook if cracked, distorted or broken (refer to Para 5.D) or return the cargo net to Cargo Systems Inc. for snap hook replacement.

2) Socket Ring Assemblies (installed inside each beam)

a) Inspect the MS21209F4–15 helicoil inside each socket ring assembly for damage or corrosion. Remove and replace the helicoil, if necessary.

b) If removal of the socket ring assemblies from inside the vertical beam is required, contact Viking Air Limited.

3) Stud and Ring Assemblies

a) Inspect the stud and ring assemblies for security and condition.

b) Inspect the stud and ring assemblies for corrosion, distortion and function.

c) Ensure the ring can rotate freely within the stud and is mounted in the horizontal direction so the ring hangs down.

d) Replace stud and ring assemblies if cracked, distorted or broken.

D. STA 332 Modified Bulkhead

1) Vertical Beams

a) Inspect the aft strap doublers on both vertical beams for condition and wear at each of the stud and ring assembly locations.

b) Ensure the aft strap doublers are not gouged or distorted from contact with the rings from the cargo net stud and ring assembly.

Note If there's any damage to the aft strap doublers of the vertical beams, contact Viking Air Limited.

c) Inspect the vertical beam gussets (which attach the vertical beams to the STA 332 bulkhead stiffeners) for distortion and/or loose rivets. If there is any damage to the gussets of the vertical beams, contact Viking Air Limited.

5. Repairs

Note All repairs MUST always be accomplished by a qualified and approved repair facility.

A. Mesh Repair Materials (available from Cargo Systems Inc.)

- Thread: CSI P/N 29520–BLK, 29523–BLK if single layered mesh; or thick 29525–BLK if double layer mesh is used or overlapping webbing
- Stitching: 6 - 8 stitches per inch per MIL–STD-751, Type 301 Straight Stitch
- Mesh: CSI P/N A48280705 (Double layer recommended for durability; folded edge not required)

Note Material substitutions for the mesh and thread are not permitted unless approved by Cargo Systems Inc.

B. Cargo Net Mesh Repair

Cargo net mesh repair is possible providing:

1) Complete mesh panel replacement is supplied by Cargo Systems Inc. only.

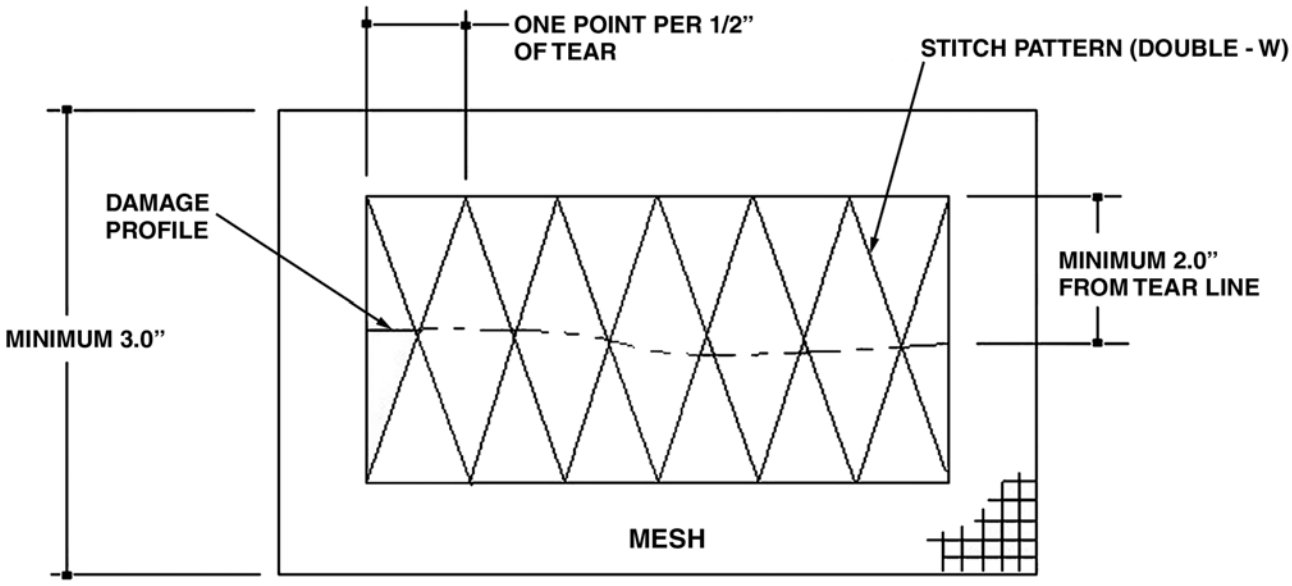
2) All mesh repair materials (P/N A48280705) are obtained from Cargo Systems Inc.

3) For tears that are in close proximity to the edge of the webbing, mesh patches may overlap webbing.

4) All mesh patches are installed on the side OPPOSITE (backside) of the webbing.

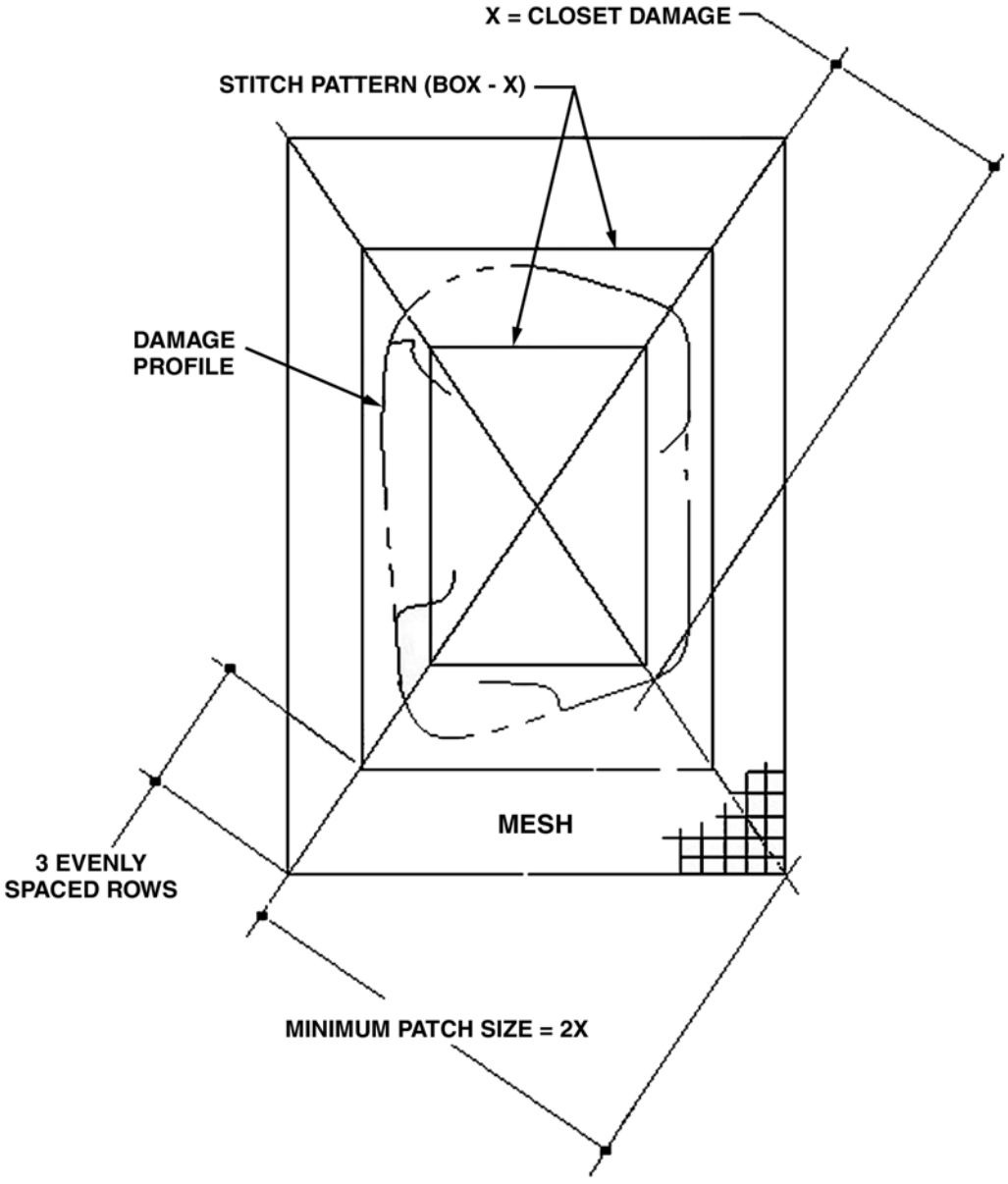
5) No more than two field repair patches are permitted for each meshed panel.

6) Repairs are made in accordance with procedures and specifications prescribed by Cargo Systems Inc.



Typical Stitch Pattern Layout (Double – W) for Mesh Repair with Straight (Linear) Tears
Figure 4

Note A double layer of mesh is recommended for this repair, using thicker 29525 BLK thread available from Cargo Systems Inc.



Typical Stitch Pattern Layout (Box – X) for Mesh Repair with Non-Linear (Uneven) Tears
Figure 5

Note A double layer of mesh is recommended for this repair, using thicker 29525 BLK thread available from Cargo Systems Inc.

C. Cargo Net Webbing Repair

Note Material substitutions for the webbing and thread are not permitted unless approved by Cargo Systems Inc.

1) Loose Stitching

Stitch patterns in the webbing that exhibit broken threads may be sewn over with bonded nylon thread (available from Cargo Systems Inc.) using a minimum 4 - 6 stitches per inch per MIL–STD–751, Type 301, straight stitch. Replacement thread (P/N 29525–BLK) is available from Cargo Systems Inc.

2) Cut/Frayed Webbing

In cases where the webbing is completely cut through or when the yarn threads have been severed, the webbing may be repaired by splicing, using a piece of MIL-Spec webbing (same as existing webbing) available from Cargo Systems Inc. The webbing splice shall extend 6.0” past the tear and shall have a minimum of 5.0” of stitching (in a 4 point Double W pattern) joining the web splice to the webbing on each side of the cut or the tear. Use a minimum of 70 stitches per pattern on either side of the cut. Stitches shall be 4 - 6 stitches per inch.

Note There may be only one webbing repair splice per webbing member (one length of load bearing webbing).

D. Hardware

1) Other Attachment Hardware

Check all hardware for security and condition and replace with same, if required.

E. STA 332 Modified Bulkhead

Repairs to the STA 332 bulkhead and vertical beam are NOT permitted unless approved by Viking Air Limited.

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Emergency – General Data

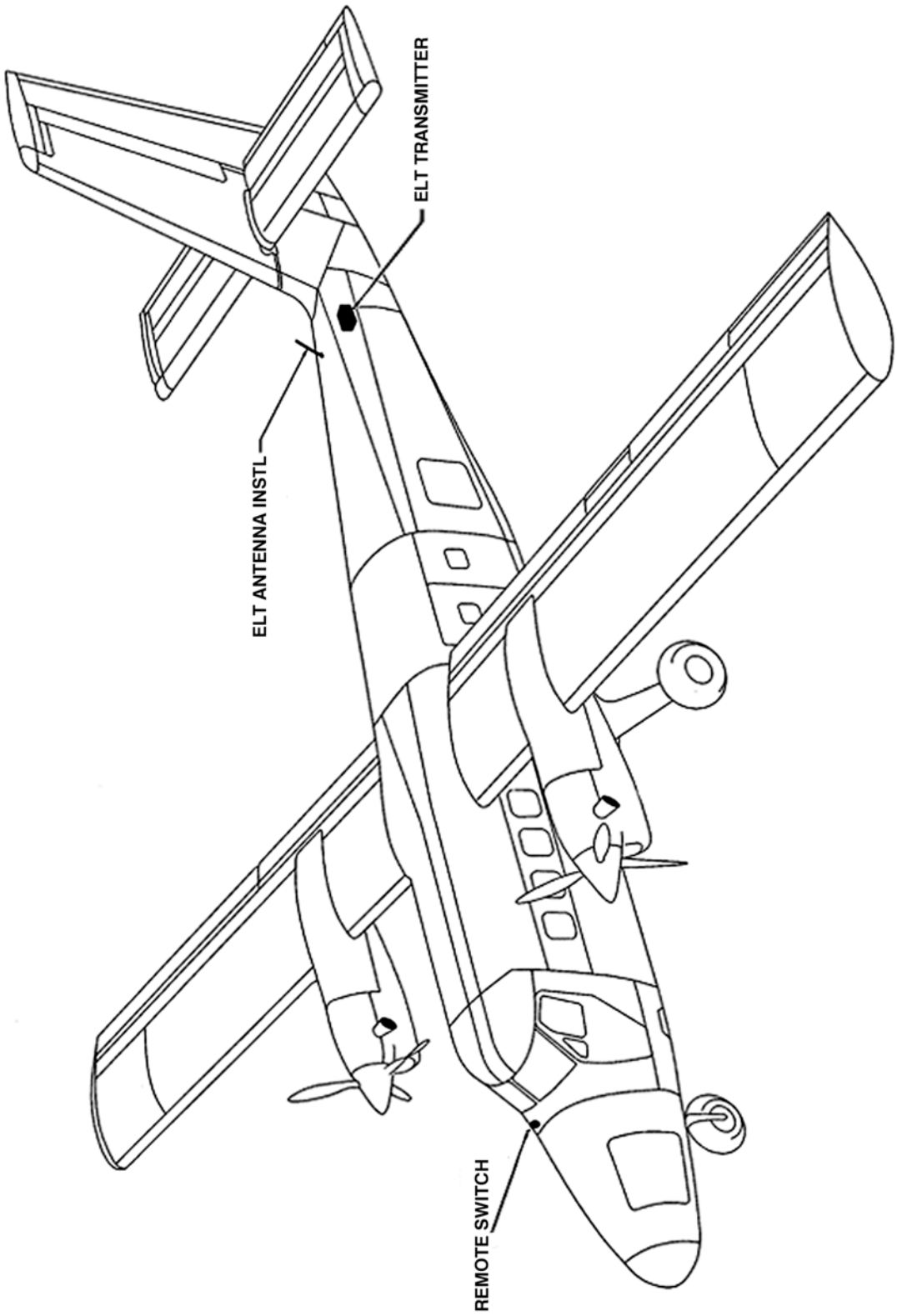
1. General

The emergency equipment consists of an emergency locator transmitter with an antenna installed (refer to 25-60-11, Emergency Locator Transmitter (ELT) – General Data) and a first aid kit, stowed on the right hand cabin side of bulkhead station 111.00.

Provisions have been made to mount a portable emergency locator transmitter (Mod S.O.O. 6270) under the co-pilot's seat in the flight compartment. (Refer to 25-10-00, Flight Compartment – General Data).

Provisions have been made for one or two stretcher rack configurations (Mod S.O.O. 6208) to be installed in the cabin compartment. (Refer to 25-60-20, Stretcher Racks (S.O.O. 6208) – General Data).

Provisions have been made to mount an axe in the flight compartment (Mod S.O.O. 6376) on the cockpit sidewall, adjacent to the pilot’s seat. (Refer to 25-60-40, Flight Compartment Axe and Axe Stowage (S.O.O. 6376) – General Data).



ELT Antenna and Transmitter
Figure 1

Emergency Locator Transmitter (ELT) – General Data

1. General

The Emergency Locator Transmitter (ELT) is used to locate an aircraft in the event of an accident.

2. Description

Refer to Figure 1 for details.

A. Artex C406–N Series ELT

An Emergency Locator Transmitter (ELT) is a self contained unit operated by an integral sealed battery pack. The ELT is installed in the aircraft on the lower avionics shelf between station 406.00 and 421.00. The ELT is labelled with an arrow marked DIRECTION OF FLIGHT, to ensure correct installation direction for operation of G-switch in the event of impact.

In the event of impact, the G-switch on sensing deceleration of 4.5 fps ± 0.5 fps with a duration of 11 ± 5 milliseconds will operate within the transmitter and activate its signal. The ELT is connected to an antenna installed on the top fuselage just forward of the fin. The ELT will transmit continually until the G-switch is deactivated or the battery is exhausted. The ELT transmits the standard swept tone on 121.5 and 243.0 MHz. Every 50 seconds for 520 milliseconds (long message protocol) the 406.025 MHz transmitter turns on. During that time an encoded digital message is sent to the COPAS-SARSAT satellite.

The navigation portion of the ELT continually receives GPS derived aircraft position information from the APEX system through the modular avionics unit. If the ELT is activated, the ELT will transmit the last recorded latitude and longitude of the aircraft in addition to data identifying the aircraft.

The information contained in the digital message includes: Serial Number of the Transmitter or Aircraft ID, Country Code, I.D. Code and Position Coordinates (Country specific ELT registration requirements may vary).

When mounted in the aircraft and connected to the electrical connector, the transmitter is armed whenever the ELT switch is selected to OFF, and the cockpit remote switch is selected to ARM.

Note Should the transmitter be activated inadvertently, set the ELT or cockpit switch to the ON position, then back to the OFF/ARM position. The ELT will be deactivated and the G-switch reset. If either switch is in the ON position, the other switch will not reset the ELT.

An audio buzzer is installed adjacent to the ELT. The buzzer sounds whenever the ELT is activated.

The buzzer is powered by the ELT unit and therefore is not dependent upon the aircraft battery for operation. It is not designed to operate continuously but sounds at predetermined intervals, and runs for shorter periods towards the end of battery life.

28 Volt DC power is provided from the left bus through a 1 amp ELT NAV (K5) circuit breaker on the avionics circuit breaker panel refer to 24-00-00, Electrical Power – General Data.

Note If the ELT Assembly is moved to a different aircraft, the ELT assembly must be given a new registration number. (Refer to manufacturer's component maintenance manual for more details).

B. Cockpit Control Panel

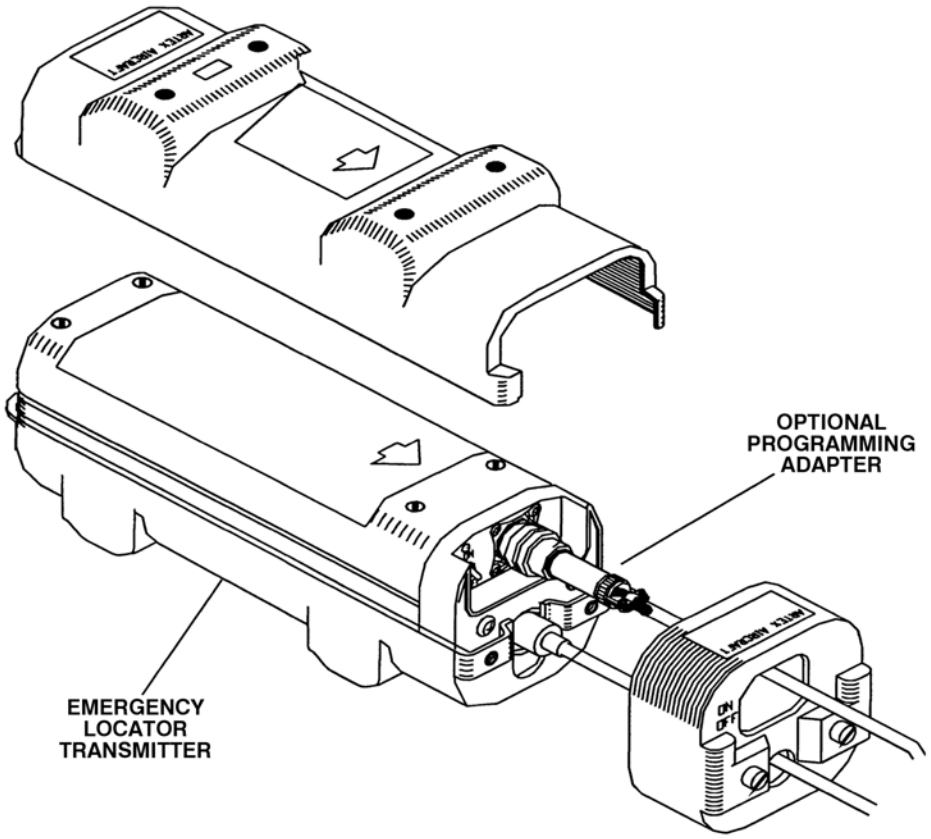
An ELT remote control panel is installed on the outboard side of the right hand sub-panel. This unit has a guarded switch with the positions ON and ARM. If the ELT is accidentally activated, either by tripping the G-switch or with the ELT switches, the indicator light will flash and ELT Active caution CAS message will be displayed on the APEX screen. The indicator light is also used during self test to verify proper operation of circuitry.

28 Volt DC power is provided from the hot bus through a 1 amp ELT SWITCH (U1) circuit breaker on the rear circuit breaker panel refer to 24-00-00, Electrical Power – General Data.

C. Programming Adapter

The Programming Adapter (PA) is an optional accessory for the C406–N series of ELT's. The PA is contained within the ELT's round connector backshell. The PA is approximately 2 inches long (50mm) and has a silver label affixed which records the programming information, refer to Figure 1.

For aircraft with the ELT programming adapter installed, if the ELT Assembly is replaced, the PA will transfer the information held within it to the replacement ELT assembly. (Refer to manufacturer's component maintenance manual for more details).



Emergency Locator Transmitter Assembly
Figure 1

3. Adjustment/Test

A. Transmitter Test

WARNING **DO NOT ALLOW TEST TO EXCEED 5 SECONDS IN DURATION. ACTIVATION OF THE ELT SIGNAL, OUTSIDE OF THE SPECIFIED TIME WINDOW AND LONGER THAN THE 5 SECOND DURATION IS CONSIDERED BY SATELLITE SYSTEMS TO BE A VALID TRANSMISSION SIGNAL.**

Note Always do a transmitter test within the first 5 minutes of the hour (UTC). Ensure you have notified the local control tower, local air traffic of your intentions to do a transmitter test.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Ensure that ELT NAV (K5) circuit breaker on the avionics circuit breaker panel and ELT SWITCH (U1) circuit breaker on the rear circuit breaker panel are closed.

4) Ensure the ELT switch on the ELT assembly is OFF.

5) Ensure the remote ELT switch, located on the co-pilot's sub panel, is set to ARM.

   Note Ensure the aircraft's navigational system is active at least 30 seconds prior to doing the ELT transmitter test.

6) Tune the one VHF transceiver to 121.5 MHz.

7) Select remote ELT switch to ON.

8) Confirm that the ELT buzzer adjacent to the ELT assembly sounds.

9) Wait approximately 1 second, or for 3 sweeps of the VHF receiver, then select the ELT remote switch to the ARM position, paying special attention to the LED when ARM is selected.

   Note A 406.028 MHz test signal will be transmitted during the reset cycle. It will be coded as a test signal and will not contain position data if viewed with a test set.

10) If the ELT is working properly, the LED will remain ON for approximately 1 second, then extinguish. If a problem is detected, it will be annunciated using a code following the initial pulse, refer to the troubleshooting section, Para 5. for more information.

11) Continue monitoring the VHF transceiver for a few seconds to verify ELT does not continue to transmit.

B. Electrical Bonding Test – ELT Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to empennage through rear baggage compartment.

3) Locate ELT Antenna aft of station 421.00 on the ceiling of the aircraft.

4) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the ELT Antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

5) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

6) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

4. Removal/Installation

WARNING INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND LIGHTNING. ENSURE INTEGRITY TO THE HIRF/LIGHTNING PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:

SHIELD GROUNDING – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

CAUTION **DO NOT USE CONTACT CLEANER ON ELT COMPONENTS. CHEMICAL CLEANERS CAN BE HIGHLY DESTRUCTIVE TO THE MOUNTING HARDWARE AND ELT HOUSING, AND MAY CAUSE CRACKING, FRACTURING AND BREAKAGE.**

A. ELT

1) Removal of ELT

a) Ensure the aircraft electrical system is OFF.

b) Open the ELT NAV (K5) circuit breaker and fit safety clip.

c) Gain access to the ELT, which is accomplished through the aft baggage compartment into the rear avionics compartment.

d) Loosen the (2) thumb screws which secure the mounting cap to end of ELT assembly.

e) Pull mounting cap away from unit.

f) Lift the top protective cover assembly off the unit.

g) Disconnect antenna coax connector and electrical connector.

h) Lift the connector end of the ELT up to disengage the locking ears at the aft end of the mounting tray. Lift the ELT out of the bottom protective cover.

Note Stow connectors as required.

2) Installation of ELT

CAUTION **IF PROGRAMMING ELT ASSY IS INSTALLED ON AIRCRAFT WHICH HAS A PROGRAMMING ADAPTER INSTALLED, THE DATA HELD IN THE PROGRAMMING ADAPTER WILL OVERWRITE ELT ASSY DATA.**

a) Ensure the aircraft electrical system is OFF.

b) Ensure the ELT NAV (K5) circuit breaker is opened and fitted with safety clip.

c) Reconnect antenna coax connector and electrical connector.

d) Fit the top protective cover assembly onto the unit.

e) Install mounting cap onto the front of unit.

f) Tighten the (2) thumb screws which secure the mounting cap to end of ELT assembly. Do not tighten over 18 in–lbs.

g) Close access panels as required.

h) Remove safety clip and close the ELT NAV (K5) circuit breaker.

i) Do a ELT transmitter test, refer to Para 3.A.

B. ELT Battery Pack

CAUTION **THE BATTERY PACK CONTAINS STATIC SENSITIVE PARTS, TAKE ESD PRECAUTIONS BEFORE HANDLING.**

1) Removal of ELT Battery

a) Remove the ELT, refer to Para 4.A.1).

b) Remove the (4) screws from the bottom of the ELT.

c) Lay ELT on side, carefully lift battery pack away from ELT unit, set it alongside the ELT.

d) Carefully disconnect the 8-pin connector on the small circuit board in the battery pack. Do not short connector pins.

e) Disconnect 2-pin harness from the ELT body.

f) Inspect battery pack for signs of damage, wear and corrosion including the battery cells, the pins, around the circuit board area and outside surface of ELT unit.

g) Inspect the integrity of the wires and connections. Ensure none are damaged or broken.

2) Installation of ELT Battery

a) Connect the 8-pin connector to the header on the circuit board in the battery pack.

b) Connect the 2-pin power harness to the ELT. This will cause the ELT to activate.

c) Reset the ELT by toggling the switch on the ELT from ON to OFF.

d) Lay wires away from the standoffs to avoid pinching wires between standoffs and battery pack.

e) Install (4) screws on the bottom of the ELT.

f) Install the ELT, refer to Para 4.A.2).

C. ELT Mounting Tray

1) Removal of ELT Mounting Tray

a) Remove the ELT, refer to Para 4.A.1).

b) Remove (4) screws and washers holding the mounting tray to the equipment shelf.

c) Remove the mounting tray from the aircraft.

2) Installation of ELT Mounting Tray

a) Place the ELT mounting tray on the equipment shelf.

b) Install (4) screws and washers.

c) Install the ELT, refer to Para 4.A.2).

D. ELT Antenna

1) Removal of ELT Antenna

a) Ensure the aircraft electrical system is OFF.

b) Open the ELT NAV (K5) circuit breaker and fit safety clip.

c) Gain access to the ELT, which is accomplished through the aft baggage compartment into the rear avionics compartment.

d) Place the ELT switch in the OFF position and remove ELT electrical connector to ensure the ELT is not transmitting.

e) Carefully cut the sealant from around the base of the antenna.

f) Remove the (4) screws from the base of the antenna.

g) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

h) Disconnect the coaxial connector.

i) Remove antenna from aircraft.

2) Installation of ELT Antenna

a) Ensure the aircraft electrical system is OFF.

b) Ensure the ELT NAV (K5) circuit breaker is opened and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) A layer of anti-corrosion bonding grease should be applied between the aircraft and the base of the antenna.

f) Secure the antenna assembly on fuselage with (4) mounting screws.

g) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

h) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

i) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching affected area.

j) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

k) Check the antenna electrical bonding as per Para 3.B.

l) Connect the electrical connector and ensure ELT is not transmitting.

m) Remove the safety clip and replace the ELT NAV (K5) circuit breaker in the avionics circuit breaker panel.

n) Do a ELT transmitter test, refer to Para 3.A.

5. Troubleshooting

A. Flashing LED Code

If a problem is detected within the operation of the ELT when the unit is reset, the LED on the remote switch, located on the co-pilot's sub panel, will display a code sequence of flashing lights. Please note that if there are more than one error type detected the coded LED sequence will be displayed in order of importance with approximately a 1 second pause between each coded sequence.

Coded LED Error sequence in order of importance:

1 flash – Indication of G-switch loop open failure.

3 flashes – Indication of a transmission failure of 406.028 Hz signal. May be caused by bad or unconnected coax, an antenna problem or low power output.

5 flashes – Indication of invalid data or no navigational data being received by the ELT. May be caused by improper wiring, improper programming, or invalid navigation data.

7 flashes – Indication of a battery problem. May be caused by battery usage time over an hour.

There is an error condition where the LED on the remote switch will cycle of 1 short and 1 long light signal. This occurs when power is applied to the ELT assembly, and the jumper between pins 3 and 4 is not installed.

B. ELT

1) Ensure ELT NAV (K5) and ELT SWITCH (U1) circuit breakers are closed.

2) Visually inspect ELT and ELT switch connectors for damaged wiring or contacts. Ensure electrical connector on ELT and ELT switch are fully seated and locked.

3) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

4) If repair or service is required, send to appropriately rated avionics repair facility.

6. Servicing

A. ELT Battery Pack

1) Change the battery pack on any of the following conditions:

a) After use in an emergency.

b) Inadvertent activation of unknown duration.

c) When the total of all known transmissions exceeds one hour.

d) On or before the battery replacement date entered on the case of the transmitter (beacon).

e) There is any evidence of corrosion or leakage of any cell or on the small interface board and connector.

Note Additional maintenance and inspection requirements for the installation of the ELT may be mandated by the regulations of the jurisdiction under which the aircraft is registered. In the event of disagreement between PSM 1-64-2 and these additional requirements, the more restrictive requirements shall take precedence unless an alternative program of maintenance has been approved by the regulatory authority.

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Stretcher Racks (S.O.O. 6208) – General Data

1. General

Provisions have been made to accommodate one stretcher (with double and single seat assemblies, rows 3 through 6 removed) or two stretchers (with double and single seat assemblies, rows 1 through 6 removed), that can be installed on the right hand side of the aircraft fuselage. There are two support assemblies, per stretcher, that can be fitted into the existing floor tracks in the cabin compartment. Each set of support assemblies provide keepers and latches to secure the stretcher for transport.

2. Removal/Installation

A. Removal of Stretcher

Refer to Figure 1 for details.

1) Unlatch the pull action clamps at the forward and aft inboard keepers (undo (4)).

2) Open the forward and aft inboard keepers (undo (3)).

3) Lift the inboard side of the stretcher frame out of the forward and aft inboard keepers (undo (2)).

4) Slide the outboard side of the stretcher frame out of the outboard forward and aft keepers (undo (1)).

5) Remove stretcher from aircraft.

B. Installation of Stretcher

Refer to Figure 1 and Figure 2 for details.

1) Load stretcher onto aircraft. The reclining headboard must be facing forward.

2) Place the outboard side of stretcher frame into the forward and aft outboard keepers ((1)).

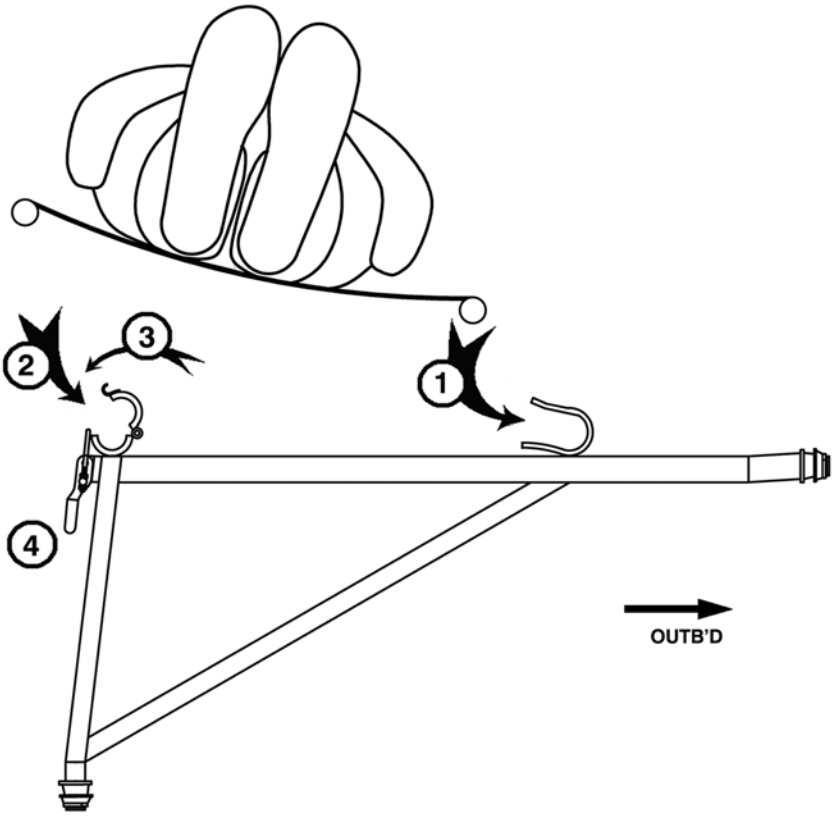
3) Lower the inboard side of the stretcher frame into the forward and aft inboard keepers ((2)).

Note Ensure the aft legs of the stretcher bear against the inboard and outboard keepers.

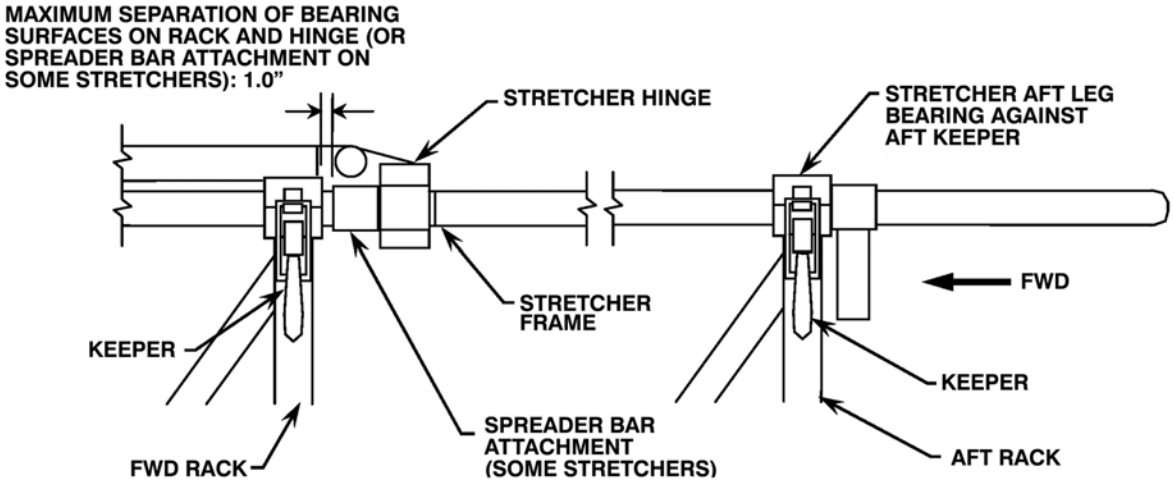
4) Close the forward and aft keepers over the stretcher frame ((3)).

5) Latch the forward and aft keepers to secure the stretcher in place, using the pull action latch clamp ((4)).

6) Ensure that the stretcher is securely attached. Adjust latch clamp as required, per Para 3.A.



Loading and Securing Stretcher
Figure 1



Stretcher Installation
Figure 2

C. Removal of Support Racks

Refer to Figure 3 and Figure 4 for details.

1) Remove the stretcher as per Para 2.A.

2) Pull up and rotate the three spring loaded lock collars on the stretcher support assembly.

3) Slide the stretcher support assembly forward or aft, until the outboard side and the leg supports are clear of the seat retaining tracks.

4) Install seating as per 25-20-01 – Cabin Compartment – Seating Arrangement, Para 3.

D. Installation of Support Racks

Refer to Figure 3 and Figure 4 for details.

Note There are two (2) Stretcher Support Assemblies per stretcher installation.

1) Remove the seating; rows 3 through 6 for a single stretcher configuration, or rows 1 through 6 for a double stretcher configuration, on the right hand side of aircraft, as per 25-20-01 – Cabin Compartment – Seating Arrangement, Para 3.

2) Pull up on the three spring loaded lock collars at the seat track attachments.

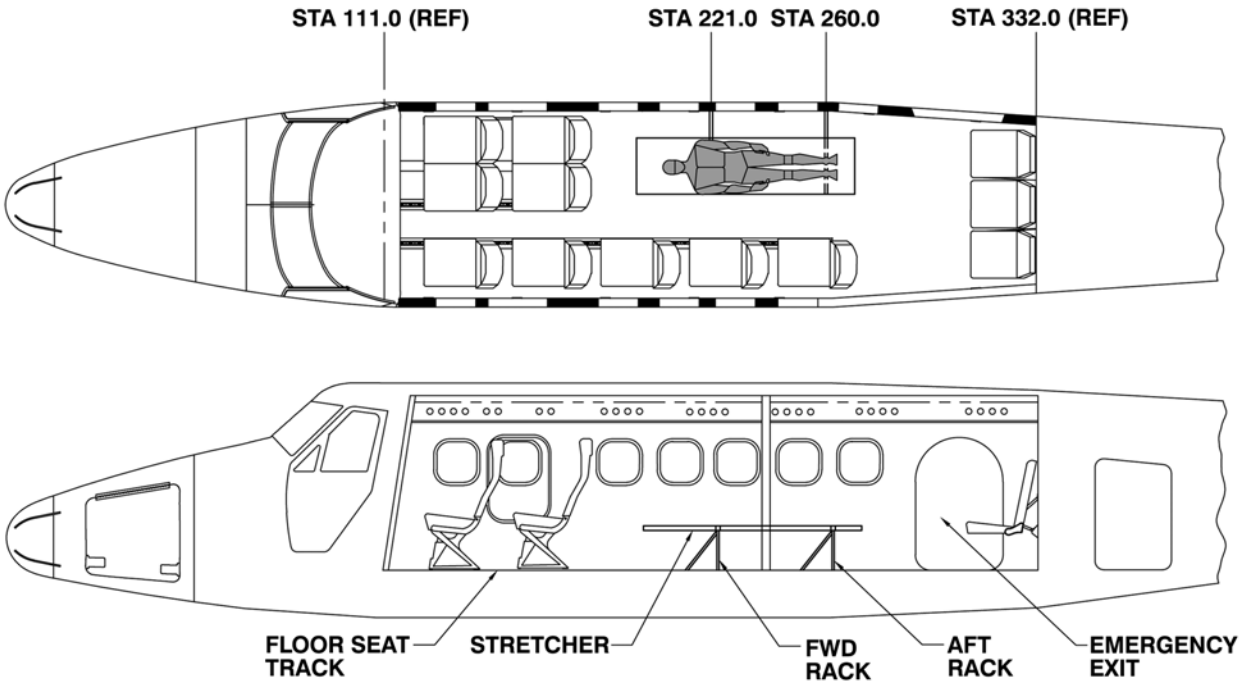
3) Slide the Stretcher Support Assemblies into the side and floor seat tracks, in the correct locations.

Note The stretcher support assemblies are installed at side rail attachment points.

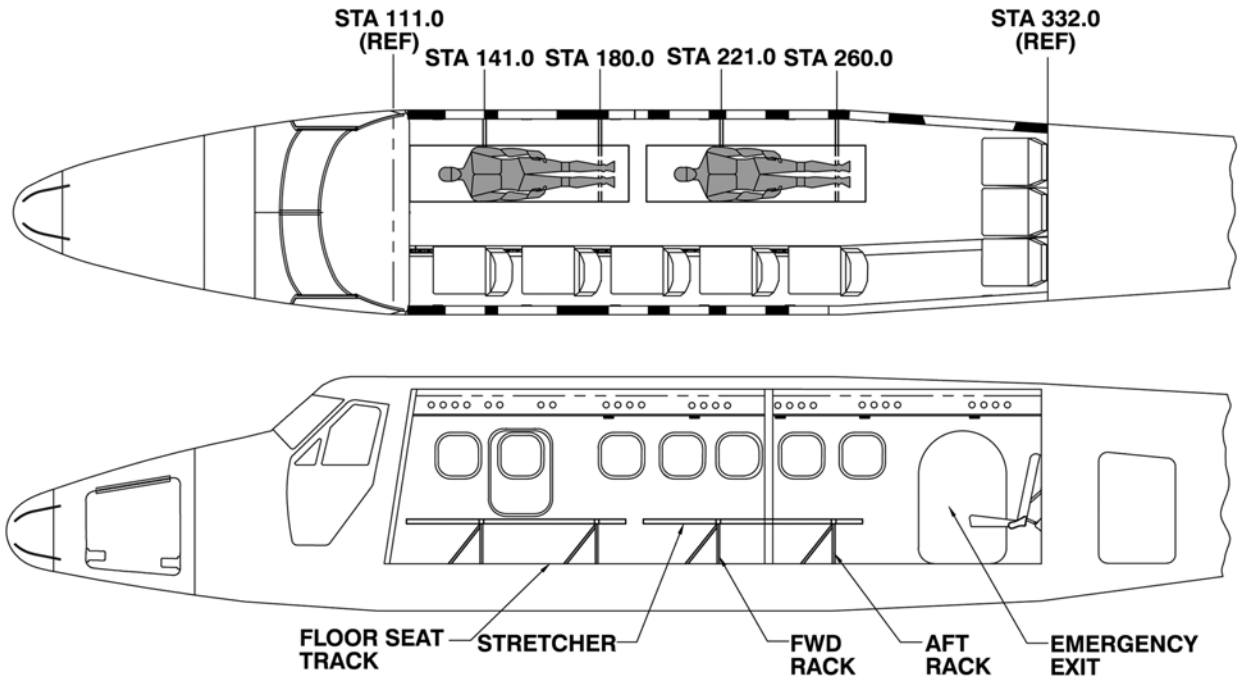
- Aft STA 221.0 and 260.0
- Forward STA 141.0 and 180.0

4) Rotate the forward, aft leg and outboard side lock collars to engage the Stretcher Support Assembly in the side and floor tracks.

5) Ensure the Stretcher Support Assembly is secure.



Single Stretcher Configuration
Figure 3



Double Stretcher Configuration
Figure 4

E. Removal of Passenger Restraint

Refer to Figure 5 for details.

1) Unhook all strap latches.

2) Unloop shoulder straps from around the stretcher frame, aft of the wheel attachment.

3) Unloop chest straps from around stretcher frame, aft of the headboard hinges.

4) Unloop lap belt from around stretcher frame, at the center fabric cut-out.

5) Unloop leg belt from around the stretcher frame, at the aft fabric cut-out or foot-end.

F. Installation of Passenger Restraint

Refer to Figure 5 for details.

1) The reclining headboard must be facing forward.

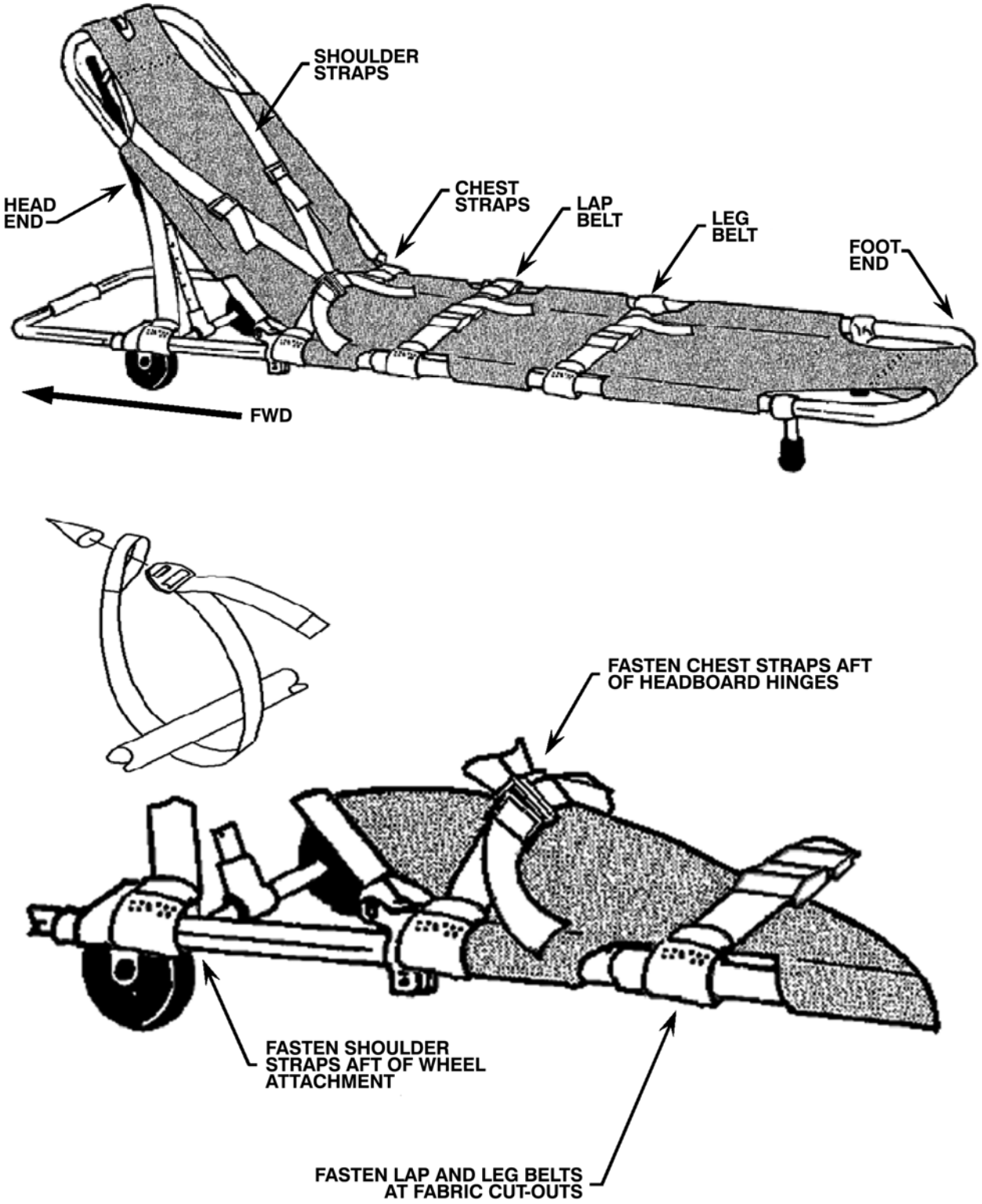
2) The shoulder straps must be looped around the stretcher frame and secured immediately aft of the wheel attachment.

3) The chest straps must be looped around the stretcher frame and secured immediately aft of the headboard hinges.

4) The lap belt must be looped around the stretcher frame and secured in the center fabric cut-out.

5) The leg belt must be looped around the stretcher frame and secured in the aft fabric cut-out or at the foot-end.

6) Connect all strap latches and tighten straps.



Patient Restraint Attachment
Figure 5

3. Adjustment/Test

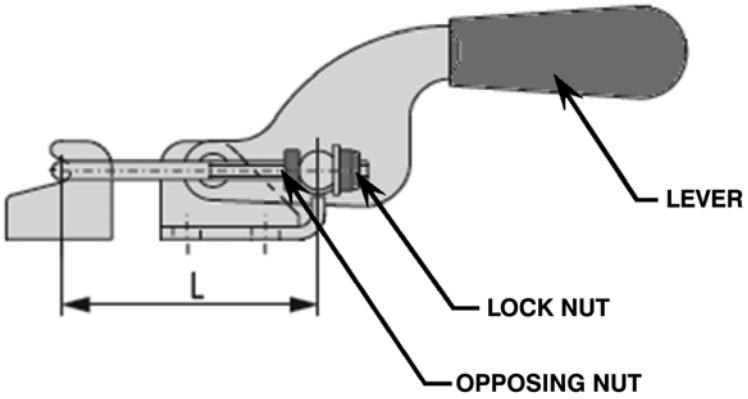
A. Pull Action Latch Clamp Adjustment

1) Ensure secure locking of clamp by adjusting clamp into the corresponding overcenter toggle position.

2) Adjust length 'L' (refer to Figure 6) of the U-shaped hook, so that 2 to 3 lbs of force are necessary to move the lever into the overcenter position.

3) Length 'L' can be adjusted by loosening or tightening the locknuts on both sides of the lever.

4) After the locknuts have been adjusted, tighten (snug + 1/4 turn) the opposing nuts to prevent rattling.



Pull Action Latch Clamp Adjustment
Figure 6

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Life Vest Provisions – General Data

1. General

On aircraft with Mod 6/2281 and Mod 6/2282 installed, provisions have been made for the installation of life vest pouches to be attached to the bottom of the passenger seats.

For aircraft incorporating S.O.O. 6309, provisions have been made for the installation of English life vest pouches and placards.

For aircraft incorporating S.O.O. 6352, provisions have been made for the installation of Chinese/English life vest pouches and placards.

2. Description

Life vest pouches are attached to the bottom of the passenger seats with four (4) 'pull the dot snaps' and a hook-loop flap that attaches to the leading edge of the seat frame. The pouch and strap are designed for a maximum life vest size of 7 x 7 x 2 inches, and a maximum packaged weight of 2.0 pounds.

Only approved life vests, TSO–C13e or later Revision, are to be used with S.O.O. 6309 or S.O.O. 6352. If approved life vests are not stowed in life vest pouch, the 'LIFE VEST UNDER SEAT' placard must be removed.

3. Removal/Installation

A. Removal of Life Vest Pouch from Station 332 Seats

1) Ensure seat bottom is in the 'tip-up' position.

2) Pull on canvas of pouch surrounding the snap; pull from 'dot' side and away from seat bottom.

Note Use caution to ensure the seat bottom is not damaged.

3) Inspect condition of seat bottom, snaps and life vest pouch.

4) Replace any damaged components.

B. Installation of Life Vest Pouch on Station 332 Seats

1) Ensure seat bottom is in 'tip-up' position.

2) With pouch flap positioned toward the forward edge of the seat, engage four (4) pouch snap halves with seat bottom snap halves.

3) Mount life vest to the life vest harness/pull strap, and place into pouch with red pull strap protruding from the pouch.

4) Engage harness strap snap with seat snap halve on the forward edge of seat bottom.

5) Engage pouch hook fastener to seat loop on the forward edge of seat bottom.

C. Removal of Life Vest Pouch from Non-Bulkhead Seats

1) Locate life vest pouch on the bottom of seat.

2) Detach pouch by pulling on the canvas of pouch surrounding the snap; pull from 'dot' side of snap and away from seat bottom.

Note Use caution to ensure the seat bottom is not damaged.

3) Inspect condition of seat bottom, snaps and life vest pouch.

4) Replace any damaged components.

D. Installation of Life Vest Pouch on Non-Bulkhead Seats

1) Locate four (4) snap halves located on the bottom of each passenger seat.

2) With pouch flap positioned towards forward edge of the seat, engage four (4) pouch snap halves with seat bottom snap halves.

3) Mount life vest to the life vest harness/pull strap, and place into pouch with red pull strap protruding from the pouch.

4) Engage harness strap snap with seat snap halve on the forward edge of seat bottom.

5) Engage pouch hook fastener to seat loop on the forward edge of seat bottom.

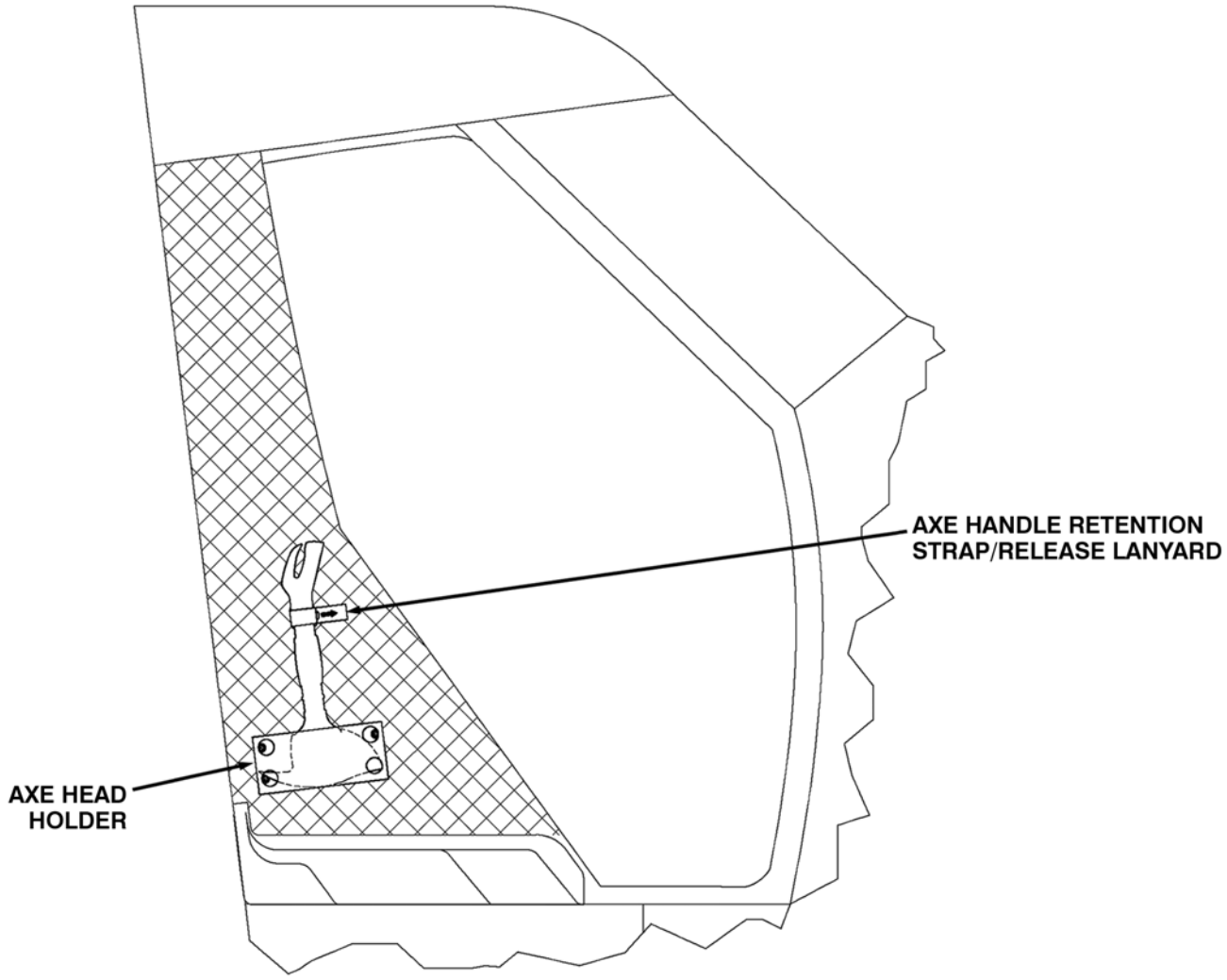
Flight Compartment Axe and Axe Stowage (S.O.O. 6376) – General Data

1. General

For aircraft incorporating S.O.O. 6376, provisions have been made for stowing an axe in the flight compartment.

2. Description

The axe and axe stowage brackets are installed in the flight compartment on the cockpit sidewall, adjacent to the pilot's seat. The axe stowage structure consists of a metal holder which encases and supports the axe head, and a metal bracket with a nylon retention strap which retains the axe handle. The axe is removed by firmly pulling on the retention strap lanyard.



Flight Compartment Axe (S.O.O. 6376)
Figure 1

3. Removal/Installation

A. Removal of Axe and Axe Stowage Brackets

1) Remove axe by firmly pulling on the axe handle retention/release strap.

2) Carefully lift axe out of holder. Inspect the cutting edge and pike for signs of corrosion and loosening.

3) Remove the (4) screws securing the axe head holder to the cockpit sidewall. Retain all hardware.

4) Remove the axe head holder from the aircraft.

5) Remove the (2) screws and (2) spacers securing the axe handle bracket to the cockpit sidewall. Retain all the hardware.

6) Inspect the nylon retention strap/lanyard for signs of wear or deterioration of elastic and nylon webbing. Replace if worn or damaged.

7) Remove the axe handle bracket from the aircraft.

B. Installation of Axe and Axe Stowage Brackets

1) Install (4) screws which secure the axe head holder to the cockpit sidewall.

2) Install (2) screws and (2) spacers which secure the axe handle bracket to the cockpit sidewall.

3) Install axe head into axe head holder.

4) Secure the axe handle to axe handle bracket, using the retention nylon strap. A firm, audible click or snap will indicate the strap is properly fastened.

5) Ensure that the axe can be released when lanyard is firmly pulled.

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Cold Weather Operations Kit (S.O.O. 6276 or S.O.O. 6287) – General data

1. General

The Cold Weather Operations Kit 115 VAC (S.O.O. 6287) or 230 VAC (S.O.O. 6276) is used to heat the engines, engine oil, Primary Flight Displays (PFDs), and the batteries. The system reduces thermal stress, warm-up times, and increases the reliability of safety operations in cold environments.

2. Description

Cold Weather Operations are facilitated by incorporating a series of heating elements onto engines, oil reservoirs, PFDs and batteries for use when the aircraft is on ground. Power is provided through a power source and shore power line supplied by the operator to the system, which is separated from, and independent of aircraft systems. Additionally, provisions are provided for an outlet to power a portable heater inside the cockpit. The system is not operational during flight.

An option for either 115 VAC (S.O.O. 6287) or 230 VAC (S.O.O. 6276) is provided depending on the requirements of the operator. For 230 VAC operations, adaptor plugs are provided to replace the ends of power cords used for the system. Power cords are to be supplied by the operators.

System heat is provided using electrical resistance heating elements in the form of thin pads, sized for the various locations. Heating elements are bonded to the engine, below the PFDs and to the bottom of the emergency lighting battery (when equipped), with a high temperature fluorosilicone adhesive. The main battery is heated using a battery blanket wrapped around the battery. Both battery and PFD elements are controlled using a thermostatic switch set to turn on heating below 32°F (0°C) and turn off heating once temperatures have risen back above 32°F (0°C), when the system is energized. All other elements are actively heating whenever the system is energized.

Power is routed to the elements through a dedicated wiring harness, routed alongside aircraft power wiring and stood off using nylon d-blocks. Power enters the system through a dedicated ground power port located in the hydraulic bay access door. System activation is identified by LED lights mounted above the door for easy visibility.

3. Materials and Equipment

A. Materials

- DC 730, Dow Corning High temperature fluorosilicone adhesive/sealant
- Industrial solvent
  (For cleaning gluing surfaces prior to application of fresh adhesive, refer to 20-40-01, Silicone Adhesives – General Data, Table 1 – Solvent Selection for General Materials, or MEK, or CB911, Click Bond™ Solvent Wipes)
  (For removing old adhesive use Acetone or DOW CORNING® DS–2025 Silicone Cleaning Solvent)
- Securing Material/Fixture, for element placement during cure
- Optional primer: DOW CORNING® DC 1200 Prime Coat, adhesion promoter.
  (Not required when using the CB911 Solvent Wipe, though suggested in rough service application)
- Abrasive cleaning pad (For example, Scotch Brite™, green or maroon)

B. Equipment

- Clean, lint free rags or cloth conforming to AMS 3819
- Plastic Scraper (putty knife)
- Calibrated multi-meter

4. Requirements

A. Element Location

1) The installation site is to be smooth and clear of obstructions.

2) The mounting site and the area immediately adjacent to the pad element perimeter must be clean in order to allow for proper sealant and tape adhesion, and/or fixture attachment.

3) Pre-fitting trial installation without sealant is suggested for all installations. This should include the fitting of fixtures for first time and complex installations.

4) Consider wire lead orientation, system installation instruction, and cable routing.

5) Tape or custom formed fixture(s) must be capable of maintaining pressure on the pad element throughout the cure process.

B. Adhesive/Sealant Cure Process

WARNING **DO NOT TOUCH ANY OF THE ELEMENTS WHILE THE SYSTEM IS IN OPERATION AS ELEMENTS MAY BURN EXPOSED SKIN.**

CAUTION **DO NOT ATTEMPT TO CURE THE BONDING AGENT BY APPLYING POWER TO THE ELEMENT OR THE PREHEAT SYSTEM.**

1) A minimum undisturbed full cure time of 8 hours is recommended in standard conditions of 21°C (70°F) with humidity level of 50% before the removal of tape, installation blocks, or other securing devices.

2) Cure time is dependant on a number of variables such as thickness of the bonding agent, temperature, and humidity levels. Due to the many variables under standard conditions it is possible to return to service with a minimum undisturbed initial cure time of 2 hours with the completion of the final cure process under engine operating temperatures.

3) If the short initial cure is used, removal of securing materials must be done in a manner that avoids disturbing the heat element and sealant.

4) The processing temperature (cure temp) for DC 730 is between 0°C (32°F) and 80°C (175°F). The operating temperature of the aircraft engine can complete the cure of the adhesive/sealant as long as the above value 80°C (175°F) is not exceeded.

5. Surface Preparation

Surfaces can vary greatly. This can affect sealant adhesion. It is recommended that adhesion characteristics to a specific surface be determined prior to application on parts or assemblies.

A. Cleaning

1) A progressive cleaning procedure should be employed using appropriate solvents (refer to Para 3.A) and a new lint free cloth conforming to AMS 3819. Contaminants such as dirt, grease, and/or processing lubricants must be removed prior to adhesive/sealant application. Always pour solvent on the cloth to avoid contaminating the solvent supply. Wash one small area at a time.

a) Clean surface with solvent, using rolling motion to remove surface contamination. Inspect cloth and repeat if contamination on wiping cloth is visible. Repeat using clean solvent wipe until solvent wiping cloth remains clean upon inspection.

b) Immediately after solvent wipe, use second clean dry cloth and wipe surface using rolling motion to remove residual solvent prior to the solvent evaporating. This is done to prevent redepositing contaminants on the surface. Do not allow solvent to evaporate.

c) Once surface is clean, abrade using a non-metallic abrasive cleaning pad and solvent.

d) After abrading, outline where the pad is to be installed.

e) Aggressively wipe down the installation site and the bonding side of pad element, with the solvent, refer to Para 3.A, followed immediately by dry wipe.

6. Removal/Installation

Refer to Figure 1 and Figure 2 for details.

A. Removal of Heating Pads

CAUTION **DO NOT USE ANY OTHER BONDING AGENTS, SILICONE OR RTV. TO AVOID DAMAGE TO THE SURFACE OR FINISH OF THE INSTALLATION SITE, USE APPROPRIATE SOLVENT. REFER TO Para 3.A FOR SOLVENT SELECTION.**

1) Engine Heating Pads

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the engines by opening the lower cowls.

c) Locate the heating pad and remove it with a non-marring scraper that will not damage the surface or the finish that the heating pad is bonded to.

d) Use the scraper to lift and release a corner of the heating pad. Grip the released corner firmly, and slowly pull the heating pad away from the mounted surface.

e) Use the scraper to separate the heating pad from the surface where required.

f) Use of a solvent during the removal process may ease removal of the pad. Refer to Para 3.A.

g) Disconnect the heating pad's wire from the junction block. Pay special attention to the wire routing and attachment details for heating pad reinstallation.

h) Clean the surface with appropriate solvent and scraper. Refer to Para 3.A.

Note It is not recommended that removed elements be reused.

2) Emergency Battery Heating Pad

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the battery bay by removing the floor panel. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram).

c) Disconnect the heating pad's connector from the electrical system. Pay special attention to the wire routing and attachment details for heating pad reinstallation.

d) Remove the battery. (Refer to 33-50-01, Emergency Lighting Battery – General Data).

e) Locate and remove the heating pad with non-marring scraper that will not damage the surface or the finish that the heating pad is bonded to.

f) Use the scraper to lift and release a corner of the heating pad.

g) Grip the released corner firmly, and slowly pull the heating pad away from the mounted surface.

h) Use the scraper to separate the heating pad from the surface where required.

i) Use of solvent during the removal process may ease removal of the pad. Refer to Para 3.A.

j) Clean the surface with an appropriate solvent and tool.

k) Reinstall the battery. (Refer to 33-50-01, Emergency Lighting Battery – General Data).

Note It is not recommended that removed elements be reused.

3) Main Battery Heating Belt

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the battery bay by removing the floor panel. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram).

c) Disconnect the heating belt's connector from the electrical system. Pay special attention to the wire routing and attachment details for heating pad reinstallation.

d) Remove the battery. (Refer to 24-30-31, Main Battery System – Maintenance Practices).

e) Cut the cable ties that secure the battery belt to the battery.

f) Remove the battery heating belt.

g) Reinstall the battery. (Refer to 24-30-31, Main Battery System – Maintenance Practices).

4) Primary Flight Display Heating Pads

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove the pilot and co-pilot's PFDs, refer to 31-11-00, Primary Flight Display (PFD), Para 4.

c) Disconnect the heating pad's connector from the electrical system. Pay special attention to the wire routing and attachment details for heating pad reinstallation.

d) Locate the heating pad and remove it with a non-marring scraper that will not damage the surface or the finish that the heating pad is bonded to.

e) Use the scraper to lift and release a corner of the heating pad.

f) Grip the released corner firmly, and slowly pull the heating pad away from the mounted surface.

g) Use the scraper to separate the heating pad from the surface where required.

h) Use of solvent during the removal process may ease removal of the pad. Refer to Para 3.A.

i) Clean the surface with an appropriate solvent and tool.

Note It is not recommended that removed elements be reused.

B. Installation of Heating Pads

CAUTION DO NOT USE ANY OTHER BONDING AGENTS, SILICONE OR RTV. TO AVOID DAMAGE TO THE SURFACE OR FINISH OF THE INSTALLATION SITE, USE APPROPRIATE SOLVENT. REFER TO Para 3.A FOR SOLVENT SELECTION.

CAUTION DO NOT COAT THE TOP SURFACE OF THE PAD ELEMENT WITH ADHESIVE/SEALANT.

1) Engine Heating Pads

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to both engines by opening the lower cowls.

c) Locate and clean each installation site with an appropriate solvent and tool as per Para 5.A. Refer to Para 3.A..

d) Refer to Table 1 — Heating Pad Lead Direction for lead positioning.

e) Confirm resistance values. Refer to Table 2 — 115V System Values (±7%) and Table 3 — 230V System Values (±7%).

f) Spread DC730 Flourosilicone adhesive into a thin layer between 0.020 - 0.032 inches thick to the heating pad contact surface (flat, unmarked side).

Note The heating pad must be in complete bonded contact with the mounting surface through the adhesive/sealant. Cavities in the installation site directly under the heating pad must be filled with adhesive/sealant in order to provide proper thermal transfer.

g) Place the heating pad in location and bond, using a roller or plastic putty applicator.

h) Starting at the center of the pad, draw the roller across the surface of the pad to the outer edge, expelling air bubbles, cavities, and imperfections. When the proper amount of sealant is used, very little sealant will be expelled from under the pad.

i) When cleaning excess sealant after rolling, use extra care, and sealant, as required to seal the edges of the pad.

j) Use tooling (vacuum bags, tape, foam, straps, etc.) to ensure that constant and even pressure is applied to the pad during curing.

k) Once the bonding agent has cured (refer to Para 5.A) remove bag, tape and/or fixtures.

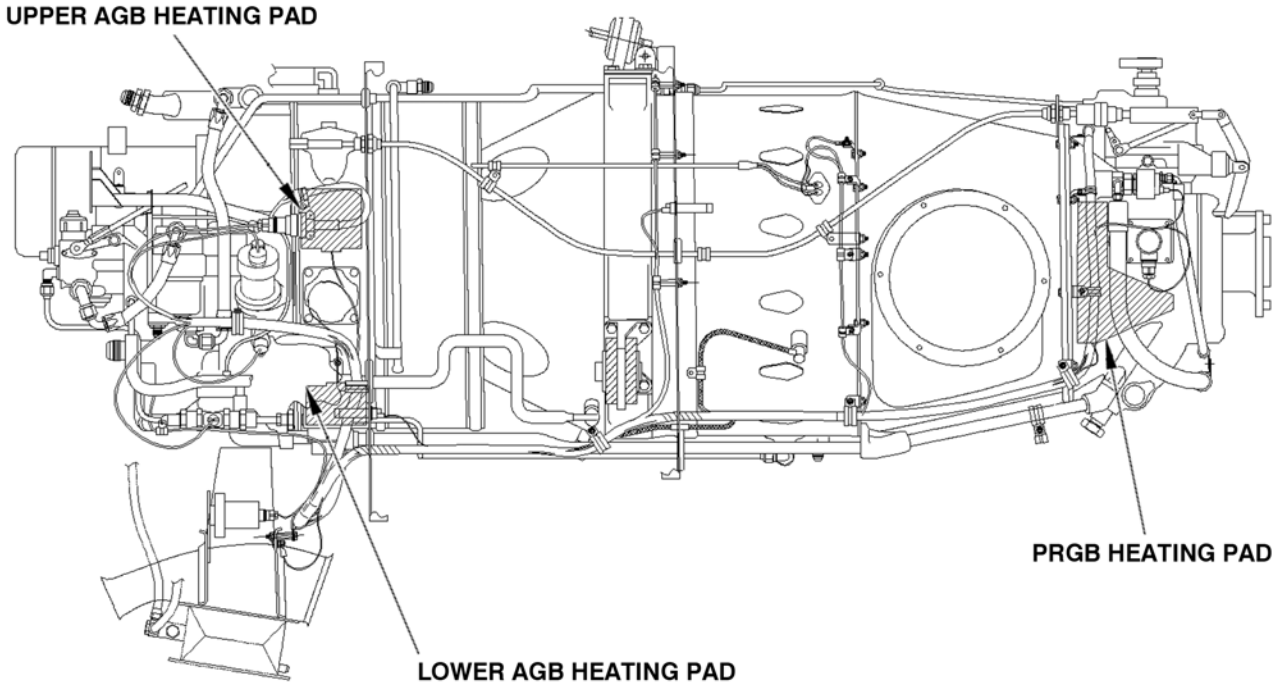
l) Apply a dollop of the adhesive/sealant under the base of the element lead (where lead enters element). This reduces wire fatigue and guards against the incursion of fluids at this location.

m) Seal the perimeter of the element with adhesive/sealant. Use just enough of the sealant to create a smooth transition along the edge of the element to mounting surface. This transition is not to extend beyond one quarter of an inch (.25”) from the element and not more than one eighth of an inch (0.125”) onto the elements upper surface.

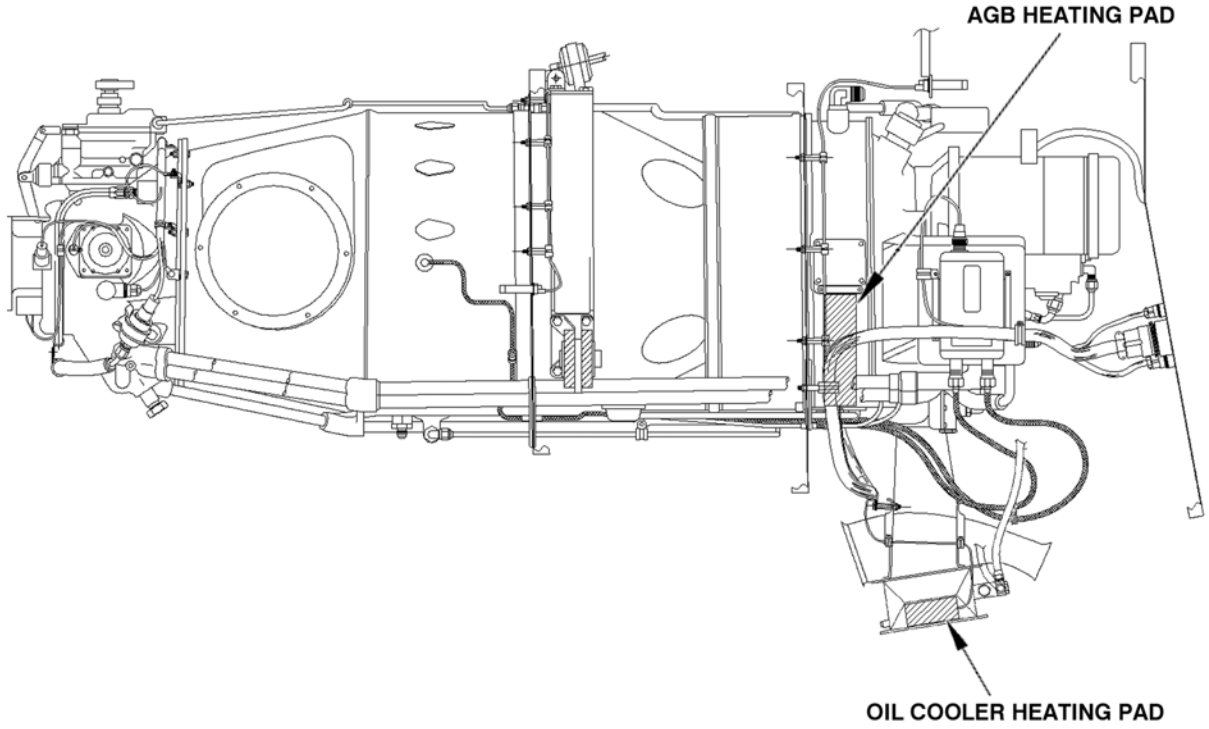
n) Refer to PSM 1-64-2W, 25-70-00, Cold Weather Operations Kit (S.O.O.s 6287 or 6276) for wire routing and terminations. If reinstalling a heating pad, route wires to match those previously removed in Para 6.A.4).

Table 1 Heating Pad Lead Direction

Heating Pad	Lead Direction
Left Side AGB	Position heating pad with the lead down
Right Side AGB, Upper Pad	Position heating pad with the lead down
Right Side AGB, Lower Pad	Position heating pad with the lead up
Right Side PRGB	Position heating pad with the lead down
Left Side of Engine Oil Cooler	Position heating pad with the lead aft



Right Hand Side of Engine
Figure 1



Left Hand Side of Engine
Figure 2

2) Emergency Lighting Battery Heating Pad (If Installed)

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the battery bay by removing the floor panel. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram).

c) Remove the battery. (Refer to 33-50-01, Emergency Lighting Battery – General Data) (if installed).

d) Clean the installation site with an appropriate solvent and non-marring scraper as per Para 5.A. Refer to Para 3.A for materials and equipment.

e) Confirm resistance values. Refer to Table 2 — 115V System Values (±7%) and Table 3 — 230V System Values (±7%).

f) Position the lead in the direction of existing wiring, refer to Figure 3.

g) Spread DC730 Flourosilicone adhesive into a thin layer between 0.020 - 0.032 inches thick to the heating pad contact surface (flat, unmarked side).

Note The heating pad must be in complete bonded contact with the mounting surface through the adhesive/sealant. Cavities in the installation site directly under the heating pad must be filled with adhesive/sealant in order to provide proper thermal transfer.

h) Place the heating pad in location and bond, using a roller or plastic putty applicator.

i) Starting at the center of the pad, draw the roller across the surface of the pad to the outer edge, expelling air bubbles, cavities, and imperfections. When the proper amount of sealant is used, very little sealant will be expelled from under the pad.

j) When cleaning excess sealant after rolling, use extra care, and sealant, as required to seal the edges of the pad.

k) Use tooling (vacuum bags, tape, foam, straps, etc.) to ensure that constant and even pressure is applied to the pad during curing.

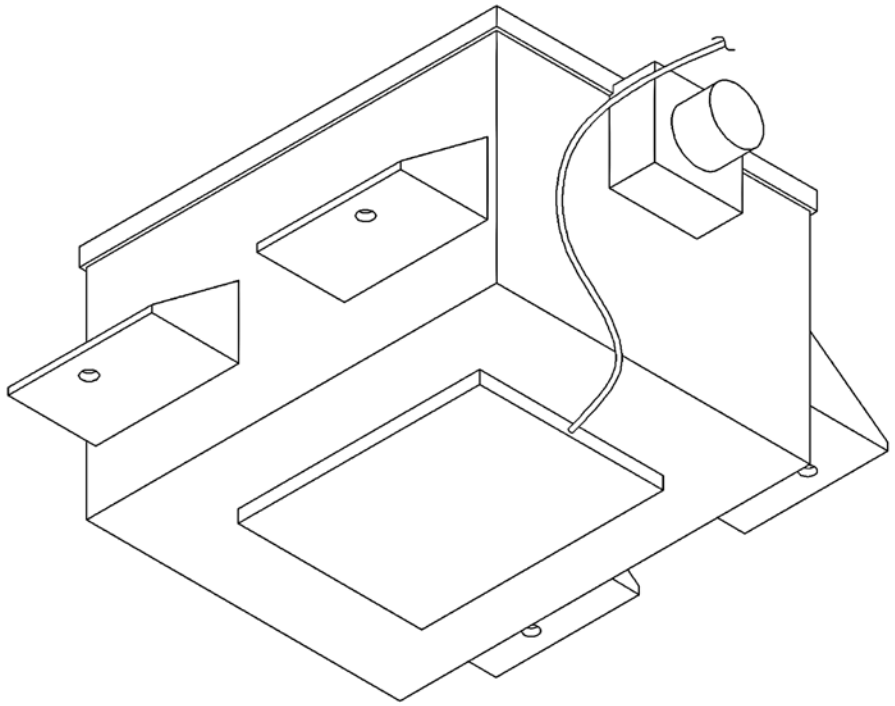
l) Once the bonding agent has cured (refer to Para 5.A) remove bag, tape and/or fixtures.

m) Apply a dollop of the adhesive/sealant under the base of the element lead (where lead enters element). This reduces wire fatigue and guards against the incursion of fluids at this location.

n) Seal the perimeter of the element with adhesive/sealant. Use just enough of the sealant to create a smooth transition along the edge of the element to mounting surface. This transition is not to extend beyond one quarter of an inch (.25”) from the element and not more than one eighth of an inch (0.125”) onto the elements upper surface.

o) Install the battery. (Refer to 33-50-01, Emergency Lighting Battery – General Data).

p) Connect the heating pad electrical connector to the appropriate mating wire harness connector. Route wires to match those previously removed in Para 6.A.4).



Emergency Lighting Battery
Figure 3

3) Main Battery Heating Belt

   a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

   b) Gain access to the battery bay by removing the floor panel. (Refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram).

   c) Remove the battery. (Refer to 24-30-31, Main Battery System – Maintenance Practices).

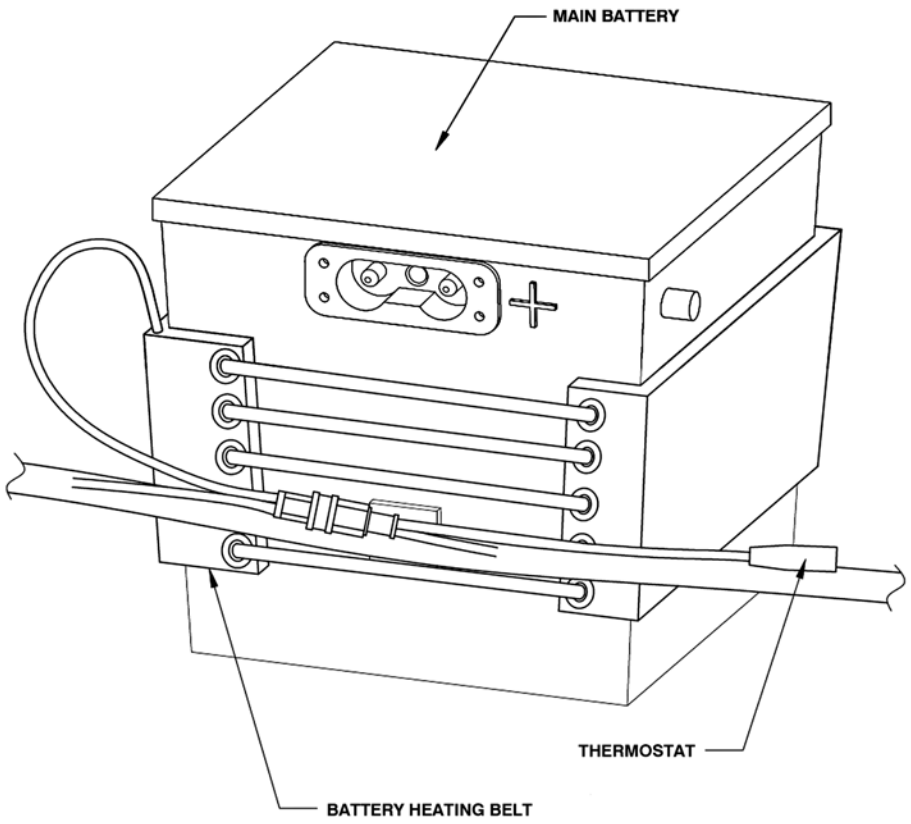
   d) Wrap the battery belt around the main battery. Position the eyelets on either side of the battery's main terminal, refer to Figure 4.

   e) Secure heating belt around the battery by looping cable ties through parallel eyelets, refer to Figure 4.

   f) Gently tighten the cable ties and alternate tension between the ties. Do not pull too hard as this could damage the belt or remove eyelets.

   g) Reinstall the battery. (Refer to 24-30-31, Main Battery System – Maintenance Practices).

   h) Connect the heating belt electrical connector to the appropriate mating wire harness connector. Route wires to match those previously removed in Para 6.A.4).



Main Battery
Figure 4

4) Primary Flight Display Heating Pads

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove the pilot and co-pilot's PFDs, refer to 31-11-00, Primary Flight Display (PFD), Para 4.

c) Clean the top surface of the frame below both PFDs with an appropriate solvent, refer to Para 3.A, and tool as per Para 5.A.

d) Position the heating pads with the lead forward and toward the left side, refer to Figure 5.

e) Confirm resistance values. Refer to Table 2 — 115V System Values (±7%) and Table 3 — 230V System Values (±7%).

f) Spread DC730 Flourosilicone adhesive into a thin layer between 0.020 - 0.032 inches thick to the heating pad contact surface (flat, unmarked side).

Note The heating pad must be in complete bonded contact with the mounting surface through the adhesive/sealant. Cavities in the installation site directly under the heating pad must be filled with adhesive/sealant in order to provide proper thermal transfer.

g) Place the heating pad in location and bond, using a roller or plastic putty applicator.

h) Starting at the center of the pad, draw the roller across the surface of the pad to the outer edge, expelling air bubbles, cavities, and imperfections. When the proper amount of sealant is used, very little sealant will be expelled from under the pad.

i) When cleaning excess sealant after rolling, use extra care, and sealant, as required to seal the edges of the pad.

j) Use tooling (vacuum bags, tape, foam, straps, etc.) to ensure that constant and even pressure is applied to the pad during curing.

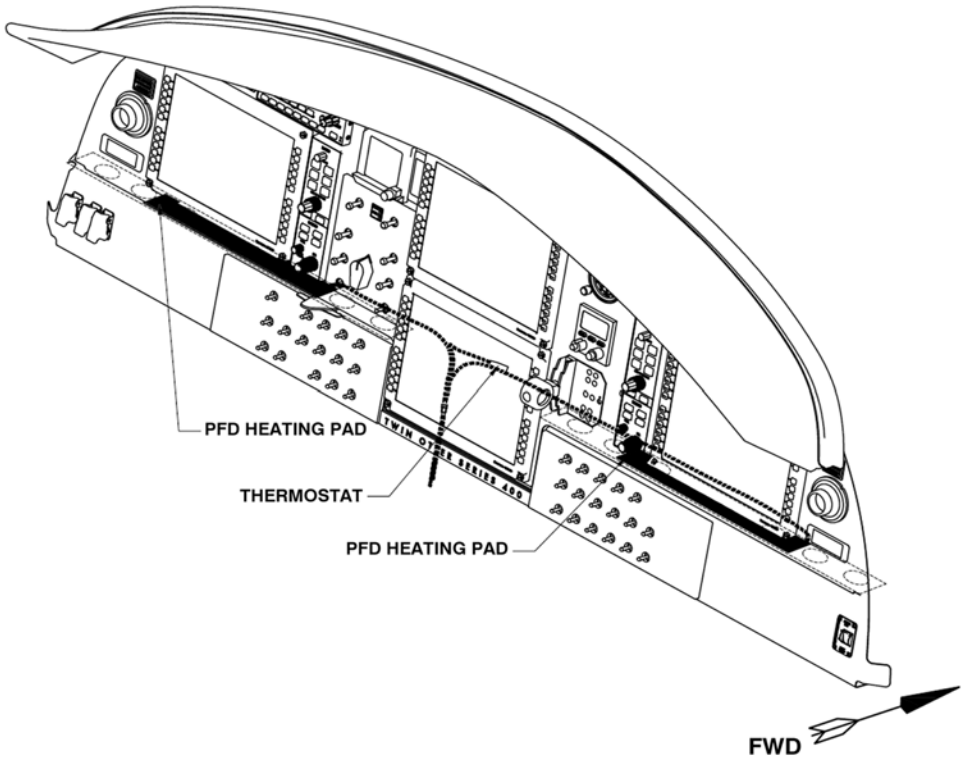
k) Once the bonding agent has cured (refer to Para 5.A) remove bag, tape and/or fixtures.

l) Apply a dollop of the adhesive/sealant under the base of the element lead (where lead enters element). This reduces wire fatigue and guards against the incursion of fluids at this location.

m) Seal the perimeter of the element with adhesive/sealant. Use just enough of the sealant to create a smooth transition along the edge of the element to mounting surface. This transition is not to extend beyond one quarter of an inch (.25") from the element and not more than one eighth of an inch (0.125") onto the elements upper surface.

n) Connect the heating pad electrical connector to the appropriate mating wire harness connector. Route wires to match those previously removed in Para 6.A.4).

o) Reinstall the pilot and co-pilot's PFDs, refer to 31-11-00, Primary Flight Display (PFD), Para 4.



Instrument Panel Frame
Figure 5

C. Ground Power Receptacle

1) Removal of Ground Power Receptacle

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Locate and open the left access panel to the hydraulic reservoir. (Refer to 12-00-00, Servicing – General Data).

c) Disconnect the wires from the fuse holders and power receptacle.

d) Remove the nuts that secure the fuse holders above the power receptacle and remove the fuse holders.

e) Remove the two nuts, four washers, and two screws that secure the power receptacle to the door.

f) Remove the power receptacle from the door.

2) Installation of Ground Power Receptacle

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Locate and open the left access panel to the hydraulic reservoir. (Refer to 12-00-00, Servicing – General Data).

c) Position the power receptacle into the door and secure the receptacle in place with two screws, four washers, and two nuts.

d) Position the fuse holders in place above the power receptacle. Secure the fuses in place with the provided nut.

e) Refer to PSM 1-64-2W, 25-70-00, Cold Weather Operations Kit (S.O.O.s 6287 or 6276) for wire routing and terminations. If reinstalling a power receptacle, route wires to match those previously removed in Para 6.C.1).

D. Cockpit Power Outlet

1) Removal of the Cockpit Power Outlet

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Locate the power outlet under the pilot's seat from inside the hydraulic bay. (Refer to 12-00-00, Servicing – General Data).

c) Disconnect the wires from the power receptacle.

d) Remove the large nut that secures the outlet in place.

e) Remove the power outlet.

2) Installation of the Cockpit Power Outlet

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Locate the installation location under the pilot's seat from inside the hydraulic bay. (Refer to 12-00-00, Servicing – General Data).

c) Position the power outlet in the channel and secure it in place with the Teflon washer and large nut provided with the outlet.

d) Refer to PSM 1-64-2W, 25-70-00, Cold Weather Operations kit (S.O.O.s 6287 or 6276) for wire routing and terminations. If reinstalling the outlet, route wires to match those previously removed in Para 6.D.1).

E. Indicator Lights

1) Removal of the Indicator Lights

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Locate the indicator lights on the left side of the aircraft.

c) Remove the electrical connector from the end of the indicator light leads by freeing the pins.

d) Remove the indicator lights by pressing in the tabs on the sides of the light's body to release the light from its location.

Note The light has been sealed in place and may require careful persuasion to free it from its boots.

2) Installation of the Indicator Lights

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Locate the installation location on the left side of the aircraft.

c) Position the indicator lights into the installation hole. Place a small amount of appropriate sealant around the head of the indicator light, behind the chrome ring. Snap the light into place by pushing it into the installation hole. Wipe off any excess sealant.

d) Install the electrical connector to the end of the indicator light leads. Connect the indicator light connector to the appropriate mating wire harness connector.

7. Adjustment/Test

A. Battery/Avionics Heat Test

1) Locate the two thermostats and disconnect from the heating pads. One is behind the instrument panel, refer to 31-11-00, Multifunction Display (MFD), Para 4, to gain access. The other is in the battery bay, refer to 12-00-00, Servicing – General Data, Figure 2 – Servicing Diagram for the location.

2) Working on one thermostat at a time, cool it below 32°F (0°C) using an ice pack, circuit cooler spray, or other appropriate means. Check the resistance across the two power contacts with an ohmmeter, the circuit should be closed. Do the same test for the other thermostat.

3) With only one thermostat connected, warm it to above 32°F (0°C) using an appropriate means. Check the resistance across the two power contacts with an ohmmeter, circuit should be open. Do the same test for the other thermostat.

4) If the thermostats do not meet either condition outlined in steps 2) and 3) above, then they need to be replaced.

B. Functional System Test

1) Cool both thermostats below 32°F (0°C) using an ice pack, circuit cooler spray, or other appropriate means.

2) Plug the system into appropriate voltage power source, 115 VAC (S.O.O. 6287) or 230 VAC (S.O.O. 6276).

3) Wait 5 to 10 minutes then check the areas next to the heating pads, the areas should feel warm to the touch.

WARNING POWERED HEATING PADS CAN BURN BARE SKIN.

8. Inspection/Check

It is recommended that the system operational status be checked when any of the following events occurs:

- Hard landing
- Lighting strike
- Water immersion
- Removal/replacement of a heated part or component

A. Heating Pad System Inspection

1) Clean as required before the inspection.

2) Inspect the wiring for secure attachments by following the cables to each heating pad.

3) Examine the power receptacle, power outlet, cable leads, and junctions. If any portion of the wiring shows signs of fatigue, chafing, flexing, airflow, or vibration, then re-secure and repair as required.

4) Inspect the heating pads for signs of de-bonding. If any portion of the pad has become loose, then it may be re-bonded, refer to Para 6.

5) Inspect heating pads for areas developing gray or yellow. These areas indicate that a pad has failed or is in the process of failing due to poor contact with the adhesive. Heating pads showing signs of gray/yellow need to be replaced, refer to Para 6.

6) Do a function test as per Para 7.B.

9. Troubleshooting

The system has been wired in parallel. When the system's power is disconnected an ohmmeter can be used to check and troubleshoot functional concerns.

Note Before proceeding, ensure the aircraft electrical system is OFF and no external power supplies are connected. Disconnect the main battery quick disconnect and check that all of the heating pads are properly connected.

A. Cold Weather Operations System

1) Verify engine to airframe bonding is as per 71-00-00, Power Plant (Mod 6/2012) – General Data.

2) Verify heating system grounding by checking for continuity between the power plug ground pin, engine, and airframe.

3) Verify that there is no continuity between the power contacts and the airframe ground.

4) Measure the resistance between the power pin with an ohmmeter and record the value. Compare the value with the total system readout obtained at the time of initial installation.

5) Connect the system to an appropriate 115 VAC (S.O.O. 6287) or 230 VAC (S.O.O. 6276) voltage power source.

6) Verify that the indicator light is illuminated.

7) While the system is warming up, freeze the battery and avionics thermostat and check the heating pads for warmth. The heating pads have not warmed up significantly at this point and can be touched with bare skin, but use caution.

8) Wait 5 to 10 minutes. Check the areas next to the heating pads, the area should feel warm.

WARNING POWERED HEATING PADS CAN BURN BARE SKIN.

Heating pads can be tested individually with an ohmmeter by disconnecting the pad from the wiring harness and measuring the resistance across the pad's leads. Resistance values for heating pads are below.

Table 2 115V System Values (±7%)

Description	Quantity	Wattage	Ohms
Battery Heating Belt	1	74	178.7
AGB Heating Pad	2	120	110.2
Oil Cooler Heating Pad	2	40	330.6
PRGB Heating Pad	2	240	55.1
AGB Heating Pad	4	90	146.9
PFD Heating Pad	2	40	330.6
Emergency Battery Heating Pad	1	4	3306.3
Total 115V System	**1318 Watts, 11.6 Amps, 10.7 to 9.2 Ohms**		

Table 3 230V System Values (±7%)

Description	Quantity	Wattage	Ohms
Battery Heating Belt	1	74	714.9
AGB Heating Pad	2	120	440.8
Oil Cooler Heating Pad	2	40	1322.5
PRGB Heating Pad	2	240	220.4
AGB Heating Pad	4	90	587.8
PFD Heating Pad	2	40	1322.5
Emergency Battery Heating Pad	1	4	13225.0
Total 230V System	**1318 Watts, 5.7 Amps, 42.5 to 36.9 Ohms**		

Additional troubleshooting information is provided in Table 4 — Troubleshooting.

Table 4 Troubleshooting

Trouble	Possible Cause	Correction
Indicator light does not illuminate when power is connected.	Circuit protection is “blown”	Troubleshoot AC power source or replace the fuse
	Voltage too far out of range	Troubleshoot AC power source or replace the fuse
System does not produce heat	Circuit protection is “blown”	Troubleshoot AC power source or replace the fuse
	Ground power cord is not providing power	Connect cord and check power at source (wall).
	Wire is broken	Check connections and wire junctions.
	Ground power plug is damaged	Repair and/or replace plug.
Some of the heating pads are not warming up	Bad heating pad(s)	Check heating pad with ohmmeter, refer to Table 2 — 115V System Values (±7%) and Table 3 — 230V System Values (±7%) for resistance values.
	Voltage is too far out of range	Connect to a better AC power source.
	Wire is broken	Check connections and wire to the heating pad.
Smoke or odor occurs from a newly installed system	Off gassing occurs normally on newly installed systems	Check the system for proper installation.
Smoke or odor occurs from system that has been installed for one month or longer	Heating pad is failing (yellow/gray areas appearing on a pad)	Refer to Para 6. for pad replacement. Inspect the rest of the system for proper installation.
	Heat element is dirty	Refer to Para 6. for pad removal. Clean the pad in an appropriate method.
Circuit protection GFI on ground power continually trips	Damaged system (shorted wire)	Inspect the system for damage and shorts.
PFD and Battery heating pads do not warm up	Temperature is above freezing	None. PFD and Battery heaters are thermostatically controlled and not designed to turn on above freezing.
	Thermostat is broken	Check function of the thermostat, refer to Para 8.

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CHAPTER 26

FIRE PROTECTION

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CHAPTER 26 – FIRE PROTECTION

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

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Fire Protection – General Data

1. General

Fire detecting and extinguishing systems are installed to combat fire in both engine nacelle accessory compartments. Visual indication of a fire condition in an engine nacelle is provided by a warning light, which illuminates the appropriate fire extinguisher push button, annunciation also available on Primary Flight Displays (PFDs), Crew Alert System (CAS) message and via aural warning. The two warning light/push buttons, when operated, cause the associated fire extinguisher bottle in the engine nacelles to be discharged into its accessory compartment (refer to 26-10-00, Fire Detecting System – General Data and 26-20-00, Fire Extinguishing System – General Data).

A hand-operated HALON fire extinguisher is installed for use within the aircraft. On special order an additional hand operated HALON fire extinguisher may be installed and located in the cabin. (Refer to 26-25-00, Hand-operated Fire Extinguishers – General Data).

For the fire protection system wiring diagram refer to PSM 1-64-2W, 26-10-00 for details.

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Fire Detecting System – General Data

1. General

A fire detection system is provided for each engine. In the event of an engine fire, thermal detector units installed in the engine nacelle actuate and complete an electrical circuit to illuminate a red warning light for the appropriate fire extinguisher push button, located on the top of the instrument panel in the flight compartment. Annunciation is available on Apex as well as the Primary Flight Displays (PFDs), Crew Alerting System (CAS) and an aural warning.

2. Description

Each engine fire detecting system comprises four thermal units connected to two wire loops, a fire warning light, magnetic and thermal circuit breakers, the associated electrical wiring, and a circuit test switch which is common to both engine systems. Aural warning consisting of a tone, the word “FIRE” is annunciated, and Left and/or Right Engine Fire warning CAS messages are shown on the systems MFD. The location of all system components is shown on Figure 1.

A. Fire Detecting Thermal Units

The four fire detecting thermal units for each engine, and their connecting wire loops, are located on the forward and rear fireseals in the engine nacelle, and on a nacelle frame at station 54.60. The two forward thermal units actuate at a temperature of 450°F, and the two rear thermal units at a temperature of 300°F.

B. Fire Detection Warning Lights and Switches

The fire detection lights are integral with the fire extinguisher push button located on the top of the instrument panel, the warning light/push button are annotated LEFT ENGINE and RIGHT ENGINE. Four circuit breakers, two for each engine system, are provided on the circuit breaker panel; two are thermal circuit breakers labeled FIRE DETECTION L and R, and two are magnetic circuit breakers labeled FIRE DET FAULT IND L and R.

C. Fire Detecting System Magnetic Circuit Breakers

Two of the system circuit breakers, one in each engine circuit, are of the magnetic type, having a single set of switch contacts with a center moving arm mechanically linked to its magnetic coil.

3. Operation

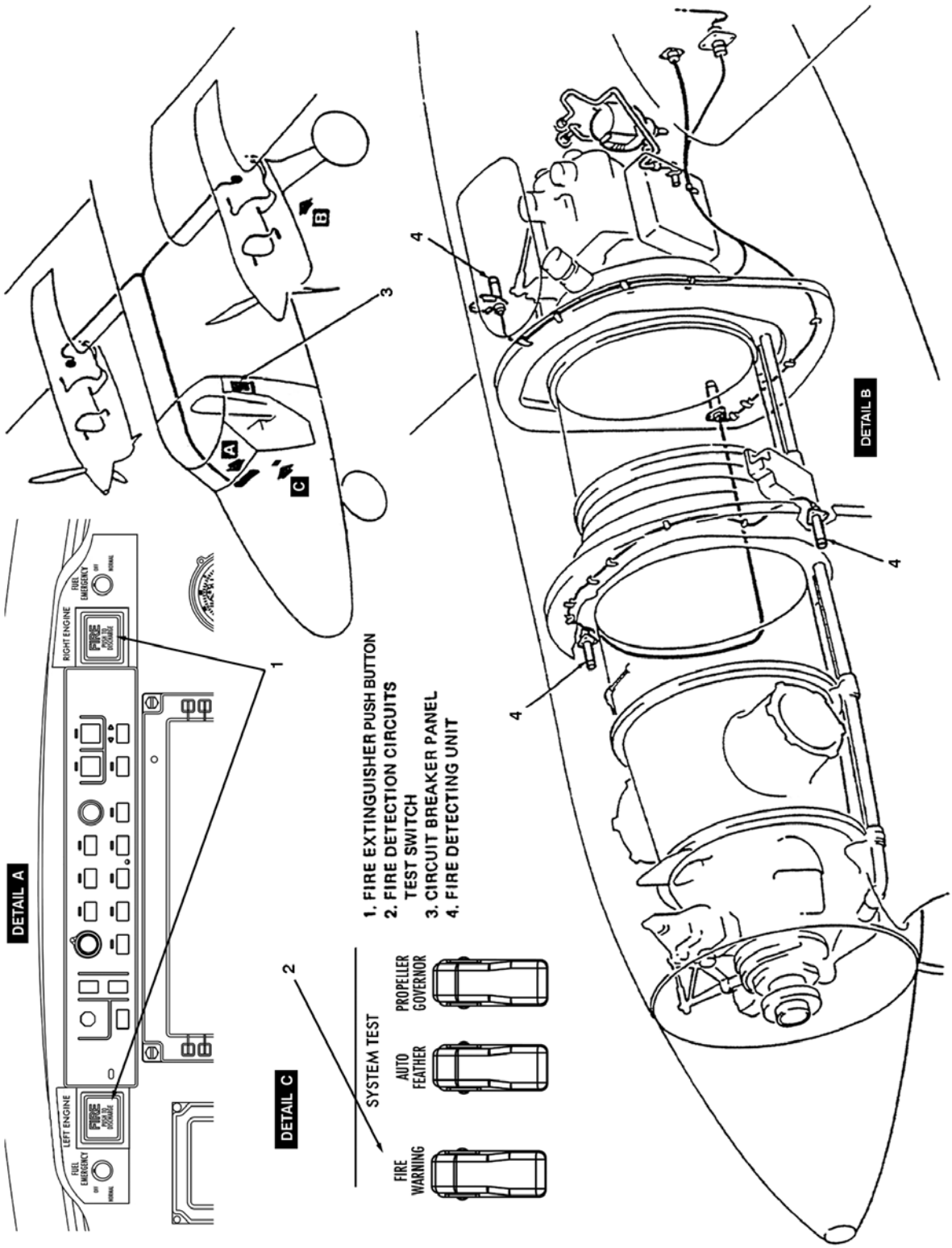
With the aircraft 28 Volt DC bus bars energized and the system circuit breakers engaged, the magnetic circuit breaker in each engine circuit supplies power to one of the two thermal detecting loops. Simultaneously the center arm of the magnetic circuit breaker, which is connected to the second of the two thermal detecting loops through the warning lamp, moves to the ground position.

When the contacts within anyone of the thermal units is actuated (due to a fire condition) the contacts close and complete a circuit between the two loops, so causing the warning lamp to illuminate and the “FIRE” warning is annunciated.

In the event of a ground fault (short circuit) occurring in the loop supplied through the magnetic circuit breaker, the circuit breaker will disengage, the center arm within the breaker moves to disconnect the ground and provide an alternative power supply to the second loop.

The TEST switch when selected to TEST, connects the two thermal-detecting loops of each engine system in series and completes a circuit through the warning light to ground, thus allowing the continuity of the loops, and the serviceability of the lights to be verified.

The aural warning is continuous except when muted by depressing the MASTER WARNING switch.



Fire Detection System – Component Locations
Figure 1

4. Adjustment/Test

A. Operational Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Engage FIRE DETECTION L and R and FIRE DET FAULT IND L and R circuit breakers.

3) Select FIRE WARNING switch to up position and check warning lights in both FIRE warning lights/push buttons illuminate.

4) Short out one of the overheat sensors in the LH nacelle by placing a suitable jumper across the two terminals of the sensor and check that the left engine fire warning light/push button is illuminated. Remove jumper and replace rubber cap over sensor terminals.

5) Repeat Para 4.A.4) at the RH nacelle and check that the right engine FIRE warning light/push button illuminates.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

7) Disengage FIRE DETECTION L and R circuit breakers.

B. Bench Test Fire Detecting Thermal Units

Equipment required:

- Fenwall field test kit 80001–0 or equivalent
- General Radio 500-volt AC Megger or equivalent

1) Test Insulation Resistance

a) Connect a General Radio 500-volt Megger or equivalent between the shell of fire detecting unit and both its leads tied together.

b) Check that minimum insulation resistance is 3 Megohms.

2) Test Control Set Point

a) Install the fire detecting unit in the test block of Fenwall field test kit 80001–0 or equivalent in a manner simulating as closely as possible the normal installation of the detecting units (especially in regard to restraint by clamps, brackets and external connections).

b) Set ON–OFF switch to OFF and connect test unit to 115-volt AC outlet.

c) Set INVERSE–REGULAR switch to REGULAR.

d) Set HIGH–LOW switch to HIGH.

e) Place thermometer in appropriate well in test block.

f) Connect receptacle leads to the fire detecting unit leads.

g) Set ON–OFF switch to ON and wait 15 minutes after cycling begins for system to reach a stable condition.

h) Calculate the control set point by taking the average of three consecutive readings of test block temperature at the instant of contact closure during stable operation.

Note All temperature readings are to be made at the instant the heater pilot lamps come “on”.

i) Check that tolerances are within ± 5°F or 2% of the designated control set point, whichever is the greater.

Note PDM17343–62–300 Control set point is 300°F.
PDM17343–62–450 Control set point is 450°F.

j) If unit is not within tolerances above, replace with serviceable unit.

Fire Extinguishing System – General Data

1. General

The fire extinguishing system is comprised of two independent systems, one for each engine. Each system has a warning light/push button located in the flight compartment; which when pushed, electrically discharges a fire extinguisher bottle. The extinguishing agent from the bottle is directed by a discharge nozzle into the appropriate engine accessory compartment.

2. Description

Each engine fire extinguishing system consists of a fire extinguisher bottle with discharge pipes and indicator discs, and a fire extinguisher Warning Light/Push Button. Figure 1 shows component locations and system details.

A. Fire Extinguisher Bottles

A fire extinguisher bottle containing HALON 1301, is installed in each engine nacelle behind the firewall. A main discharge pipe extends from a valve on the bottom of the bottle through the firewall to the engine accessory compartment. Two additional pipes, one connected to the bottle and the other to the main discharge pipe, terminate at indicator discs on the side of the nacelle.

B. Fire Extinguisher Bottle Indicating Discs

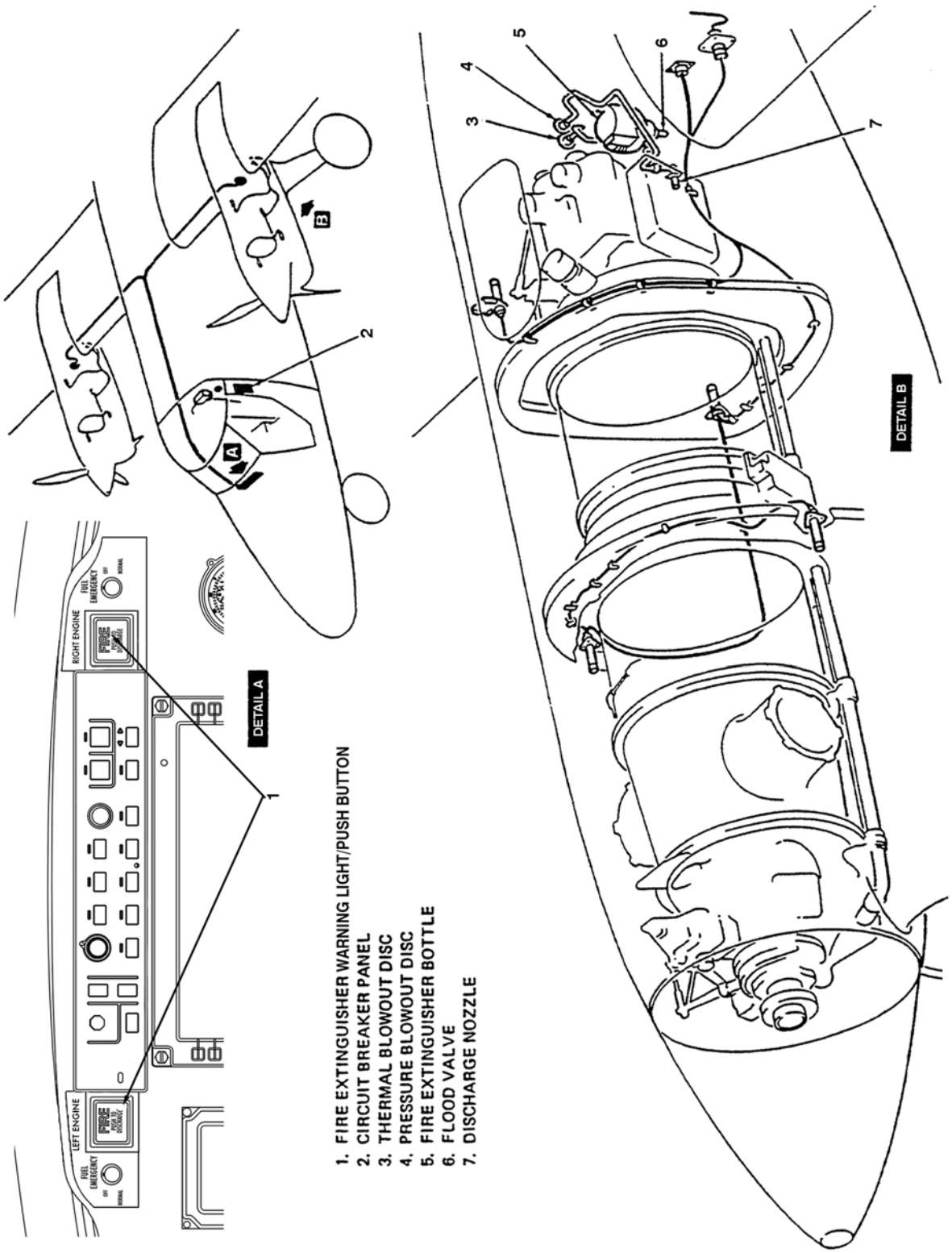
Two fire extinguisher bottle indicating discs are provided for each bottle, one yellow, and the other red; they are located on the inboard side of each engine nacelle.

C. Fire Extinguisher Warning Light/Push Buttons

A fire extinguisher warning light/push button, identified FIRE PUSH TO DISCHARGE, is provided for each engine system.

3. Operation

The warning light/push button operated switch of each system is supplied with 28 Volt DC from its respective FIRE EXTINGUISHER L or FIRE EXTINGUISHER R 5-ampere circuit breaker. When the FIRE push button is pushed, the switch is operated, and the circuit is completed to the appropriate bottle valve which is opened by the action of a squib assembly. The fire extinguishing agent is then expelled through the valve and main discharge pipe into the engine accessory compartment. The yellow indicator disc, which is backed up by a piston assembly in the pipe connected to the main discharge pipe, is punched out when the extinguisher operates. When the fire bottle is discharged due to thermal expansion, the red disc is blown out.



Fire Extinguishing System – Component Locations
Figure 1

4. Adjustment/Test

A. Function Test Fire Extinguishing System

WARNING DO NOT USE A VOLT-OHMMETER, BATTERY POWERED CONTINUITY LIGHT, OR ANY SIMILAR DEVICE IN AN ATTEMPT TO TEST THE SQUIB ASSEMBLY. USE OF A MULTIMETER, WHICH PASSES MORE THAN 25 MILLIAMP, MAY DETONATE THE SQUIB ASSEMBLY.

Equipment Required:

- Multimeter – that electrically limits test current to 25 mA, maximum
- Pressure test rig (refer to 21-40-00, Heating System – General Data)
- Test lamp-28 volt

Note Before carrying out the following tests ensure fire extinguisher system is completely and correctly installed.

1) Safety Precautions

a) All personnel shall stand clear of the fire extinguisher discharge nozzles at all times when FIREX system tests are being carried out.

b) Only a multimeter that electrically limits test current to 25 milliamps maximum shall be used to check firing cartridge continuity.

c) The use of a volt-ohmmeter, battery powered continuity light or any similar device to test the firing cartridge is prohibited as these devices may detonate the cartridge.

d) If a FIREX bottle is accidently discharged in an aircraft, the aircraft shall be well ventilated before personnel are allowed to enter.

e) Frostbite or low-temperature burns will result if the fire extinguishing agent (HALON 1301) is allowed to contact the skin.

2) Pressure Test

a) Remove split ring and the yellow indicator disc from discharge indicator on the inboard side of the right engine nacelle.

b) Connect pressure hose from test rig to end of discharge nozzle just forward of firewall in right engine nacelle.

c) Adjust test rig to supply 25 psi, check that piston at discharge indicator moves outwards as pressure applied, and reseats when pressure is reduced.

d) Disconnect test rig, replace yellow indicator disc and split ring in discharge indicator.

e) Ensure split ring is fully seated in its retaining groove.

f) Repeat Para 4.A.2).a) through Para 4.A.2).e) for left engine system.

3) System Circuit Test

a) Ensure that all aircraft electrical power is switched OFF and that all circuit breakers are disengaged (refer to 24-00-00, Electrical Power – General Data).

b) Disconnect the positive electrical wire from the insulated post on the LH and RH FIREX bottle firing cartridge. Leave the ground wire connected to the bottles. Connect a 28 Volt test lamp between the disconnected wiring at each FIREX bottle and airframe ground.

c) Engage electrical power circuit breakers. Select POWER SOURCE switch to BATTERY, and MASTER switch to ON (Pre Mod 6/2314); or select BATTERY switch to ON (Post Mod 6/2314). Engage FIRE EXTINGUISHER L and FIRE EXTINGUISHER R circuit breakers.

d) Push the RIGHT ENGINE–FIRE warning push button located on the emergency panel and check that RH test lamp illuminates. Release and check that test lamp is extinguished.

e) Push the LEFT ENGINE–FIRE warning push button located on the emergency panel and check that LH test lamp illuminates. Release and check that test lamp is extinguished.

f) Switch off aircraft DC power, disengage FIRE EXTINGUISHER L and R circuit breakers and remove test lamps from FIREX bottle wiring.

g) Reconnect the electrical wire to the FIREX bottles. Ensure that ground wire (identified by suffix letter N in wire code) is connected to ground terminal and positive wire is connected to actuate terminal.

h) Engage FIRE EXTINGUISHER L and FIRE EXTINGUISHER R circuit breakers.

4) Continuity Test

WARNING **DO NOT USE A VOLT-OHMMETER, BATTERY POWERED CONTINUITY LIGHT, OR ANY SIMILAR DEVICE IN AN ATTEMPT TO TEST THE SQUIB ASSEMBLY. USE OF A MULTIMETER, WHICH PASSES MORE THAN 25 MILLIAMP, MAY DETONATE THE SQUIB ASSEMBLY.**

a) Immediately prior to using the multimeter, short out the test leads, and check that the meter current is no more than 10 – 25 milliamps.

b) Connect leads from the multimeter to the firing cartridge terminal studs on the right engine FIREX bottle, and check that meter indicates continuity.

c) Disconnect test box and reconnect FIREX bottle wiring to firing cartridge terminals.

d) Ensure that ground wire (identified by suffix letter N in wire code) is connected to ground terminal and positive wire is connected to actuate terminal.

e) Repeat Para 4.A.4).b) through Para 4.A.4).d) for left engine FIREX bottle.

f) Check pressure gauges on FIREX bottles each indicated fully charged condition in accordance with the manufacturers' data plate (450 psi at 70°F).

g) Ensure firing cartridge has been wirelocked to its FIREX bottle.

h) At completion of test, position the rubber nipple on the actuate wire and secure with a plastic tie.

Fire Extinguishing Bottle – Maintenance Practices

1. Servicing

   A. Servicing of Fire Extinguisher Bottle

      1) Ensure that no leakage or damage to bottle or attachments is evident.

      2) Remove bottle and weigh, check that weight of charged bottle is within 0.25 pounds of total weight stamped on data plate.

         Note Damaged or underweight bottles must be replaced.

      3) Install serviceable fire extinguisher bottle.

2. Removal/Installation

   A. Remove Fire Extinguisher Bottle

      WARNING FROSTBITE OR LOW TEMPERATURE BURNS WILL RESULT IF HALON 1301 (TRIFLUOROBROMOMETHANE CBRF3) IS ALLOWED TO CONTACT THE SKIN. STORE CBRF3 BOTTLES IN AN ISOLATED AND WELL VENTILATED AREA. EACH BOTTLE MUST BE STORED WITH NECK FACING UP AND PROTECTIVE CAPS INSTALLED TO NECK AND SAFETY OUTLETS.

      1) Switch OFF aircraft electrical power (refer to 24-00-00, Electrical Power – General Data).

      2) Lower the engine cowling.

      3) Disconnect electrical connector from extinguisher bottle valve.

      4) Disconnect discharge pipe to engine accessory compartment, and thermal pressure pipe to red disc, at fire extinguisher bottle.

      5) Support bottle, remove bolts securing clamps and remove bottle.

   B. Install Fire Extinguisher Bottle

      CAUTION CHARGED BOTTLES MUST BE HANDLED WITH CARE TO AVOID DAMAGING THE SEAL. BOTTLES MUST NOT BE DISCHARGED IN CONFINED AREAS. IF A BOTTLE IS ACCIDENTALLY DISCHARGED, THE AREA MUST BE WELL VENTILATED BEFORE PERSONS ARE ALLOWED TO ENTER.

      1) Check replacement fire extinguisher bottle for evidence of leaks and damage. Check that weight is within 0.25 pounds of total weight stated on data plate.

      2) Check that bottle pressure is acceptable in accordance with the manufacturers' data plate (450 psi at 70°F).

      3) Position fire extinguisher bottle in aircraft.

      4) Install the two bolts securing clamps.

      5) Connect and secure discharge pipe to engine accessory compartment, and thermal pressure pipe to red disc, at fire extinguisher bottle.

      6) Replace indicating discs as necessary (refer to 26-20-21, Indicating Discs – Maintenance Practices).

      7) Connect and secure electrical connector to extinguisher bottle valve.

      8) Secure engine cowlings.

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Indicating Discs – Maintenance Practices

1. Servicing

   A. Replace Fire Extinguisher Bottle Indicating Discs

      1) Remove split ring from disc assembly.

      2) In the case of a red disc, replace disc only, but for a yellow disc, replace disc and retainer assembly.

      3) Reinstall split ring, ensuring that it is correctly seated in groove.

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Hand-operated Fire Extinguishers – General Data

1. General

The hand-operated extinguisher is located on the aisle floor beside the co-pilot's seat. The extinguisher contains a frangible disc which, when broken, allows HALON contents to discharge under pressure. Mod 6/2092 provides an additional HALON extinguisher is installed and located in the cabin, forward of the airstair door.

2. Servicing

A. Flag Halon 1211 Extinguisher Model HAL–035W

1) Inspect extinguisher monthly and ensure pressure gauge indicates correct operating range with needle in green sector. Check red disc at rear of operating head, and if disc is broken, extinguisher has been used. Replace extinguisher as follows:

a) Release safety catch, and squeeze lever to empty container.

b) Unscrew container, and discard container or return to F.P.S. International approved agency for recharge.

c) Reset safety catch.

d) Replace red (tamper) disc.

e) Fit properly charged container to operating head.

f) Ensure nozzle is unobstructed.

2) Weigh unit every six months. Ensure weight is within four ounces of weight indicated on extinguisher label. Replace extinguisher or renew container (refer to Para 2.A.1)) if weight falls below limit.

B. H3R Aviation 1211 Extinguisher Model A1600 (Mod 6/2060)

1) Inspect extinguisher monthly and ensure pressure gauge indicates correct operating range with needle in green sector. Check red disc at rear of operating head, and if disc is broken, extinguisher has been used. Replace extinguisher as follows:

a) Release safety catch, and squeeze lever to empty container.

b) Unscrew container, and discard container or return to F.P.S. International approved agency for recharge.

c) Reset safety catch.

d) Replace red (tamper) disc.

e) Fit properly charged container to operating head.

f) Ensure nozzle is unobstructed.

2) Weigh unit every six months. Ensure weight is within four ounces of weight indicated on extinguisher label. Replace extinguisher or renew container (refer to Para 2.A.1)) if weight falls below limit.

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CHAPTER 27

FLIGHT CONTROLS

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FLIGHT CONTROLS – CHAPTER 27

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Flight Controls – General Data

1. General

The flight control surfaces are operated in the conventional manner by cables from a dual control column and dual rudder pedals, allowing the aircraft to be flown from either the pilot's or co-pilot's position. The ailerons, elevators and rudder are conventional, but the ailerons are hinged to arms at the trailing edge of each outboard fore flap. The elevators and rudders are aerodynamically horn balanced and, together with the ailerons, are internally mass balanced. Geared tabs are fitted to the ailerons and rudder to provide an aerodynamic assist to control surface movement. Trim tabs, which are adjustable in flight, are fitted to the left aileron, left elevator, and rudder. The right elevator incorporates a trim tab which is interconnected with the wing flap system, and trims the elevators in proportion to wing flap movement. Each cable circuit is provided with conveniently located turnbuckles, and push-pull rods which are adjustable by means of screw threaded end adapters. The wing flaps are interconnected by push-pull rods which are operated by a single hydraulically-operated actuator mounted in the cabin roof.

2. Servicing

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. General Rigging Instructions

The rigging of each control system of the flight controls is detailed in the Maintenance Practices of the individual systems. For each procedure it is assumed that all components are correctly installed, that the cables are fitted but not tensioned, and that all jam nuts on push-pull rods and turnbuckles are locked on the completion of rigging. It is also assumed that access panels and upholstery are removed, as required, for accomplishing these procedures and are replaced on completion of the operation. Figure 1 in 12-00-00, Servicing – General Data, provides locations of access and inspection provisions. The hydraulic system must be fully serviceable before wing flap, aileron, or flap/elevator interconnect trim rigging is started.

B. Tensioning of Control Cables

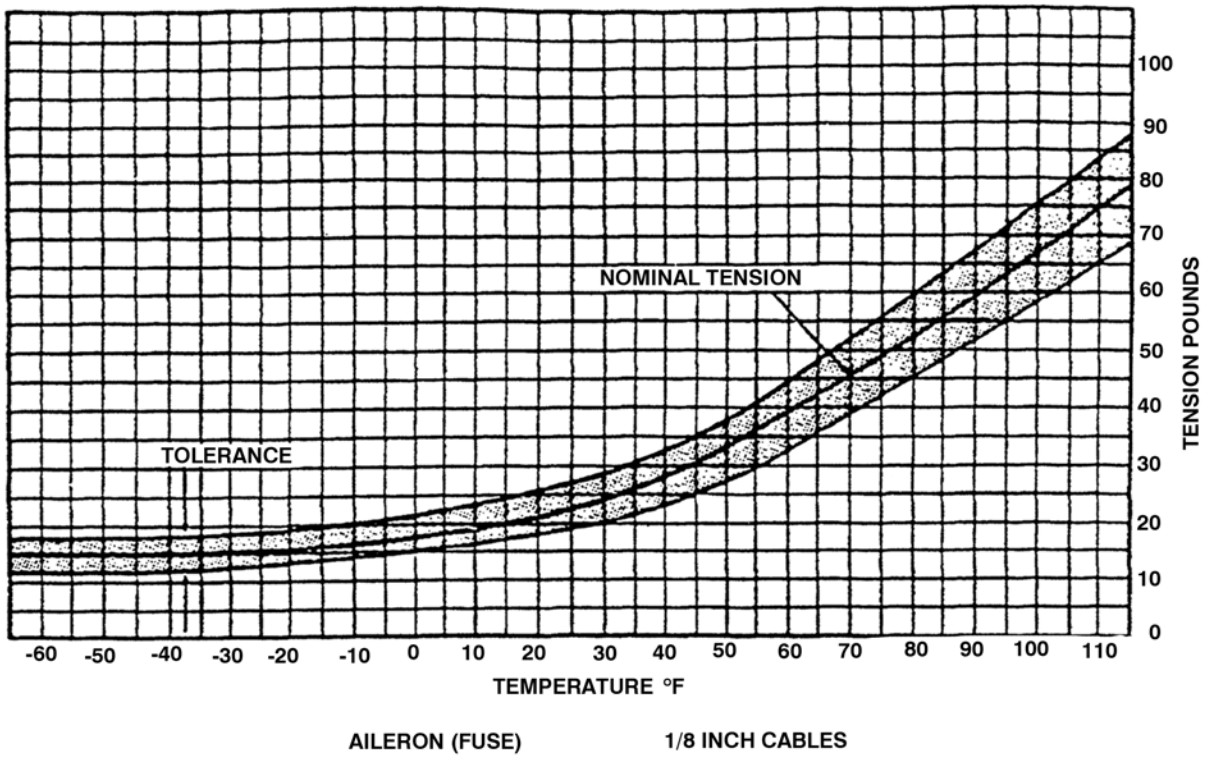
Whenever flight control system cables are slackened or replaced, they must be retensioned to the value shown on the appropriate graph (refer to Figure 1). Where possible, tensiometer readings are to be taken on bare cable. For cables fully covered in nylon, nominal values 15% lower than those shown on Figure 1 are to be used. When checking cable tensions, observe the following precautions:

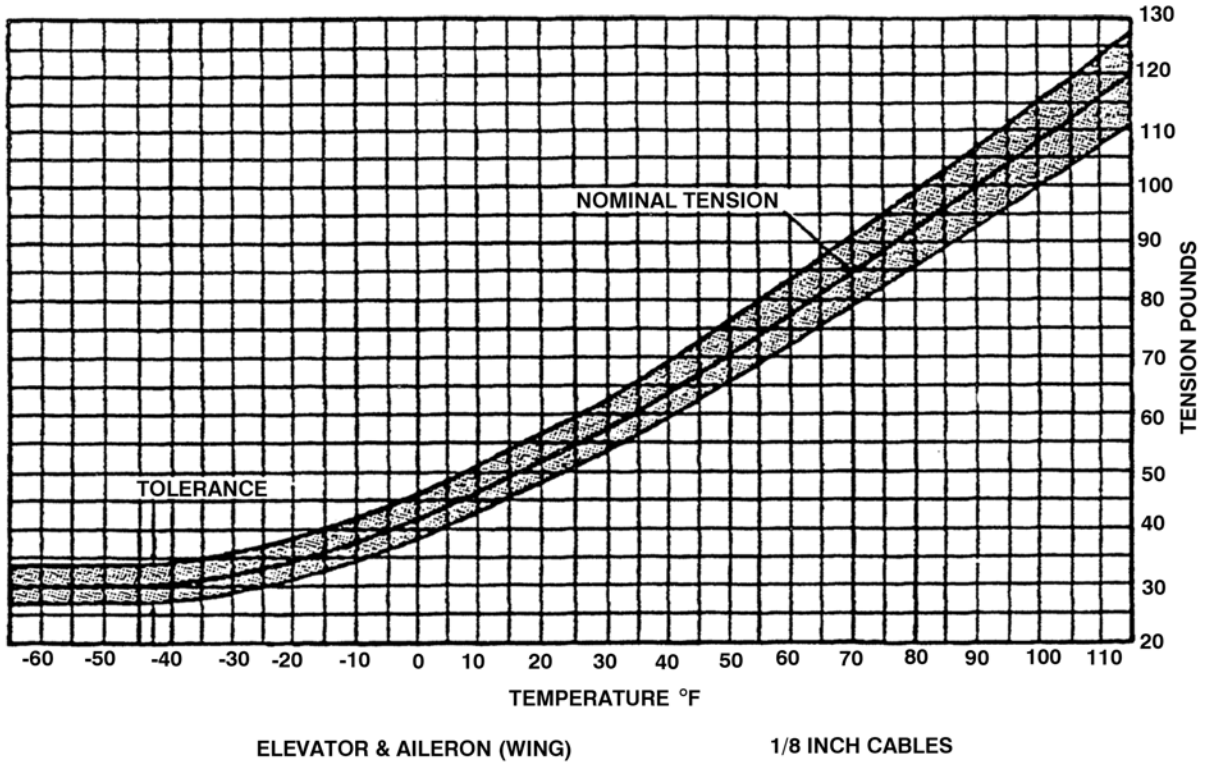
1) Ensure that tensiometer is of correct type for cable.

2) Vibrate cables to prevent an incorrect reading due to friction in circuit.

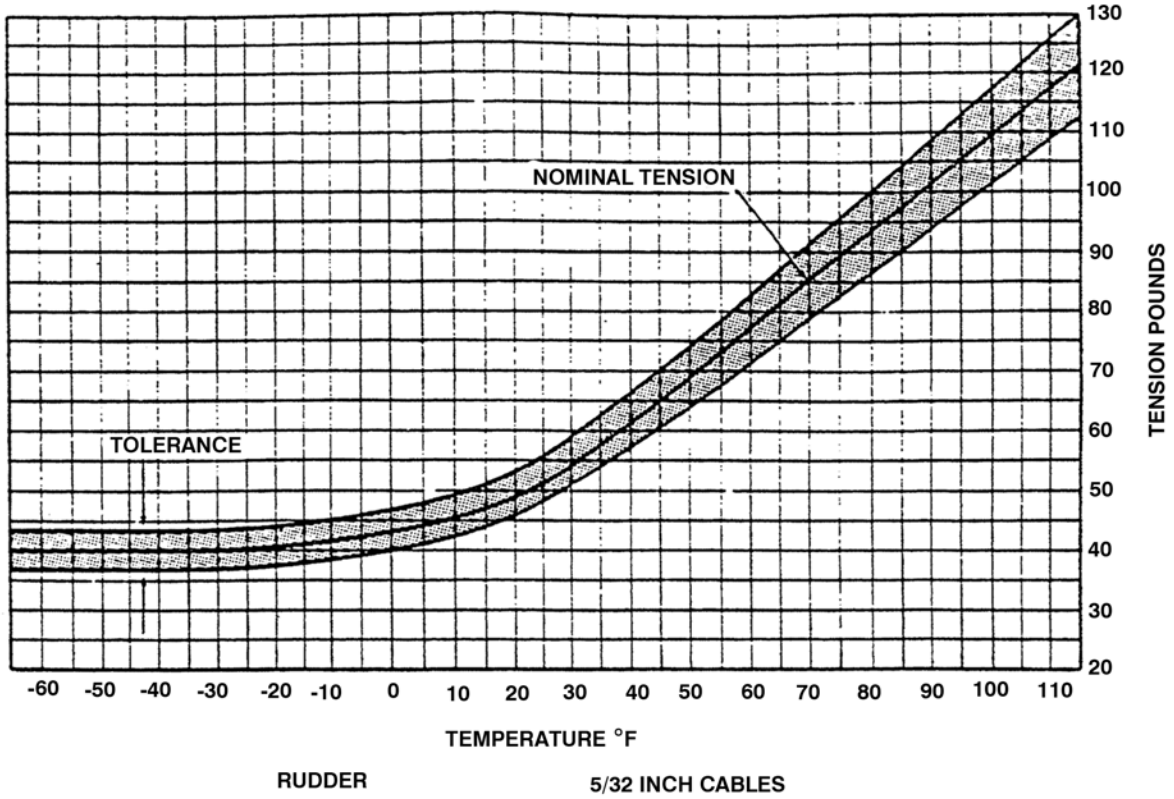
3) Position tensiometer on center portion of cable and not close to a quadrant, bellcrank, turnbuckle, etc.

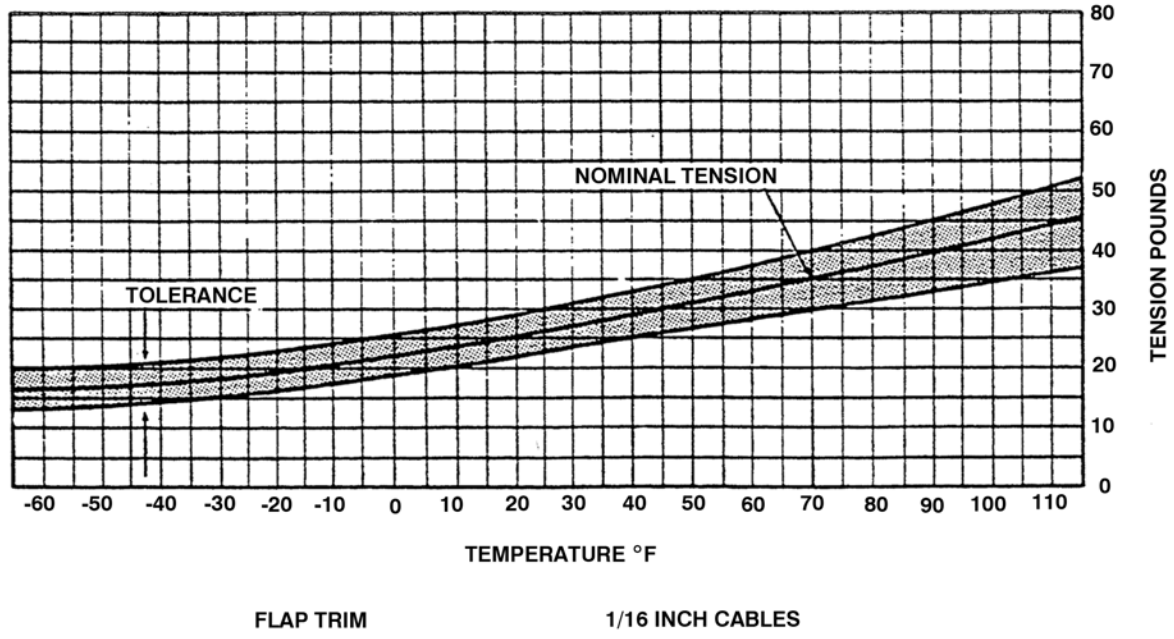
4) Recheck cable tensions after locking turnbuckles.



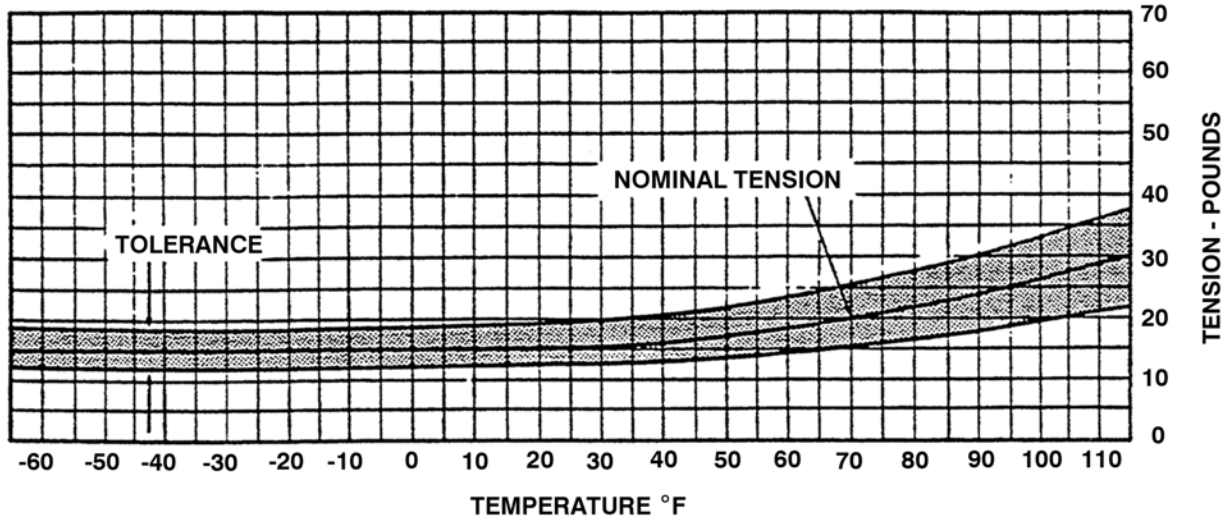


Cable Tension Graphs
Figure 1 (Sheet 1 of 3)





Cable Tension Graphs
Figure 1 (Sheet 2 of 3)



ELEVATOR TRIM
RUDDER TRIM
FLAP FOLLOW-UP
FLAP SELECTOR

1/16 INCH CABLES

ON AIRCRAFT FITTED WITH AN AUTOPILOT AUTOMATIC PITCH TRIM ACTUATOR, READ NOMINAL TO MAX. TENSION FOR ELEVATOR TRIM e.g. TENSION AT 70°F IS 20^{+5}_{0} POUNDS.

Cable Tension Graphs
Figure 1 (Sheet 3 of 3)

3. Inspection/Check

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. Inspection of Control Cables

Inspect control cables at prescribed inspection intervals to ensure their serviceability. Before inspecting cables installed in an aircraft, ensure that all accessible portions of the cable run are clean. Dust and metal particles will readily adhere to grease and impair smooth operation of the controls. Cables in the cabin area of fuselage, though not accessible along their entire run, can be cleaned and inspected at access points where they pass over pulleys or through fairleads where broken wires and wear are most likely to occur. Move flight controls through their full range of travel to allow for maximum cable length inspection. A control tending to foul momentarily is often an indication of a frayed cable fouling on a fairlead or pulley. A simple method of detecting broken wires is by running a lint-free rag along the cable. The rag will snag on any projections. If worn, frayed or corroded cables are suspected, slacken turnbuckles to allow careful twisting of cables to open strands for a more detailed inspection. Replace cables if worn or corroded and if wear limits in 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data are exceeded. Always replace cables where serviceability is marginal.

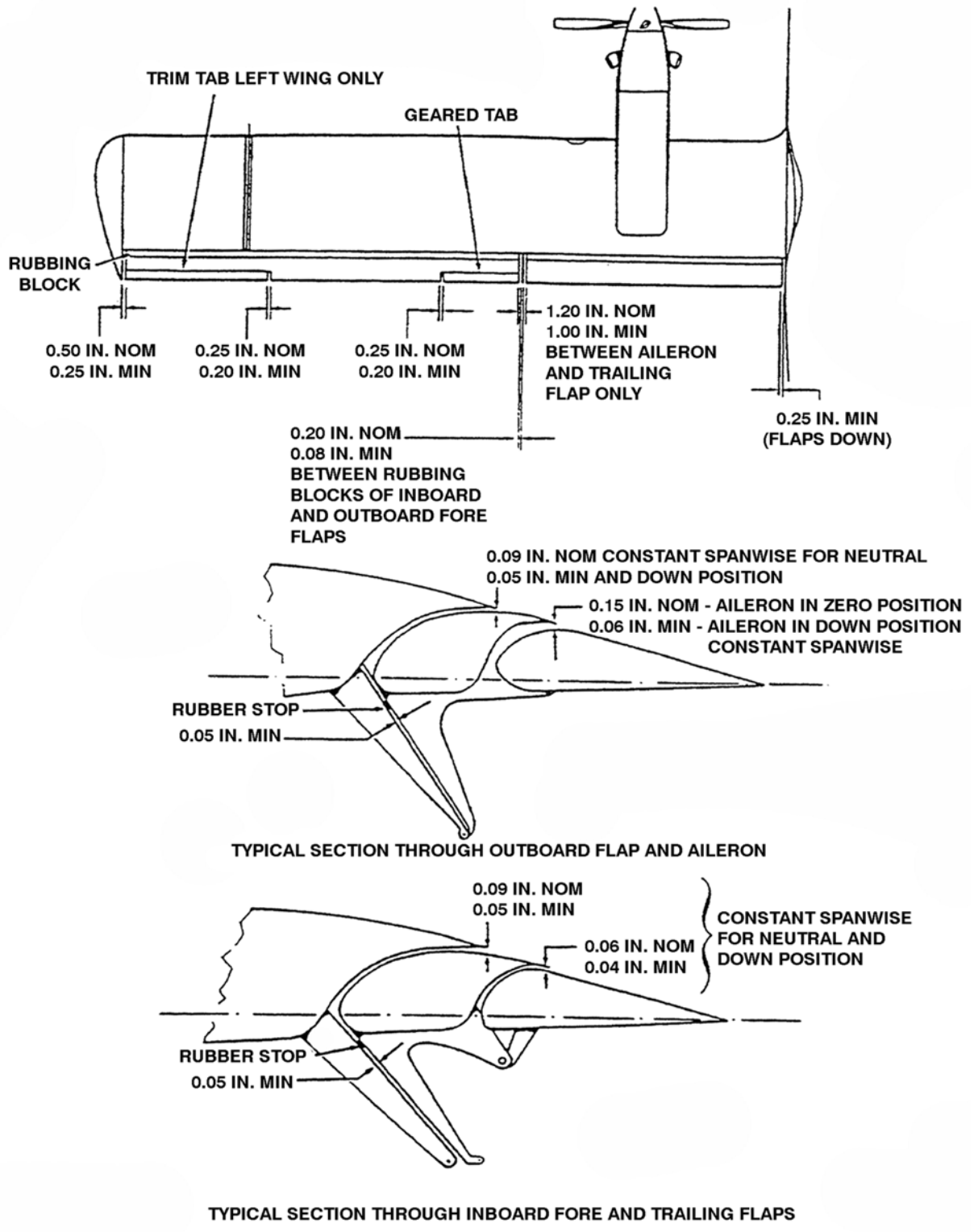
Remove corrosion and or zinc dust from the external surfaces in accordance with 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data.

B. Checking Control Surface Clearances

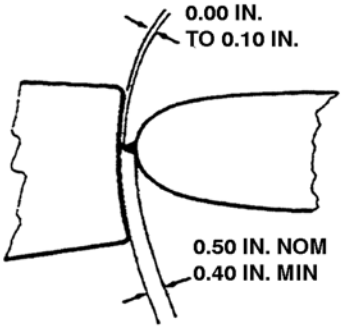
To check individual control surface clearances, refer to dimensions given in Figure 2.

C. Checking Control Components

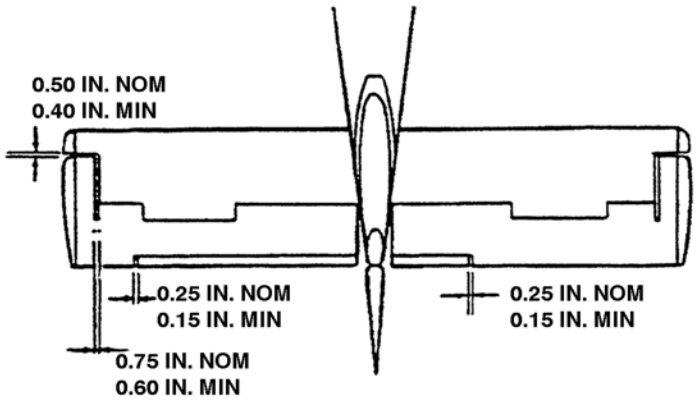
Inspect all control surfaces for damage and freedom of movement. Inspect individual control components for security. Refer to Inspection Requirements Manual for periodic and internal inspection requirements.

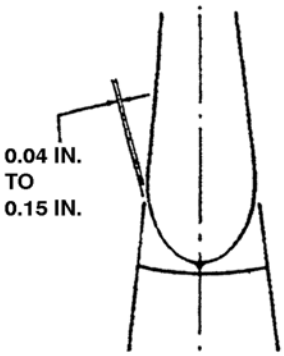


Control Surface Clearances
Figure 2 (Sheet 1 of 2)

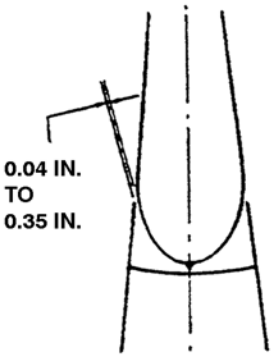


TYPICAL SELECTION
THROUGH ELEVATOR

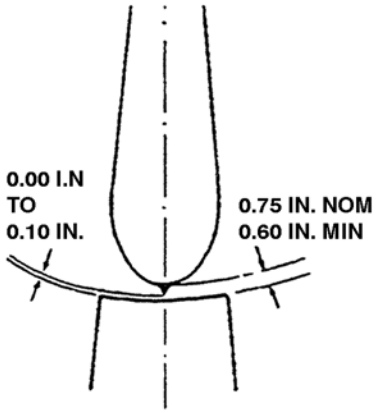




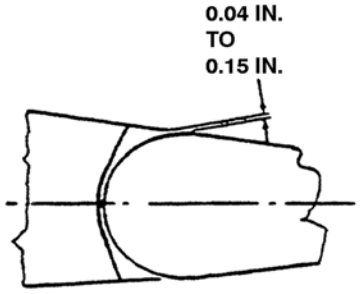
RUDDER/SHROUD CLEARANCE



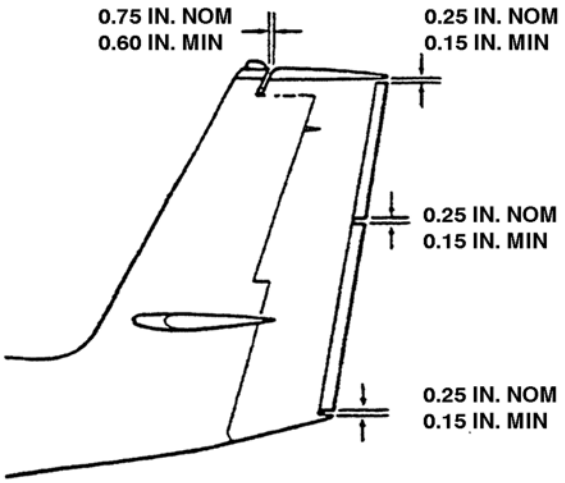
RUDDER/FUSELAGE
FAIRING CLEARANCE



TYPICAL SECTION
THROUGH RUDDER



ELEVATOR/SHROUD CLEARANCE



Control Surface Clearances
Figure 2 (Sheet 2 of 2)

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Aileron Control System – General Data

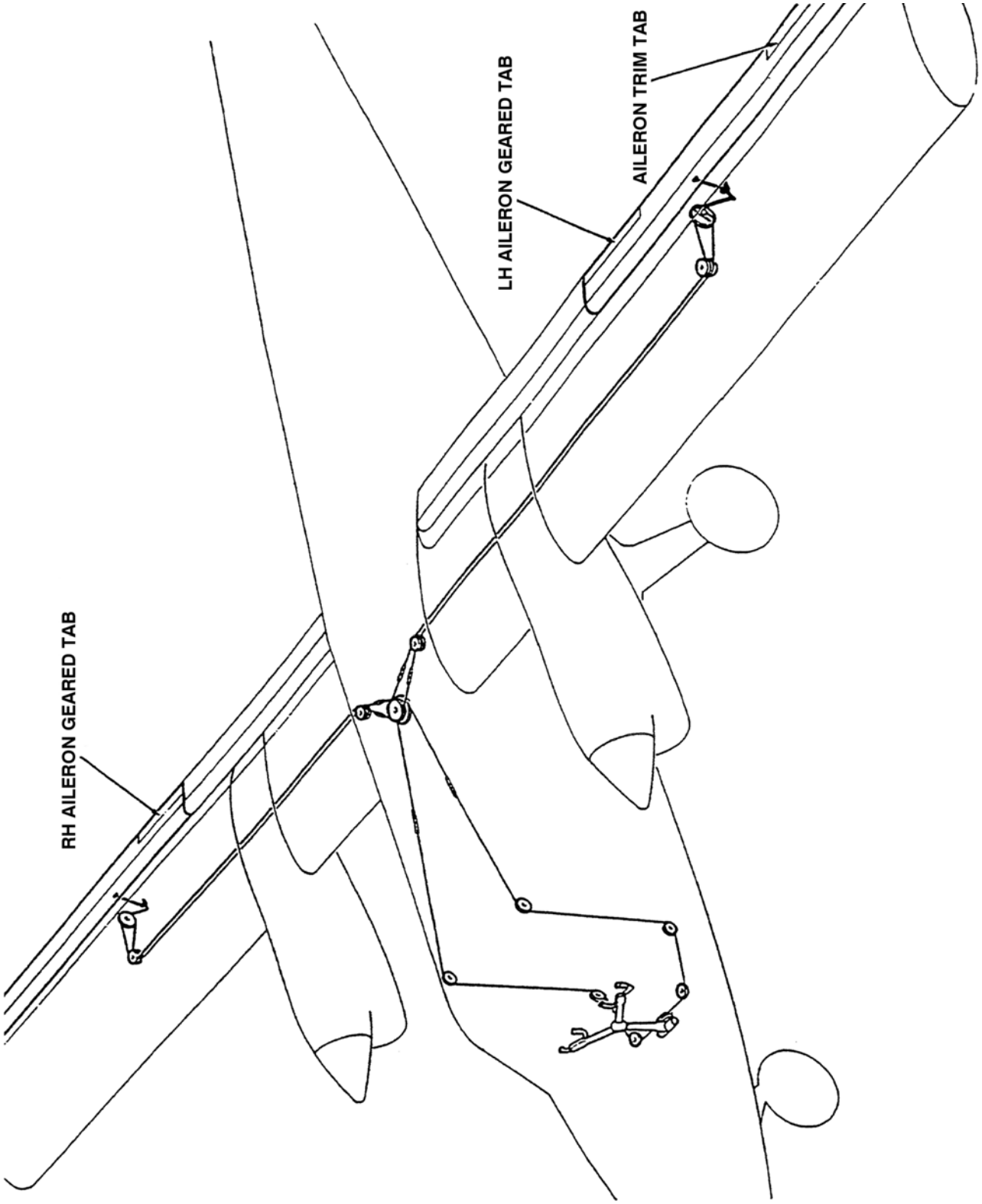
1. General

Rotary movement of the control column aileron control wheels, is transmitted by a chain and sprocket mechanism to a quadrant at the base of the column. Cables extending from the control column quadrant, are routed under the flight compartment floor and up both sides of the forward face of the flight compartment/cabin bulkhead, to transmit motion to an aileron quadrant located in the cabin roof aft of the wing front spar. From the aileron quadrant, a separate cable circuit is routed through each wing to linkage quadrants, located at the left and right second outboard fore flap hinge arm attachment brackets. A push-pull rod connects between each linkage quadrant and a bellcrank lever on the respective fore flap hinge arm. The ailerons attach to the trailing edge of the corresponding outboard flap by four aileron arms protruding from the outboard fore flap hinge arms. A push-pull rod from each aileron, connects to the bellcrank lever in the corresponding fore flap hinge arm. Refer to Figure 1 for details.

Because of the method of attachment, the ailerons droop with the wing flaps, and the geometric arrangement of the control linkage is such that the aileron range of movement increases proportionally as the flaps go down. The ailerons move differentially at any flap position.

Each aileron carries a geared tab, and the left aileron carries an electrically-operated trim tab (refer to 27-13-00, Aileron Trim Tab System – General Data and 27-16-00, Aileron Geared Tab System – General Data).

Note For general cable rigging instructions refer to 27-00-00, Flight Controls – General Data Para 2.A.



Aileron System
Figure 1

2. Adjustment/Test

A. Rig Aileron Control System

Refer to Figure 2 and Figure 3 for details.

Note Before attempting to rig the aileron control system ensure that wing flap system is correctly rigged (refer to 27-50-00, Wing Flaps System – General Data).

1) Pump flaps to fully up position.

2) Disconnect spring strut from co-pilot's handwheel.

3) Install rigging pin SD10544 (0.25 inch dia.) in base of control column.

4) Check that axis of each handwheel is horizontal. If necessary, remove covers from control column arms and adjust chain turnbuckles until handwheels are aligned. Set chain tension to value shown in Figure 2.

5) Check aileron quadrant for neutral setting by installing rigging pin SD10543 (0.375 inch dia.). Adjust cable circuit turnbuckles, if necessary, to obtain neutral position, maintaining correct cable tension (refer to 27-00-00, Flight Controls – General Data).

6) Check neutral position of aileron linkage by inserting rigging pin SD10542 (0.25 inch dia.) in linkage pulleys. Adjust circuit turnbuckles, if necessary, to obtain neutral position, maintaining correct cable tension (refer to 27-00-00, Flight Controls – General Data).

7) Remove all rigging pins and operate system through several cycles, then check that all rigging pins enter rigging holes freely.

8) With rigging pin installed in aileron linkage pulleys, check that trailing edge of aileron aligns with trailing edge of flap; adjust push-pull rod at aileron if necessary.

9) Remove rigging pins and operate controls over full range of travel with flaps in up position, then in down position. Adjust stops at base of control column, if necessary, to obtain desired travel (refer to Figure 3).

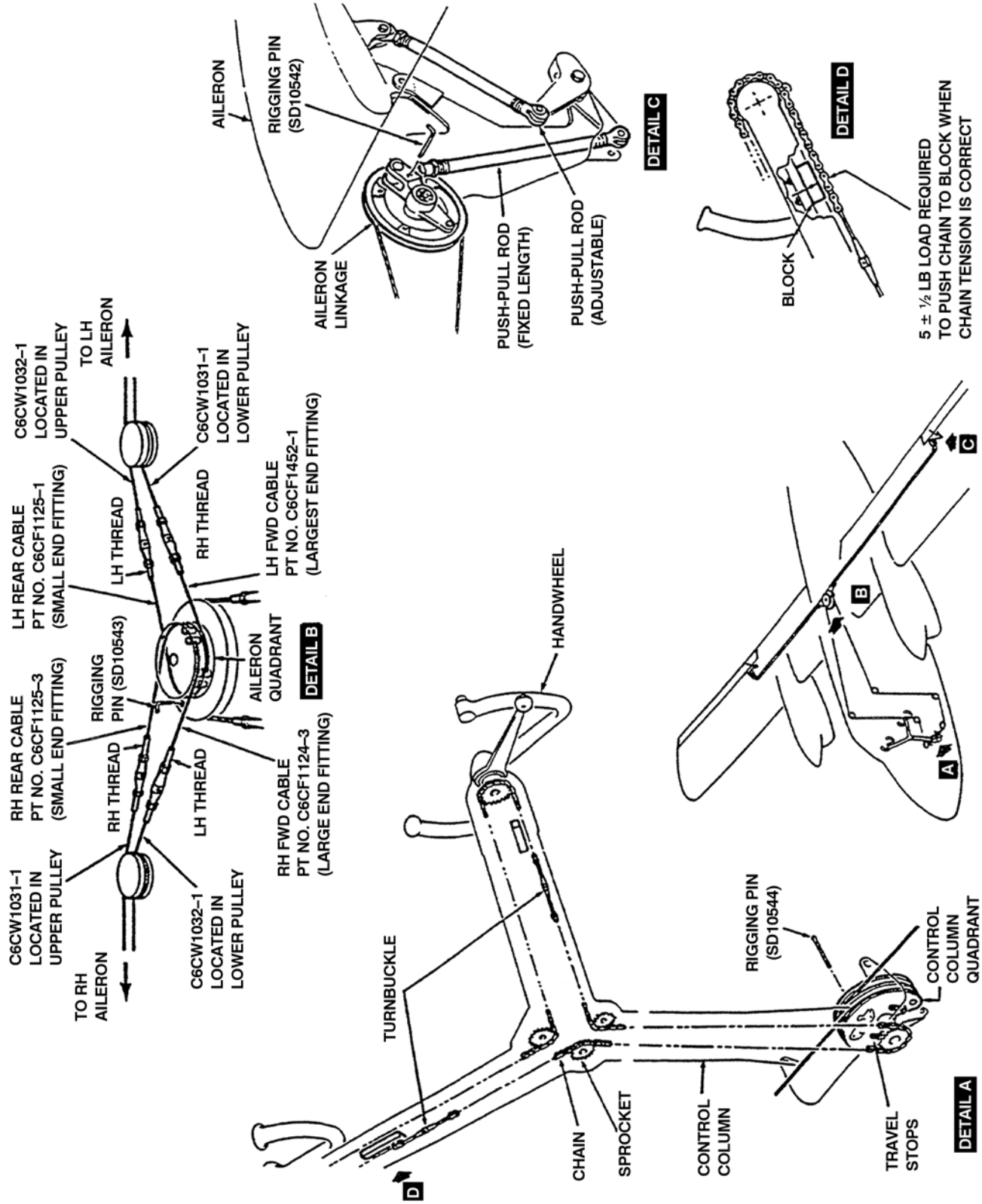
Note Flaps must be in fully up position when checking aileron travel.

10) Reconnect spring strut to co-pilot's handwheel. Adjust, if necessary, to maintain handwheel neutral position with no spring load.

11) Check aileron control surface clearances (refer to 27-00-00, Flight Controls – General Data), and aileron geared tab rigging (refer to 27-16-00, Aileron Geared Tab System – General Data).

12) Operate ailerons and check for full and free range of movement.

NOTE: Please see the TEMPORARY REVISION that revises this page.



Aileron System – Rigging
Figure 2

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR27-33**

Insert in 27-10-00 facing Page 5

AILERON CONTROL SYSTEM

REASON FOR ISSUE:

Added new Detail E to clarify installation of the aileron control cables on the Control Column Quadrant.

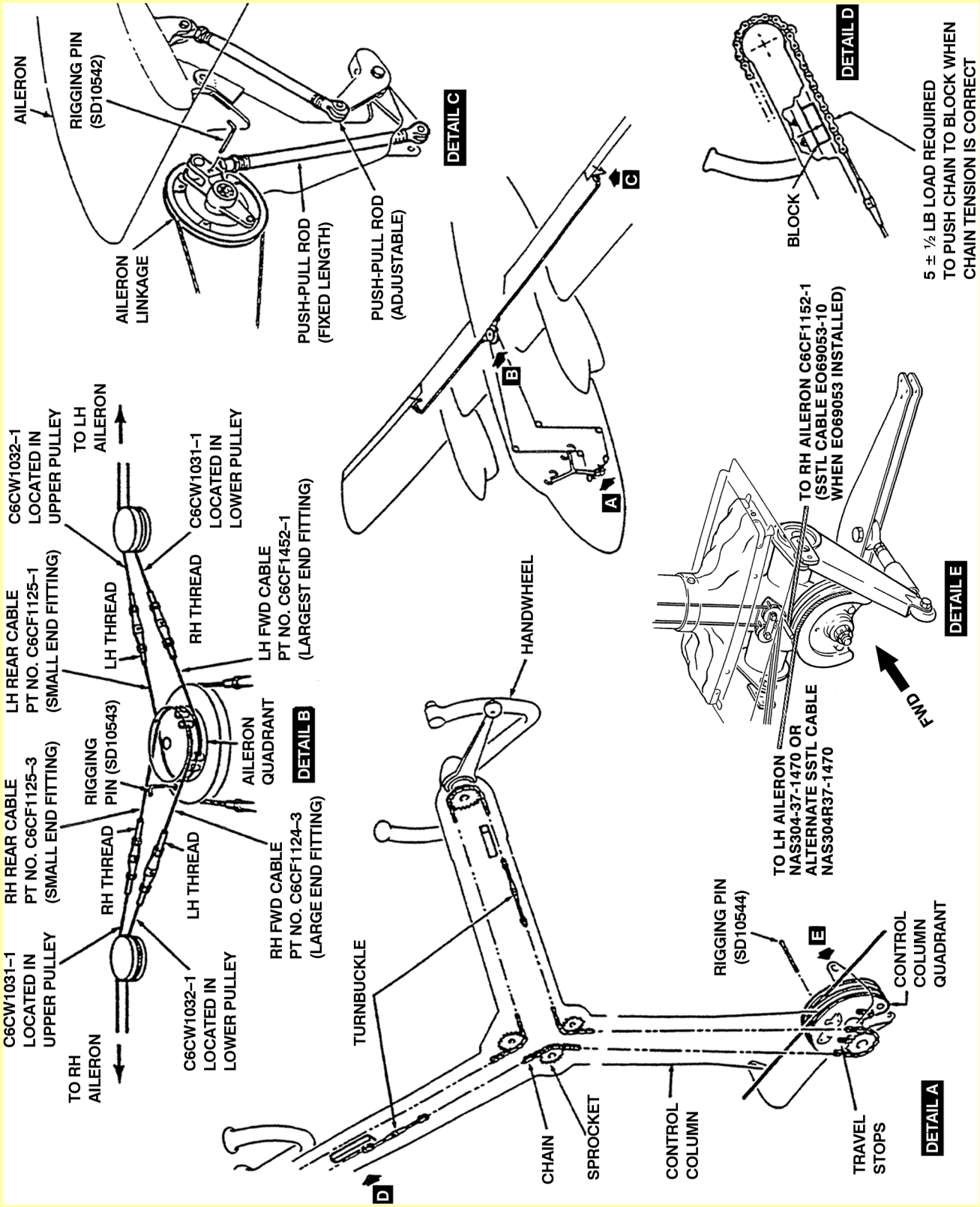
ACTION:

In Chapter 27, Section 27-10-00, ignore existing Figure 2 and insert revised Figure, as shown on Page 2 of this Temporary Revision.

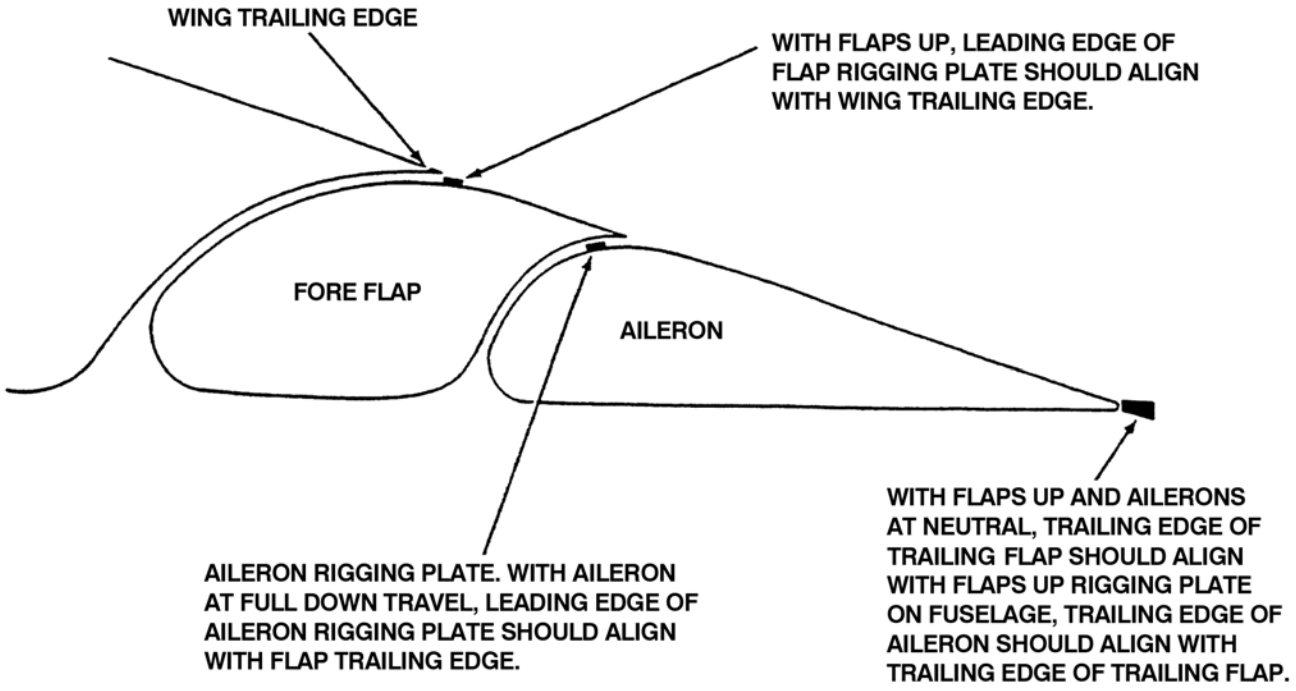
-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.



Aileron System – Rigging
Figure 2



Aileron Rigging Marks
Figure 3

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Aileron – Maintenance Practices

1. Removal/Installation

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. Remove Aileron

1) Disconnect push-pull rods at geared tab and aileron.

Note Do not alter push-pull rod adjustment as this will affect rigging.

2) On left aileron only, disconnect electrical wiring at aileron trim actuator, and sensor wiring at wing station 247.35.

3) Support aileron and remove four hinge bolts and bonding strips.

B. Install Aileron

CAUTION **ENSURE THAT AILERON IS OF THE CORRECT TYPE. AILERONS INSTALLED ON AIRCRAFT WITH WING FUEL TANKS (MOD S.O.O. 6247), ARE BALANCED IN ACCORDANCE WITH A SEPARATE STATIC BALANCE PROCEDURE.**

Note If the aileron to be installed is a replacement or repaired aileron, or if it has been painted or repainted, rebalancing must be carried out in accordance with the Structural Repair Manual PSM 1–6–3.

1) Examine replacement aileron for signs of damage (refer to Inspection Requirement Manual, Flight Controls, Ailerons and Tabs).

2) Support aileron and connect to aileron hinge brackets with four hinge bolts. Connect bonding strips on left aileron only, disconnect electrical wiring at aileron trim actuator, and sensor wiring at wing station 247.35 and secure hinge bolts with washers, nuts and cotter pins.

3) Connect push-pull rods at geared tab and aileron, install bolts and secure with nuts and cotter pins.

4) Check aileron rigging. (Refer to 27-10-00, Aileron Control System – General Data).

5) Check aileron gear tab rigging. (Refer to 27-16-00, Aileron Geared Tab System – General Data).

6) On left aileron only, check operation of aileron trim tab. (Refer to 27-13-00, Aileron Trim Tab System – General Data).

7) Check aileron control surface clearances are within limits. (Refer to 27-00-00, Flight Controls – General Data).

8) Operate ailerons and check for full and free range of movement.

9) Do operational test of the aileron angle position indicator as per 27-10-31, Aileron Angle Position Indicator – General Data, Para 3.B.

10) If adjustment is needed for the aileron angle position indicator adjust as per 27-10-31, Aileron Angle Position Indicator – General Data, Para 3.A.

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Control Column – General Data

1. General

The dual control column consists of an upper and lower section. The upper section is bolted to the lower section and consists of a fork assembly, two W-type handwheels, control chains and sprockets. The control chains driven by sprockets splined to the handwheel shafts, are routed over idler sprockets in the vee of the fork through the lower upright section to a bottom sprocket at the aileron quadrant. A torque tube is welded to the lower section carrying an elevator control arm, control column bearing assemblies, and the housing for the aileron quadrant, sprocket and stop lever. A centering device, in the form of a spring strut, is attached to a lever on the sprocket shaft of the co-pilot's handwheel, to overcome circuit friction. The left arm of the upper section includes the nose wheel steering lever and cable assembly (refer to 32-50-00, Nose Wheel Steering System – General Data). A luminous tape nose wheel position indicator (Mod 6/1396) fixed to the lever and cover provides a neutral position reference. A nose wheel position indicator light (Mod 6/1478) mounted on the cover provides illumination of the indicator (refer to Figure 1).

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Control Column Assembly

1) Remove hood assembly from base of control column and flight compartment floor.

2) Gain access to under flight compartment floor by removing forward fuselage access panels.

3) Disconnect pilot's and co-pilot's microphone switch wiring at terminal block.

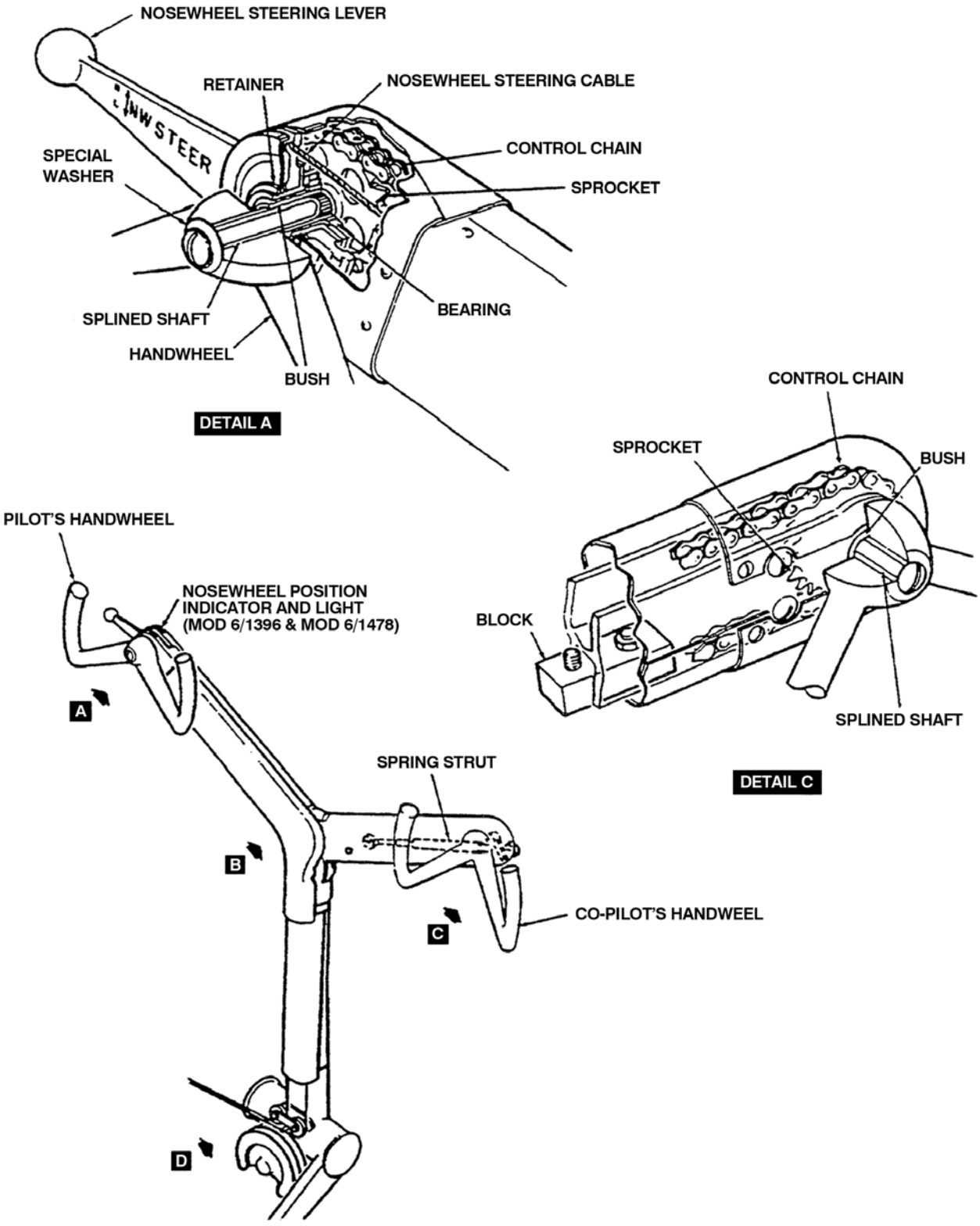
4) Disconnect nose wheel steering cables at turnbuckles and release cables from pulleys at airframe structure.

5) Remove cotter pin and pin guard from lower end of control column at aileron quadrant.

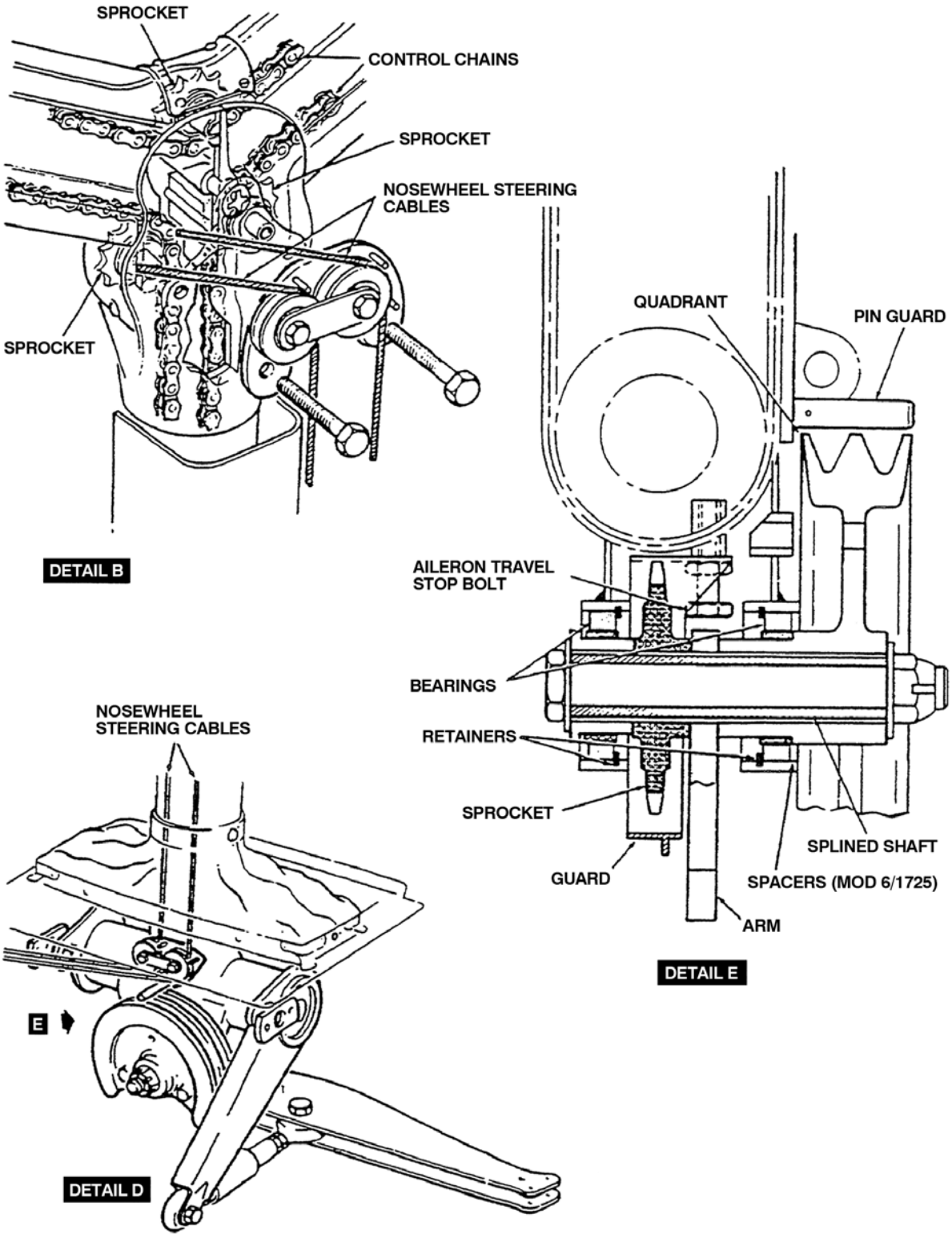
6) Slacken aileron control cables at turnbuckles and disconnect from aileron quadrant.

7) Disconnect connecting rod from control column elevator control arm.

8) Support control column and remove two bolts securing column to structure mounted housing assemblies, Remove control column from aircraft.



Control Column
Figure 1 (Sheet 1 of 2)



Control Column
Figure 1 (Sheet 2 of 2)

B. Install Control Column Assembly

Note Coat bolt shanks with corrosion preventive compound MIL–C–16173 Grade 2 or equivalent, before installation.

1) Position control column in aircraft and install bolts, washers and nuts to secure column to housing assemblies. Tighten nuts and secure with cotter pins.

2) Install bolt, washer and nut to secure connecting rod to elevator control arm. Tighten nut and secure with cotter pin.

3) Connect aileron control cables to aileron quadrant at base of control column; install pin guard and cotter pin.

4) Route nose wheel steering cables through pulleys on left side of aircraft structure, outboard of control column. If disturbed secure pulleys to bracket.

5) Connect nose wheel steering cable turnbuckles.

6) Connect pilot's and co-pilot's microphone switch cables at terminal block and test audio system (refer to 23-50-00, Audio Integration – General Data).

7) Check aileron control handwheel alignment (refer to 27-10-00, Aileron Control System – General Data).

8) Rig aileron control system (refer to 27-10-00, Aileron Control System – General Data).

9) Rig and test nose wheel steering system (refer to 32-50-00, Nose Wheel Steering System – General Data).

10) Rig elevator control system (refer to 27-30-00, Elevator Control System – General Data).

11) Secure underfloor access panels removed in Para 2.A.2).

12) Install hood assembly at base of control column.

13) Ensure control column and handwheel freedom of movement throughout fore and aft control column travel.

Aileron Angle Position Indicator – General data

1. General

A potentiometer is mounted on the port side aileron hinge arm at station W 247.35 and sends a signal of the aileron angle position through the DAU and onto the Flight Data Recorder (if installed).

2. Removal/Installation

Equipment and Materials:

- Corrosion preventative compound MIL–C–16173 Grade 2 or equivalent.

A. Remove the Aileron Angle Position Potentiometer

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

3) Disconnect the electrical connection at the terminal block.

4) Loosen the nut and screw that hold the lever assembly to the potentiometer shaft mounted in mounting bracket. Remove the lever assembly.

5) Remove the safety wire and the (3) screws and clamps which secure the potentiometer onto the mounting bracket.

6) Remove potentiometer.

B. Install the Aileron Angle Position Potentiometer

Note Coat all bolt shanks with corrosion preventative compound MIL–C–16173 Grade 2 or equivalent before installation.

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Ensure the FLIGHT CONTROLS INDICATION (A14) circuit breaker is pulled and fitted with safety clip.

3) Position the potentiometer on to the mounting bracket. Secure with the (3) clamps and screws. Safety wire screws.

4) Position the mounting bracket on aileron hinge arm and secure in place with (3) bolts, washers and nuts.

5) Locate the lever assembly on potentiometer shaft leaving a gap of 0.02 to 0.25 inch between the lever assembly and the potentiometer housing to minimize the push rod assembly misalignment.

6) Secure with a washer, nut and screw.

7) Install bolt through the push rod assembly end, install washer, and then bolt through the lever assembly end. Install washer, nut and cotter pin.

8) Adjust the aileron angle position potentiometer. (Refer to Para 3.A).

9) Reconnect the electrical connection at the terminal block.

10) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

3. Adjustment/Test

A. Adjustment Aileron Angle Position Potentiometer

Equipment and Material:

- DC Voltmeter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with safety clips.

4) Move flaps to the full up position.

5) Move ailerons to the neutral position and lock in place with aileron rigging pin.

6) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.

7) Loosen the screw and nut on the lever assembly at the potentiometer shaft enough that the shaft turns freely.

8) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker and monitor DC voltage from terminal 2 to aircraft ground.

9) Rotate potentiometer shaft until DC Voltage is 50% ± 2% of Ground Power Voltage (V_{GP}) (approximately 14 volts).

10) Reconnect pushrod and lever assembly to potentiometer shaft.

Note Do not rotate potentiometer shaft during this process.

11) Tighten screw and nut on the lever assembly leaving a gap of 0.02 to 0.05 of an inch between lever assembly and potentiometer housing.

12) Confirm voltmeter reading is unchanged.

13) Remove aileron rigging pin.

14) With voltmeter connected, function test aileron control position per the following:

a) Deflect ailerons to the full left position and confirm DC Voltage is 8.3% ± 2% of Ground Power Voltage (V_{GP}).

b) Deflect ailerons to the full right position and confirm DC Voltage is 79.2% ± 2% of Ground Power Voltage (V_{GP}).

15) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

16) Remove voltmeter and reconnect terminal 2 of the potentiometer electrical connections at the terminal block.

17) Remove safety clips and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

18) Deflect the ailerons full left and ensure DAU Maintenance Status CAS message does not activate.

19) Deflect the aileron full right and ensure DAU Maintenance Status CAS message does not activate.

20) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

B. Aileron Position Voltage Check

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with safety clips.

4) Move flaps to the full up position.

5) Move ailerons to the neutral position and lock in place with aileron rigging pin.

6) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.

7) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker and monitor DC voltage from terminal 2 to aircraft ground

8) If DC Voltage is not 50% ± 2% of V_{GP}, do adjustment steps per Para 3.A.

9) Remove aileron rigging pin.

10) With voltmeter connected, function test aileron control position per the following:

a) Deflect ailerons to the full left position and confirm DC Voltage is 8.3% ± 2% of Ground Power Voltage (V_{GP}).

b) Deflect ailerons to the full right position and confirm DC Voltage is 79.2% ± 2% of Ground Power Voltage (V_{GP}).

11) If DC Voltage does not correspond, do adjustment steps per Para 3.A.

12) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

13) Remove voltmeter and reconnect terminal 2 of the potentiometer electrical connections at terminal block.

14) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

15) Deflect the ailerons full left and ensure DAU Maintenance Status CAS message does not activate.

16) Deflect the ailerons full right and ensure DAU Maintenance Status CAS message does not activate.

17) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

Aileron Trim Tab System – General Data

1. General

An electrically-operated aileron trim tab, hinged to the trailing edge of the outboard end of the left aileron, is controlled by an aileron trim switch mounted on the trim console.

2. Description

The aileron trim tab system consists of an aileron trim tab, actuator, switch, and position transmitter and indicator. Refer to Figure 1 for details.

A. Trim Tab

The aileron trim tab is hinged to the left aileron top skin at the trailing edge. A hinge arm, riveted to the lower surface of the tab, connects to a push-pull rod which attaches to a trim tab actuator.

B. Trim Tab Actuator

The aileron trim tab actuator is an electro-mechanical unit powered by a DC motor, which, when operating, derives its power from the DC bus. Limit switches in the actuator operate at both extremities of travel to switch off the motor.

C. Trim Tab Switch

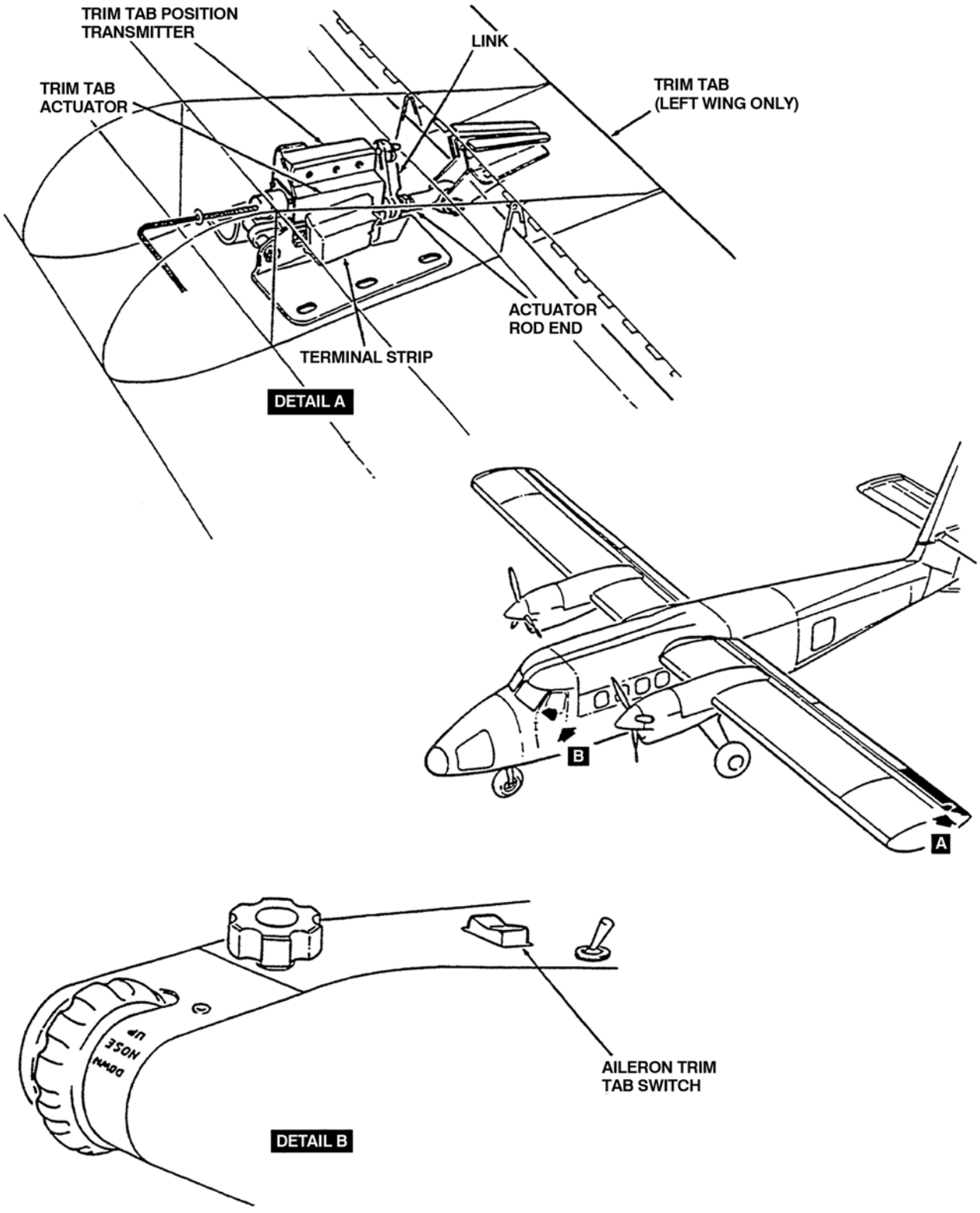
The aileron trim tab switch, mounted on the trim console, is marked LW DN and RW DN. It is a tilt type switch which, when released after making a trim selection, returns to its center off position to stop actuator movement at the selected position.

D. Trim Tab Position Transmitter

The aileron trim tab position transmitter installed on the trim tab actuator body, is an electro-mechanical unit which transmits aileron trim tab position to Apex, through the DAUs. The transmitter is a linear potentiometer with the contact arm connected to the actuator push-pull rod by a link. Extension or retraction of the actuator push-pull rod, causes a parallel movement of the contact arm resulting in a change of voltage being transmitted to the indicator.

E. Trim Tab Position Indicator

The aileron trim tab position is integrated into the APEX flight controls window and provides a visual indication of the trim tab position.



Aileron Trim Tab System
Figure 1

3. Operation

When the aileron trim tab switch is pressed to LW DN the actuator motor rotates to retract the actuator rod, which through the push-pull control rod, moves the trim tab down. The actuator will continue to retract until the actuator limit switch operates, or the trim switch is released; in either case the actuator will remain in the selected position until a further selection is made. During actuator rod movement, the position transmitter will have relayed the actuator position to Apex in respect of the amount of trim tab movement. The same sequence of events will occur with a RW DN selection, except that the actuator rod will extend, and the trim tab will move up.

4. Adjustment/Test

A. Rig Aileron Trim Tab System

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Operate aileron to neutral position.

4) Operate trim tab until trailing edges of aileron and trim tab align.

5) Check that Roll trim display indication on the Apex flight controls window reads centered.

6) If Roll trim display indication does not read centered, disconnect actuator rod from trim tab, operate actuator until centered, adjust actuator rod length and reconnect to trim tab.

Note If a zero reading is not obtained with the actuator in its mid-travel position, the trim tab position transmitter should be readjusted (refer to 27-13-21, Aileron Trim Tab Position Transmitter – Maintenance Practices, Para 2.).

7) Do a function test for aileron trim tab position potentiometer as per 27-13-21, Aileron Trim Tab Position Transmitter – Maintenance Practices, Para 3.A.)

8) Operate trim tab through its ranges of travel and check for full and free range of movement.

9) Check trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

10) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

5. Inspection/Check

A. Free Play Check – Aileron Trim Tab

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Operate ailerons to neutral position.

3) Operate trim tab to neutral position.

4) Move trim tab manually and check that free play (backlash), measured at trailing edge, does not exceed 0.10–inch.

Note Measurement to be taken adjacent to control rod attachment point.

5) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Aileron Trim Tab Actuator – Maintenance Practices

1. Removal/Installation

   A. Remove Aileron Trim Tab Actuator Assembly

      1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      2) Operate aileron trim tab switch to LW DN until trim tab actuator is fully retracted.

      3) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      4) Disconnect actuator rod end from aileron trim tab.

      5) Supporting actuator, remove six screws securing actuator bracket assembly to aileron.

      6) Gently lower actuator and disconnect electrical wiring from actuator. Remove actuator.

   B. Install Aileron Trim Tab Actuator Assembly

      1) Check replacement actuator for signs of damage and ensure actuator rod is fully retracted.

      2) Position actuator and connect electrical wiring.

      3) Secure actuator to aileron using six screws.

      4) Secure actuator rod end to trim tab using bolt, washer and nut. Secure nut with cotter pin.

      5) Rig aileron trim tab system (refer to 27-13-00, Aileron Trim Tab System – General Data).

      6) Do a function test for aileron trim tab position potentiometer as per section 27-13-21, Aileron Trim Tab Position Transmitter – Maintenance Practices, Para 3.A.

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Aileron Trim Tab Position Transmitter – Maintenance Practices

1. Description

A linear potentiometer is mounted on the outboard, port side of the aileron at approximately station 571.978 and sends a signal of the aileron trim position through the DAU and onto the Flight Data Recorder (if installed) and the aileron trim position is indicated on the system screen.

2. Removal/Installation

A. Remove Aileron Trim Tab Position Indicator Potentiometer

1) With aircraft power OFF, remove trim tab actuator assembly (refer to 27-13-11, Aileron Trim Tab Actuator – Maintenance Practices).

2) Remove actuator rod end and locknut.

3) Remove and retain nut securing transmitter lever. Slacken lever clamp screw and, by turning potentiometer shaft, unscrew lever from potentiometer shaft. Remove lever.

4) Remove the four screws securing bracket and actuator connector or, if same actuator is to be used, remove the two screws securing potentiometer to transmitter bracket. Remove potentiometer.

Note Do not discard attachment items until new potentiometer has been installed on new or replacement actuator (refer to Para 2.B).

B. Install Aileron Trim Tab Position Indicator Potentiometer

Note Coat all bolt shanks with corrosion preventative compound MIL–C–16173 Grade 2 or equivalent before installation.

1) If fitted, remove rod end, rod end locknut and lever nut from new or replacement actuator.

2) Using a 28 Volt DC power supply connected to the appropriate terminals, operate actuator between fully retracted and fully extended, measure overall travel. Operate actuator to the mid-travel position.

3) If actuator is not provided with transmitter attachment bracket, remove the four screws securing actuator connector to actuator body, and install bracket, using bracket and four screws (AN–500A–2–4) retained from actuator previously removed. Tighten and lockwire screws.

4) Install potentiometer to bracket with two screws. Ensure potentiometer shaft is parallel with actuator shaft in both planes. Tighten and lockwire screws.

5) Locate lever on actuator shaft, screwing transmitter potentiometer shaft into lever, until lever contacts actuator shaft shoulder. Install and tighten lever nut, and lockwire to lever.

6) Using a volt/ohm meter, adjust potentiometer shaft (with actuator maintained in the mid-travel position) to give an equal resistance reading between terminals 1 – 2 and 2 – 3. When satisfactory, tighten lever clamp screw.

7) Screw rod end locknut and rod end onto actuator shaft. With actuator fully retracted, adjust rod end until distance between bolt hole centers is 5.19 inches. Tighten rod end locknut.

8) Install trim tab actuator assembly (refer to 27-13-11, Aileron Trim Tab Actuator – Maintenance Practices).

9) Do a test of the aileron trim tab system as per 27-13-00, Aileron Trim Tab System – General Data, Para 4.A.

3. Adjustment/Test

A. Function Test Aileron Trim Tab Position Potentiometer

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Deflect aileron trim to the RW down position (Reference 15 degrees) and note Roll trim display indication correlates and does not become invalid (indicated by an amber 'X').

4) Deflect aileron trim to the neutral position (Reference 0 degrees) and note the Roll trim display indication correlates.

5) Deflect aileron trim to the LW down position (Reference 15 degrees) and note the Roll trim display indication correlates and does not become invalid (indicated by an amber 'X').

6) If the Roll trim display does not match the aileron trim surface position, repeat the adjustment steps in Para 2..

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Aileron Trim Tab – Maintenance Practices

1. Removal/Installation

   A. Remove Trim Tab

      1) Disconnect trim actuator rod end from trim tab.

      2) Cut and remove locking wire securing end of hinge pin to aileron.

      3) Support tab and withdraw hinge pin.

   B. Install Trim Tab

      1) Lightly lubricate hinge pin. Refer to 12-20-10, Lubrication – General Data for lubricant specification.

      2) Align hinge components on tab and aileron.

      3) Insert hinge pin from outboard end of aileron and lockwire end of hinge to holes in adjacent aileron structure.

      4) Connect trim actuator rod end to tab with bolt, washer, nut and cotter pin.

      5) Check trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

      6) Operate trim tab and check for full and free range of movement.

         Note If tab or aileron has been repaired, or has had additional paint applied, the aileron assembly must be balanced as detailed in Structural Repair Manual PSM 1–6–3.

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Aileron Geared Tab System – General Data

1. General

The aileron geared tab is hinged to the trailing edge of the inboard end of each aileron. A tab hinge arm, riveted to the inboard end of the tab, is connected to the flap hinge bracket by adjustable control rods which are linked to the outboard fore flap hinge arm by a lever link.

2. Adjustment/Test

A. Rig Aileron Geared Tab

Refer to Figure 1 for details.

Note Before attempting to rig the aileron geared tab, ensure that aileron and wing flap systems are correctly rigged (refer to 27-10-00, Aileron Control System – General Data and 27-50-00, Wing Flaps System – General Data).

1) Lower flaps to full down position.

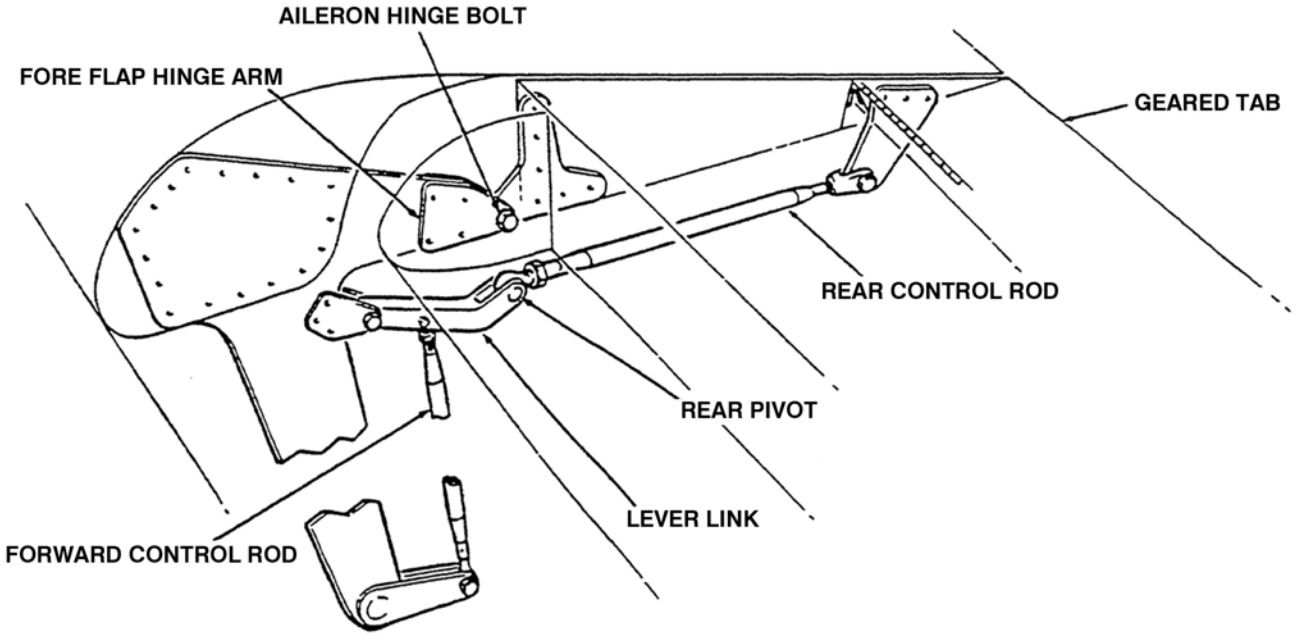
2) With ailerons in neutral, check that lever link rear pivot is aligned with aileron hinge center and that tab is in neutral.

3) Adjust forward control rod, as necessary, to align the pivot, then adjust rear control rod to align tab.

4) Secure control rod nuts and locknuts where disturbed, and install cotter pins as necessary.

5) Check geared tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

6) Operate ailerons and check for full and free range of movement.



Aileron Geared Tab Control
Figure 1

3. Inspection/Check

A. Free Play Check – Aileron Geared Tab

1) Operate ailerons (and geared tab) to neutral position.

2) Move geared tab manually and check that free play (backlash), measured at trailing edge, does not exceed 0.10–inch.

Note Measurement to be taken adjacent to control rod attachment point.

Aileron Geared Tab – Maintenance Practices

1. Removal/Installation

   A. Remove Geared Tab

      1) Disconnect control rod at geared tab.

      2) Cut and remove locking wire securing end of hinge pin to aileron.

      3) Support tab and withdraw hinge pin.

   B. Install Geared Tab

      1) Lightly lubricate hinge pin. Refer to 12-20-10, Lubrication – General Data for lubricant specification.

      2) Align hinge components on tab and aileron.

      3) Insert hinge pin from inboard end of aileron and lockwire end of hinge to holes in adjacent aileron structure.

      4) Connect control rod to tab with bolt, washer, nut and cotter pin.

      5) Check geared tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

      6) Operate ailerons and check geared tab for full and free range of movement.

         Note If tab or aileron has been repaired, or has had additional paint applied, the aileron assembly must be balanced as detailed in Structural Repair Manual PSM 1–6–3.

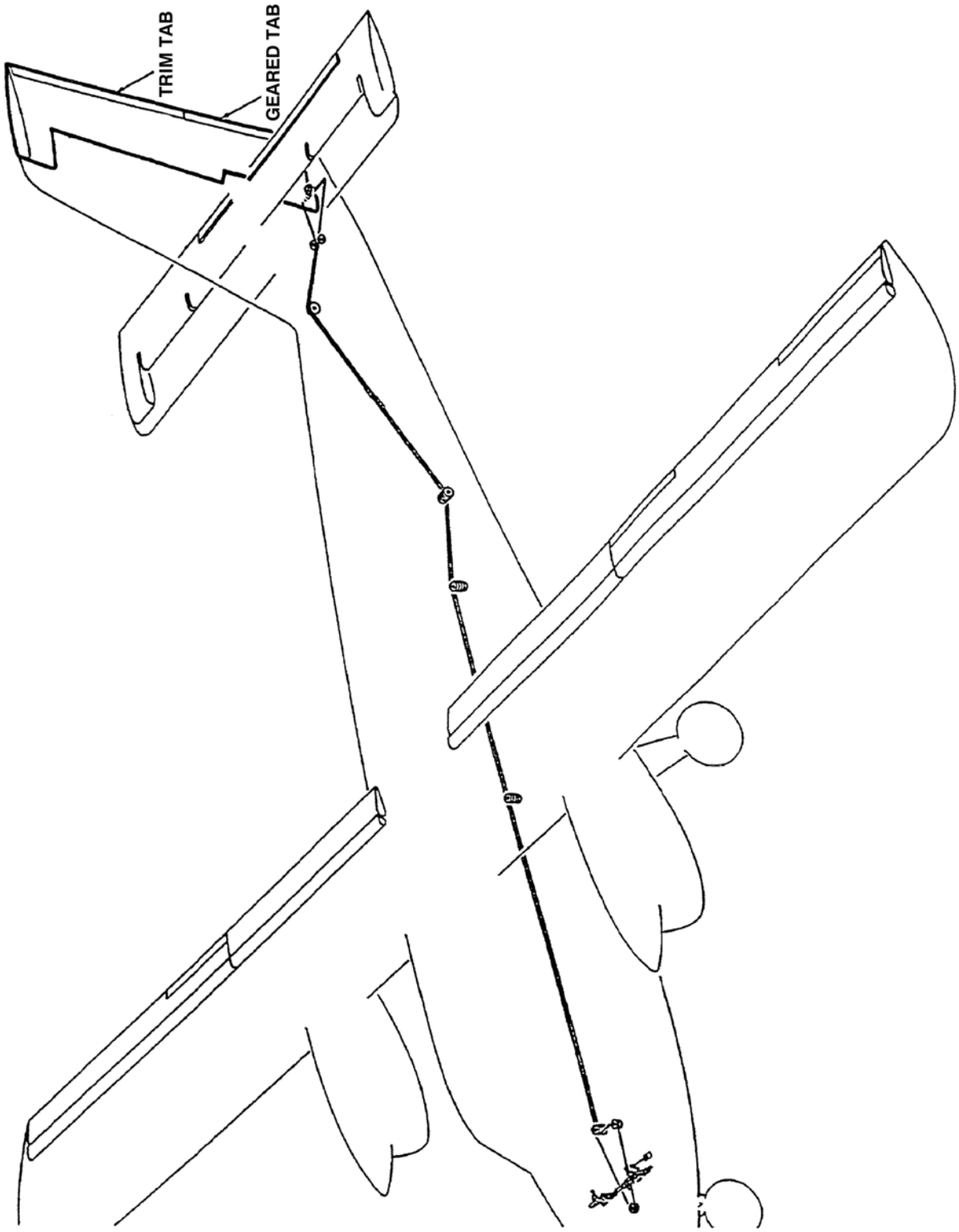
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Rudder Control System – General Data

1. General

The rudder system provides operation of the rudder from the pilot's and co-pilot's positions. It consists of two rudder pedal assemblies, a torque tube, a rudder control quadrant, rudder lever, and rudder. The rudder pedals transmit the movement through the torque tube and rudder control quadrant located under the flight compartment floor. From the quadrant, two cables convey the movement to the rudder lever mounted on the rudder. The rudder cables are routed along the right side of the aircraft using the same banks of pulleys as for the elevator system, then change direction into the center of the rear fuselage. The rudder is hinged to the vertical stabilizer and rear fuselage at three points. Left and right rudder travel is limited by rudder stops, but, because left and right travel is unequal, the stops are of a different size. As an aid to identification the left stop is coloured blue, and the right stop grey. Two tabs are hinged to the rudder trailing spar, the upper being the trim tab, and the lower the geared tab (refer to 27-23-00, Rudder Trim Tab System – General Data and 27-26-00, Rudder Geared Tab System – General Data). Refer to Figure 1 for details.

Note For general cable rigging instructions, refer to 27-00-00, Flight Controls – General Data, Para 2.A.



Rudder System
Figure 1

2. Adjustment/Test

A. Rig Rudder Control System

Refer to Figure 2 for details.

1) Set rudder pedals to highest position by pulling out adjustment catch, and engaging in bottom hole.

2) Engage rudder gust lock.

3) Using scale SD12568, check rudder neutral position by measuring 18.73 ± 0.06 inches between centers of index screw and bolt in rudder lever (dimension L, Figure 2).

4) If measurement is not within limits, adjust turnbuckles in rear fuselage as necessary to obtain dimension; tension cables to correct values. (Refer to 27-00-00, Flight Controls – General Data).

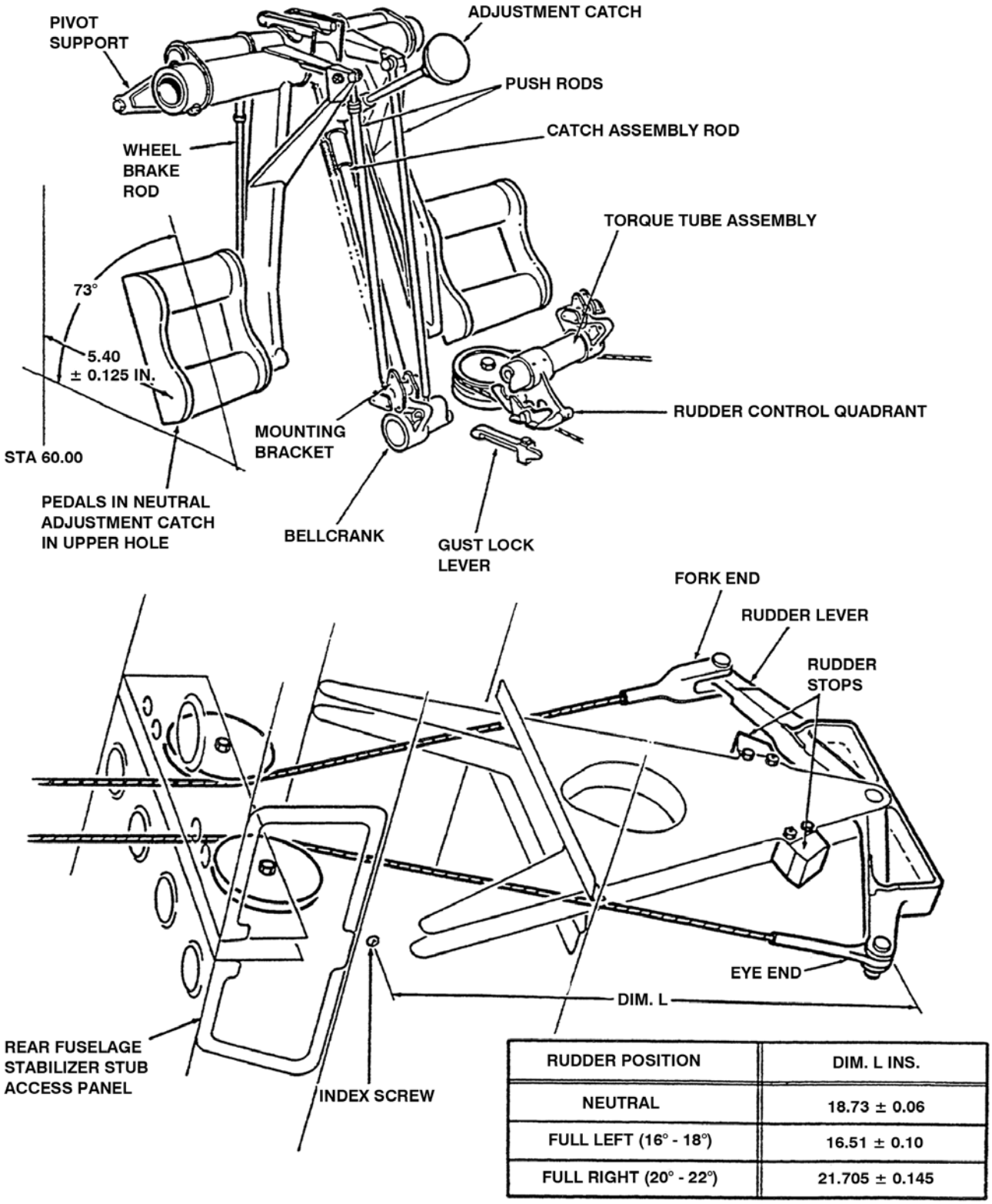
5) Set rudder pedal catch in highest hole, check with rudder in neutral, pedals align and angle of pedals is 73°, with a measurement of 5.40 ± 0.125 inches between bulkhead at station 60.00 and rudder pedal center pivots. Adjust push rods as necessary. Tighten nuts if disturbed.

6) Disengage rudder gust lock. Lock turnbuckles using locking clips.

7) Using scale SD12568, check dimension L (Figure 2) for full left and right rudder positions.

8) Operate rudder and check for full and free range of movement.

9) Repeat Para 2.A.7) and Para 2.A.8) with rudder pedals adjusted to each of the other three positions.



RUDDER POSITION	DIM. L INS.
NEUTRAL	18.73 ± 0.06
FULL LEFT (16° - 18°)	16.51 ± 0.10
FULL RIGHT (20° - 22°)	21.705 ± 0.145

Rudder System Rigging
Figure 2

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR27-35**

Insert in 27-20-00 after Page 4
RUDDER CONTROL SYSTEM

REASON FOR ISSUE:

To introduce Viking Part Numbers in Table 1 - Rudder Cable Assy - Flight Compartment and Passenger Cabin.

ACTION:

In Chapter 27, Section 27-20-00 ignore existing Table 1 and insert revised Table, as shown on Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

Table 1 Rudder Cable Assy - Flight Compartment and Passenger Cabin

		CARBON STEEL	STAINLESS STEEL
Rudder quadrant inboard sheave to station 455.00	Pre Mod 6/2483	NAS305–34–4201	NAS305R34–4201
	Post Mod 6/2483	C6CF1511–3	C6CF1511–7
Rudder quadrant outboard sheave to station 455.00	Pre Mod 6/2483	NAS305–35–4266	NAS305R35–4266
	Post Mod 6/2483	C6CF1511–1	C6CF1511–5

NOTE: Please see the TEMPORARY REVISION that revises this page.

3. Inspection/Check

A. Job Set-up

1) Either pull and clip breakers required to isolate affected circuits or placard and select aircraft electrical power to OFF.

2) Obey all relevant WARNINGS and CAUTIONS detailed in Maintenance Manual PSM 1–64–2, Chapters 12 and 24.

3) Placard flight controls in flight compartment.

B. Flight Compartment

1) Open access doors 15 to gain access to rudder cables (refer to Table 1 — Rudder Cable Assy – Flight Compartment and Passenger Cabin and Figure 2).

2) Do a visual inspection of control cables (non-jacketed) visible at this section of fuselage (refer to Table 1 — Rudder Cable Assy – Flight Compartment and Passenger Cabin and Figure 2). Observe carefully areas where cables pass over pulleys, quadrants, drums, or through fairleads.

3) Replace cables not acceptable to criteria in Inspection of Control Cables (refer to 27-00-00, Flight Controls – General Data). Make sure the new cable part numbers are identical to the discarded cable part numbers (refer to Table 1 — Rudder Cable Assy – Flight Compartment and Passenger Cabin).

4) Lubricate new or existing non-jacketed Carbon Steel cables (refer to 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data).

Note Do not lubricate stainless steel cables.

5) Replace cable as required.

6) Install and rig replaced cable assemblies (refer to 27-00-00, Flight Controls – General Data).

7) Close access doors 15.

Table 1 Rudder Cable Assy – Flight Compartment and Passenger Cabin

	CARBON STEEL	STAINLESS STEEL
Rudder quadrant inboard sheave to station 455.0	NAS305–34–4201	NAS305R34–4201
Rudder quadrant outboard sheave to station 455.0	NAS305–35–4266	NAS305R35–4266

C. Passenger Cabin

1) Remove panel 43 (refer to Figure 1, 12-00-00, Servicing – General Data) to gain access to rudder cables and pulleys (refer to Table 1 — Rudder Cable Assy – Flight Compartment and Passenger Cabin and Figure 2).

2) Release, attach string and pull back rudder cables to allow visual inspection (refer to Table 1 — Rudder Cable Assy – Flight Compartment and Passenger Cabin, Figure 2 and 27-00-00, Flight Controls – General Data Para 3.A).

3) Observe carefully areas where cables pass over pulleys, quadrants, drums, or through fairleads.

4) Replace cables not acceptable to criteria in Inspection of Control Cables (refer to 27-00-00, Flight Controls – General Data). Make sure the new cable part numbers are identical to the discarded cable part numbers per Table 1 — Rudder Cable Assy – Flight Compartment and Passenger Cabin.

5) Lubricate new or existing non-jacketed Carbon Steel cables (refer to 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data).

Note Do not lubricate stainless steel cables.

6) Install and rig cable assemblies (refer to 27-00-00, Flight Controls – General Data).

7) Install panel 43.

D. Rear Fuselage and Tail Area

CAUTION **WHEN INSTALLING ELEVATOR AND RUDDER PRIMARY FLIGHT CONTROL CABLES, CARGO AREA FLOOR BOARDS (BETWEEN STA 332.0 TO 376.0) AND AFT CARGO AREA FLOOR BOARDS (AFT OF STA 376.0), MUST BE REMOVED TO ENSURE PROPER INSTALLATION. THIS WILL REDUCE THE POSSIBILITY OF MISSRIGGING AND CABLE CROSSING, WHICH COULD LEAD TO ACCELERATED CABLE WEAR TO THE ELEVATOR AND RUDDER PRIMARY FLIGHT CONTROL SYSTEMS.**

1) Remove floor panels to gain access to rudder cable assemblies in baggage compartment rear fuselage.

2) Remove panels 11 and 12 (refer to 12-00-00, Servicing – General Data, Figure 1) to gain access to rudder cable assemblies in the tail area.

3) Do a visual inspection of control cables (non-jacketed) visible at this section of fuselage (refer to Table 2 — Rudder Cable Assy – Rear Fuselage and Tail Area, Figure 2 and 27-00-00, Flight Controls – General Data Para 3.A). Observe carefully areas where cables pass over pulleys, quadrants, drums, or through fairleads and cable ends. Place special emphasis on elevator and rudder cable inspection near cable cluster at station 376 (refer to Figure 2).

4) Replace cables not acceptable to criteria in Inspection of Control Cables (refer to 27-00-00, Flight Controls – General Data). Make sure the new cable part numbers are identical to the discarded cable part numbers per Table 2 — Rudder Cable Assy – Rear Fuselage and Tail Area.

5) Lubricate new or existing non-jacketed Carbon Steel cables only (refer to 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data).

Note Do not lubricate stainless steel cables.

6) Install and rig replaced cable assemblies (refer to 27-00-00, Flight Controls – General Data).

7) Install baggage compartment floor panels and panels 11 and 12.

Table 2 Rudder Cable Assy – Rear Fuselage and Tail Area

	CARBON STEEL	STAINLESS STEEL
Station 455.0 to right-hand side rudder control attachment	C6CF1150–1	E.O. 69053–6
Station 455.0 to left-hand side rudder control attachment	C6CF1151–1	E.O. 69053–7

E. Close-out:

1) Remove placards on the flight controls in flight compartment.

2) Obey all relevant WARNINGS and CAUTIONS detailed in Maintenance Manual PSM 1–64–2, Chapters 12 and 24.

3) Either remove clip and push breakers that were required to isolate affected circuits or remove placards and select aircraft electrical power to ON.

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TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR27-34**

Insert in 27-20-11 before Page 1

RUDDER

REASON FOR ISSUE:

To introduce new sling SD12517 for hoisting the rudder.

ACTION:

In Chapter 27, Section 27-20-11, ignore existing Section 27-20-11 and insert revised section, as shown on Pages 3 and 4 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

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Rudder – Maintenance Practices

1. Removal/Installation

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. Remove Rudder

1) Remove fairings at bottom of vertical stabilizer.

2) Disconnect electrical wiring to tail light, at quick disconnect (refer to 33-00-00, Lights – General Data) and sensor wiring at WL 50.15 FWD of rudder hinge line.

3) Slacken and disconnect rudder control cables at rudder lever and slacken rudder trim cables at turnbuckles in rear fuselage.

4) Release rudder trim cables from screw jack.

5) Disconnect geared tab control rod and link (refer to 27-26-11, Rudder Geared Tab – Maintenance Practices).

6) Attach sling SD12517 to rudder and take weight (refer to 07-00-00, Lifting and Shoring – General Data).

7) Support rudder and remove lower and center hinge bolts and bonding. At top hinge position, remove two bolts from upper hinge plate. Lower rudder to ground and remove upper hinge attachment bolt to remove upper hinge plate.

8) Disconnect rudder trim tab connecting rod and remove trim screw jack.

B. Install Rudder

Note If the rudder to be installed has been repaired, painted or repainted, check the balance in accordance with Structural Repair Manual PSM 1–6–3.

1) Examine replacement rudder for signs of damage.

2) Install hinge plate to rudder upper hinge using attachment bolt.

3) Install rudder trim screw jack in rudder.

4) Attach sling SD12517 to rudder (refer to 07-00-00, Lifting and Shoring – General Data).

5) Position rudder and secure to vertical stabilizer lower and center hinges, using bolts, washers and nuts.

Note Pass bolt through bonding jumper terminal before inserting in lug; use two washers when installing each bolt, one between bonding jumper terminal and lug, sensor wiring at WL 50.15 FWD of rudder hinge line, and one under nut.

6) Secure upper hinge plate to vertical stabilizer, using two bolts, washers and nuts.

7) Tighten all rudder attachment bolts.

8) Install trim tab connecting rod.

9) Connect geared tab control rod and link (refer to 27-26-11, Rudder Geared Tab – Maintenance Practices).

10) Connect rudder cables to rudder lever, and secure trim cables to trim screw jack.

11) Rig rudder control system (refer to 27-20-00, Rudder Control System – General Data).

12) Rig rudder trim tab system (refer to 27-23-00, Rudder Trim Tab System – General Data).

13) Check rudder and rudder trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

14) Connect tail light electrical wiring at quick disconnect (refer to 33-00-00, Lights – General Data).

15) Operate rudder and rudder trim controls and check for full and free range of movement.

16) Do an operational test for the rudder angle position indicator as per section 27-20-31, Rudder Angle Position Indicator – General Data, Para 3.B.

17) If adjustment is required for the rudder angle position potentiometer adjust as per 27-20-31, Rudder Angle Position Indicator – General Data, Para 3.A.

Rudder – Maintenance Practices

1. Removal/Installation

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. Remove Rudder

1) Remove fairings at bottom of vertical stabilizer.

2) Disconnect electrical wiring to tail light, at quick disconnect (refer to 33-00-00, Lights – General Data) and sensor wiring at WL 50.15 FWD of rudder hinge line.

3) Slacken and disconnect rudder control cables at rudder lever and slacken rudder trim cables at turnbuckles in rear fuselage.

4) Release rudder trim cables from screw jack.

5) Disconnect geared tab control rod and link (refer to 27-26-11, Rudder Geared Tab – Maintenance Practices).

6) Support rudder and remove lower and center hinge bolts and bonding. At top hinge position, remove two bolts from upper hinge plate. Lower rudder to ground and remove upper hinge attachment bolt to remove upper hinge plate.

7) Disconnect rudder trim tab connecting rod and remove trim screw jack.

B. Install Rudder

Note If the rudder to be installed has been repaired, painted or repainted, check the balance in accordance with Structural Repair Manual PSM 1–6–3.

1) Examine replacement rudder for signs of damage.

2) Install hinge plate to rudder upper hinge using attachment bolt.

3) Install rudder trim screw jack in rudder.

4) Position rudder and secure to vertical stabilizer lower and center hinges, using bolts, washers and nuts.

Note Pass bolt through bonding jumper terminal before inserting in lug; use two washers when installing each bolt, one between bonding jumper terminal and lug, sensor wiring at WL 50.15 FWD of rudder hinge line, and one under nut.

5) Secure upper hinge plate to vertical stabilizer, using two bolts, washers and nuts.

6) Tighten all rudder attachment bolts.

7) Install trim tab connecting rod.

8) Connect geared tab control rod and link (refer to 27-26-11, Rudder Geared Tab – Maintenance Practices).

9) Connect rudder cables to rudder lever, and secure trim cables to trim screw jack.

10) Rig rudder control system (refer to 27-20-00, Rudder Control System – General Data).

NOTE: Please see the TEMPORARY REVISION that revises this page.

11) Rig rudder trim tab system (refer to 27-23-00, Rudder Trim Tab System – General Data).

12) Check rudder and rudder trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

13) Connect tail light electrical wiring at quick disconnect (refer to 33-00-00, Lights – General Data).

14) Operate rudder and rudder trim controls and check for full and free range of movement.

15) Do an operational test for the rudder angle position indicator as per section 27-20-31, Rudder Angle Position Indicator – General Data, Para 3.B.

16) If adjustment is required for the rudder angle position potentiometer adjust as per 27-20-31, Rudder Angle Position Indicator – General Data, Para 3.A.

Rudder Pedal Assembly – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Rudder Pedal Assembly

1) Remove clevis pins from wheel brake lever rods, located at bulkhead station 60.00.

2) Disconnect left and right push rods from rudder torque tube bellcrank.

3) Disconnect pedal adjustment catch assembly at lower bolt.

4) Support rudder pedal assembly and remove bolts securing support pivots to bulkhead brackets.

5) Remove rudder pedal assembly from aircraft.

B. Install Rudder Pedal Assembly

1) Position rudder pedal assembly in aircraft and support.

2) Install bolts, spacers, washers and nuts to secure support pivots to bulkhead brackets. Position additional spacing washers as required to obtain maximum end play of 0.03 inch. Tighten nuts and secure with cotter pins.

3) Connect pedal adjustment catch assembly lower end using bolt, washer and nut. Catch assembly must be free to rotate on bolt. Secure nut with cotter pin.

4) Using clevis and cotter pins connect wheel brake lever rods to levers at bulkhead station 60.00.

5) Position rudder pedals to obtain a dimension of 5.4 ± 0.125 inches between bulkhead at station 60.00 to center of pedal pivots, with adjustment catch in upper hole.

6) With rudder set to neutral and pedals set as in Para 1.B.5), connect left and right push rods to rudder torque tube bellcrank. Position washer between rod and bellcrank to ensure clearance of parking brake tube.

7) Rig rudder control system (refer to 27-20-00, Rudder Control System – General Data).

8) Adjust wheel brake rods forward of rudder pedals to obtain pedal angle of 73° in the neutral position.

Note Angle measured between a line taken from heel and toe rubbers and flight compartment floor.

9) Check wheel brake rigging (refer to 32-40-00, Wheels and Brakes (Mod 6/2002) – General Data).

2. Inspection/Check

A. Wear Check – Rudder Push Rod and Wheel Brake Rod Bearings

Refer to Figure 1 for details.

1) Gain access to pilot's LH and RH rudder push rods.

2) Remove cotter pins, nuts, washers and bolts attaching upper end of rudder push rods to rudder pedal support assembly.

3) Check for bearing wear at lower end of rudder push rods, as indicated by play between ball and housing.

4) If evidence of wear exists, omit Para 2.A.5) through Para 2.A.10) and proceed as follows:

a) Remove push rods from aircraft.

b) Check that play between ball and housing, in upper and lower bearings, does not exceed 0.020–inch and that radial play does not exceed 0.006–inch.

c) Install serviceable pushrods in aircraft. Connect with bolts, washers and nuts and lock with cotter pins.

Note Washer to be positioned between push rod and rudder torque tube bellcrank, as required, to ensure push rod clearance of parking brake tube.

5) If no evidence of wear exists, connect upper end of push rods to rudder pedal support assembly, using bolts, washers and nuts. Position one washer between support assembly and each rod end and remaining washer(s) under nut.

Note Do not lock with cotter pins at this stage.

6) Remove cotter pins, nuts, washers and bolts attaching lower end of rudder push rods to rudder torque tube bellcrank.

7) Check for bearing wear at upper end of rudder push rods (refer to Para 2.A.3)).

8) If evidence of wear exists, proceed as follows:

a) Remove push rods from aircraft.

b) Check that play between ball and housing does not exceed 0.020–inch, and radial play does not exceed 0.006–inch.

9) If no evidence of wear exists, connect lower end of push rods to rudder pedal support assembly, using bolts, washers and nuts.

Note Washer to be positioned between push rod and rudder torque tube bellcrank, as required, to ensure push rod clearance of parking brake tube.

10) Install cotter pins and lock LH and RH push rod upper and lower attachment bolts and nuts.

11) Remove cotter pins, washers and clevis pins attaching upper end of LH and RH wheel brake rods to brake lever assembly.

12) Check for bearing wear at lower end of wheel brake rods (refer to Para 2.A.3)).

13) If evidence of wear exists, omit Para 2.A.14), and proceed as follows:

a) Remove wheel brake rods from aircraft.

b) Check that play between ball and housing does not exceed 0.020–inch, and radial play does not exceed 0.006–inch.

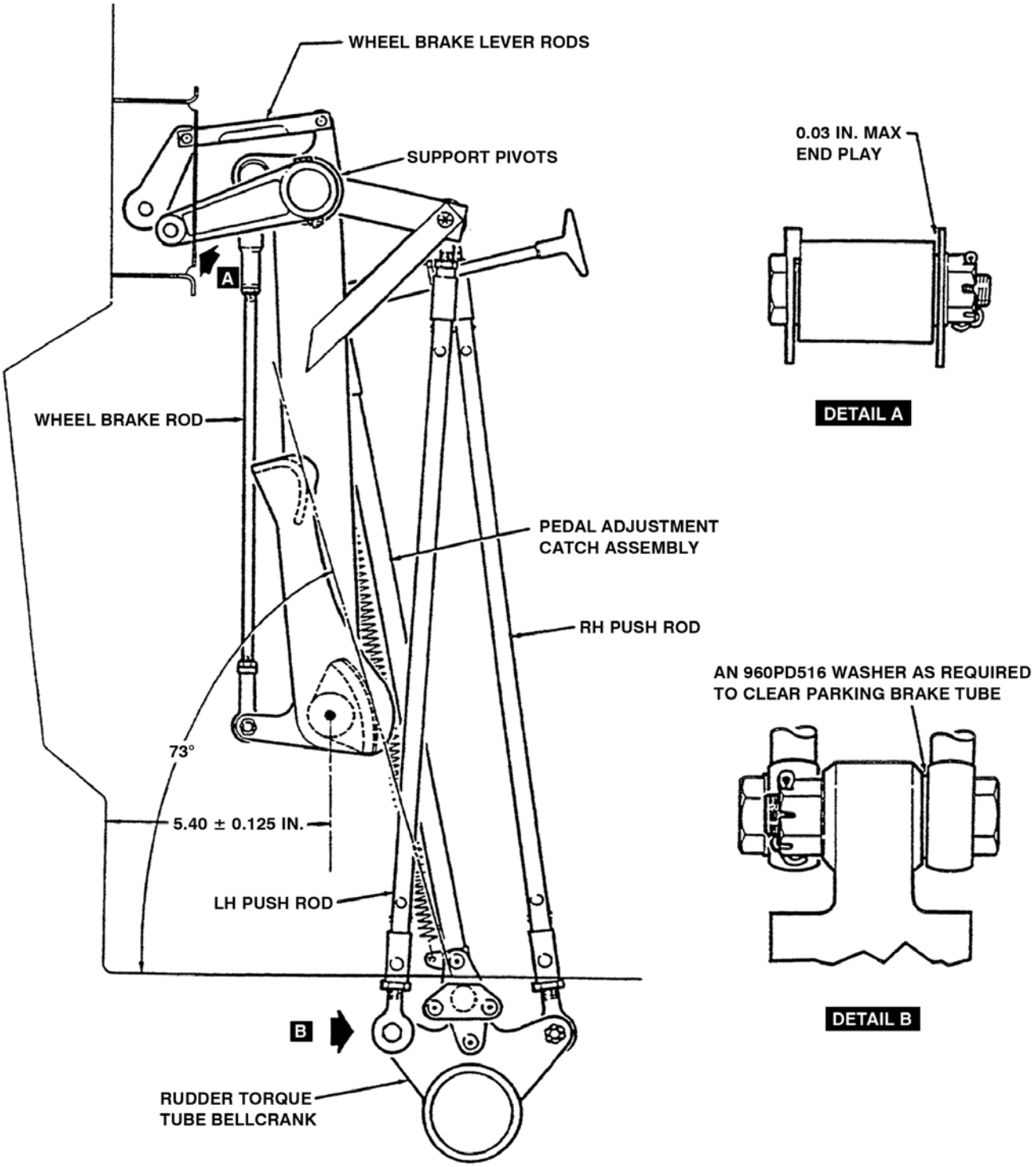
c) Check that clearance between clevis pin and wheel brake rod fork end hole does not exceed 0.018–inch.

d) Install serviceable wheel brake rods in aircraft. Connect with bolts, washers, nuts and clevis pins and lock with cotter pins.

14) If no evidence of play exists, check that clearance between clevis pin and wheel brake rod fork end hole does not exceed 0.018–inch.

15) If clearance does not exceed 0.018–inch, connect wheel brake rods to brake lever assembly, using clevis pins, washers and cotter pins.

16) Repeat Para 2.A.1) through Para 2.A.5) and check co-pilot's rudder push rods and wheel brake rod bearings.



Rudder Pedal Installation
Figure 1

Rudder Angle Position Indicator – General data

1. Description

A potentiometer is mounted forward of the rudder hinge bracket assembly near station WL 50.15 and sends a signal of the rudder angle position through the DAU and onto the Flight Data Recorder (if installed).

2. Removal/Installation

Equipment and Materials:

- Corrosion preventative compound MIL–C–16173 Grade 2 or equivalent

A. Remove the Rudder Angle Position Potentiometer

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

3) Disconnect the electrical connection at the terminal block.

4) Loosen the nut and screw that hold the lever assembly to the potentiometer shaft mounted in mounting bracket. Remove lever assembly.

5) Remove the safety wire and the (3) screws and clamps which secure the potentiometer onto the mounting bracket.

6) Remove potentiometer.

B. Install the Rudder Angle Position Potentiometer

Note Coat all bolt shanks with corrosion preventative compound MIL–C–16173 Grade 2 or equivalent before installation.

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Ensure the FLIGHT CONTROLS INDICATION (A14) circuit breaker is pulled and fitted with safety clip.

3) Position the potentiometer on to the mounting bracket. Secure with the (3) clamps and screws. Safety wire screws.

4) Position the mounting bracket on channel and secure in place with (4) bolts, washers and nuts.

5) Locate the lever assembly on potentiometer shaft leaving a gap of 0.02 to 0.25 inch between the lever assembly and the potentiometer housing to minimize the push rod assembly misalignment.

6) Secure with a washer, nut and screw.

7) Install bolt through the push rod assembly end, install washer, and then bolt through the lever assembly end. Install washer, nut and cotter pin.

8) Adjust the rudder angle position potentiometer. (Refer to Para 3.A).

9) Reconnect the terminals 1, 2 and 3 to the electrical connection at terminal block.

10) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

11) Do a Rudder Position Voltage Check per Para 3.B.

3. Adjustment/Test

A. Adjustment Rudder Angle Position Potentiometer

Equipment and Material:

- DC Voltmeter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

4) Move the rudder to neutral and lock in place (refer to 27-20-00, Rudder Control System – General Data).

5) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION **ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.**

6) Loosen screw and nut on lever at the potentiometer shaft enough that the shaft turns freely.

7) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker.

8) Rotate potentiometer shaft until measured DC Voltage is precisely 50% of total aircraft voltage V_{GP} (approximately 14 volts).

9) Reconnect pushrod and lever assembly to potentiometer shaft.

Note Do not rotate potentiometer shaft during this process.

10) Tighten screw and nut on lever leaving a gap of 0.02 to 0.05 between lever assembly and potentiometer housing.

11) Remove rudder control lock.

12) With voltmeter connected, function test rudder control position per the following:

a) Deflect rudder to the full left position (Reference 17 degrees) and verify measured voltage is 78.3% of V_{GP}.

b) Deflect rudder to the full right position (Reference 21 degrees) and verify measured voltage is 15.0% of V_{GP}.

Note Rudder may be configured for 25 degrees full left/right deflection. For 25 degrees full left, measured voltage is 91.7% of V_{GP} and for 25 degrees full right, measured voltage is 8.3% of V_{GP}. Adjust procedure per applicable configuration.

13) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with safety clip. Disconnect voltmeter from potentiometer.

14) Reconnect the potentiometer electrical connections at the terminal block.

15) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

16) Deflect the rudder full left and ensure DAU Maintenance Status CAS message does not activate.

17) Deflect the rudder full right and ensure DAU Maintenance Status CAS message does not activate.

18) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

B. Rudder Position Voltage Check

Equipment and Materials:

- DC Voltmeter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

4) Move the rudder to neutral and lock in place (refer to 27-20-00, Rudder Control System – General Data).

5) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.

6) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker and monitor DC voltage from terminal 2 to aircraft ground.

7) If DC Voltage is not 50% of V_{GP}, do adjustment steps per Para 3.A.

8) Remove rudder control locks.

9) With voltmeter connected, function test rudder control position per the following:

a) Deflect rudder to the full left position (Reference 17 degrees) and verify measured voltage is 78.3% of V_{GP}.

b) Deflect rudder to the full right position (Reference 21 degrees) and verify measured voltage is 15.0% of V_{GP}.

Note Rudder may be configured for 25 degrees full left/right deflection. For 25 degrees full left, measured voltage is 91.7% of V_{GP} and for 25 degrees full right, measured voltage is 8.3% of V_{GP}. Adjust procedure per applicable configuration.

10) If DC Voltage does not correspond, do adjustment steps per Para 3.A.

11) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with safety clip.

12) Disconnect voltmeter from potentiometer.

13) Reconnect the potentiometer electrical connections at the terminal block.

14) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

15) Deflect the rudder full left and ensure DAU Maintenance Status CAS message does not activate.

16) Deflect the rudder full right and ensure DAU Maintenance Status CAS message does not activate.

17) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

Rudder Trim Tab System – General Data

1. General

The rudder trim tab system is hand-operated from a trim tab handwheel, which is part of the trim control assembly mounted in the trim console to the right of the pilot's seat. Rotary movement of the handwheel is transmitted by a drum and cable system to a similar drum on a trim screw jack mounted in the upper portion of the rudder. From the screw jack the movement is transmitted by control rod to the rudder trim tab.

2. Description

The rudder trim tab system consists of the following components:

A. Trim Tab

The rudder trim tab is hinged to the trailing edge of the rudder upper portion and is connected to a screw jack by an adjustable control rod.

B. Trim Tab Screw Jack

The rudder trim tab screw jack converts rotary movement of the cable drum into linear motion to operate a push-pull rod. The screw jack is mounted on the front spar of the rudder structure in the upper portion of the rudder. The rudder trim tab screw jack is identical to the flap/elevator interconnect trim tab screw jack.

C. Rudder Trim Tab Handwheel

The rudder trim tab handwheel is hand-operated and is mounted on the trim controls console at the right of the pilot's seat. Rotation of the handwheel rotates a cable drum to drive the trim tab screw jack and an indicator pointer.

3. Operation

Rotation of the trim tab handwheel turns the cable drum, cables attached to the handwheel drum and the screw jack drum, move and rotate the screw jack. With rotation the screw jack either extends or retracts to operate a push-pull rod and so move the rudder trim tab. Extension of the screw jack moves the tab to the right (nose left trim), and retraction moves the tab to the left.

4. Adjustment/Test

A. Rig Rudder Trim Tab System (Mod 6/1219)

Refer to Figure 1 for details.

1) Rotate rudder trim tab handwheel until cables are arranged as in “Detail D”, and cable terminals are in position shown in “Detail C” of Figure 1.

2) Maintain position as in Para 4.A.1) and set rudder trim screw jack to 0.62 inch extension with cables arranged as shown in “Detail A” of Figure 1.

3) Maintaining positions as in Para 4.A.1) and Para 4.A.2), adjust and tension cables in rear fuselage. For cable tension values refer to 27-00-00, Flight Controls – General Data. Lock turnbuckles using locking clips.

Note The two cables from turnbuckles to screw jack are nylon covered, therefore tensiometer readings must be taken on the bare cables forward of the rear turnbuckles.

4) Disconnect connecting rod between screw jack and trim tab at trim tab attachment.

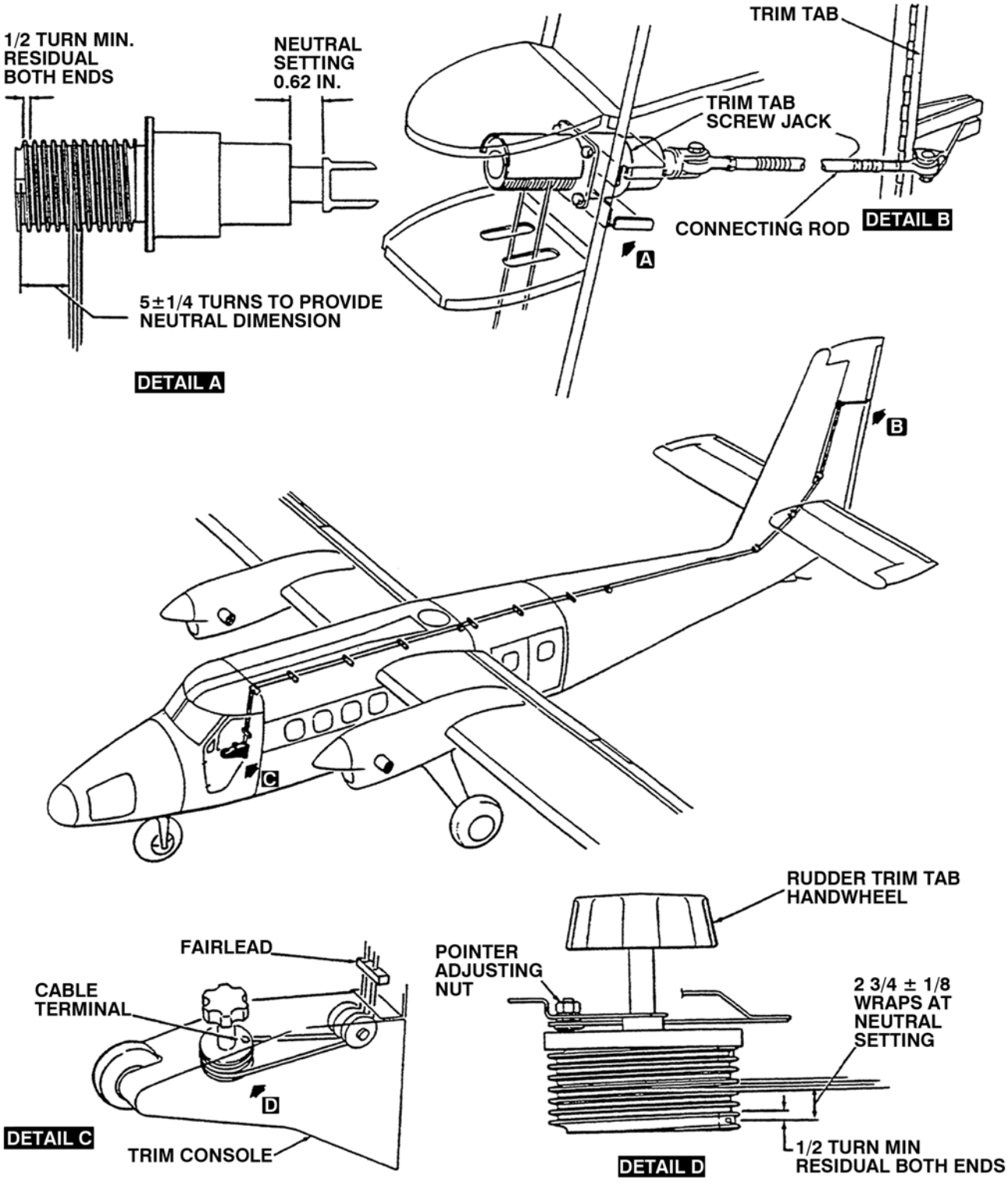
5) Set trim tab in neutral position by aligning trim tab and rudder trailing edges.

6) Maintaining position of screw jack as in Para 4.A.2), adjust and connect connecting rod to trim tab. Release nut securing handwheel pointer and align pointer with neutral position on trim console. Tighten pointer nut and connecting rod lock nut.

7) Rotate rudder trim handwheel to full nose left position, and to full nose right position. Check at extremities of travel that handwheel and screw jack drums have a minimum of half a turn of cable on each drum.

8) Check trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

9) Operate rudder and trim tab system and check for full and free range of movement.



Rudder Trim Tab System (Mod 6/1219)
Figure 1

B. Alignment of Mechanical Rudder Trim Pointer and EFIS Rudder (Yaw) Trim Display

1) Install the control lock to set rudder to a neutral position.

2) Adjust Rudder Trim Tab pointer to full left position while monitoring the Electronic Flight Instrument System (EFIS) Rudder (Yaw) Trim Display.

3) Ensure that the EFIS display indicates full left at the same time the mechanical pointer indicates full left. Confirm Rudder Trim Tab is fully deflected right.

4) Adjust Rudder Trim Tab pointer to full right position while monitoring the EFIS display.

5) Ensure that the EFIS display indicates full right position at the same time the mechanical pointer indicates full right. Confirm Rudder Trim Tab is fully deflected left.

6) Adjust Rudder Trim Tab pointer to neutral position, while monitoring the EFIS display.

7) Ensure that the EFIS display indicates neutral position at the same time mechanical pointer indicates neutral. Confirm Rudder Trim Tab is at neutral.

8) If adjustment of the EFIS Rudder (Yaw) Trim Display Potentiometer is required (refer to 27-23-31, Rudder Trim Position Transmitter – General Data, Para 3.A.). If adjustment of the mechanical rudder trim display is required reference the mechanical trim system rigging, Para 4.A.

5. Inspection/Check

A. Free Play Check – Rudder Trim Tab

Refer to Figure 1 for details.

1) Operate rudder to neutral position.

2) Operate trim tab to neutral position.

3) Move trim tab manually and check that free play (backlash), measured at trailing edge, does not exceed 0.10 inch.

Note Measurement to be taken adjacent to control rod attachment point.

Trim Control Assembly – Maintenance Practices

1. Removal/Installation

A. Remove Trim Control Assembly

1) Place elevator and rudder trim tabs in neutral.

2) Remove pilot's seat (refer to 25-10-00, Flight Compartment – General Data).

3) Remove plexiglass panel from upper surface of trim control assembly.

4) Detach and support fuse and rheostat panel on upper forward face of flight compartment/cabin bulkhead.

5) Remove cabin ceiling panels and flight compartment trim cable cover, to gain access to trim cables.

6) Retain tension in rear trim cables (to prevent unwrapping at screw jack), and disconnect cables at turnbuckles located in rear fuselage.

CAUTION **OWING TO THE VERY SMALL DIAMETER OF THE INDIVIDUAL WIRES OF THE NYLON COVERED 1/16–INCH REAR TRIM CABLES, EXTREME CARE MUST BE TAKEN AT ALL TIMES WHEN HANDLING, AS DAMAGE CAN BE TRANSMITTED THROUGH THE NYLON COVER.**

7) Release forward trim cables from fairleads and pulleys on airframe structure along cable route.

8) Support trim control assembly and remove twelve screws securing assembly to trim console.

9) Withdraw trim control assembly sufficiently to gain access to interior of trim console, and release trim cables from exposed pulleys.

10) Remove trim control assembly from aircraft, withdrawing cables through aircraft structure.

B. Install Trim Control Assembly

1) Ensure elevator trim cables are arranged around cable drum of trim control assembly in approximately neutral position (refer to Figure 1, Detail A).

2) Ensure rudder trim cables are arranged around cable drum of trim control assembly in approximately neutral position (refer to Figure 2, Detail A).

3) Route trim control assembly cables through pulleys and fairleads on airframe structure. Secure pulleys, guard pins and fairleads, where disturbed.

4) Install trim control assembly in trim console. Secure with twelve screws.

5) Connect forward and rear trim cables at turnbuckles (refer to CAUTION in Para 1.A.5)). Release cable tensioning device used to prevent unwrapping at screw jack.

6) Install plexiglass panel on upper surface of trim control assembly.

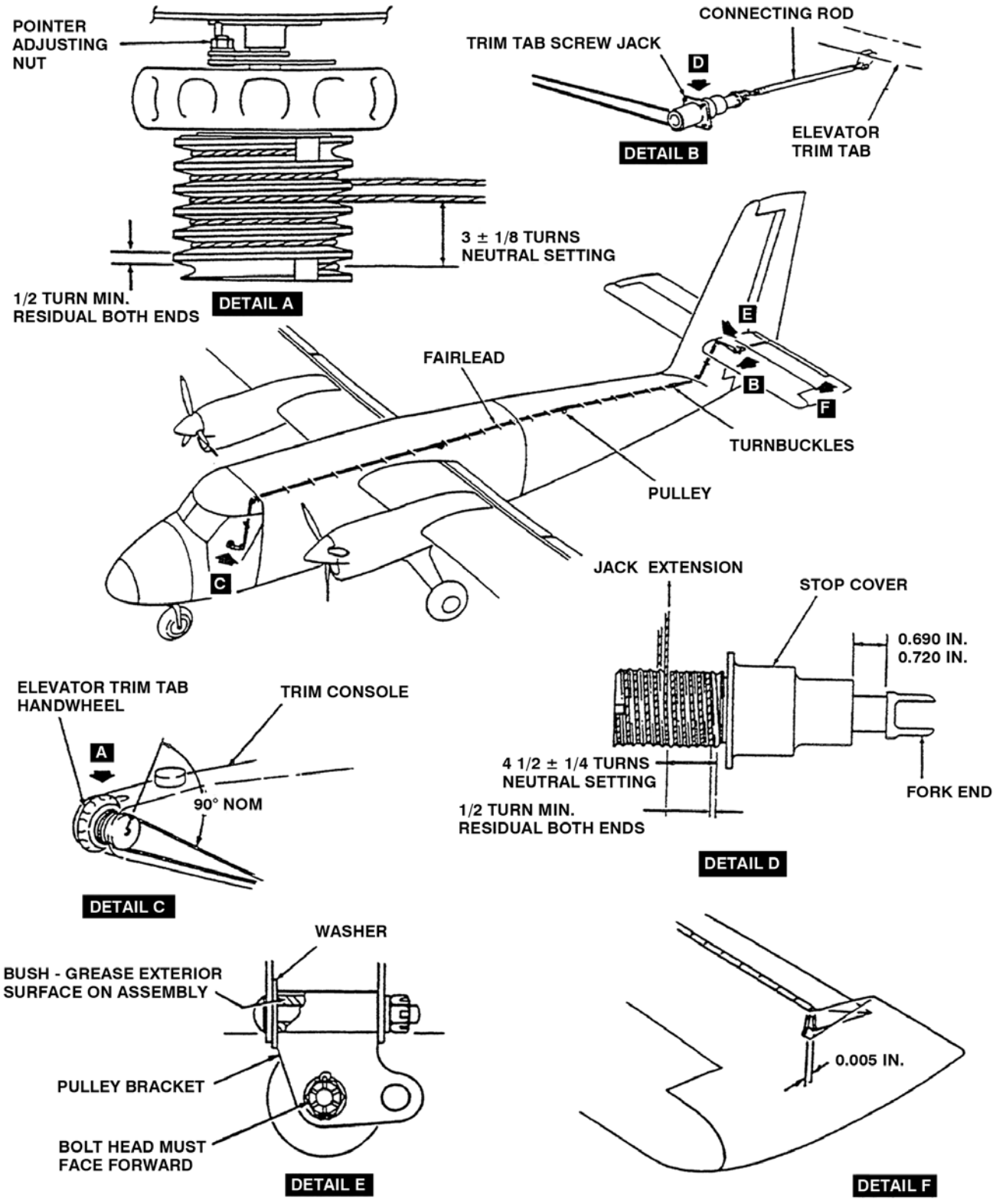
7) Rig elevator trim tab system (refer to 27-33-00, Elevator Trim Tab System – General Data).

8) Rig rudder trim tab system (refer to 27-23-00, Rudder Trim Tab System – General Data).

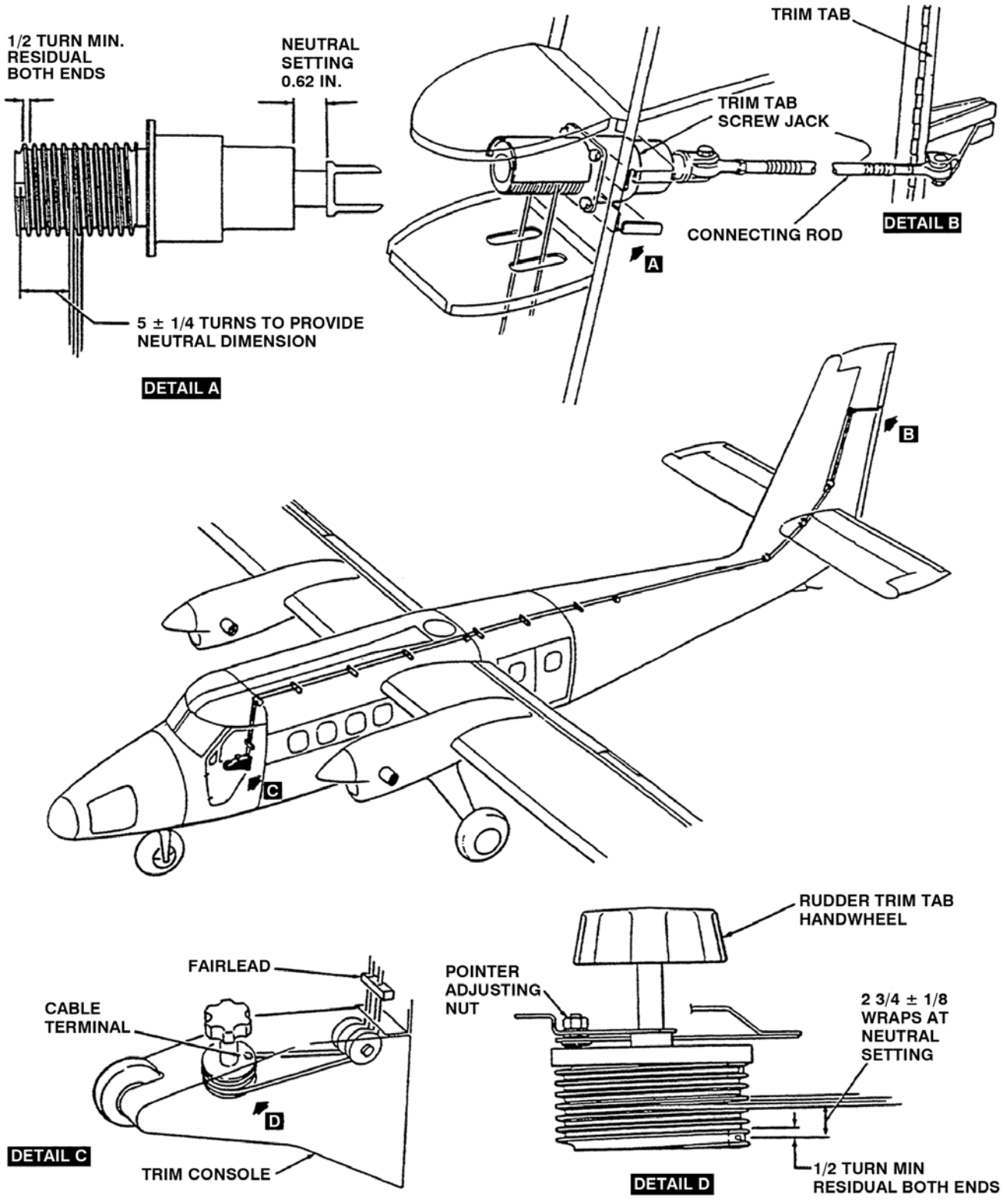
9) Install fuse and rheostat panel on flight compartment/cabin bulkhead.

10) Install removed cabin ceiling panels and flight compartment/cabin bulkhead trim cable cover.

11) Install pilot's seat (refer to 25-10-00, Flight Compartment – General Data).



Elevator Trim Tab System – Rigging (Mod 6/1219)
Figure 1



Rudder Trim Tab System (Mod 6/1219)
Figure 2

Rudder Trim Tab – Maintenance Practices

1. Removal/Installation

A. Remove Trim Tab

1) Disconnect control rod at trim tab and sensor wiring at WL 134.55.

2) Disengage lower end of hinge pin from retaining clip on rudder.

3) Support tab and withdraw hinge pin.

B. Install Trim Tab

1) Lightly lubricate hinge pin. Refer to 12-20-00, Lubrication – General Data for lubricant specification.

2) Align hinge components on tab and rudder.

3) Insert hinge pin from lower end of tab and engage lower end of pin in retaining clip on rudder.

4) Connect control rod to tab with bolt, washer, nut and cotter pin and sensor wiring at WL 134.55.

5) Check trim tab control surface clearances (refer to 27-10-00, Flight Controls – General Data).

6) Operate rudder and trim tab system and check for full and free range of movement.

7) Do function test for the rudder trim tab potentiometer indicator as per 27-23-31, Rudder Angle Position Indicator - General data, Para 3.A.12).

8) If adjustment is required for the rudder trim tab potentiometer adjust as per 27-23-31, Rudder Angle Position Indicator - General data, Para 3.A.

Note If tab or rudder has been repaired, or has had additional paint applied, the rudder assembly must be balanced as detailed in Structural Repair Manual PSM 1–6–3.

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Rudder Trim Position Transmitter – General data

1. General

A potentiometer is mounted on the starboard side of the rudder internally at approximately station 571.978 and sends a signal of the rudder trim position through the DAU and onto the Flight Data Recorder (if installed) and the rudder trim position is indicated on the system screen.

2. Removal/Installation

Equipment and Materials:

- Corrosion preventative compound MIL–C–16173 Grade 2 or equivalent.

A. Remove the Rudder Trim Indicator Potentiometer

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

3) Disconnect the electrical connection at the terminal block.

4) Loosen the nut and screw that hold the lever assembly to the potentiometer shaft mounted in mounting bracket. Remove lever assembly.

5) Remove the (2) bolts, nuts and washers that hold the potentiometer mounting bracket. Remove the potentiometer and the mounting bracket assembly.

6) Remove the safety wire and the (3) screws and clamps which secure the potentiometer onto the mounting bracket.

7) Remove potentiometer.

B. Install the Rudder Trim Indicator Potentiometer

Note Coat all bolt shanks with corrosion preventative compound MIL–C–16173 Grade 2 or equivalent before installation.

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Ensure the FLIGHT CONTROLS INDICATION (A14) circuit breaker is pulled and fitted with safety clip.

3) Position the potentiometer on to the mounting bracket. Secure with the (3) clamps and screws. Safety wire screws.

4) Position the mounting bracket on channel and secure in place with bolt, (2) washers and nut in two places.

5) Install the lever assembly on the potentiometer shaft ensuring the mechanism can move freely through the full range of motion and there is a minimum gap of 0.04 inch between the lever assembly and the potentiometers housing.

6) Secure with a washer, nut and screw.

7) Install the clevis pin, (2) washers cotter pin from the end of the lever assembly to the link halves.

8) Install the clevis pin, (2) washers cotter pin from the end of the link halves to the end of the push rod assembly.

9) Ensure that the mechanism can move freely through the full range of motion.

10) Adjust the rudder trim indicator potentiometer. (Refer to Para 3.A).

11) Reconnect the electrical connection at terminal block.

12) Remove the safety clip and replace circuit breaker for the FLIGHT CONTROLS INDICATION (A14) circuit breaker.

13) Do a function test of the rudder trim tab indicator potentiometer per Para 3.B.

3. Adjustment/Test

A. Adjustment Rudder Trim Tab Indicator Potentiometer

Equipment and Material:

- DC Voltmeter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

3) Move rudder to neutral and lock in place (refer to 27-20-00, Rudder Control System – General Data). Set rudder trim tab to the neutral position.

4) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION **ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.**

5) Loosen screw and nut on lever assembly at the potentiometer shaft enough so that the shaft turns freely.

6) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker.

7) Rotate potentiometer shaft until DC Voltage is precisely 50% of total aircraft voltage (approximately 14 volts), pull FLIGHT CONTROLS INDICATION (A14) circuit breaker.

8) Disconnect voltmeter from potentiometer. Reconnect pushrod and lever assembly to potentiometer shaft.

Note Do not rotate potentiometer shaft during this process.

9) Tighten screw and nut on lever assembly leaving a gap of 0.02 to 0.05 of and an inch between lever assembly and potentiometer housing.

10) Reconnect the potentiometer electrical connections at the terminal block.

11) Remove safety clip and replace FLIGHT CONTROLS INDICATION (A14) circuit breaker.

12) Remove rudder control lock.

13) Do a function test per Para 3.B.

14) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

B. Function Test Rudder Trim Tab Indicator Potentiometer

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Deflect rudder trim to the full nose left position (Reference 25 degrees) and note yaw trim display indication correlates and does not become invalid (indicated by an amber ‘X’).

4) Deflect rudder trim to the neutral position (Reference 0 degrees) and note yaw trim display indication correlates.

5) Deflect rudder trim to the full nose right position (Reference 15 degrees) and note yaw trim display indication correlates and does not become invalid (indicated by an amber ‘X’).

6) If the rudder trim display does not match the rudder trim surface, repeat the adjustment steps in Para 3.A.

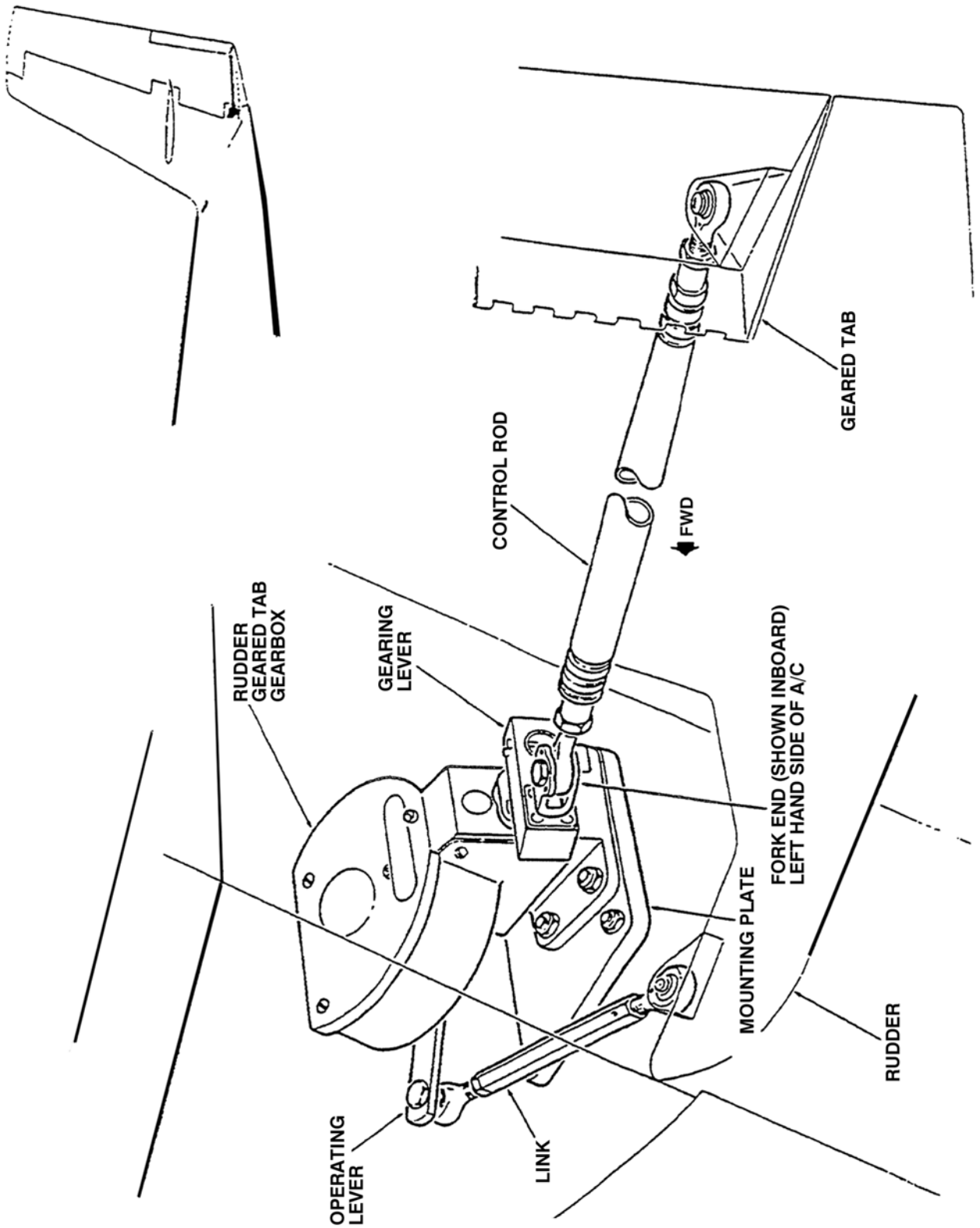
7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Rudder Geared Tab System – General Data

1. General

A geared tab is installed on the lower portion of the rudder trailing edge to provide aerodynamic assistance to rudder movement. The geared tab system consists of control rods, levers, gearbox and tab. The gearbox, which provides a non-linear deflection of the tab in relation to rudder movement, is bolted to a mounting plate installed on the bottom rudder hinge bracket of the vertical stabilizer. A link connects the rudder to the gearbox operating lever, and the gearbox gearing lever is connected to the tab by a control rod. Movement of the rudder is transmitted through the gearbox, where the mechanism changes the ratio of movement of tab to rudder, as it moves the tab in the opposite direction to the rudder. Refer to Figure 1 for details.



Rudder Geared Tab System
Figure 1

2. Adjustment/Test

   A. Rig Rudder Geared Tab

      Refer to Figure 1 for details.

      1) Check that geared tab is neutral with rudder neutral (refer to 27-20-00, Rudder Control System – General Data).

      2) Check that gearbox gearing lever is horizontal with fork end inboard, and that rod attachment bolt is vertical.

      3) Adjust link to obtain gearing lever position, then adjust control rod to set tab in neutral.

      4) Tighten and wirelock control rod lock nuts as necessary.

      5) Check rudder geared tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

      6) Operate rudder and check for full and free range of movement (refer to 27-20-00, Rudder Control System – General Data). Make sure that the direction of geared tab movement is opposite rudder movement.

3. Inspection/Check

   A. Free Play Check – Rudder Geared Tab Gearbox

      Refer to Figure 1 for details.

      1) Remove rudder gearbox access panels at rear fuselage.

      2) Disconnect control rod from fork end, and operating lever from link.

      3) Move gearing lever fore and aft and check that free play at shaft of gearing lever does not exceed 0.002 inch. Move operating lever up and down and check that free play at gearbox bolt does not exceed 0.002 inch.

      4) Reconnect operating lever to link, and control rod to fork end.

      5) Install rudder gearbox access panels.

   B. Free Play Check – Rudder Geared Tab

      1) Operate rudder (and geared tab) to neutral position.

      2) Move geared tab manually and check that free play (backlash), measured at trailing edge, does not exceed 0.10–inch.

         Note Measurement to be taken adjacent to control rod attachment point.

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Rudder Geared Tab – Maintenance Practices

1. Removal/Installation

A. Remove Geared Tab

1) Disconnect control rod at geared tab.

2) Disengage upper end of hinge pin from retaining clip on rudder.

3) Support tab and withdraw hinge pin.

B. Install Geared Tab

1) Lightly lubricate hinge pin. Refer to 12-20-10, Lubrication – General Data for lubricant specification.

2) Align hinge components on tab and rudder.

3) Insert hinge pin from upper end of tab and engage upper end of hinge pin in retaining clip on rudder.

4) Connect control rod to tab with bolt, washer, nut and cotter pin.

5) Check geared tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

6) Operate rudder and check for full and free range of movement.

Note If tab or rudder has been repaired, or has had additional paint applied. If the rudder assembly must be balanced as detailed in Structural Repair Manual PSM 1–6–3.

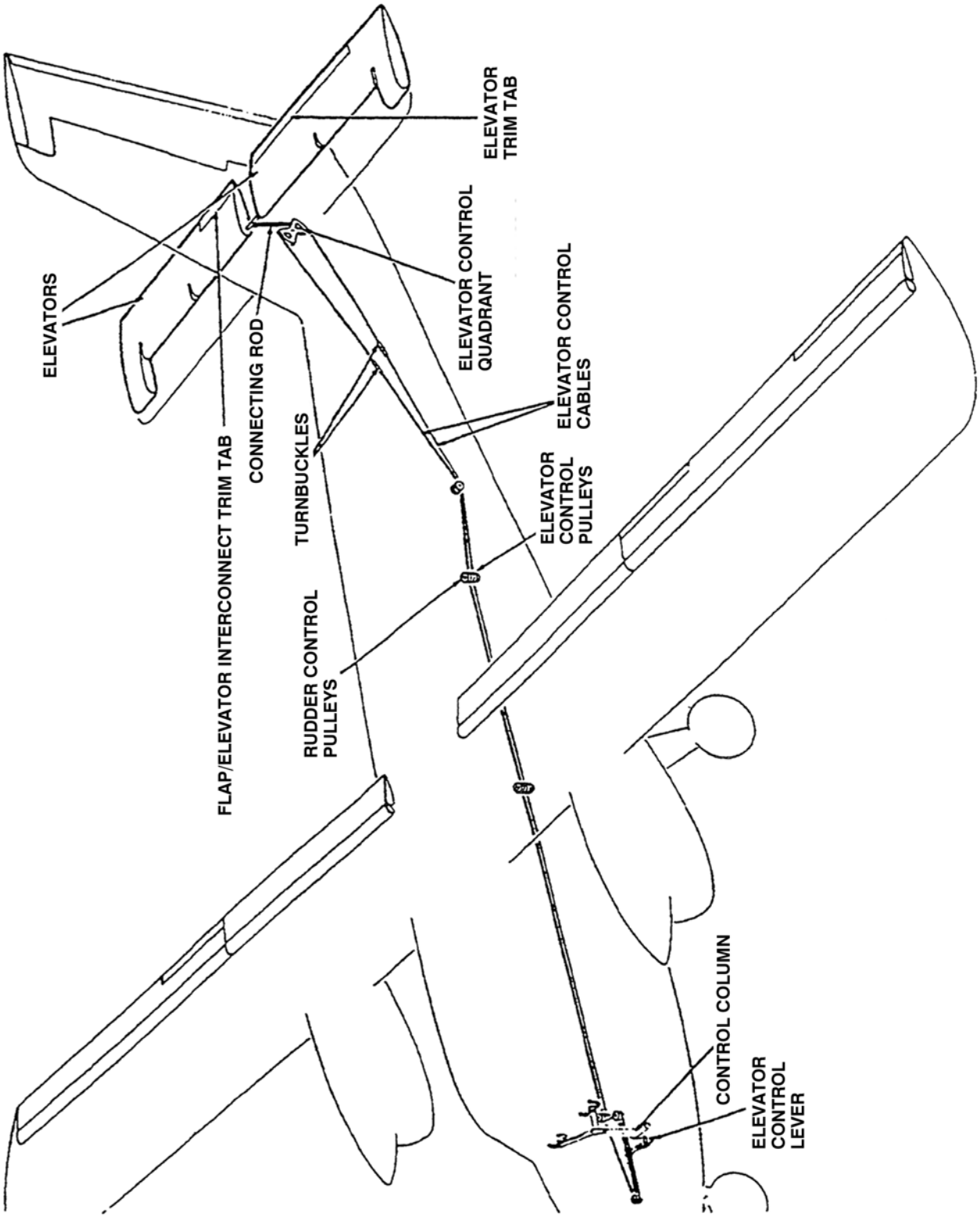
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Elevator Control System – General Data

1. General

Elevator deflection is achieved through fore and aft movement of the control column. A connecting rod (located below the flight compartment floor) joins the control column to the elevator control lever which transmits movement to the elevator control cables. Movement of the elevator control lever is limited by a cable with swaged stops which passes through the outboard end of the control lever; the cable is secured at each end to elevator control pulley attachment bolts. On aircraft with Mod 6/1747 incorporated, end straps are installed on the stop cables to provide for cable tension adjustment. From the control lever, movement is transmitted through cables, routed through pulleys and fairleads on the right side of the fuselage under the cabin floor, to an elevator quadrant in the rear fuselage. An adjustable connecting rod transmits movement from the quadrant to the elevator torque tube and elevators. On aircraft with Mod 6/1739 incorporated, drainage holes are provided in the connecting rod. Two hinge attachments are mounted on the elevator front spar and a third is located on the flange at the end of the elevator torque tube. When the elevators are installed, the torque tube flanges, when aligned and bolted together, provide the maximum up and down travel stops. The left elevator carries an elevator trim tab and the right elevator carries a flap/elevator interconnect tab. Refer to Figure 1.

Note For general cable rigging instructions, refer to 27-00-00, Flight Controls – General Data.



Elevator Control System
Figure 1

2. Adjustment/Test

Refer to Figure 2 for details.

A. Rig Elevator Control System

1) Lock control column in neutral by installing gust locks. On aircraft with Mod 6/1676 incorporated, obtain neutral by installing elevator rigging tool C6GT1047–1 and accompanying rigging pins C6GT1048–1 between control column and plate on instrument panel frame.

2) On aircraft with Mod 6/1747 incorporated, check stop cable for minimum slackness. Adjust tension of stop cable by using alternate mounting holes in straps to achieve minimum slackness, maintaining ease of insertion of stop cable mounting bolts. End strap hole pitch may be adjusted between +0.100 and –0.150 inch in increments of 0.050 inches.

Note Inspect elevator quadrant mounting bracket for cracks, and the quadrant for distortion and loose attachment bolts (refer to Inspection Requirements Manual PSM 1-6-7, Basic and Special Inspections).

3) Check that top surface of right elevator horn and lower edge of rigging disc at horizontal stabilizer tip are aligned within ± 0.030 inches (neutral position).

Note If Para 2.A.3) is satisfactory, ignore Para 2.A.4), Para 2.A.5) and Para 2.A.6).

4) Insert rigging pin SD10542 (0.25 inch dia.) in elevator control quadrant in rear fuselage.

a) If rigging pin enters easily, adjust connecting rod at elevator as necessary to comply to Para 2.A.3).

b) If rigging pin does not enter quadrant, adjust turnbuckles in rear fuselage until quadrant setting is obtained; tension cables to values given in 27-00-00, Flight Controls – General Data.

5) Check rigging disc at right elevator horn for alignment, as in Para 2.A.3), and make final adjustment on connecting rod as necessary. Safety connecting rod. Lock turnbuckles using locking clips as necessary.

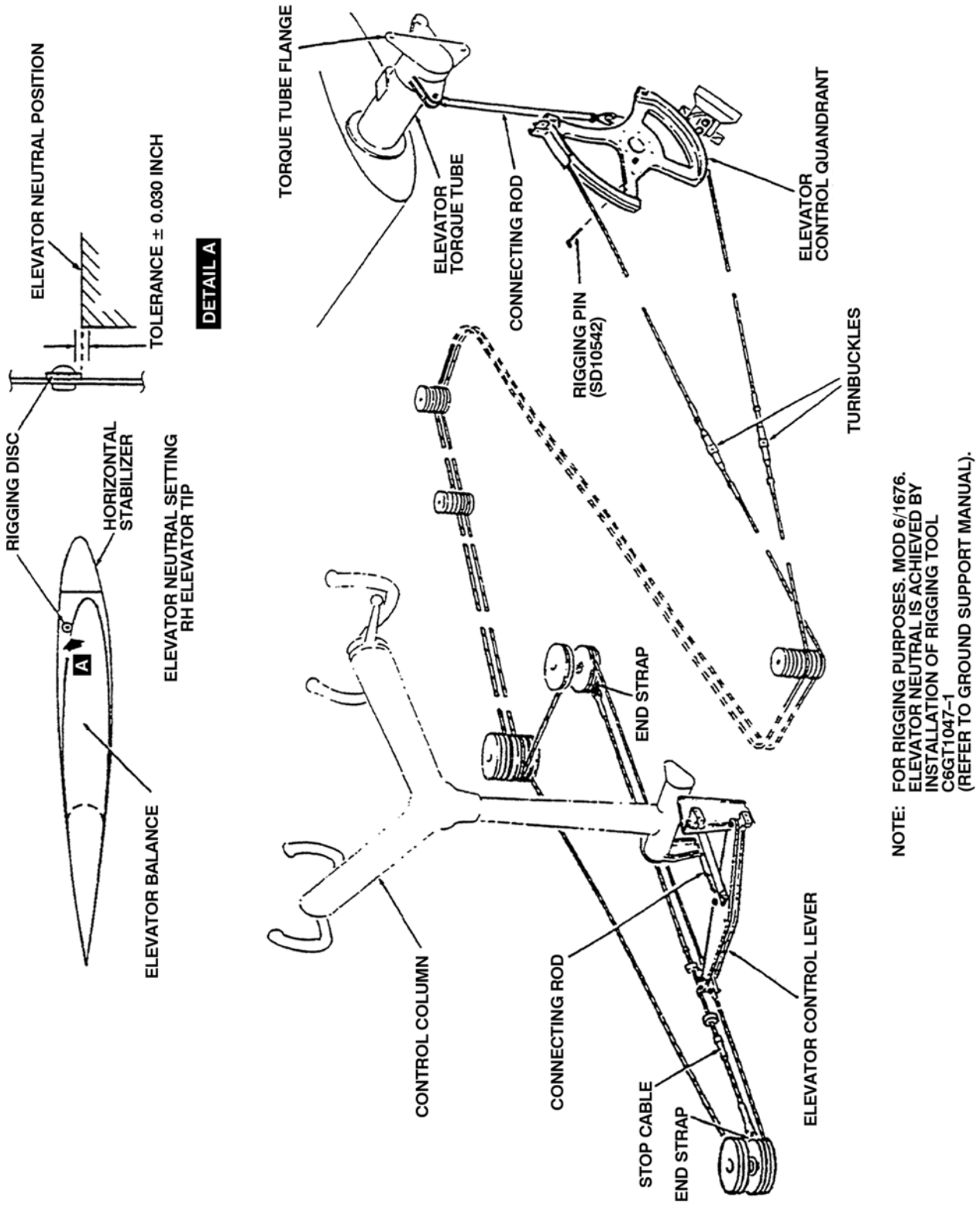
6) Remove rigging pins.

7) Remove rigging tool (Mod 6/1676).

8) Check elevator control surface clearances (refer to 27-00-00, Flight Controls – General Data).

9) Operate elevator controls and check for full and free range of movement.

10) Check rigging of elevator trim tab and flap/elevator interconnect trim tab systems (refer to 27-33-00, Elevator Trim Tab System – General Data and 27-36-00, Flap/Elevator Interconnect Trim System – General Data).



Elevator System Rigging
Figure 2

3. Inspection/Check

A. Job Set-up

1) Either pull and clip breakers required to isolate affected circuits or placard and select aircraft electrical power to OFF.

2) Obey all relevant WARNINGS and CAUTIONS detailed in Maintenance Manual PSM 1-64-2, Chapters 12 and 24.

3) Placard flight controls in flight compartment.

B. Flight Compartment

1) Open access doors 15 to gain access to elevator cables (refer to Table 1 — Elevator Cable Assy – Flight Compartment and Passenger Cabin and Figure 2).

2) Do a visual inspection of control cables (non-jacketed) visible at this section of fuselage (refer to Table 1 — Elevator Cable Assy – Flight Compartment and Passenger Cabin and Figure 2). Observe carefully areas where cables pass over pulleys, quadrants, drums, or through fairleads.

3) Replace cables not acceptable to criteria in Inspection of Control Cables (refer to 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data). Make sure the new cable part numbers are identical to the discarded cable part numbers (refer to Table 1 — Elevator Cable Assy – Flight Compartment and Passenger Cabin).

4) Lubricate new or existing non-jacketed Carbon Steel cables (refer to 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data).

Note Do not lubricate stainless steel cables.

5) Replace cable as required.

6) Install and rig replaced cable assemblies (refer to 27-00-00, Flight Controls – General Data).

7) Close access doors 15.

Table 1 Elevator Cable Assy – Flight Compartment and Passenger Cabin

	CARBON STEEL	STAINLESS STEEL
Front of elevator lever to station 426.75	C6CF1146–1	E.O. 69053–1
Front of elevator lever to station 426.75	C6CF1147–1	E.O. 69053–2
Elevator lever stop cable	C6CF1100–11	E.O. 69053–5

C. Passenger Cabin

1) Remove panel 43 (refer to 12-00-00, Servicing – General Data, Figure 1) to gain access to elevator cables and pulleys (refer to Table 1 — Elevator Cable Assy – Flight Compartment and Passenger Cabin and Figure 2).

2) Release, attach string and pull back elevator cables to allow visual inspection (refer to Table 1 — Elevator Cable Assy – Flight Compartment and Passenger Cabin, Figure 2 and Inspection of Control Cables PSM 1-64-2, Section 27-00-00).

3) Observe carefully areas where cables pass over pulleys, quadrants, drums, or through fairleads.

4) Replace cables not acceptable to criteria in Inspection of Control Cables (refer to 27-00-00, Flight Controls – General Data). Make sure the new cable part numbers are identical to the discarded cable part numbers per Table 1 — Elevator Cable Assy – Flight Compartment and Passenger Cabin.

5) Lubricated replaced or existing non-jacketed Carbon Steel cables (refer to 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data).

Note Do not lubricate stainless steel cables.

6) Install and rig cable assemblies (refer to 27-00-00, Flight Controls – General Data).

7) Install panel 43.

D. Rear Fuselage and Tail Area

CAUTION **WHEN INSTALLING ELEVATOR AND RUDDER PRIMARY FLIGHT CONTROL CABLES, CARGO AREA FLOOR BOARDS (BETWEEN STA 332.00 TO 376.00 AND AFT CARGO AREA FLOOR BOARDS (AFT OF STA 376.00, MUST BE REMOVED TO ENSURE PROPER INSTALLATION. THIS WILL REDUCE THE POSSIBILITY OF MISS-RIGGING AND CABLE CROSSING, WHICH COULD LEAD TO ACCELERATED CABLE WEAR TO THE ELEVATOR AND RUDDER PRIMARY FLIGHT CONTROL SYSTEMS.**

1) Remove floor panels to gain access to elevator cable assemblies in baggage compartment rear fuselage.

2) Remove panels 11 and 12 (refer to Figure 2) to gain access to elevator cable assemblies in the tail area.

3) Do a visual inspection of control cables (non-jacketed) visible at this section of fuselage (refer to Table 2 — Elevator Cable Assy – Rear Fuselage and Tail Area, Figure 2 and 27-00-00, Flight Controls – General Data, Para 3.A). Observe carefully areas where cables pass over pulleys, quadrants, drums, or through fairleads and cable ends. Place special emphasis on elevator and rudder cable inspection near cable cluster at station 376.0 (refer to Figure 2).

4) Replace cables not acceptable to criteria in Inspection of Control Cables (refer to 27-00-00, Flight Controls – General Data). Make sure the new cable part numbers are identical to the discarded cable part numbers per Table 2 — Elevator Cable Assy – Rear Fuselage and Tail Area.

5) Lubricate new or existing non-jacketed Carbon Steel cables only (refer to 20-60-01, Inspection, Cleaning and Lubrication of 7x19 and 7x7 Control Cables – General Data).

Note Do not lubricate stainless steel cables.

6) Install and rig replaced cable assemblies (refer to 27-00-00, Flight Controls – General Data).

7) Install baggage compartment floor panels and panels 11 and 12.

Table 2 Elevator Cable Assy – Rear Fuselage and Tail Area

	CARBON STEEL	STAINLESS STEEL
Upper cable, station 426.75 to elevator quadrant	C6CF1148–1	E.O. 69053–3
Lower cable, station 426.75 to elevator quadrant	C6CF1149–1	E.O. 69053–4

E. Close-out:

1) Remove placards on the flight controls in flight compartment.

2) Obey all relevant WARNINGS and CAUTIONS detailed in Maintenance Manual PSM 1-64-2, Chapters 12 and 24.

3) Either remove clip and push breakers that were required to isolate affected circuits or remove placards and select aircraft electrical power to ON.

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Elevator – Maintenance Practices

1. Removal/Installation

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. Remove Left Elevator

1) Disconnect elevator trim tab system cables in rear fuselage.

2) Remove elevator trim tab system rear pulley assembly (refer to 27-33-00, Elevator Trim Tab System – General Data).

3) Remove three bolts from torque tube flanges. Disconnect bonding strip.

4) Support elevator, remove two hinge bolts and disconnect bonding strips and sensor wiring at WL 58.875. Lower elevator to ground.

B. Install Left Elevator

Note If the elevator to be installed is a replacement or repaired elevator, or if it has been painted or repainted, rebalancing of elevators must be carried out in accordance with Structural Repair Manual PSM 1–6–3.

1) Position elevator, install and secure two hinge bolts. Connect and secure bonding strips and sensor wiring at WL 58.875.

2) Install and secure bolts to torque tube flanges. Connect and secure bonding strip.

3) Install elevator trim tab system rear pulley (refer to 27-33-00, Elevator Trim Tab System – General Data).

4) Connect elevator trim tab system cables.

5) Rig elevator control system (refer to 27-30-00, Elevator Control System – General Data).

6) Do an elevator position operational test as per 27-30-21, Elevator Angle Position Indicator – General Data, Para 3.B.

7) If adjustment is required, adjust the elevator angle position indicator as per 27-30-21, Elevator Angle Position Indicator – General Data, Para 3.A.

C. Remove Right Elevator

1) Disconnect flap/elevator interconnect trim system cables in rear fuselage, or at quick disconnects as applicable.

2) Disconnect connecting rod from elevator torque tube.

3) Remove flap/elevator interconnect system rear pulley assembly (refer to 27-36-00, Flap/Elevator Interconnect Trim System – General Data).

4) Remove three bolts from torque tube flanges. Disconnect bonding strip and sensor wiring at WL 58.875.

5) Support elevator, remove two hinge bolts and disconnect bonding strips. Lower elevator to ground.

6) Do an elevator position operational test as per 27-30-21, Elevator Angle Position Indicator – General Data, Para 3.B.

7) If adjustment is required, adjust the elevator angle position indicator as per 27-30-21, Elevator Angle Position Indicator – General Data, Para 3.A.

D. Install Right Elevator

Note If the elevator to be installed is a replacement or repaired elevator, or if it has been painted or repainted, rebalancing of elevators must be carried out in accordance with Structural Repair Manual PSM 1–6–3.

1) Position elevator, install and secure two hinge bolts. Connect and secure bonding strips.

2) Install and secure bolts to torque tube flanges. Connect and secure bonding strip and sensor wiring at WL 58.875.

3) Install flap/elevator interconnect system rear pulley assembly (refer to 27-36-00, Flap/Elevator Interconnect Trim System – General Data).

4) Connect flap/elevator interconnect system cables.

5) Connect connecting rod to elevator torque tube.

6) Rig elevator control system (refer to 27-30-00, Elevator Control System – General Data).

7) Do an elevator position operational test as per 27-30-21, Elevator Angle Position Indicator – General Data, Para 3.B.

8) If adjustment is required, adjust the elevator angle position indicator as per 27-30-21, Elevator Angle Position Indicator – General Data, Para 3.A.

Elevator Angle Position Indicator – General data

1. General

A potentiometer is mounted on the port side elevator hinge bracket assembly at station WL 58.875 and sends a signal of the elevator angle position through the DAU and onto the Flight Data Recorder (if installed).

The potentiometer can be accessed by removing the cover panel (Mod 6/2417). The panel is secured to the aircraft by screws.

2. Removal/Installation

Equipment and Materials:

- Corrosion preventative compound MIL–C–16173 Grade 2 or equivalent

A. Remove the Elevator Angle Position Potentiometer

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

3) Remove the cover panel fasteners and remove the panel (Mod 6/2417).

4) Disconnect the electrical connection at the terminal block.

5) Remove the cotter pin, nut, (2) washers and bolt from the push rod end at the elevator hinge bracket.

6) Remove the two bolts which hold the potentiometer mounting bracket to the structure.

7) Remove the potentiometer and mounting bracket assembly.

8) Loosen the nut and screw that hold the lever assembly to the potentiometer shaft mounted in mounting bracket.

9) Remove the safety wire and the (3) screws and clamps which secure the potentiometer onto the mounting bracket.

10) Remove the potentiometer.

B. Install the Elevator Angle Position Potentiometer

Note Coat all bolt shanks with corrosion preventative compound MIL–C–16173 Grade 2 or equivalent before installation.

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Ensure the FLIGHT CONTROLS INDICATION (A14) circuit breaker is pulled and fitted with safety clip.

3) Position the potentiometer on to the mounting bracket. Secure with the (3) clamps and screws. Safety wire screws.

4) Locate the lever assembly on potentiometer shaft leaving a gap of 0.02 to 0.05 inch between the lever assembly and the potentiometer housing to minimize the push rod assembly misalignment.

5) Secure with a screw, washer and nut.

6) Install bolt through the push rod assembly at lever end, install washer, nut and cotter pin.

7) Position the mounting bracket and secure in place with bolt, and washers in two places.

8) Install nut and cotter pin bolt through the elevator hinge end. Install washer, nut and cotter pin.

9) Adjust the elevator angle position potentiometer. (Refer to Para 3.A).

10) Replace the cover panel (Mod 6/2417) and re-install fasteners. Do not overtighten.

11) Reconnect the electrical connection at terminal block.

12) Remove the safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

13) Do an Elevator Position Voltage Check per Para 3.B.

3. Adjustment/Test

A. Adjustment Elevator Angle Position Potentiometer

Equipment and Material:

- DC Voltmeter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

4) Move the elevator to the neutral position and lock in place.

5) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block (TB5) and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION **ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.**

Note There are a set of drop-down resistors at the DAU. If reading the voltage at the DAU or on the ASCB bus, this voltage is reduced to 0.5 times of the values listed in this section.

6) Loosen screw and nut on the lever at the potentiometer shaft. Remove lever from shaft.

7) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker.

8) Rotate potentiometer shaft until the measured DC Voltage is precisely 50% of total aircraft voltage V_{GP} (approximately 14 volts).

9) Reconnect pushrod and lever assembly to potentiometer shaft.

10) Tighten screw and nut on lever leaving a gap of 0.02 to 0.05 of an inch between lever assembly and potentiometer housing.

Note Do not rotate potentiometer shaft during this process.

11) Remove elevator control lock.

12) With voltmeter connected, function test elevator control position per the following:

a) Deflect elevator to the full up position (Reference 25 degrees) and verify measured voltage is 91.70% of V_{GP}.

b) Deflect elevator to the full down position (Reference 16 degrees) and verify measured voltage is 23.30% of V_{GP}.

13) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip..

14) Disconnect voltmeter from potentiometer.

15) Reconnect the potentiometer electrical connections at the terminal block.

16) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

17) Deflect the elevator full up and ensure DAU Maintenance Status CAS message does not activate.

18) Deflect the elevator full down and ensure DAU Maintenance Status CAS message does not activate.

19) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

B. Elevator Position Voltage Check

Equipment and Material:

- DC Voltmeter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

4) Move the elevator to the neutral position and lock in place.

5) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block (TB5) and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.

Note There are a set of drop-down resistors at the DAU. If reading the voltage at the DAU or on the ASCB bus, this voltage is reduced to 0.5 times of the values listed in this section.

6) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker and confirm that DC Voltage is precisely 50% of total aircraft voltage V_{GP} (approximately 14 volts).

7) If DC Voltage is not 50% of V_{GP}, do adjustment steps per Para 3.A.

8) Remove elevator control lock.

9) With voltmeter connected, function test elevator control position per the following:

a) Deflect elevator to the full up position (Reference 25 degrees) and verify measured voltage is 91.70% of V_{GP}.

b) Deflect elevator to the full down position (Reference 16 degrees) and verify measured voltage is 23.30% of V_{GP}.

10) If DC Voltage does not correspond, do adjustment steps per Para 3.A.

11) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with safety clip.

12) Disconnect voltmeter from potentiometer.

13) Reconnect the potentiometer electrical connections at the terminal block.

14) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

15) Deflect the elevator full up and ensure DAU Maintenance Status CAS message does not activate.

16) Deflect the elevator full down and ensure DAU Maintenance Status CAS message does not activate.

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

C. Alignment of Mechanical Pitch Trim Pointer and EFIS Pitch Trim Display Operational Test

1) Rotate the pitch trim handwheel until the pointer indicates the aircraft full nose up position while monitoring the Electronic Flight Instrument System (EFIS) Pitch Trim Display.

2) Ensure that the EFIS display indicates full pitch up (25 degrees ± 1 degree) at the same time the mechanical pointer indicates full nose up. Visually confirm the Elevator Trim Tab is fully deflected down.

3) Rotate the pitch trim handwheel until the pointer indicates the full nose down position while monitoring the EFIS Pitch Trim Display.

4) Ensure that the EFIS display indicates full pitch down (20 degrees ± 1 degree) at the same time as the mechanical pointer indicates full nose down. Visually confirm the Elevator Trim Tab is fully deflected up (20 degrees ± 1 degree).

5) Adjust the pitch trim handwheel until the pointer indicates the neutral position while monitoring the EFIS Pitch Trim Display.

6) Ensure that the EFIS display indicates the neutral position (0 degrees) at the same time the mechanical pointer indicates the neutral pitch trim. Confirm the Elevator Trim Tab is at the neutral position.

7) If adjustment of the EFIS Pitch Trim Display Potentiometer is required refer to 27-33-21, Elevator Trim Position Transmitter – General Data Para 3.A. If adjustment of the mechanical pitch trim display is required, reference the mechanical trim system rigging, in accordance with 27-33-00, Elevator Trim Tab System – General Data Para 5.A.

Note The mechanical pitch trim display is the primary trim indication system.

Elevator Trim Tab System – General Data

1. General

An elevator trim tab, attached to the left elevator trailing edge, is operated by a handwheel which is part of the trim control assembly mounted in the trim console to the right of the pilot's seat.

2. Description

The elevator trim tab system consists of a trim tab, trim handwheel, trim screw jack and trim cables. The cables, connected to a drum attached to the trim handwheel, are routed through pulleys and fairleads up the forward face of the sloping bulkhead, along the cabin roof to the rear fuselage, to connect to a cable drum attached to the screw jack drive shaft.

A. Trim Tab

The elevator trim tab is hinged to the trailing edge of the left elevator. A bracket on the lower surface of the tab provides for the attachment of an adjustable connecting rod which connects to the trim screw jack. Drain holes are provided in the tab bottom skin.

B. Trim Tab Handwheel

The elevator trim tab handwheel is mounted in the trim console, a cable drum to which the trim cables connect, is attached to the handwheel. Mounted on the inboard end of the cable drum is a spiral grooved plate, which through a pin and lever arrangement, drives an indicator pointer.

C. Trim Tab Screw Jack

The elevator trim tab screw jack, mounted on the front spar of the left elevator, converts rotary motion of the cable drum into linear movement of the tab connecting rod attachment. The cable drum is rotated by the trim cables secured to the drum.

3. Operation

Rotary movement of the elevator trim handwheel turns the cable drum, which through the cables turns the screw jack cable drum to extend or retract the screw jack. The screw jack operates the connecting rod, to either raise or lower the trim tab. The trim tab moves up for a nose down selection, or down for a nose up selection of the trim tab handwheel.

4. Removal/Installation

A. Remove Elevator Trim Tab System Rear Pulley Assembly

1) Release elevator trim tab system cables.

2) Remove rear pulley bracket from elevator torque tube.

B. Install Elevator Trim Tab System Rear Pulley Assembly

1) Install rear pulley bracket assembly, referring to Figure 1 for assembly details. Lubricate pulley bush (refer to 12-20-10, Lubrication – General Data). Ensure that after tightening the pivot bolt there is free movement of bracket on bushing.

2) Connect and tension elevator trim tab cables. Rig elevator trim tab system, refer to Para 5.A.

3) Do a function test of the elevator trim tab indicator potentiometer as per 27-33-21, Elevator Trim Position Transmitter, Para 3.B.

4) If adjustments are required to the elevator trim tab indicator potentiometer adjust as per 27-33-21, Elevator Trim Position Transmitter, Para 3.A.

5. Adjustment/Test

A. Rig Elevator Trim Tab System (Mod 6/1219)

Refer to Figure 1 for details.

1) Rotate elevator trim tab handwheel two turns (approximately) from full nose down position until cable terminals are located as shown in Figure 1, "Detail C", and cables are arranged as in "Detail A".

2) Maintain handwheel as in Para 5.A.1) and set trim screw jack in left elevator to 0.690 to 0.720 inches extended, and arrange cables as shown in Figure 1, "Detail D".

3) Maintaining handwheel and screw jack cable drum settings as in Para 5.A.1) and Para 5.A.2), adjust and tension cables in rear fuselage. For cable tension values refer to 27-00-00, Flight Controls – General Data. Lock turnbuckles using locking clips.

4) Disconnect connecting rod between screw jack and tab attachment bracket at tab attachment.

5) Set trim tab in neutral position by aligning trim tab and elevator trailing edges.

6) Maintaining position of screw jack as in Para 5.A.2), adjust and connect trim tab connecting rod.

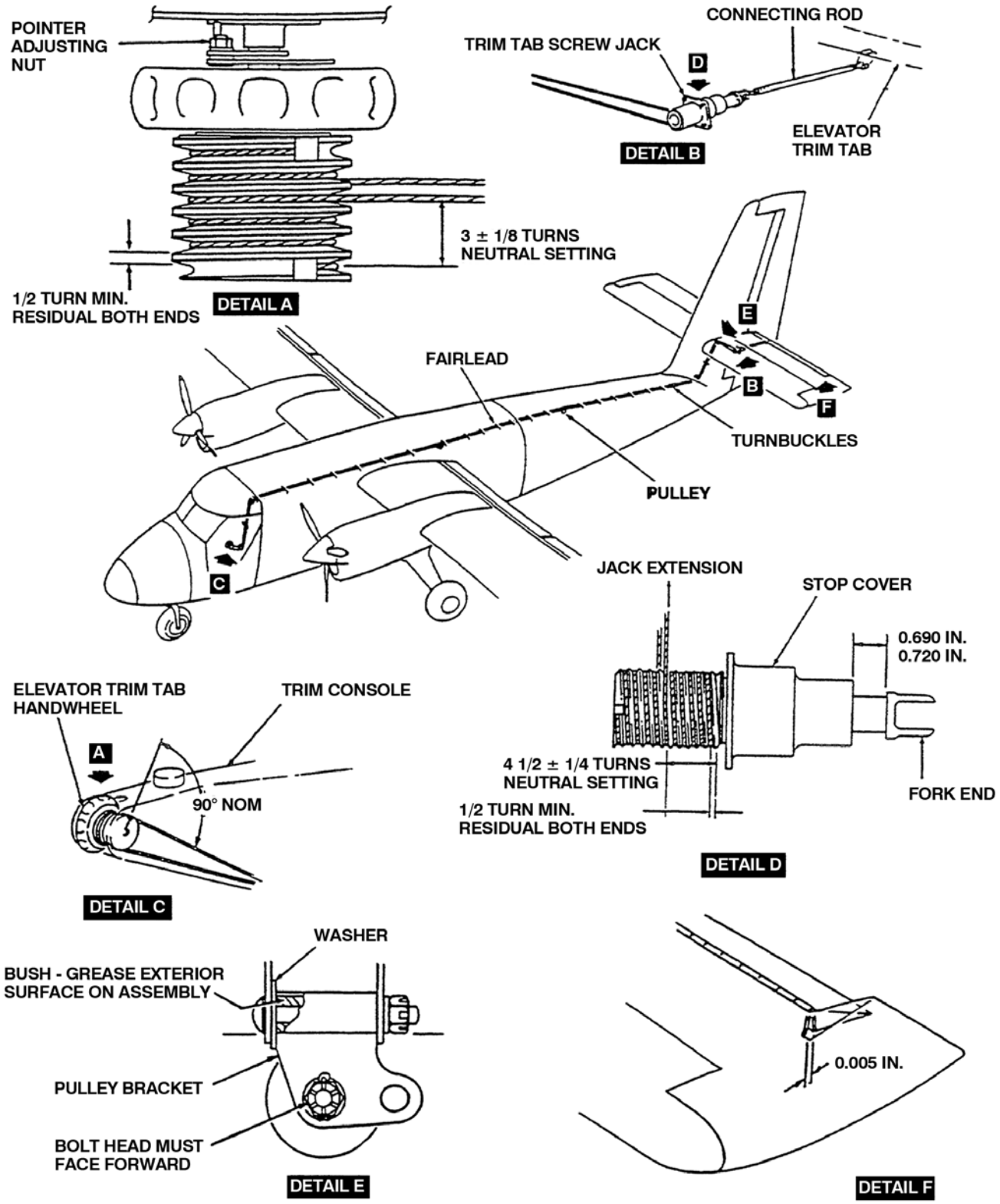
7) Rotate elevator trim handwheel to nose full up position; ensure that screw jack is fully retracted. Check that 0.005 inch exists between trim tab hinge lugs (refer to Figure 1, "Detail F"), if not, lengthen connecting rod to obtain clearance.

8) Tighten and lock connecting rod adjustable end.

9) With elevator trim handwheel still in nose full up position, slacken pointer nut and position pointer to coincide with nose up position. Tighten pointer nut.

10) Check elevator trim tab control surface clearance (refer to 27-00-00, Flight Controls – General Data).

11) Operate trim tab and check for full and free range of movement. Check at extremities of travel that handwheel and screw jack drums have a minimum of half a turn of cable on each drum.



Elevator Trim Tab System – Rigging
Figure 1

6. Inspection/Check

A. Free Play Check – Elevator Trim Tab

Refer to Figure 1 for details.

1) Operate elevators to neutral position.

2) Operate trim tab to neutral position.

3) Move trim tab manually and check that free play (backlash), measured at trailing edge does not exceed 0.10–inch.

Note Measurement to be taken adjacent to control rod attachment point.

Elevator Trim Tab – Maintenance Practices

1. Removal/Installation

   A. Remove Trim Tab

      1) Disconnect control rod at trim tab and sensor wiring at WL 54.00.

      2) Ease back plastic sleeve at junction of the two hinge pins to enable hinge pins to be withdrawn.

         Note If replacement sleeve is available, existing sleeve maybe cut and discarded.

      3) Support tab and withdraw inboard and outboard hinge pins.

   B. Install Trim Tab

      1) Ensure that ends of hinge pins are free from burrs or sharp edges and lightly lubricate with lubricant specified in 12-20-10, Lubrication – General Data.

      2) Align hinge components on trim tab and elevator.

      3) Insert inboard hinge pin at hinge gap and install plastic sleeve.

      4) Insert outboard hinge pin and locate inboard end in sleeve. Ensure that ends of pins are aligned.

      5) Connect control rod to tab with bolt, washer, nut and cotter pin and sensor wiring at WL 54.00.

      6) Check trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

      7) Operate trim tab and check for full and free range of movement.

         Note If tab or elevator has been repaired, or has had additional paint applied the elevator assembly must be balanced as detailed in Structural Repair Manual PSM 1-6-3.

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Elevator Trim Position Transmitter – General data

1. General

A potentiometer is mounted on the port side elevator at approximately station WL 54.00 and sends a signal of the elevator trim tab position through the DAU and onto the Flight Data Recorder (if installed) and the elevator trim tab position is indicated on the system screen.

2. Removal/Installation

Equipment and Materials:

- Corrosion preventative compound MIL–C–16173 Grade 2 or equivalent

A. Remove the Elevator Trim Indicator Potentiometer

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

3) Disconnect the electrical connection at the terminal block.

4) Remove the cotter pin, (2) washers and clevis pin from the push rod end and link halves.

5) Remove the cotter pin, (2) washers and the clevis pin from the link halves to the lever assembly.

6) Loosen the nut, and screw that hold the lever assembly to the potentiometer shaft mounted in mounting bracket. Remove lever assembly.

7) Remove two bolts holding the potentiometer mounting bracket to the bracket assembly.

8) Remove the safety wire and the (3) screws and clamps which secure the potentiometer onto the mounting bracket.

9) Remove potentiometer.

B. Install the Elevator Trim Indicator Potentiometer

Note Coat all bolt shanks with corrosion preventative compound MIL–C–16173 Grade 2 or equivalent before installation.

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Ensure the FLIGHT CONTROLS INDICATION (A14) circuit breaker is pulled and fitted with safety clip.

3) Position the potentiometer on to the mounting bracket. Secure with the (3) clamps and screws. Safety wire screws.

4) Position the mounting bracket on the bracket assembly and secure in place with bolt and washers in two places.

5) Locate the lever assembly on potentiometer shaft leaving a gap of 0.02 to 0.05 of an inch between lever assembly and the potentiometer housing to minimize the pushrod assembly misalignment.

6) Secure with a washer nut and screw.

7) Install the clevis pin, (2) washers cotter pin from the end of the lever assembly to the link halves.

8) Install the clevis pin, (2) washers cotter pin from the end of the link halves to the end of the push rod assembly.

9) Ensure that the mechanism can move freely through the full range of motion.

10) Adjust the elevator trim tab indicator potentiometer. (Refer to Para 3.A).

11) Reconnect the electrical connection at the terminal block.

12) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

13) Do a function test per Para 3.B.

3. Adjustment/Test

A. Adjustment Elevator Trim Tab Indicator Potentiometer

Equipment and Material:

- DC Voltmeter

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with safety clip.

4) Move elevator to neutral position and lock in place. Set elevator trim tab to the neutral position.

5) Disconnect terminal 2 of the potentiometer electrical connection at the terminal block and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION **ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.**

6) Loosen screw and nut on lever at the potentiometer shaft enough that the shaft turns freely.

7) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker.

8) Rotate potentiometer shaft until the measured DC Voltage is precisely 50% of total aircraft voltage V_{GP} (approximately 14 volts).

9) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with a safety clip.

10) Disconnect voltmeter from potentiometer. Reconnect pushrod and lever assembly to potentiometer shaft.

Note Do not rotate potentiometer shaft during this process.

11) Reconnect the potentiometer electrical connections at the terminal block.

12) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

13) Remove elevator control lock.

14) Do a function test (refer to Para 3.B).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

15) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Function Test Elevator Trim Tab Indicator Potentiometer

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Deflect elevator trim to the full up position and note the Pitch Trim Display Indication correlates to approximately 20 degrees and does not become invalid (indicated by an amber 'X').

4) Deflect elevator trim to the neutral position and note the Pitch Trim Display Indication correlates to approximately 0 degrees.

5) Deflect elevator trim to the full down position and note the Pitch Trim Display Indication correlates to approximately 25 degrees and does not become invalid (indicated by an amber 'X').

6) If the Pitch Trim Display does not match the elevator trim surface position, repeat adjustment steps in Para 3.A.

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Flap/Elevator Interconnect Trim System – General Data

1. General

The flap/elevator interconnect trim system is a remote mechanical operation in which wing flap actuator travel is used to apply compensating nose down trim proportional to the amount of flap extension.

2. Description

The system comprises a ball screw jack, a cable system, trim tab screw jack, connecting rod and trim tab.

A. Ball Screw Jack

The flap/elevator interconnect ball screw jack, mounted in the cabin roof structure and connected to the flap operating mechanism, converts flap system linear motion into rotary motion. The ball screw jack comprises a ball screw assembly, a spindle and an end support. The ball screw assembly consists of a threaded shaft, which engages ball bearings enclosed in a tube integral with a ball cage. The ball cage is keyed into a spindle which has an integral cable drum, and the spindle rotates on a bearing mounted on the end support bracket. The end support bracket is bolted to the cabin roof structure and a fork end installed on the screw shaft connects to the flap operating mechanism. Linear movement of the screw shaft causes the ball cage and spindle to rotate and drive the cable drum.

B. Trim Tab Screw Jack

The flap/elevator interconnect trim tab screw jack mounted on the front spar of the right elevator, converts rotary motion into linear motion. The screw jack comprises a cable drum spindle which rotates in bearings in a stop cover, and a slide screw assembly. The cable drum is internally threaded and engages with external threads on the slide screw. A fork end is threaded and riveted into the slide screw.

C. Trim Tab

The flap/elevator interconnect trim tab is hinged to the trailing edge of the right elevator. A bracket on the lower surface of the tab provides for the connection of a connecting rod, from the screw jack slide screw fork end.

3. Operation

During flap operation, movement of the right inboard operating bellcrank (refer to 27-50-00, Wing Flaps System – General Data, Para 2. and Para 3.) causes the ball screw jack spindle and cable drum to rotate. The cables attached to the ball screw jack drum transmits this movement to turn the interconnect trim screw jack cable drum, which extends or retracts the screw jack slide screw and operates the interconnect trim tab through the connecting rod. For a flaps down selection, the interconnect trim tab moves up, to provide a nose down trim proportional to flap movement.

4. Removal/Installation

A. Install Flap/Elevator Interconnect System Rear Pulley Assembly

1) Install rear pulley bracket assembly, referring to, Figure 1 or Figure 2 for assembly details. Lubricate pulley bush (refer to 12-20-10, Lubrication – General Data). Ensure after tightening pivot bolt there is free movement of the bracket on the bushing.

2) Connect flap/elevator interconnect cables and rig trim system, refer to Para 5.A or Para 5.B.

5. Adjustment/Test

A. Rig Flap/Elevator Interconnect Trim System (Mod 6/1219)

Note Before attempting to rig the flap/elevator interconnect trim system, ensure that the wing flap system is correctly rigged. (Refer to 27-50-00, Wing Flaps System – General Data).

Refer to Figure 1 for details.

1) Ensure wing flaps are pumped fully up.

2) Set cables on interconnect ball screw jack cable drum as in Figure 1, "Detail A", with inboard cable terminal uppermost.

3) Set trim screw jack extension to between 0.030 and 0.070 inch and arrange cables on screw jack drum as shown in Figure 1, "Detail B".

4) Maintaining positions of ball screw jack and screw jack as in Para 5.A.2) and Para 5.A.3), adjust and tension trim cables at turnbuckles in rear fuselage. For cable tension values refer to 27-00-00, Flight Controls – General Data. Lock turnbuckles using locking clips.

5) Disconnect connecting rod between screw jack and tab attachment bracket, at tab attachment.

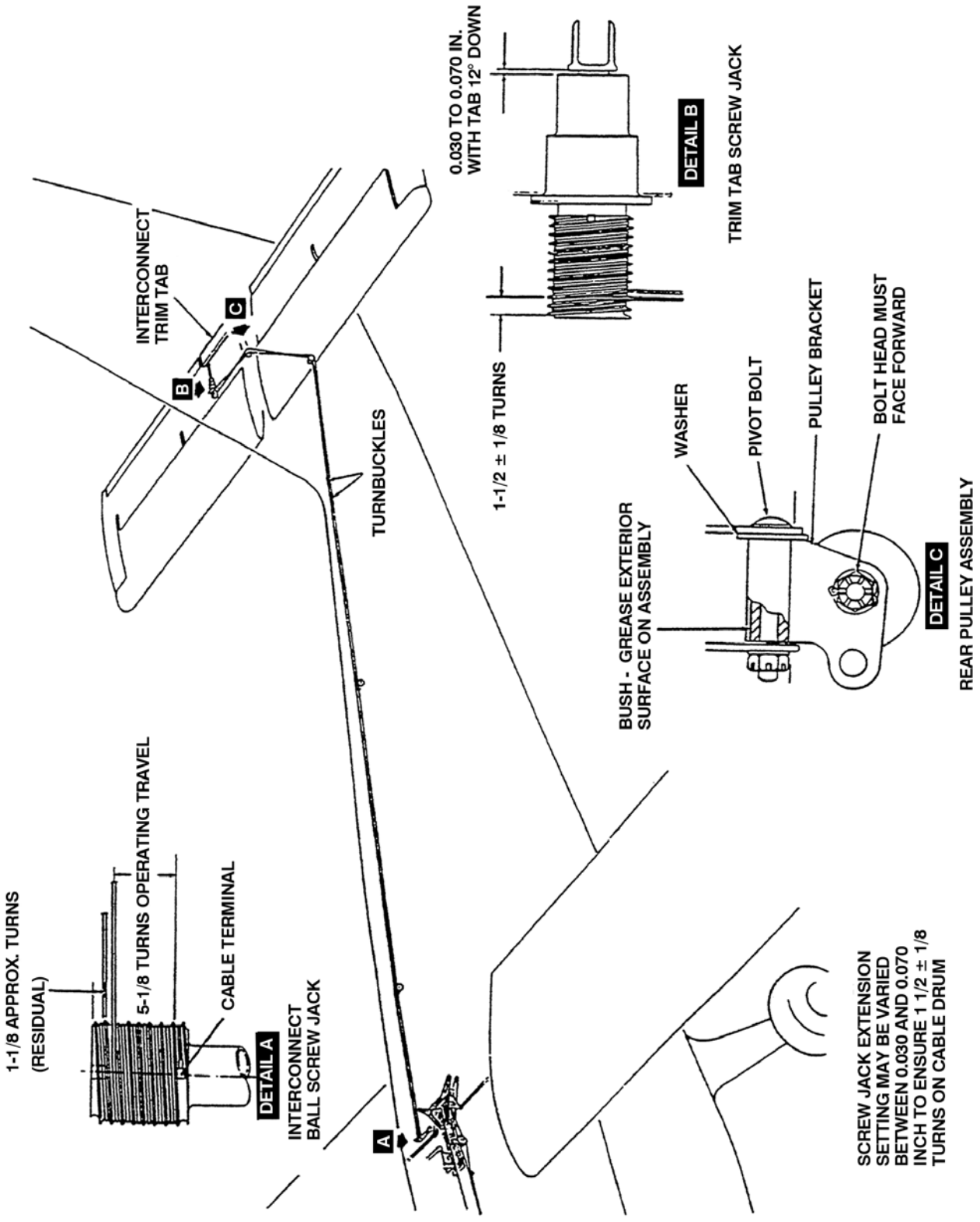
6) Set the trim tab 12° down. Measure angular difference along upper skin, or measure 0.85 ± 0.05 inch from elevator trailing edge to trim tab trailing edge with tab down.

7) With trim screw jack set as in Para 5.A.3) and tab set as in Para 5.A.6), adjust and connect trim tab connecting rod. Lock connecting rod lock nut.

8) Pump flaps fully down, ensure that at extremities of travel, ball screw jack and trim screw jack cable drums have a minimum of half a turn of cable left on drum.

9) Check flap/elevator interconnect trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

10) Operate flaps up and down and check interconnect system for full and free range of movement.



Flap/Elevator Trim Interconnect System – Rigging (Mod 6/1219)
Figure 1

B. Rig Flap/Elevator Interconnect Trim System (Mod 6/1775)

Note Before attempting to rig the flap/elevator interconnect trim system, ensure that the wing flap system is correctly rigged. (Refer to 27-50-00, Wing Flaps System – General Data).

Refer to Figure 2 for details.

1) Ensure wing flaps are pumped fully up.

2) Set cables on interconnect ball screw jack cable drum as in Figure 2, "Detail A", with inboard cable terminal uppermost.

3) Disconnect connecting rod between screw jack and tab attachment bracket at tab attachment.

4) Set trim screw jack extension to 0.050 inch and arrange cables on screw jack drum as shown in Figure 2, "Detail B".

5) Connect turnbuckles and tighten to approximately equal lengths.

6) Check there is clearance at trim tab screw jack as shown in Figure 2, "Detail B", of 0.030/0.070 inch. If faces touch, dismount jack, rotate its mounting flange 180° and reinstall jack.

7) Measure clearance at trim tab screw jack as shown in Figure 2, "Detail B".

8) If clearance is between 0.030 and 0.070 inch, omit Para 5.B.9) or Para 5.B.10).

9) If clearance is between 0.071 and 0.084 inch:

   a) Dismount ball screw jack (Figure 2, "Detail A"), and rotate 180° clockwise (looking outboard). Transfer the 0.100 inch thick packing from outboard to inboard side of structure, to place it under jack.

   b) Dismount trim tab screw jack (Figure 2, "Detail B"), and rotate 180° counterclockwise (looking aft).

   c) Reinstall both jacks.

10) If clearance is between 0.001 and 0.029 inch:

   a) Dismount ball screw jack (Figure 2, "Detail A"), and rotate 180° clockwise (looking outboard). Reinstall 0.100 inch thick packing on outboard side of structure.

   b) Dismount trim tab screw jack (Figure 2, "Detail B"), and rotate 180° counterclockwise (looking aft).

   c) Reinstall both jacks.

11) Adjust turnbuckles to obtain correct cable tension values. For cable tension values, refer to 27-00-00, Flight Controls – General Data. Maintain trim jack setting obtained in Para 5.B.9) or Para 5.B.10).

12) Lock both turnbuckles.

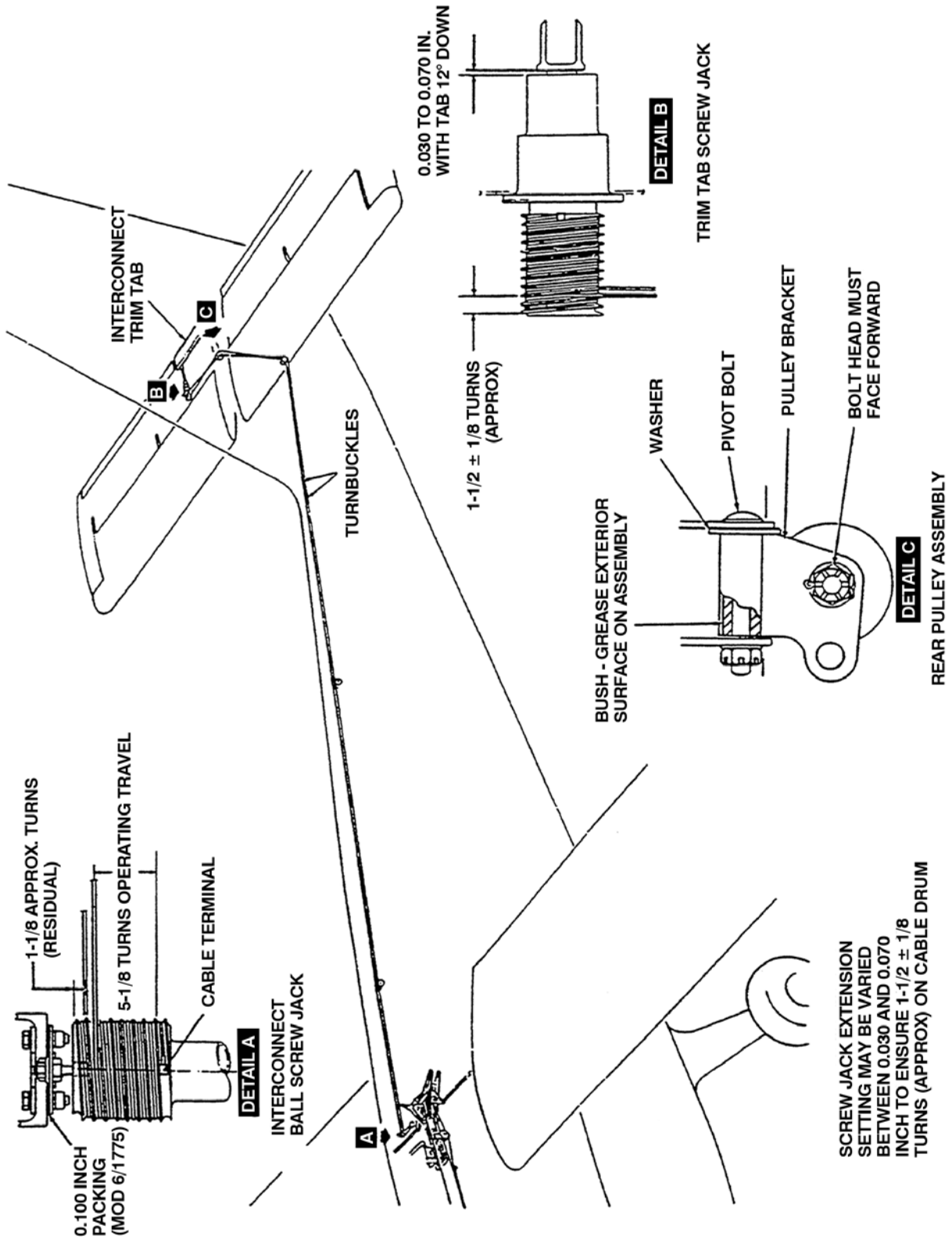
13) Set the trim tab 12° down. Measure angular difference along upper skin, or measure 0.85 ± 0.05 inch from elevator trailing edge to trim tab trailing edge with tab down.

14) With trim screw jack set as in Para 5.B.9) or Para 5.B.10) and tab set as in Para 5.B.13), adjust and connect trim tab connecting rod. Lock connecting rod lock nut.

15) Pump flaps fully down, ensure that at extremities of travel, ball screw jack and trim screw jack cable drums have a minimum of half a turn of cable left on drum.

16) Check flap/elevator interconnect trim tab control surface clearance (refer to 27-00-00, Flight Controls – General Data).

17) Operate flaps up and down and check interconnect system for full and free range of movement.



Flap/Elevator Trim Interconnect System – Rigging (Mod 6/1775)
Figure 2

6. Inspection/Check

A. Free Play Check – Flap/Elevator Interconnect Tab

1) Select and pump flaps slowly down until flap/elevator interconnect tab is aligned with elevator trailing edge.

2) Operate elevators to neutral position.

3) Move trim tab manually and check that free play (backlash), measured at trailing edge, does not exceed 0.10–inch.

Note Measurement to be taken adjacent to control rod attachment point.

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Flap/Elevator Interconnect Trim Tab – Maintenance Practices

1. Removal/Installation

CAUTION **IF REMOVING OR INSTALLING CONTROL CABLES, DISCONNECT MAIN BATTERY (REFER TO 24-30-31, MAIN BATTERY SYSTEM – MAINTENANCE PRACTICES, PARA 2.), AVIONICS START BATTERY (IF INSTALLED, REFER TO 24-30-32, AVIONICS START BATTERY (MOD 6/2315) – GENERAL DATA, PARA 4.) AND EMERGENCY BATTERY (IF INSTALLED, REFER TO 33-50-01, EMERGENCY LIGHTING BATTERY – GENERAL DATA, PARA 1.) AND DISCONNECT GROUND POWER.**

A. Remove Trim Tab

1) Disconnect control rod at trim tab.

Note If replacement sleeve is available, existing sleeve may be cut and discarded.

2) Ease back plastic sleeve at junction of the two hinge pins to enable hinge pins to be withdrawn.

3) Support tab and withdraw inboard and outboard hinge pins.

B. Install Trim Tab

1) Ensure that ends of hinge pins are free from burrs or sharp edges and lightly lubricate with lubricant specified in 12-20-10, Lubrication – General Data.

2) Align hinge components on trim tab and elevator.

3) Insert inboard hinge pin at hinge gap and install plastic sleeve.

4) Insert outboard hinge pin and locate inboard end in sleeve. Ensure that ends of pins are aligned.

5) Connect control rod to tab with bolt, washer, nut and cotter pin.

6) Check trim tab control surface clearances (refer to 27-00-00, Flight Controls – General Data).

7) Operate flaps up and down and check interconnect system for full and free range of movement.

Note If tab or elevator has been repaired, or has had additional paint applied, the elevator assembly must be balanced as detailed in Structural Repair Manual PSM 1-6-3.

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Stall Warning System – General Data

1. General

The aircraft stall warning system is triggered from the lift detector vanes located in the leading edge of the left wing. An active stall warning will trigger the Apex Stall Monitor and the independent Stall light located on the main instrument panel just below the ESIS. The Stall Monitor will trigger an Aural “Stall” annunciation, and the annunciation of Stall on the pilot's and co-pilot's PFDs.

Each lift detector has an integral anti-condensation heater operated by the pitot heater switch. Power to operate the system is from the 28 Volt left DC bus, routed through a 7.5 ampere PITOT HEAT L circuit breaker on the circuit breaker panel.

2. Adjustment/Test

A. Adjust Lift Detecting Vanes

Refer to Figure 1 for details.

Note The positions of the lift detecting vanes on the mounting plate assembly determine the airspeed at which stall warning is given. Therefore, whenever a vane is installed, an air test is required to determine the correct position of the vane on the mounting plate. The lower vane determines the airspeed at which warning is given in the range of 0° to 12° flaps down, and the upper vane determines the airspeed at which warning is given in the range of 13° to 40° flaps down. The data in Figure 1 gives the amount of change in airspeed produced by 1/8-inch movement of vane position for various flap settings.

1) Remove six screws securing lift detector vane mounting plate assembly to wing leading edge.

2) Carefully withdraw mounting plate assembly away from leading edge sufficiently to gain access to rear of mounting plate.

3) Using suitable marking instrument, mark rear of mounting plate and flanges of vane to be moved to show original installation position.

4) Using data given in Figure 1, change position of appropriate vane in direction required to produce warning at specified air speed.

Note The specified airspeed at which stall warning is given (STALL light illuminated) is 4 to 9 knots above stalling speeds. For stalling speeds at specified aircraft gross weight with various flap settings, refer to Flight Manual PSM 1-64-1A.

5) Secure detector vane mounting hardware and reinstall mounting plate to wing leading edge with six screws removed in Para 2.A.1).

B. Adjust Upper Vane Circuit Microswitch

Note Before attempting the following adjustments, ensure that the wing flaps are correctly rigged.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Using handpump, pump wing flaps to fully up position.

3) Set BUS TIE switch to NORMAL and ensure that CAUTION LTS NORMAL circuit breaker is energized. Operate CAUTION LIGHTS DIM switch and check that indication of a STALL is provided by Apex and the single square inverse video annunciation above the fuel selector knob, and is accompanied by a persistent aural warning.

4) Pump flaps to 12° down as indicated by flap position indicator.

5) Release cabin roof upholstery sufficiently to gain access to upper vane circuit microswitch positioned adjacent to flap actuator.

CAUTION **ENSURE THAT PITOT HEATER SWITCH IS OFF BEFORE TOUCHING STALL WARNING VANE, OTHERWISE A BURN COULD RESULT.**

6) With second operator actuating upper vane, loosen two switch adjusting screws and adjust switch until indication of STALL is provided by the display and the warning is heard, then back off until STALL indication and accompanied aural warning ceases. Tighten adjusting screws at this point.

7) Return flaps to fully up position and with upper vane actuated, lower flaps smoothly and slowly until STALL indication is displayed and accompanied by the aural warning, then stop pumping.

8) Check that flap position indicator reads 12° ± 2°.

9) If STALL indication is not displayed properly, nor is there the accompanied aural warning within range specified in Para 2.B.8), repeat adjustment procedure as necessary until STALL is displayed properly, (refer to Para 1.) and is accompanied by the persistent aural warning.

10) Replace upholstery released in Para 2.B.5) and disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Function Test Stall Warning System

1) Ensure that wing flap system is operating properly and that indicating system is accurate.

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Using handpump, pump wing flaps to fully up position.

4) Set BUS TIE switch to NORMAL and ensure CAUTION LTS NORMAL circuit breaker is energized. Operate CAUTION LIGHTS TEST switch and check that STALL is provided by the display, as indicated in Para 1.

Note Two people are required to do Para 2.C.5) through Para 2.C.9).

CAUTION **ENSURE THAT PITOT HEATER SWITCH IS OFF BEFORE TOUCHING STALL WARNING VANE, OTHERWISE A BURN COULD RESULT.**

5) Actuate lower vane gently to complete circuit and hold vane in that position. Pump flaps from fully up to fully down and check that STALL is provided by the displays as indicated and the audible “stall” stall warning is heard through the headset.

6) Release lower vane and return flaps to fully up position.

7) Actuate upper vane gently and check that there is no STALL indication displayed and no warning heard.

8) With upper vane actuated, pump flaps slowly and smoothly down until STALL indication is displayed and accompanied by the aural warning, then stop pumping. Check that flap position indicator reads 12° ± 2°.

9) With upper vane actuated, continue pumping flaps to fully down. Check that STALL indication is displayed and the aural warning heard.

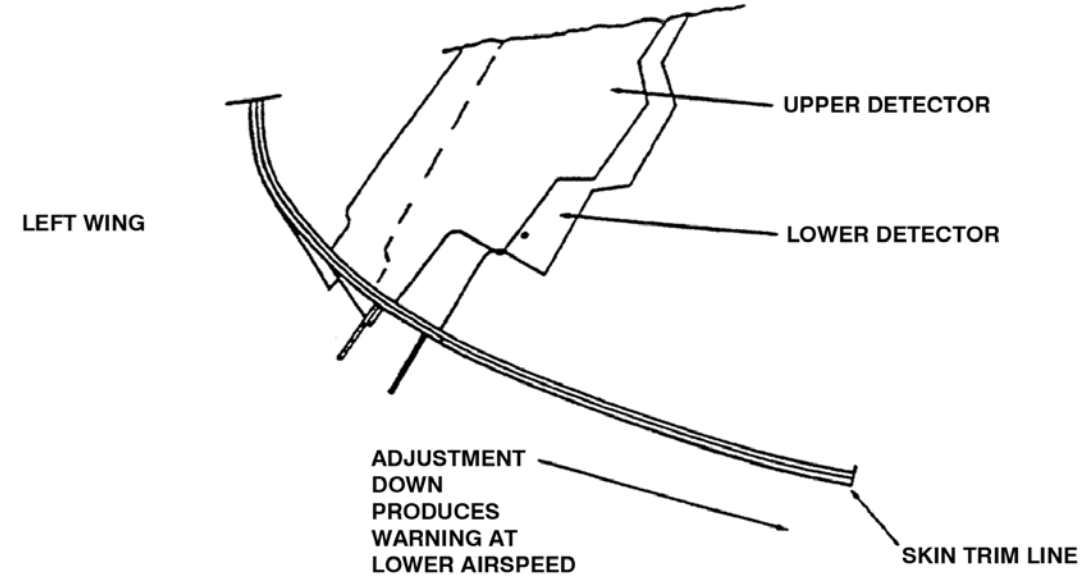
10) If light does not illuminate within specified range, adjust upper vane microswitch in accordance with procedure given in Para 2.B.

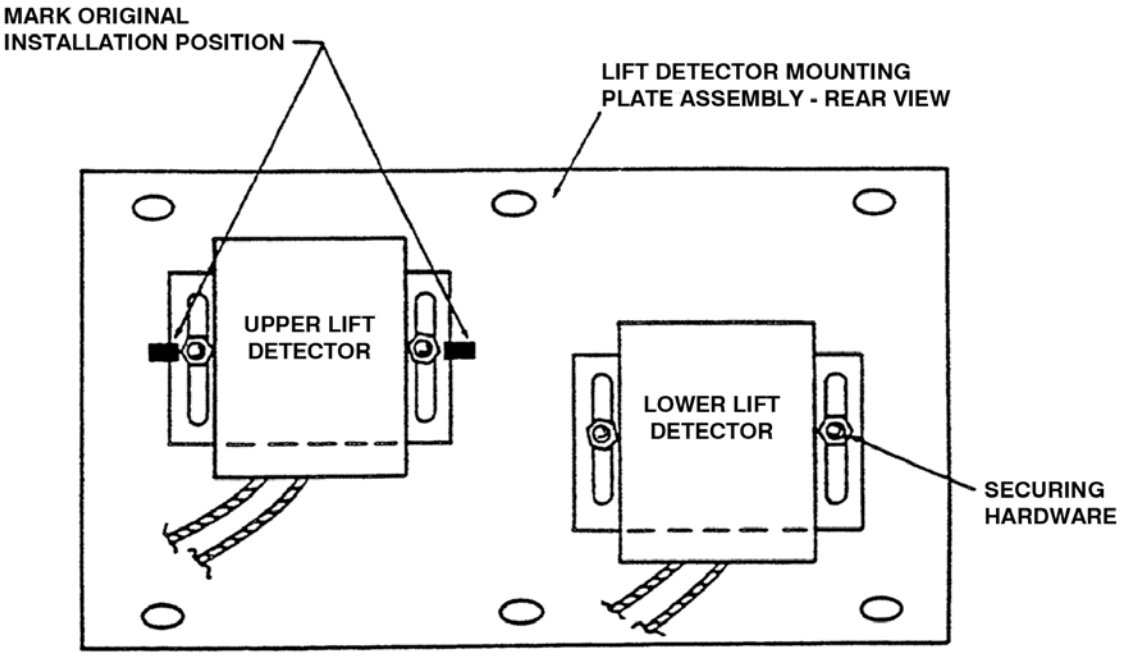
11) When the conditions of Para 2.C.10) are satisfied, select PITOT HEAT switch to ON and check that the stall warning vane heaters warm up.

CAUTION **DO NOT LEAVE PITOT HEATER SWITCHED ON FOR MORE THAN ONE MINUTE.**

12) Select PITOT HEAT switch to OFF.

13) Disconnect DC power source from aircraft (refer to 12-00-00, Servicing – General Data, Para 2.).





LIFT DETECTOR	FLAP POSITION (DEGREES DOWN)	CHANGE PRODUCED BY 1/8-INCH MOVEMENT OF LIFT DETECTOR
		KNOTS
LOWER	0	1.87
LOWER	10	1.19
UPPER	20	0.81
UPPER	40	0.47

Calibration Data for Lift Detector Vane
Figure 1

Wing Flaps System – General Data

1. General

Refer to Figure 1 and Figure 2 for details.

The wing flaps system consists of inboard and outboard flaps hinged to each wing; each outboard flap has an aileron hinged at its trailing edge and each inboard flap has a trailing flap hinged at its trailing edge. A selector lever in the flight compartment controls a hydraulic selector valve, which determines the position of a hydraulic actuator. The actuator, through linkage, bellcranks and push-pull rods, connects to and operates the wing flaps. An indicator system is provided to register the position of the flaps. Lights are provided on each side of the flap selector lever and inside the flap position indicator pointer (refer to 33-10-00, Flight Compartment Lights (Mod 6/2028) – General Data.

2. Description

The inboard (fore and trailing) and outboard flaps hinged to each wing are all interconnected by push-pull rods, idler levers and bellcranks, which are connected to a single hydraulic actuator piston rod by links. The flap selector lever in the flight compartment, is connected by cables to a selector pulley assembly mounted on the selector valve operating shaft; the selector valve being attached to the flap actuator. The selector pulley assembly comprises two pulleys mounted side by side, either of which operates the selector valve independently. The cables from the flap selector lever are connected to the outboard pulley, and a follow-up cable system from the flap actuator piston rod connects to the inboard pulley. Hydraulic pipe lines from the aircraft main hydraulic system (refer to 29-00-00, Hydraulic Power – General Data) connect to the flap selector valve. Flap position indication is provided on the Multi-Function Display. To maintain aircraft trim during flap operation, an interconnecting system to the right elevator trim tab is installed.

A. Wing Flaps

The inboard and outboard flap hinge arms, are hinged to the flap hinge arm attachment brackets, which are bolted to the wing structure. A common flap hinge arm attachment bracket is located at station 172.50 and accommodates the end hinge arms of the inboard fore and outboard flaps. Fore flap hinge arms at stations 97.50 and 247.15 (approximately) are connected to the flap system push-pull rods. The inboard trailing flap hinge arms are hinged to those of the fore flap, and a connecting rod connects both flap hinge arms at station 97.50 to control the travel of the trailing flap.

B. Wing Flap Hydraulic System

The wing flap hydraulic circuit lines are tapped into the main hydraulic system lines at the hydraulic power package located beneath the flight compartment floor, and are routed to the flap selector valve on the flap actuator. Restrictions are provided in the flaps up and flaps down ports of the actuator and a check valve is fitted in the pressure supply line at the flap selector valve. Both the flap up and down lines, incorporate a thermal relief valve.

C. Flap Actuator

The wing flap actuator, with the selector valve attached, is mounted in a frame located centrally in the cabin roof. The actuator piston rod is connected by links to the left and right operating bellcranks.

D. Flap Selector Lever

The flap selector lever assembly, mounted in the flight compartment overhead console, comprises a selector lever, quadrant, pulley and cable system, and serrated segment. The selector lever, located behind the engine control levers, is connected by a connecting rod to the quadrant immediately forward of the lever. A pawl on the lever, engages the serrated segment about which the lever pivots, for intermediate settings of the flaps. Cables connected to the quadrant are routed over pulleys to connect to the outboard pulley on the flap selector valve spindle.

E. Flap Follow-up System

A cable connected to a bracket on the actuator piston rod, is routed to the inboard pulley on the flap selector valve spindle.

F. Flap Position Indicator

The flap position indicator system comprises of a rotary potentiometer installed on the LH flap. As the flap is extended, the transducer provides an analog electrical output signal (0 to 28 Volt DC) to the Data Acquisition Units. Using an interface (ARINC429) the signal is converted and flap position is shown on the EFIS computer screen in the flight compartment. Flap position indication is provided on the Multi-Function Display in the flight compartment.

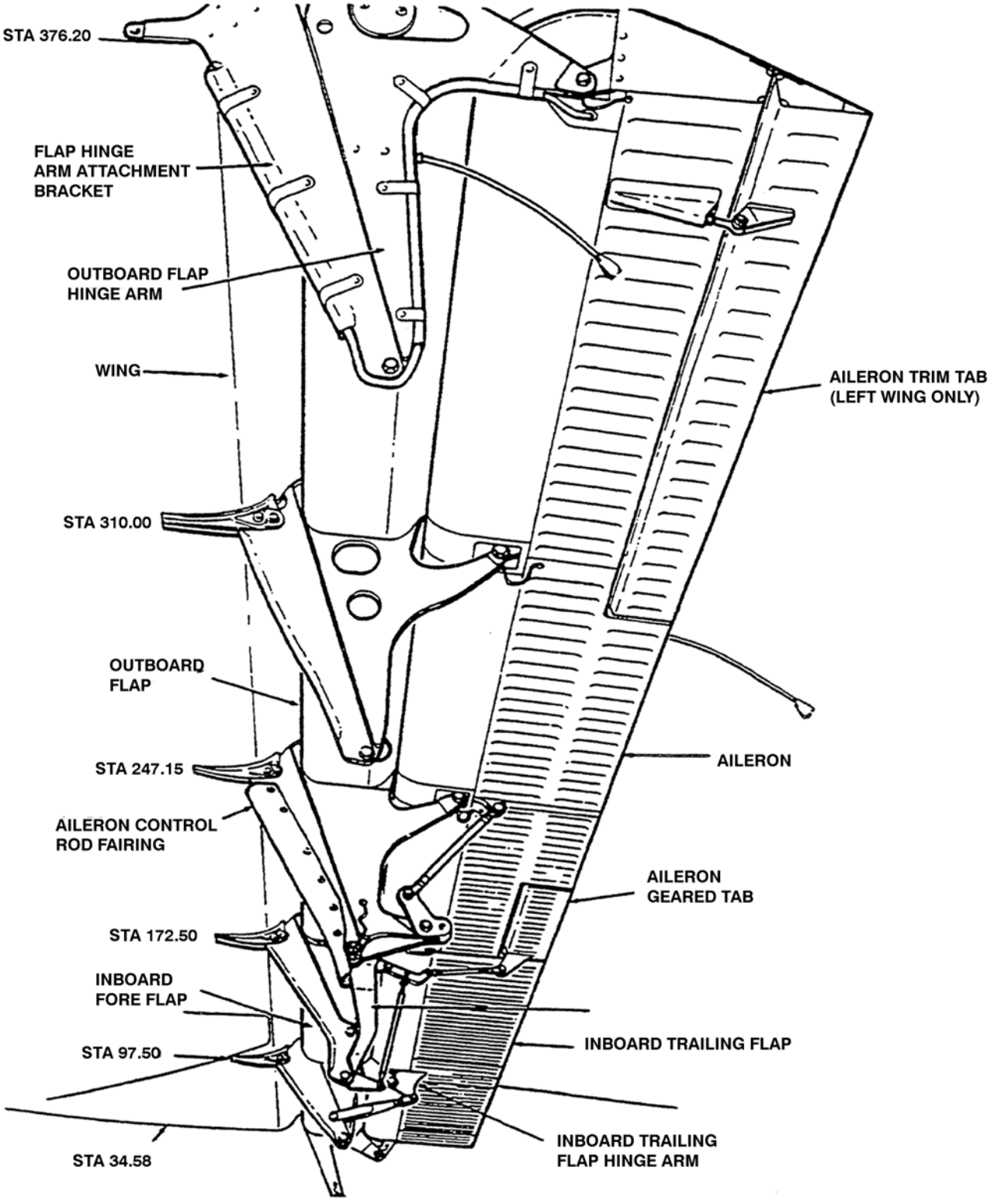
3. Operation

The wing flap hydraulic system derives its power for the wing flap actuator from the aircraft hydraulic system main pressure line, and returns displaced fluids from the actuator into the aircraft hydraulic system main return line (refer to 29-00-00, Hydraulic Power – General Data). When the aircraft hydraulic system is pressurized, and the flap selector lever is moved to the flaps down position, a corresponding amount of movement is transmitted to the quadrant, and through the cables to the outboard pulley on the selector valve shaft. Rotation of the outboard pulley opens the selector valve to permit hydraulic fluid under pressure to enter the flap actuator down port to extend the actuator piston rod. The piston rod, through the links, operates the left and right inboard operating bellcranks, which through the push-pull rods, idler levers and outboard operating bellcranks, moves the flaps to the down position. During actuator piston rod extension, the follow-up mechanism cable rotates the inboard cable pulley on the selector valve shaft, which is part of an epicyclic gear train, to close the selector valve. The reverse of the above operation applies when a flaps up selection is made.

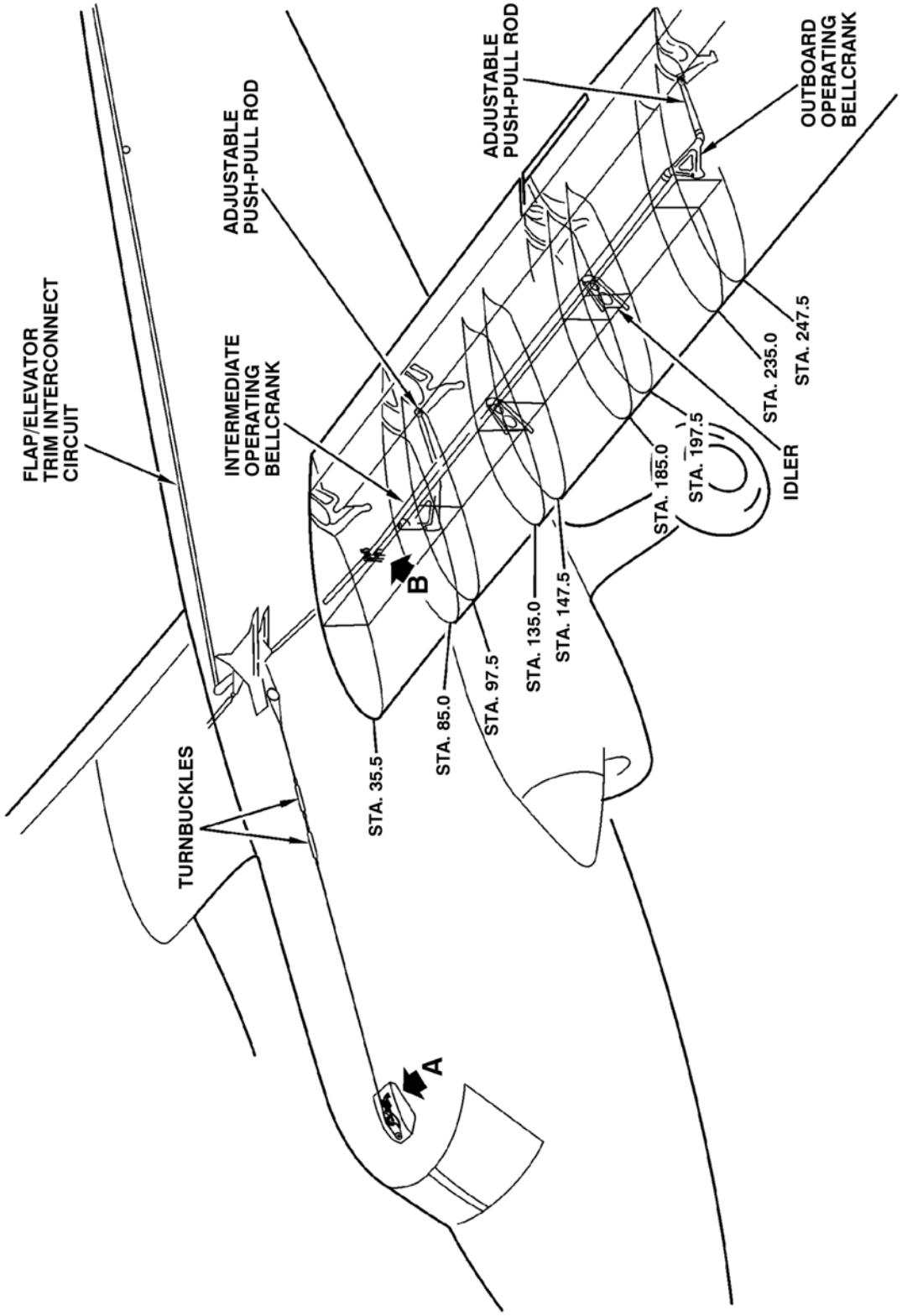
The Multi-Function Display (MFD) registers the flap position by the signal it receives from the position transducer installed on the LH flap.

A flap/elevator trim interconnect ball screw jack, attached to the airframe structure and the right inboard operating bellcrank, operates during flap movement to transmit movement to the flap/elevator trim interconnect circuit to maintain aircraft elevator trim during flap operation.

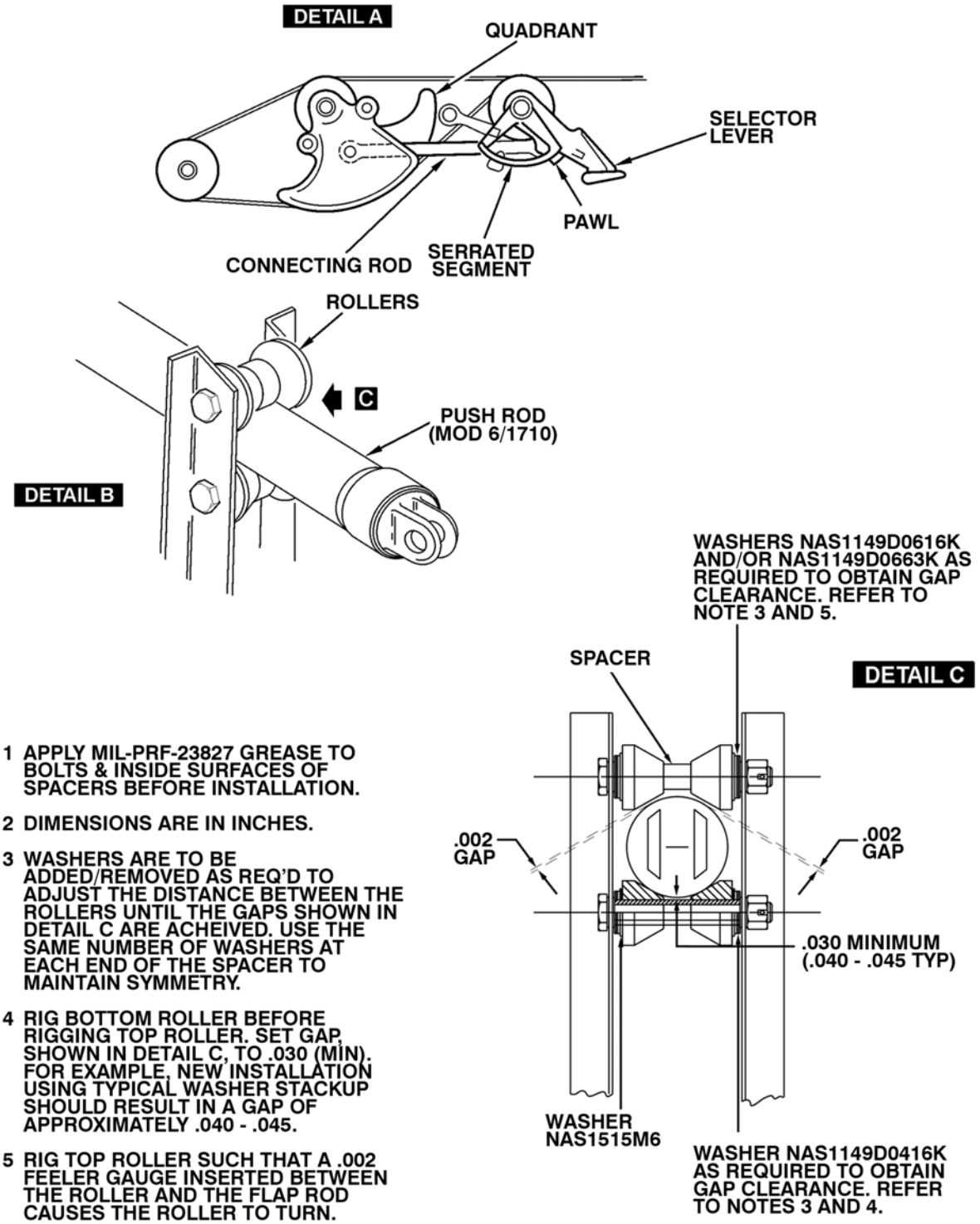
The restrictors in the actuator flaps up and down ports limit the rate of flap movement, and the check valve in the pressure supply line isolates the flap circuit in the event of a failure in the hydraulic pressure supply.



Wing Flap Arrangement
Figure 1



Wing Flap System
Figure 2 (Sheet 1 of 4)



NOTE: 1 APPLY MIL-PRF-23827 GREASE TO BOLTS & INSIDE SURFACES OF SPACERS BEFORE INSTALLATION.

2 DIMENSIONS ARE IN INCHES.

3 WASHERS ARE TO BE ADDED/REMOVED AS REQ'D TO ADJUST THE DISTANCE BETWEEN THE ROLLERS UNTIL THE GAPS SHOWN IN DETAIL C ARE ACHEIVED. USE THE SAME NUMBER OF WASHERS AT EACH END OF THE SPACER TO MAINTAIN SYMMETRY.

4 RIG BOTTOM ROLLER BEFORE RIGGING TOP ROLLER. SET GAP, SHOWN IN DETAIL C, TO .030 (MIN). FOR EXAMPLE, NEW INSTALLATION USING TYPICAL WASHER STACKUP SHOULD RESULT IN A GAP OF APPROXIMATELY .040 - .045.

5 RIG TOP ROLLER SUCH THAT A .002 FEELER GAUGE INSERTED BETWEEN THE ROLLER AND THE FLAP ROD CAUSES THE ROLLER TO TURN.

6 BOLTS SHOULD NOT BE OVER TIGHTENED. ROLLERS/SPACERS SHOULD TURN FREELY ON BOLTS.

PRE MOD 6/2186

Wing Flap System
Figure 2 (Sheet 2 of 4)

NOTE: 1 APPLY MIL-PRF-23827 GREASE TO BOLTS & INSIDE SURFACES OF SPACERS BEFORE INSTALLATION.

2 DIMENSIONS ARE IN INCHES.

3 WITH ROD PUSHED ALL THE WAY INSIDE THE WING (I.E. FLAP FULLY RETRACTED). PRE-FIT ROLLERS, SPACERS, & BOLTS (NOTE: NO WASHERS) BY CENTERING THE ROD ON ROOT RIB CUT-OUT AND BETWEEN ROLLERS/SPACERS. SPACERS SHALL BE APPROX CENTERED BETWEEN SUPPORT ANGLES.

4 SLIDE ROLLERS TO MAKE CONTACT WITH ROD, THEN MEASURE GAPS G1, G2, G3 & G4. REMOVE PRE-FIT ROLLERS, SPACERS & BOLTS.

5 DETERMINE QUANTITY OF NAS1149D0663K AND NAS1149D0616K WASHERS USING DIMENSION: Gx-.05 = Tx; CHOOSE EQUAL OR NEXT GREATER THICKNESS.

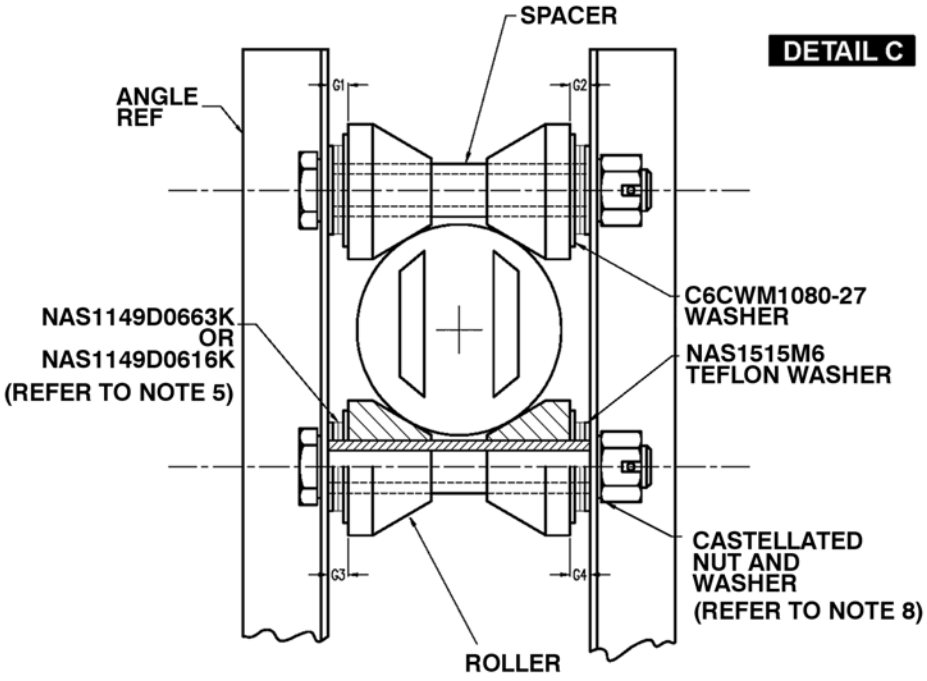
EXAMPLE:
G1=.189, THEREFORE G1-.05 = .139
REQD WASHER: 2x NAS1149D0663K = .126
1x NAS1149D0616K = .016
.142

6 INSTALL BOTTOM ROLLER/SPACER WITH WASHERS AS PER NOTE 5. ENSURE THE SPACER IS APPROX CENTERED: I.E. EQUAL GAP BETWEEN FWD & AFT SUPPORT ANGLE.

7 WHILE PUSHING THE FLAP ROD DOWNWARD TO COMPRESS THE LOWER ROLLERS, INSTALL THE UPPER ROLLERS/SPACER AS PER NOTE 5. CENTER THE SPACER ON THE FLAP ROD.

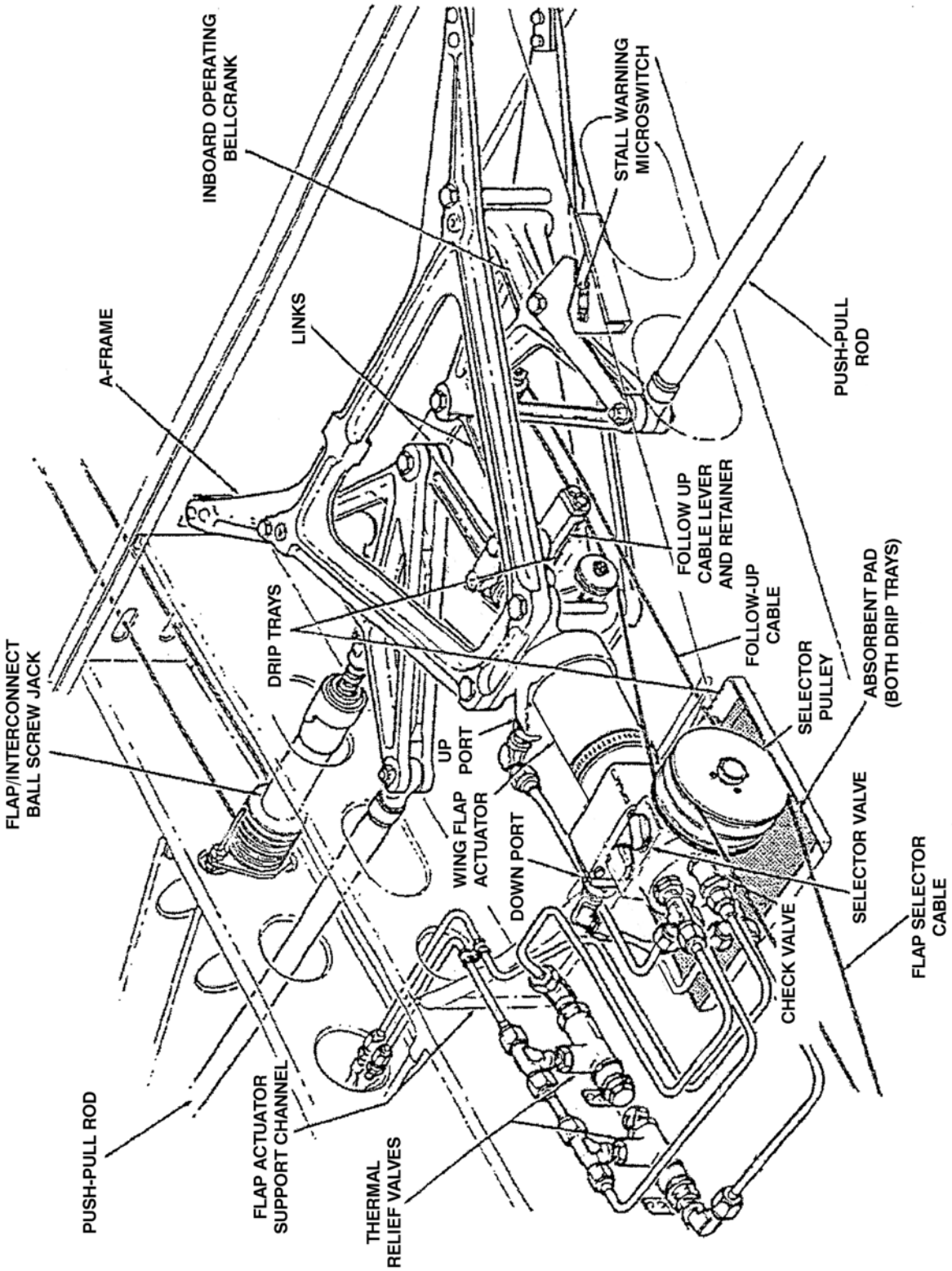
8 FIRST, TIGHTEN BOTH NUTS TO ENSURE A SNUG CONTACT AGAINST THE WASHER/BRACKET. THEN, TIGHTEN BOTH NUTS BY 1 TO 3 CASTELLATED SLOTS TO ALIGN COTTER PIN HOLE ON THE BOLT. IF MORE THAN 50% OF PIN HOLE IS ABOVE THE NUT CASTELLATION, ADD NAS1149D0463K OR NAS1149D0416 WASHER UNDER NUT.

9 MAKE SURE THE ROLLERS/SPACERS ARE TURNING FREELY ON BOLTS (ROD NOT SLIDING) WHILE THE FLAP RODS MOVE IN AND OUT.



POST MOD 6/2186

Wing Flap System
Figure 2 (Sheet 3 of 4)



Wing Flap System
Figure 2 (Sheet 4 of 4)

4. Servicing

A. Bleed Wing Flap System

Note Bleeding the wing flap system need only be accomplished if the flap operation is not smooth, a hydraulic line or component has been replaced, or the main hydraulic system has been dismantled.

1) Check that hydraulic reservoir is full.

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.), and check that BUS TIE switch is set to NORMAL.

3) Set HYDRAULIC OIL PUMP circuit breaker.

Note Observe that motor pump charges accumulators and that motor cuts out when pressure reaches 1575 ± 50 psi (Mod 6/1570).

4) Check that flap area is clear of equipment and personnel.

5) Select flaps down and allow them to travel to full down position, select flaps up and allow them to travel to full up position. Repeat for ten full selections to ensure complete bleeding of system.

6) Check flap system for smooth operation.

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

5. Adjustment/Test

Refer to Figure 3 for details.

A. Rig Wing Flaps

1) Move flap selector lever to flaps up position, using handpump; pump flaps to fully up position.

2) Check that cable terminals on flap selector and follow-up pulley are 30° from the vertical and that the distance between each terminal, measured in a horizontal plane fore and aft, is 1.75 inches.

3) If cable terminal settings are incorrect, remove attachment bolts and disconnect the left and right push-pull rods at each inboard operating bellcrank. Support flaps in down position and carry out Para 5.A.4) through Para 5.A.26) as necessary.

4) With flap selector in up position (against rear stop), operate handpump to fully retract flap actuator.

5) Slacken two 3/8–inch bolts attaching bellcrank links to flap actuator piston rod sufficiently to loosen follow-up cable lever.

6) Slacken selector cable turnbuckles in cabin roof, and set cable terminal in outboard pulley to 30° forward of vertical; check dimension between terminals, measured fore and aft, is 1.75 inches.

7) Adjust selector cable turnbuckles in cabin roof to tension cable to values given in 27-00-00, Flight Controls – General Data. Ensure that settings made to both pulleys remain unchanged.

8) Tighten two 3/8–inch bolts loosened in Para 5.A.5), and adjust follow-up cable adjusting nut on lower "A" frame to tension cable to value given in 27-00-00, Flight Controls – General Data, ensuring that settings on both pulleys remain unchanged.

9) Move selector to down position (against serrated stop) and operate handpump until actuator piston has extended exactly 3.40 inches. Adjust serrated stop to limit movement of selector lever exactly at this point.

Note If movement of 3.40 inches cannot be obtained, the follow-up valve is misaligned and Para 5.A.5) through Para 5.A.8) must be repeated to ensure correct follow-up valve alignment before proceeding.

10) Retract and extend actuator several times to ascertain piston extension is constant at 3.40 inches.

11) Connect push-pull rods disconnected in Para 5.A.3).

12) Move selector lever to up and pump flaps fully up.

13) Check that trailing edge of inboard trailing flaps align with rigging plates on each side of fuselage and rigging plates on inboard and outboard fore flaps are aligned with wing shroud trailing edge.

14) If fore flaps are not aligned with their rigging plates, remove attachment bolt and disconnect adjustable push-pull rod at each inboard and outboard fore flap on both sides and carry out Para 5.A.15) through Para 5.A.26).

15) Screw out, to full extension, push-pull rod to left inboard fore flap and reattach to flap.

16) Pump flap to full up and adjust push-pull rod, as necessary, until rigging plate is aligned.

17) Repeat Para 5.A.15) and Para 5.A.16) for right inboard fore flap, then left and right outboard fore flaps, in that order.

18) Check that all fore flap hinge arms are bearing equally on adjacent rubber pads when flaps are fully up, and a gap of 0.040/0.200 inch exists between fore flap hinge arms and flap hinge arm attachment brackets.

Note If required, trim rubber pads to suit actual gap after rigging.

19) If, with fore flaps fully up, inboard trailing flaps are not aligned with fuselage rigging plates, remove attachment bolt, and disconnect connecting rod of each trailing flap and carry out Para 5.A.20) and Para 5.A.21).

20) Hold left inboard trailing flap up in line with fuselage plate and adjust connecting rod until attachment bolt can be freely installed.

21) Repeat Para 5.A.20) for right inboard trailing flap.

22) Check left and right aileron rigging (refer to 27-10-00, Aileron Control System – General Data).

23) Ensure that all push-pull and connecting rod ends are tightened and correctly locked.

24) Check all flap control surface clearances are within limits (refer to 27-00-00, Flight Controls – General Data).

25) Operate flaps and check for full and free range of movement.

26) Check rigging of flap/elevator interconnect trim system (refer to 27-36-00, Flap/Elevator Interconnect Trim System – General Data).

27) Do a flap position potentiometer function test as per 27-50-31, Flap Angle Position Indicator – General Data, Para 3.B.

28) If adjustment is required for the flap position potentiometer, adjust as per 27-50-31, Flap Angle Position Indicator – General Data, Para 3.A.

B. Operational Test Wing Flap System

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.), and check that BUS TIE switch is set to NORMAL.

2) Set HYDRAULIC OIL PUMP circuit breaker.

   Note Observe that motor pump charges accumulators and that motor cuts out when pressure reaches 1550 ± 50 psi.

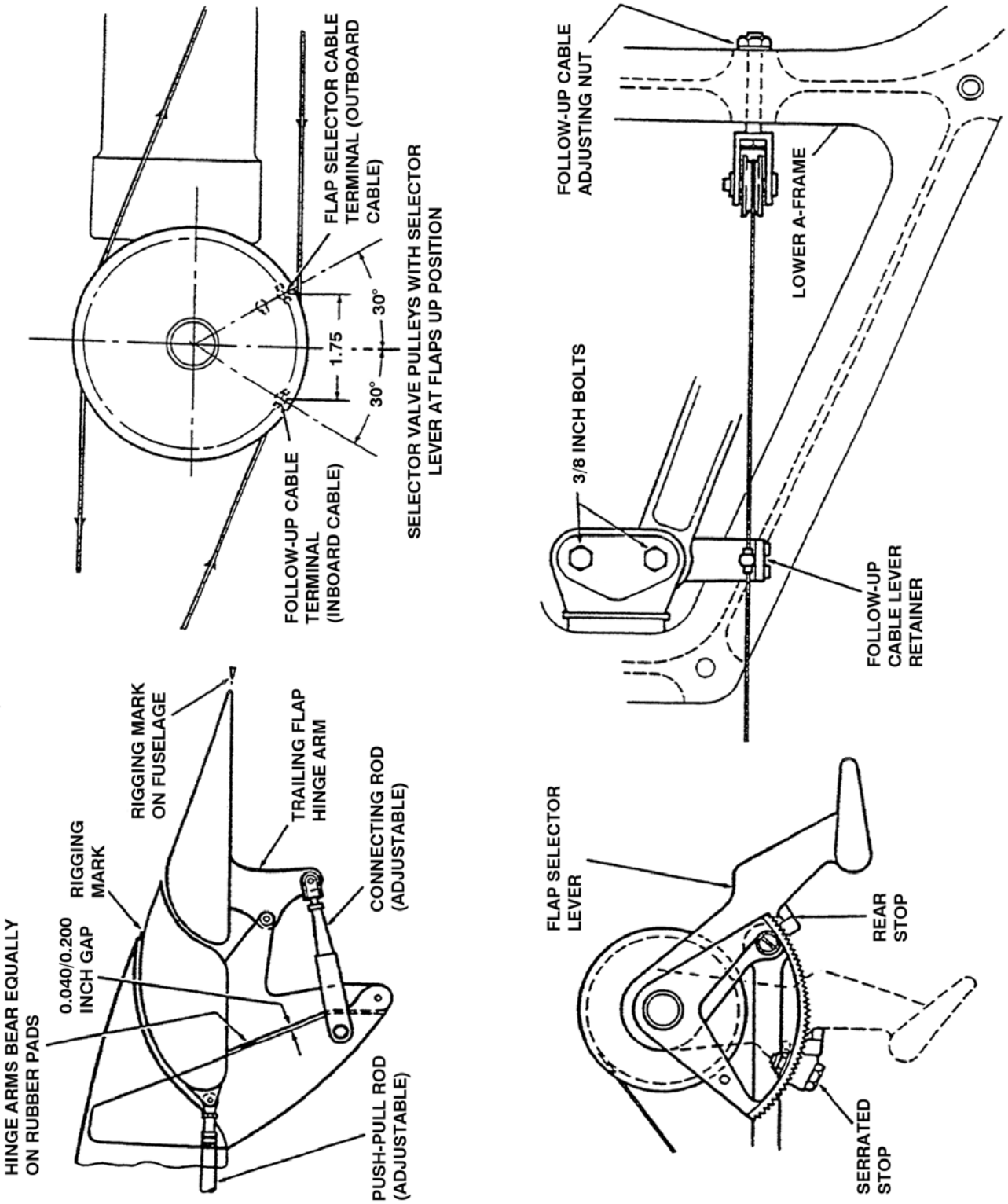
3) Check that flap area is clear of equipment and personnel.

4) Select flaps down and check that flaps are fully extended in 19 to 25 seconds.

5) Select flaps up and check that flaps are fully retracted to 35 to 50 seconds.

6) Make several selections and check for smooth operation over full travel.

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).



Wing Flaps System – Rigging
Figure 3

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Flap Actuator – Maintenance Practices

1. Removal/Installation

Note This procedure requires a minimum of two people.

A. Remove Wing Flap Actuator

Refer to Figure 1 and Figure 2 for details.

1) Remove forward cabin ceiling panels.

2) Disengage HYDRAULIC OIL PUMP circuit breaker and install locking clip. Discharge hydraulic pressure from accumulators by operating brake pedals.

3) Move flap selector lever to flaps up position, and pump flaps up, using hand pump.

4) Disconnect flap/elevator interconnect ball screw jack from right inboard bellcrank.

Note Tape cables to flap interconnect ball screw jack cable drum to prevent cable unseating from pulley grooves, and inadvertent movement of ball screw jack.

5) Move flap selector lever to 20° down position. Disengage flap/interconnect ball screw jack from bellcrank by pumping flaps down, using hand pump.

6) Support flaps and remove bolts securing left and right push-pull rods to inboard bellcranks.

7) Lift flaps manually to withdraw push-pull rods from inboard bellcranks. Support flaps at this position.

8) Move flap selector lever to up position.

9) Remove flap actuator forward drip tray.

10) Identify up and down selector cables, and remove selector cable terminals from selector valve pulley.

11) Disconnect hydraulic lines at selector valve. If fitted, remove ski hydraulic lines routed through A-frame. Cap lines (refer to 29-00-00, Hydraulic Power – General Data).

12) Release hydraulic line clips from flap actuator support channel.

13) Remove four bolts and washers securing A-frame to airframe structure.

14) Support flap actuator and A-frame assembly, and remove six bolts and washers securing flap actuator support channel to airframe structure.

15) Remove flap actuator and A-frame assembly from aircraft.

16) Slacken selector valve follow-up cable at A-frame pulley adjusting nut, and remove terminals from selector valve pulley. Release cable from flap actuator pulley and flap actuator support channel.

17) Remove follow-up cable lever, links from flap actuator piston rod end fitting.

18) Remove flap actuator support channel, cushion and clamp from flap actuator.

19) Remove A-frame assembly and aft drip tray from flap actuator.

20) Remove hydraulic lines between flap selector valve and actuator. Cap ports.

B. Install Wing Flap Actuator

Refer to Figure 1 and Figure 2 for details.

1) Position A-frame and aft drip tray on flap actuator. Coat attachment bolts with corrosion protective compound, Shell CG929 (refer to 20-30-01, Protective Coatings Hard and Soft Film Corrosion Preventive Compound – General Data).

2) Secure A-frame and aft drip tray to flap actuator, using two bolts, washers and nuts.

   Note Bolts to be installed with heads up.

3) Insert links in flap actuator piston rod end fitting, with two teflon washers in position.

   Note Install teflon washer between lower face of each link and piston rod end fitting.

4) Secure links to piston rod end fitting as follows:

   a) Position follow-up cable lever on bottom face of piston rod end fitting, with cable and retainer aligned with actuator pulley, and lever flange facing down.

   b) Coat attachment bolts with corrosion protective compound, Shell CG929. Secure indicator post, links and follow-up cable lever to piston rod end fitting using two bolts, washers, nuts and cotter pins.

      Note Bolts to be installed with heads up.

5) Install cushion and support channel on actuator. Secure with clamp but do not fully tighten at this stage. On aircraft incorporating Mod 6/1748, loosen four bolts joining support channels. Install cushion and support channels on actuator and secure with clamp. Ensure that bottom edges of support channels are parallel.

6) Ensure selector lever and selector valve pulleys are in flaps-up position, and flap actuator piston is fully retracted (refer to Figure 2).

7) Route upper portion of follow-up cable under flap actuator pulley and over selector valve pulley. Connect cable terminal to selector valve follow-up pulley.

8) Route remaining portion of follow-up cable through hole in flap actuator support channel and under flap selector valve follow-up pulley.

9) Connect follow-up cable terminal to flap selector follow-up pulley.

10) Remove caps from flap actuator ports, position suitable container to receive spillage and extend flap actuator piston approximately 2.0 inches. Reinstall caps.

11) Finally, position flap actuator and A-frame assembly in aircraft. Secure support channel with six bolts and washers, and A-frame with four bolts, washers and nuts.

12) Tighten clamps securing actuator to support channel. Clip hydraulic lines to flap actuator support channel. On aircraft incorporating Mod 6/1748, tighten four bolts joining support channels, allowing actuator to retain its free position as determined by anchor bolts at rear frame.

13) Connect inboard bellcranks to push-pull rods as follows:

   a) Remove caps from flap actuator ports, position container to receive spillage, and move flap actuator piston until bellcranks engage with push-pull rods. Replace caps.

   b) Secure bellcranks to push-pull rods with two bolts, four washers, two nuts and two cotter pins.

      Note Bolts to be installed heads down with one washer positioned under each bolt head.

14) Ensure selector valve selector pulley is in flaps-up position (refer to Figure 2). Connect flap selector cable terminals to selector pulley and lockwire in position.

    Note Flap-down selector cable is routed over selector pulley.

15) Tension flap selector and follow-up cables (refer to 27-00-00, Flight Controls – General Data). Ensure position of selector valve pulleys remain unchanged. Lock turnbuckles.

    Note Adjust tension of follow-up cables at A-frame pulley adjustment nut.

16) Remove caps, connect hydraulic lines to flap selector valve, and install hydraulic lines between actuator and selector valve.

17) Using hand pump, pump flaps up and guide right bellcrank into fork end of flap/interconnect ball screw jack.

18) Install one nylon washer between upper and lower faces of bellcrank and flap/interconnect ball screw jack fork end.

19) Coat bellcrank and flap/interconnect ball screw jack attachment bolt with corrosion, protective compound, Shell CG929. Install bolt, and secure with washer, nut and cotter pin.

    Note Bolt to be installed with head up.

20) Remove tape from flap/interconnect ball screw jack cable drum.

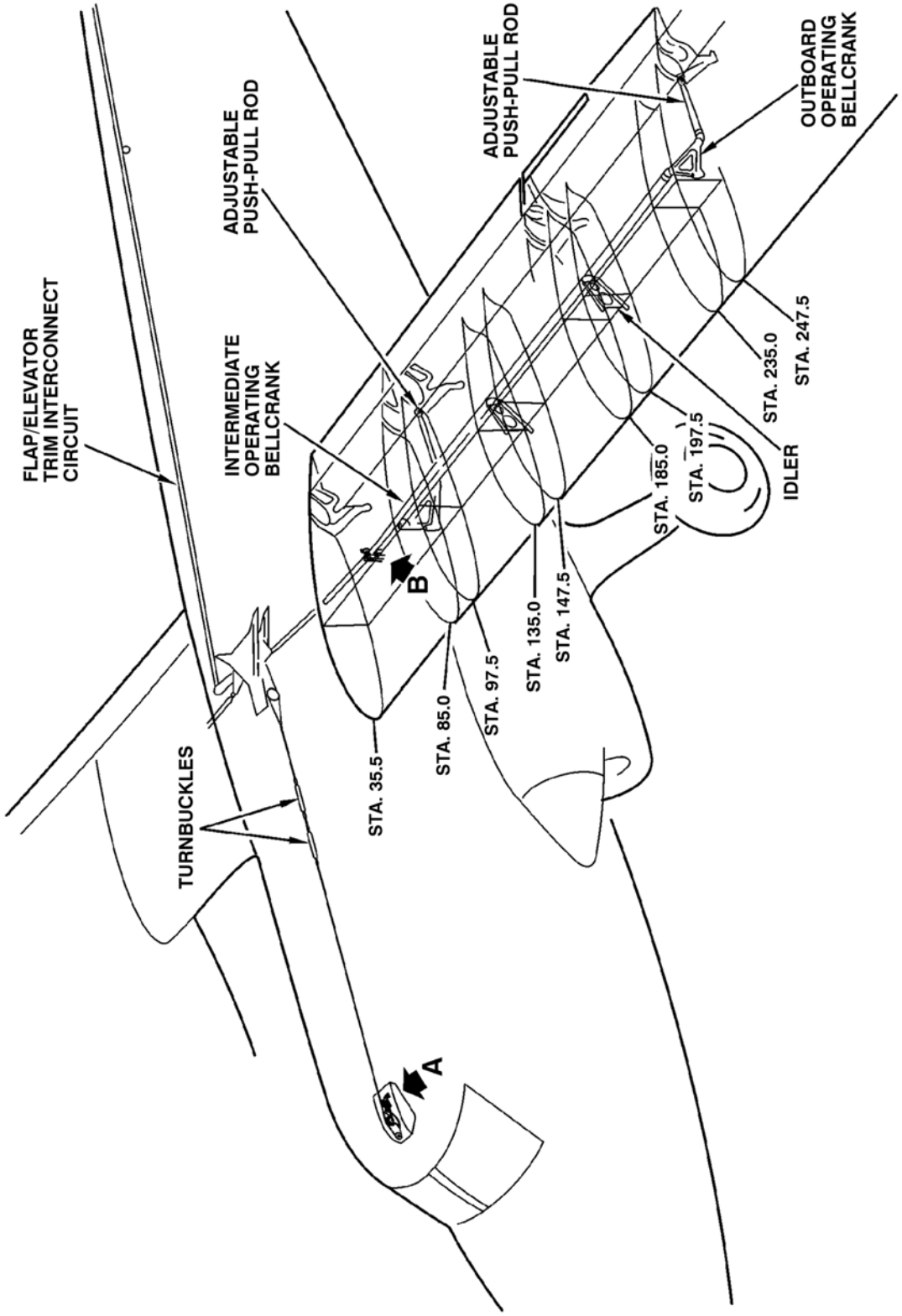
21) Rig flaps (refer to 27-50-00, Wing Flaps System – General Data).

22) Bleed wing flap system (refer to 27-50-00, Wing Flaps System – General Data).

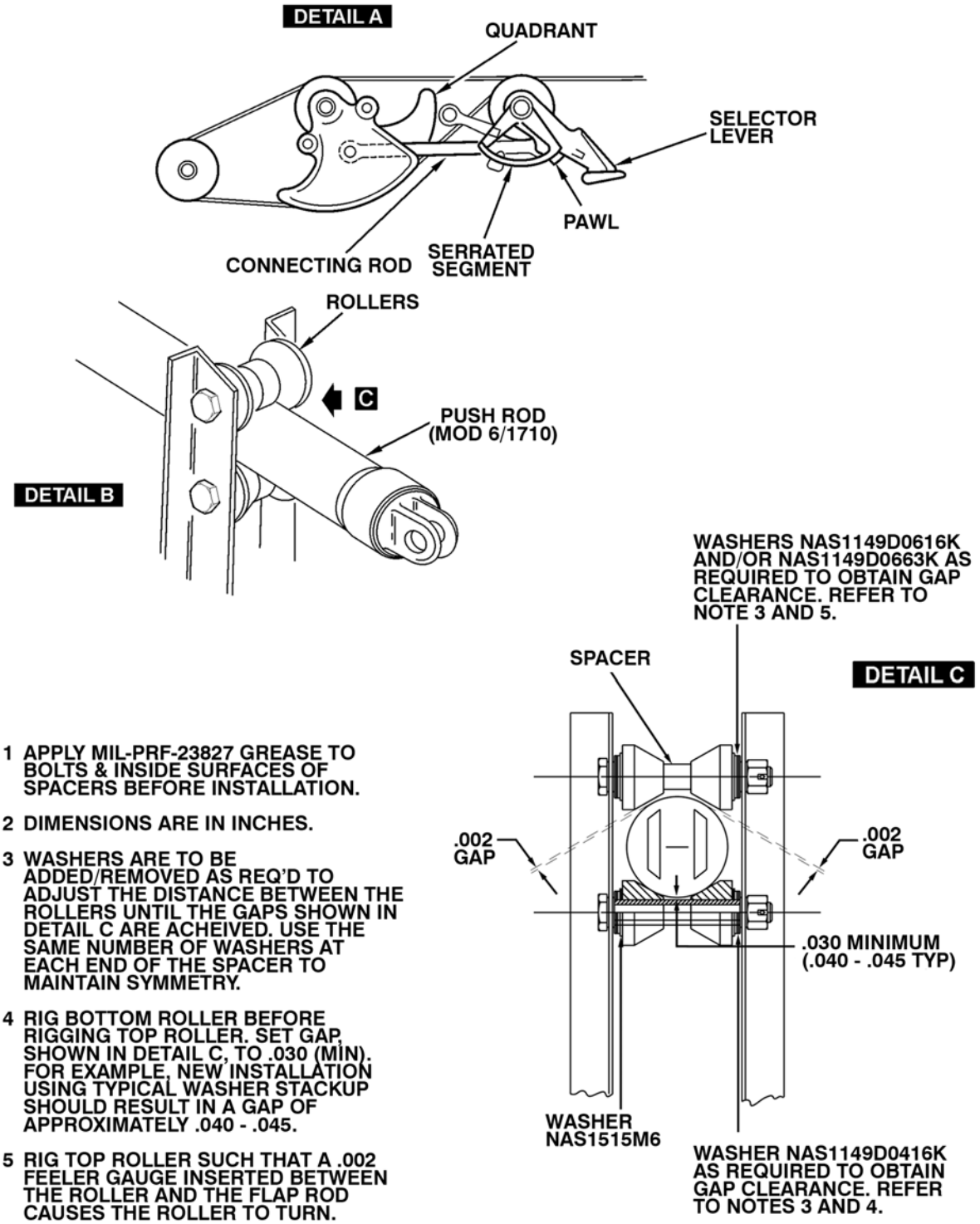
23) Do an operational test on wing flap system (refer to 27-50-00, Wing Flaps System – General Data).

24) Install forward drip tray.

25) Install removed cabin ceiling panels.



Wing Flap System
Figure 1 (Sheet 1 of 4)



NOTE: 1 APPLY MIL-PRF-23827 GREASE TO BOLTS & INSIDE SURFACES OF SPACERS BEFORE INSTALLATION.

2 DIMENSIONS ARE IN INCHES.

3 WASHERS ARE TO BE ADDED/REMOVED AS REQ'D TO ADJUST THE DISTANCE BETWEEN THE ROLLERS UNTIL THE GAPS SHOWN IN DETAIL C ARE ACHEIVED. USE THE SAME NUMBER OF WASHERS AT EACH END OF THE SPACER TO MAINTAIN SYMMETRY.

4 RIG BOTTOM ROLLER BEFORE RIGGING TOP ROLLER. SET GAP, SHOWN IN DETAIL C, TO .030 (MIN). FOR EXAMPLE, NEW INSTALLATION USING TYPICAL WASHER STACKUP SHOULD RESULT IN A GAP OF APPROXIMATELY .040 - .045.

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6 BOLTS SHOULD NOT BE OVER TIGHTENED. ROLLERS/SPACERS SHOULD TURN FREELY ON BOLTS.

PRE MOD 6/2186

Wing Flap System
Figure 1 (Sheet 2 of 4)

NOTE: 1 APPLY MIL-PRF-23827 GREASE TO BOLTS & INSIDE SURFACES OF SPACERS BEFORE INSTALLATION.

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3 WITH ROD PUSHED ALL THE WAY INSIDE THE WING (I.E. FLAP FULLY RETRACTED). PRE-FIT ROLLERS, SPACERS, & BOLTS (NOTE: NO WASHERS) BY CENTERING THE ROD ON ROOT RIB CUT-OUT AND BETWEEN ROLLERS/SPACERS. SPACERS SHALL BE APPROX CENTERED BETWEEN SUPPORT ANGLES.

4 SLIDE ROLLERS TO MAKE CONTACT WITH ROD, THEN MEASURE GAPS G1, G2, G3 & G4. REMOVE PRE-FIT ROLLERS, SPACERS & BOLTS.

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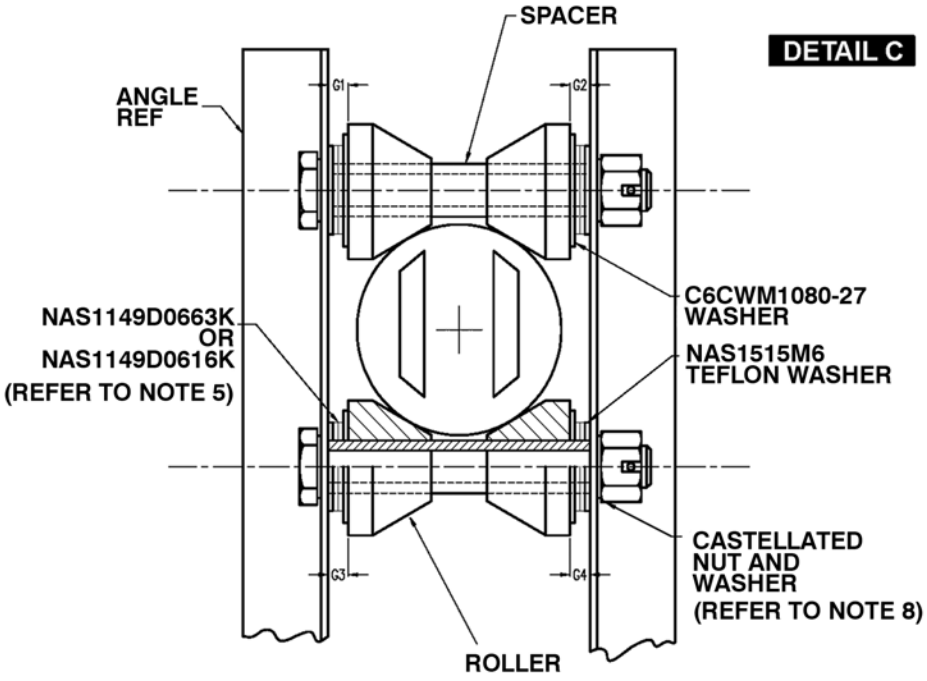
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.142

6 INSTALL BOTTOM ROLLER/SPACER WITH WASHERS AS PER NOTE 5. ENSURE THE SPACER IS APPROX CENTERED: I.E. EQUAL GAP BETWEEN FWD & AFT SUPPORT ANGLE.

7 WHILE PUSHING THE FLAP ROD DOWNWARD TO COMPRESS THE LOWER ROLLERS, INSTALL THE UPPER ROLLERS/SPACER AS PER NOTE 5. CENTER THE SPACER ON THE FLAP ROD.

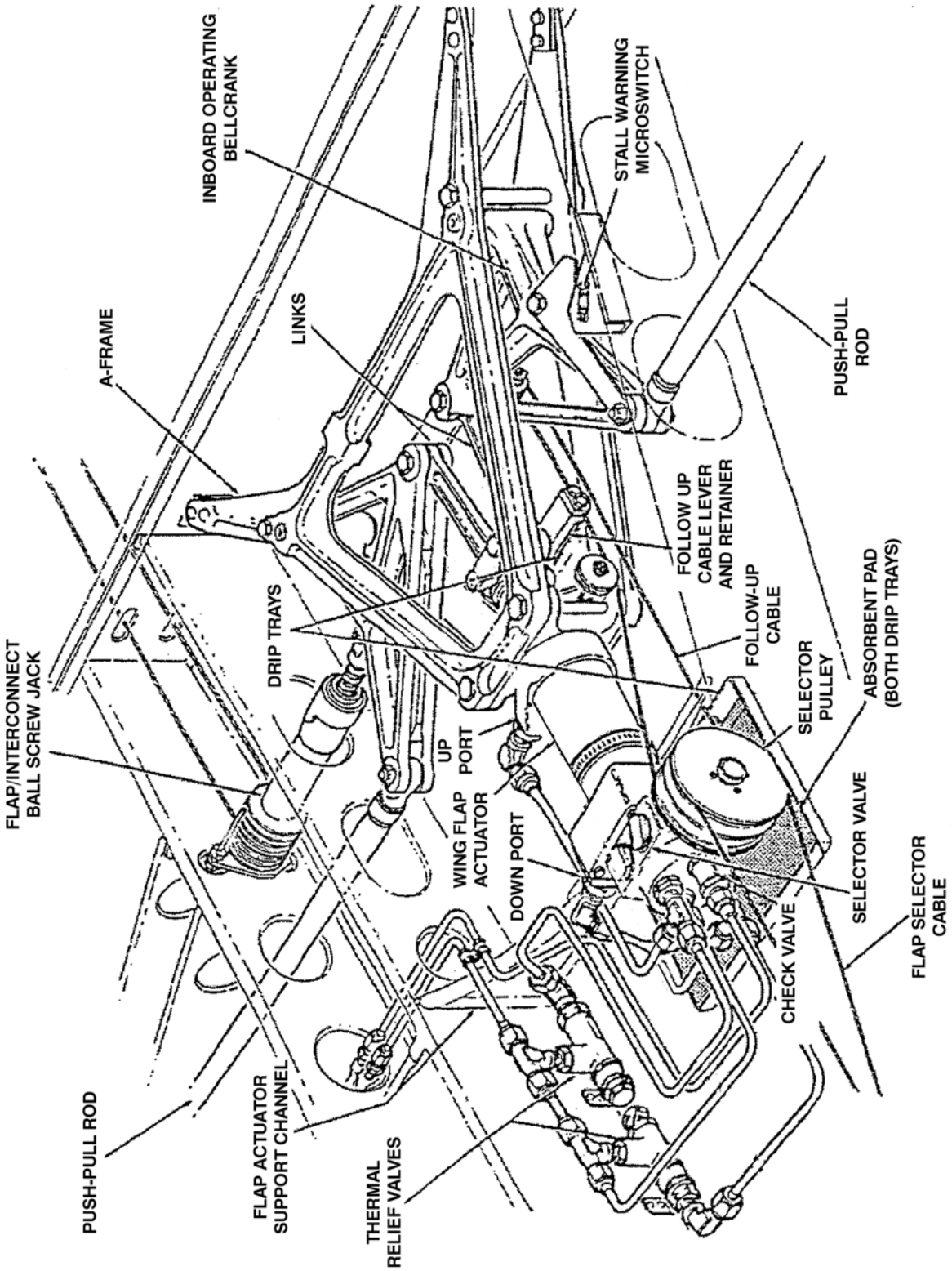
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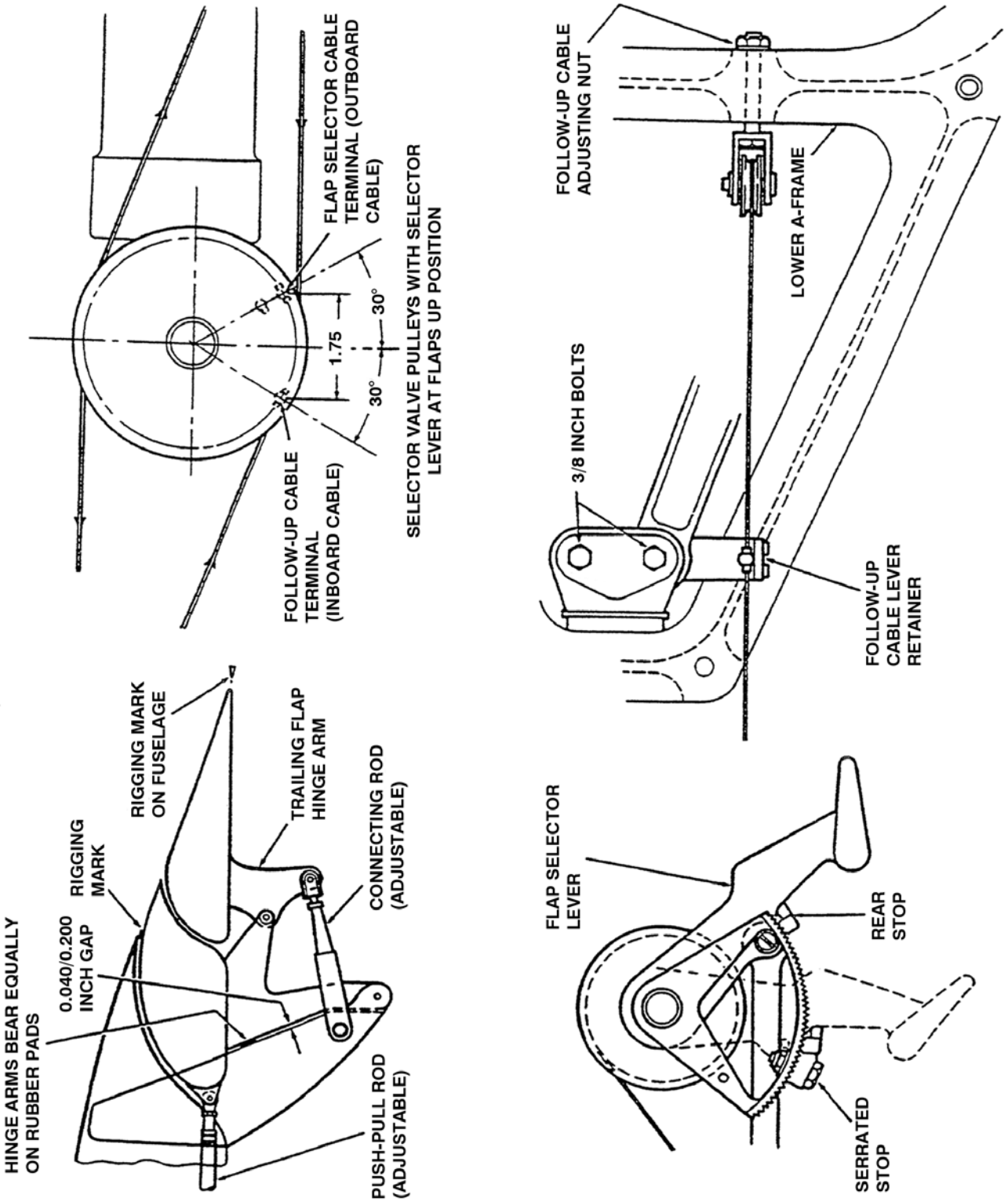


POST MOD 6/2186

Wing Flap System
Figure 1 (Sheet 3 of 4)



Wing Flap System
Figure 1 (Sheet 4 of 4)



Wing Flaps System – Rigging
Figure 2

2. Adjustment/Test

A. Bench Pressure Test

Equipment required:

- Suitable hydraulic test rig capable of supplying and maintaining 2300 psi. Clean hydraulic fluid (MIL–H–5606).

1) Compress piston rod into flap actuator; leave down port open, and connect test rig to up port. (Refer to Figure 2).

2) Apply 2300 psi hydraulic pressure for three minutes; there must be no external leakage from open down port.

3) Leave up port open, and connect test rig to down port; fully extend piston rod.

4) Apply 2300 psi hydraulic pressure for three minutes; there must be no leakage from open up port or cylinder.

5) Release hydraulic pressure and check cylinder for permanent deformation.

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Flap Selector Valve – Maintenance Practices

1. Removal/Installation

   A. Remove Flap Selector Valve

      Refer to Figure 1 and Figure 2 for details.

      1) Remove forward cabin ceiling panels.

      2) Disengage HYDRAULIC OIL PUMP circuit breaker and install locking clip. Discharge hydraulic pressure from accumulators by operating brake pedals.

      3) Move flap selector lever to flap down position and pump flaps down, using handpump.

      4) Remove flap actuator forward drip tray.

      5) Slacken flap selector cable at turnbuckles. Identify up and down cables and remove cable terminals from selector valve selector pulley.

      6) Slacken selector valve follow-up cable at A-frame pulley adjusting nut, and remove cable terminals from selector valve follow-up pulley.

      7) Position suitable container and disconnect hydraulic lines at selector valve. Cap lines and ports.

      8) Support selector valve and remove four bolts and washers securing selector valve to flap actuator. Remove selector valve from aircraft.

   B. Install Flap Selector Valve

      Refer to Figure 1 and Figure 2 for details.

      1) Position selector valve on flap actuator and secure with four bolts and washers. Lockwire bolts.

      2) Position suitable container and remove caps from hydraulic lines and selector valve ports. Connect hydraulic lines to selector valve.

      3) Install follow-up cable terminals in selector valve follow-up pulley. Lockwire terminals in position.

         Note If selector valve pulleys are in flaps up position (refer to Figure 2), rotate to permit cable terminal installation.

      4) Install flap selector cable terminals in selector valve selector pulley. Lockwire terminals in position. Refer to Note in Para 1.B.3)).

         Note Flaps down selector cable is routed over selector pulley.

      5) Tension flap selector and follow-up cables. (Refer to 27-00-00, Flight Controls – General Data).

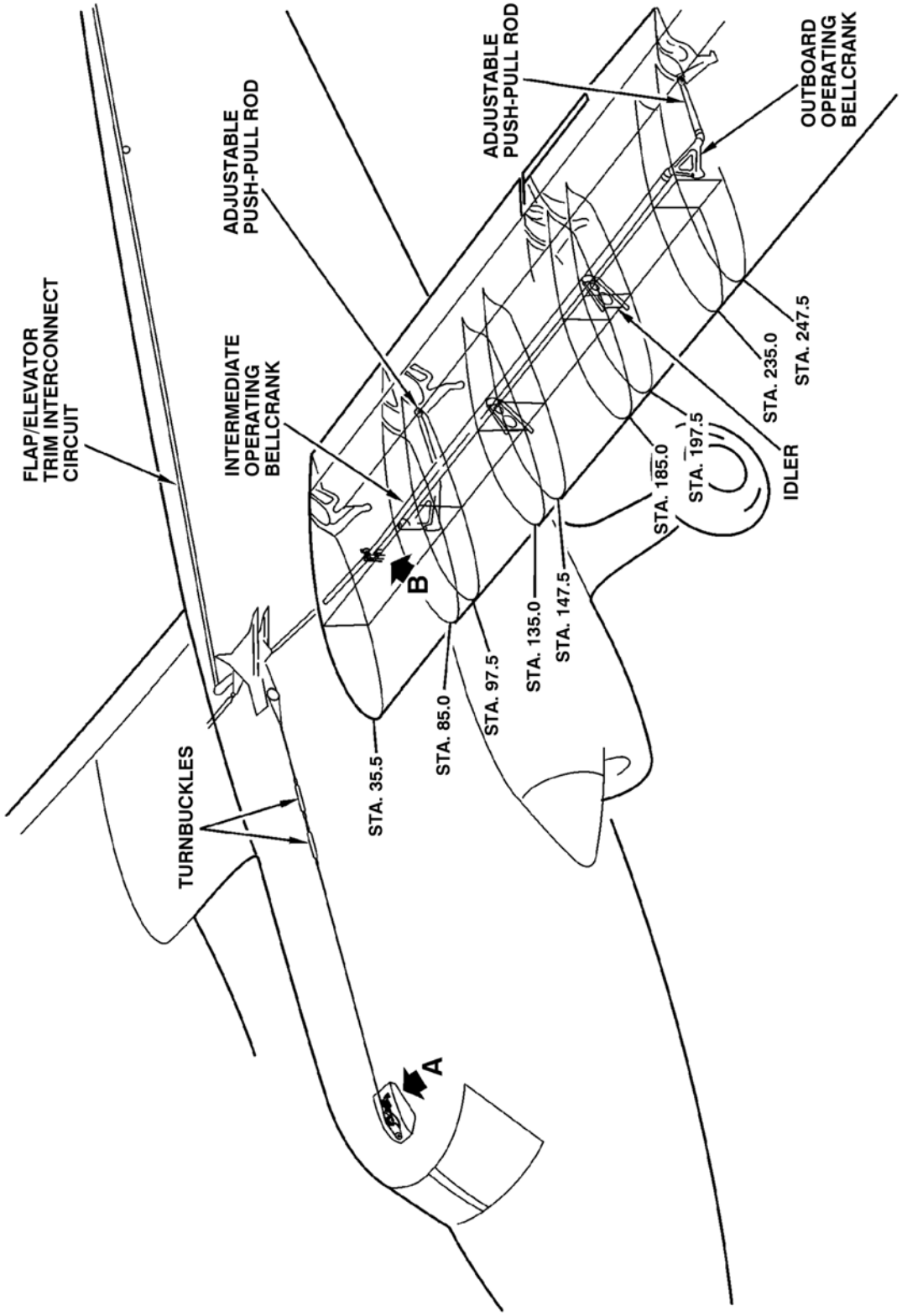
      6) Rig flaps. (Refer to 27-50-00, Wing Flaps System – General Data).

      7) Remove clip from HYDRAULIC OIL PUMP circuit breaker and bleed flap actuator. (Refer to 27-50-11, Flap Actuator – General Data).

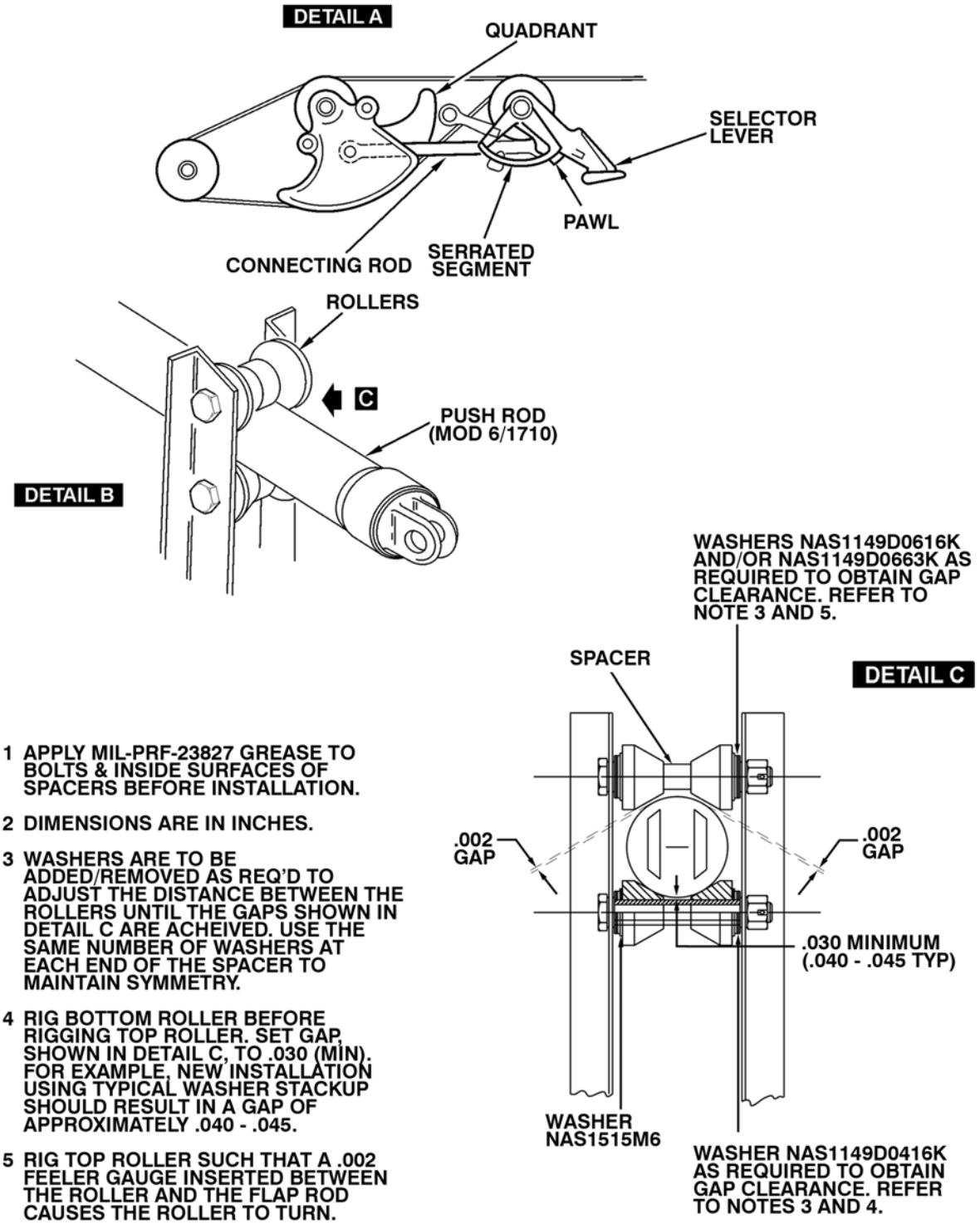
      8) Do an operational test on wing flaps. (Refer to 27-50-00, Wing Flaps System – General Data).

      9) Install flap actuator forward drip tray.

      10) Install forward cabin ceiling panels.



Wing Flap System
Figure 1 (Sheet 1 of 4)



NOTE: 1 APPLY MIL-PRF-23827 GREASE TO BOLTS & INSIDE SURFACES OF SPACERS BEFORE INSTALLATION.

2 DIMENSIONS ARE IN INCHES.

3 WASHERS ARE TO BE ADDED/REMOVED AS REQ'D TO ADJUST THE DISTANCE BETWEEN THE ROLLERS UNTIL THE GAPS SHOWN IN DETAIL C ARE ACHEIVED. USE THE SAME NUMBER OF WASHERS AT EACH END OF THE SPACER TO MAINTAIN SYMMETRY.

4 RIG BOTTOM ROLLER BEFORE RIGGING TOP ROLLER. SET GAP, SHOWN IN DETAIL C, TO .030 (MIN). FOR EXAMPLE, NEW INSTALLATION USING TYPICAL WASHER STACKUP SHOULD RESULT IN A GAP OF APPROXIMATELY .040 - .045.

5 RIG TOP ROLLER SUCH THAT A .002 FEELER GAUGE INSERTED BETWEEN THE ROLLER AND THE FLAP ROD CAUSES THE ROLLER TO TURN.

6 BOLTS SHOULD NOT BE OVER TIGHTENED. ROLLERS/SPACERS SHOULD TURN FREELY ON BOLTS.

PRE MOD 6/2186

Wing Flap System
Figure 1 (Sheet 2 of 4)

NOTE: 1 APPLY MIL-PRF-23827 GREASE TO BOLTS & INSIDE SURFACES OF SPACERS BEFORE INSTALLATION.

2 DIMENSIONS ARE IN INCHES.

3 WITH ROD PUSHED ALL THE WAY INSIDE THE WING (I.E. FLAP FULLY RETRACTED). PRE-FIT ROLLERS, SPACERS, & BOLTS (NOTE: NO WASHERS) BY CENTERING THE ROD ON ROOT RIB CUT-OUT AND BETWEEN ROLLERS/SPACERS. SPACERS SHALL BE APPROX CENTERED BETWEEN SUPPORT ANGLES.

4 SLIDE ROLLERS TO MAKE CONTACT WITH ROD, THEN MEASURE GAPS G1, G2, G3 & G4. REMOVE PRE-FIT ROLLERS, SPACERS & BOLTS.

5 DETERMINE QUANTITY OF NAS1149D0663K AND NAS1149D0616K WASHERS USING DIMENSION: Gx-.05 = Tx; CHOOSE EQUAL OR NEXT GREATER THICKNESS.

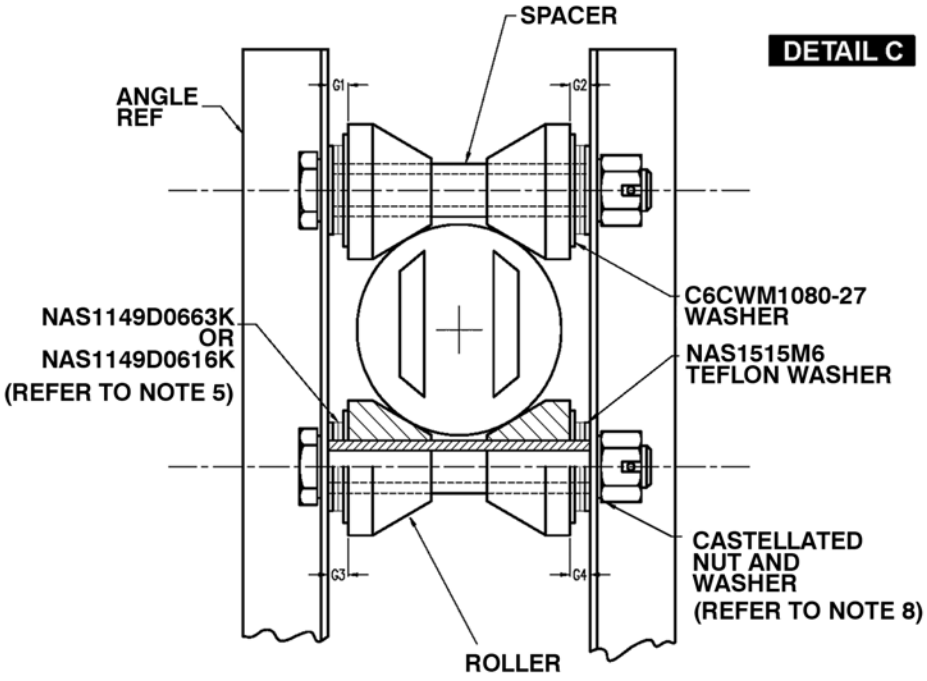
EXAMPLE:
G1=.189, THEREFORE G1-.05 = .139
REQD WASHER: 2x NAS1149D0663K = .126
1x NAS1149D0616K = .016
.142

6 INSTALL BOTTOM ROLLER/SPACER WITH WASHERS AS PER NOTE 5. ENSURE THE SPACER IS APPROX CENTERED: I.E. EQUAL GAP BETWEEN FWD & AFT SUPPORT ANGLE.

7 WHILE PUSHING THE FLAP ROD DOWNWARD TO COMPRESS THE LOWER ROLLERS, INSTALL THE UPPER ROLLERS/SPACER AS PER NOTE 5. CENTER THE SPACER ON THE FLAP ROD.

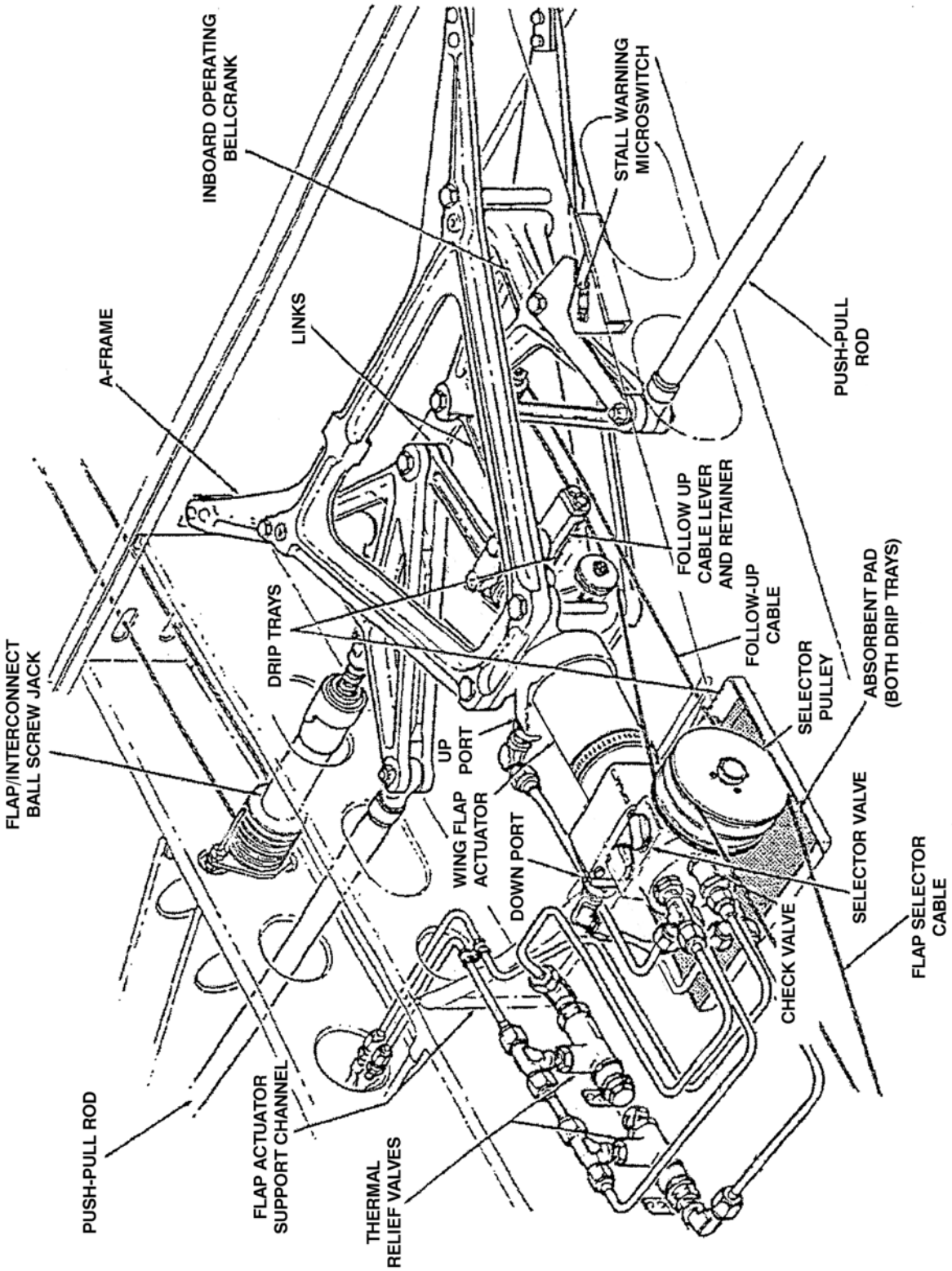
8 FIRST, TIGHTEN BOTH NUTS TO ENSURE A SNUG CONTACT AGAINST THE WASHER/BRACKET. THEN, TIGHTEN BOTH NUTS BY 1 TO 3 CASTELLATED SLOTS TO ALIGN COTTER PIN HOLE ON THE BOLT. IF MORE THAN 50% OF PIN HOLE IS ABOVE THE NUT CASTELLATION, ADD NAS1149D0463K OR NAS1149D0416 WASHER UNDER NUT.

9 MAKE SURE THE ROLLERS/SPACERS ARE TURNING FREELY ON BOLTS (ROD NOT SLIDING) WHILE THE FLAP RODS MOVE IN AND OUT.

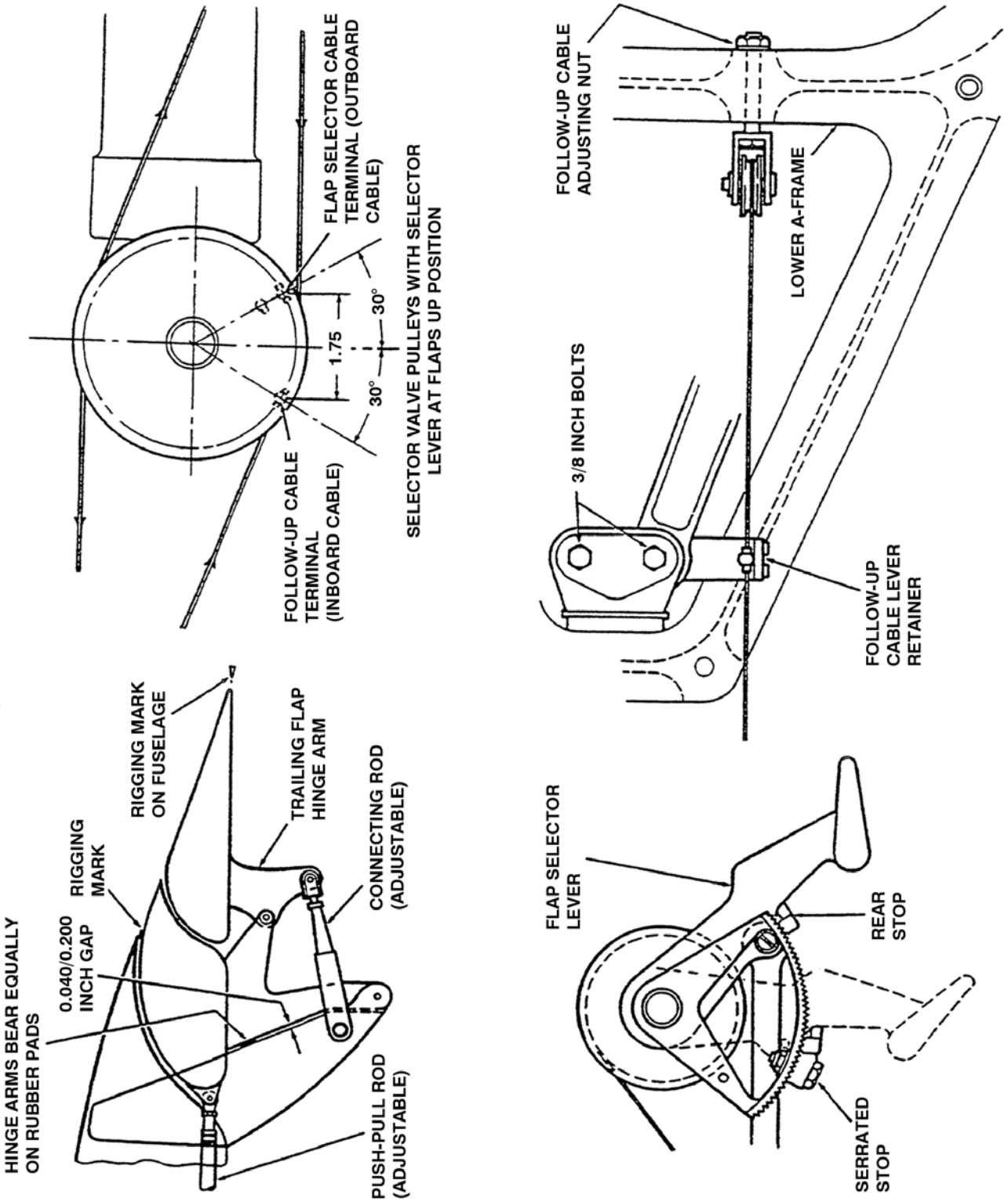


POST MOD 6/2186

Wing Flap System
Figure 1 (Sheet 3 of 4)



Wing Flap System
Figure 1 (Sheet 3 of 4)



Wing Flaps System – Rigging
Figure 2

2. Adjustment/Test

A. Bench Pressure Test

Refer to Figure 3 for details.

Equipment required:

- Hydraulic test rig capable of supplying and maintaining 2250 psi and flow rate of 1.0 gpm
- Clean hydraulic fluid (MIL–H–5606)

1) Proof pressure:

a) Connect test rig to port P, cap port R, and connect ports C1 and C2 together with shorting line.

b) Apply and maintain 2250 psi hydraulic pressure for two minutes, and rotate pulleys A and B together. Check there is no external leakage, permanent set or deformation.

c) Release hydraulic pressure, remove cap front port R, leave ports C1 and C2 connected by shorting line, rotate pulleys A and B to neutral position, apply 10 psi hydraulic-pressure and maintain for two minutes. Check there is no external leakage.

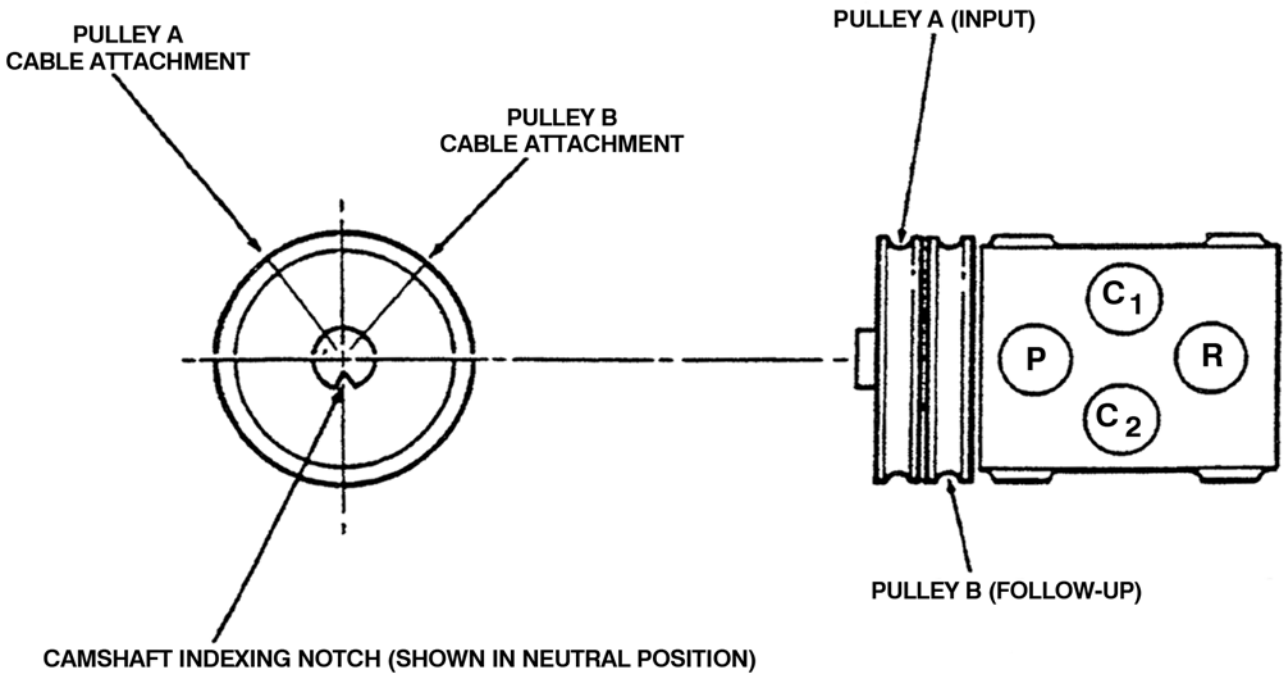
2) Internal leakage:

a) Connect test rig to port P, leave ports R, C1 and C2 open. Ensure pulleys A and B are in neutral position.

b) Apply 1500 psi hydraulic pressure at flow rate of 0.5 to 1.0 gpm, check ports R, C1 and C2 for leaks; leakage must not exceed one drop in three minutes. Release hydraulic pressure then repeat test at 500 psi.

c) Release hydraulic pressure, connect test rig to port R; leave ports P, C1 and C2 open. Ensure pulleys A and B are in neutral position.

d) Apply 1500 psi hydraulic pressure at flow rate of 0.5 to 1.0 gpm, check ports P, C1 and C2 for leaks; leakage must not exceed one drop in three minutes. Release hydraulic pressure then repeat test at 500 psi.



Flap Selector Valve – Test Ports
Figure 3

Flap Angle Position Indicator – General data

1. General

A potentiometer is mounted on the port side fuselage attached to the inboard fore flap hinge arm at fuselage station 243.00, and sends a signal of the flap angle position through the DAU and onto the Flight Data Recorder and the flap angle position is indicated on the system screen.

2. Removal/Installation

Equipment and Materials:

- Corrosion preventative compound MIL–C–16173 Grade 2 or equivalent

A. Remove the Flap Angle Position Potentiometer

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit safety clip.

3) Disconnect the electrical connection.

4) Loosen the nut and screw that hold the lever assembly to the potentiometer shaft mounted in mounting bracket. Remove lever assembly.

5) Remove the safety wire and the (3) screws and clamps which secure the potentiometer onto the mounting bracket.

6) Remove potentiometer.

B. Install the Flap Angle Position Potentiometer

Note Coat all bolt shanks with the corrosive preventative compound MIL–C–16173 Grade 2 or equivalent before installation.

1) Ensure the electrical system is OFF (refer to 24-00-00, Electrical Power – General Data).

2) Ensure the FLIGHT CONTROLS INDICATION (A14) circuit breaker is pulled and fitted with safety clip.

3) Position the potentiometer on to the mounting bracket. Secure with the (3) clamps and screws. Safety wire screws.

4) Locate the lever assembly on potentiometer shaft so that the mechanism rotates freely. Ensure clearance is maintained with the structure through full travel, and a minimum gap of 0.05 inch between the lever assembly and the fuselage skin.

5) Secure with a washer, nut and screw.

6) Install bolt through the push rod assembly end, install between the bearing and the hinge arm, and bolt through the lever assembly end. Install washer, nut and cotter pin.

7) Adjust the flap angle position potentiometer. Refer to Para 3.A.

8) Reconnect the electrical connection.

9) Replace FLIGHT CONTROLS INDICATION (A14) circuit breaker.

3. Adjustment/Test

A. Adjustment Flap Angle Position Potentiometer

Equipment and Materials:

- DC Voltmeter
- Digital Inclinometer

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Measure the Ground Power Voltage (V_{GP}) at the external power relay (or other suitable location).

3) Pull the FLIGHT CONTROLS INDICATION (A14) circuit breaker and fit with safety clip.

4) Select flaps down 20°.

5) Check that the flaps have reached 20° with the digital inclinometer. Move the flap selector up or down to get exactly 20° with inclinometer.

6) Disconnect terminal 2 of the electrical connection going to the potentiometer and connect voltmeter positive to the lead from terminal 2 while attaching voltmeter negative to aircraft ground.

CAUTION **ENSURE THE TERMINAL 2 LEAD DOES NOT COME INTO CONTACT WITH AN ELECTRICAL GROUND.**

7) Loosen screw and nut on the lever assembly at the potentiometer shaft enough that the shaft turns freely.

8) Reset the FLIGHT CONTROLS INDICATION (A14) circuit breaker.

9) Rotate potentiometer shaft until the measured DC Voltage is precisely 50% of total aircraft voltage V_{GP} (approximately 14 volts).

10) Pull FLIGHT CONTROLS INDICATION (A14) circuit breaker, fit safety clip and disconnect voltmeter from potentiometer. Tighten hardware on lever at potentiometer shaft.

Note Do not rotate potentiometer shaft during this process.

11) Reconnect the potentiometer electrical connection.

12) Remove safety clip and reset FLIGHT CONTROLS INDICATION (A14) circuit breaker.

13) Do a function test. (Refer to Para 3.B).

Note For Flight Data Recorder input checks, refer to 31-31-00, Flight Data Recorder (FDR) – General Data.

For Data Acquisition Unit input checks, refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

B. Function Test Flap Angle Position Potentiometer

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Deflect flaps to the full up position and note flap position display indication correlates to 0°.

4) Deflect flaps to the full down position (Reference inboard fore flap 40°) and note the flap position display indication correlates to 37°.

5) If the flap display indication does not match the flap position, repeat adjustment steps in Para 3.A.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Gust Lock – General Data

1. General

Gust locks are provided to secure the main flight control surfaces (ailerons, rudder and elevator) while the aircraft is parked. The rudder system is provided with a built-in system controlled by an operating lever in the flight compartment floor in front of the pilot's seat. Both aileron and elevator gust locks are removable struts (part of flyaway kit) which engage fittings on the instrument panel structure, control column and floor. On aircraft with Mod 6/1676 incorporated, the elevator gust lock strut locks the control column in a forward position. The rudder gust lock must be engaged and the handle held up until the vertical strut is installed. A pivoted flag, Mod 6/1726, provides visual warning that the gust lock is installed. Refer to Figure 1 for details.

A. Rudder Gust Lock

The rudder gust lock is a permanently installed mechanical linkage beneath the flight compartment floor which secures the rudder in the neutral position. The gust lock linkage consists of a hinged lever, connecting rods, a bellcrank and a spring-loaded lever. The hinged lever is flush with the flight compartment floor and when lifted to select rudder gust lock, operates the connecting rods to engage the spring-loaded lever in a slot in the rudder control quadrant. A spring connected to the spring-loaded lever, ensures that the lock cannot be engaged unless selected.

2. Inspection/Check

A. Check Rudder Gust Lock Operation

1) Ensure rudder control system is correctly rigged (refer to 27-20-00, Rudder Control System – General Data).

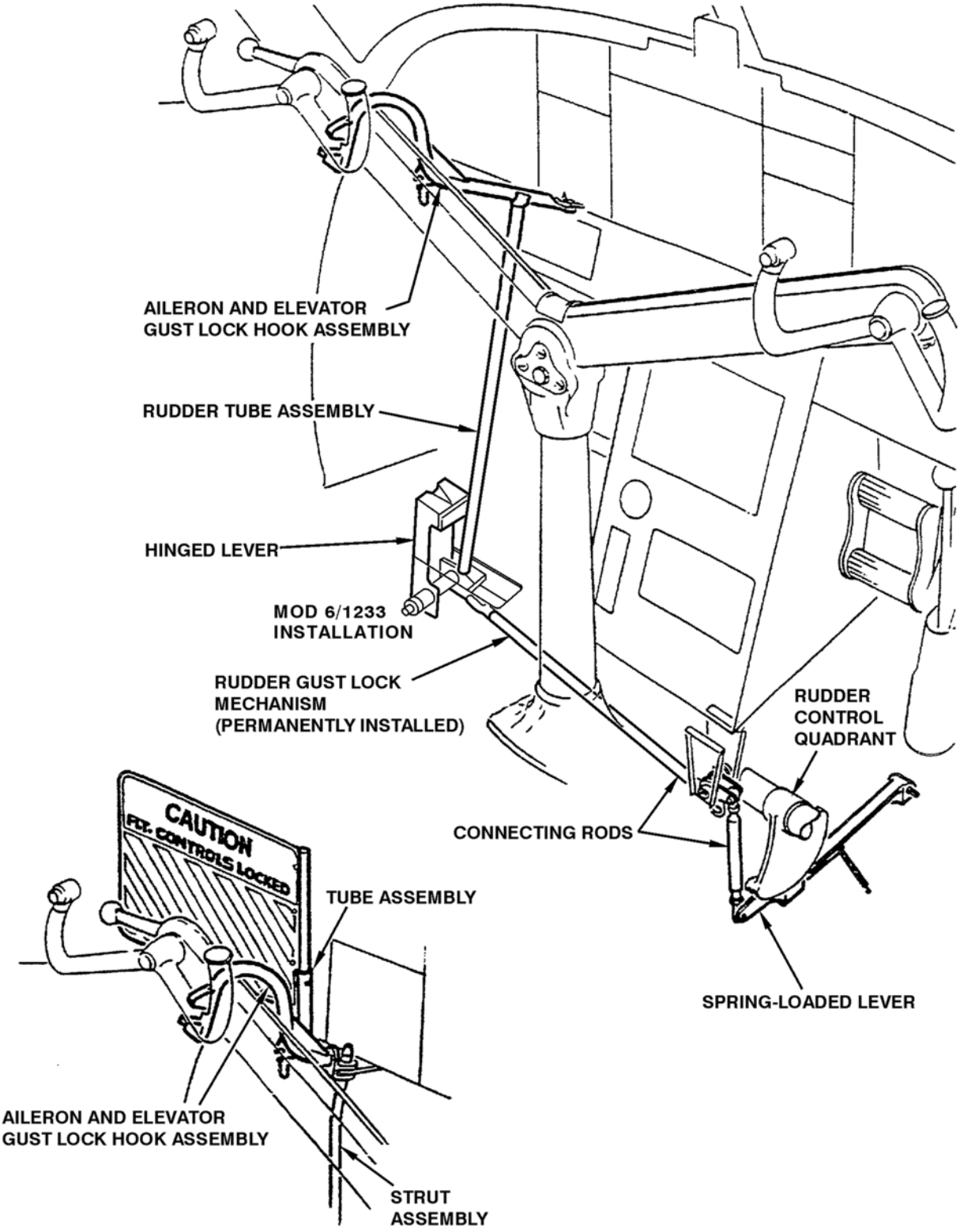
2) Set rudder, aileron and elevator controls to their neutral positions, engage rudder gust lock and install aileron and elevator gust locks.

3) Ensure that the rudder gust lock spring-loaded lever fully engages rudder quadrant.

4) Remove aileron and elevator gust lock struts and lower rudder gust lock hinged lever. Check lever is flush with floor and that spring-loaded lever completely clears rudder control quadrant.

5) On aircraft incorporating Mod 6/1755, check that clearance between plunger in spring-loaded lever and quadrant is 0.120 inch minimum. Adjust connecting rod as necessary and ensure there is no free play between plunger and quadrant when plunger is engaged in gust lock on position.

6) Operate rudder and reengage rudder gust lock several times, checking each time for full and free range of rudder movement after disengaging gust lock.



Gust Lock Installation
Figure 1

CHAPTER 28

FUEL

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CHAPTER 28 – FUEL

LIST OF EFFECTIVE PAGES

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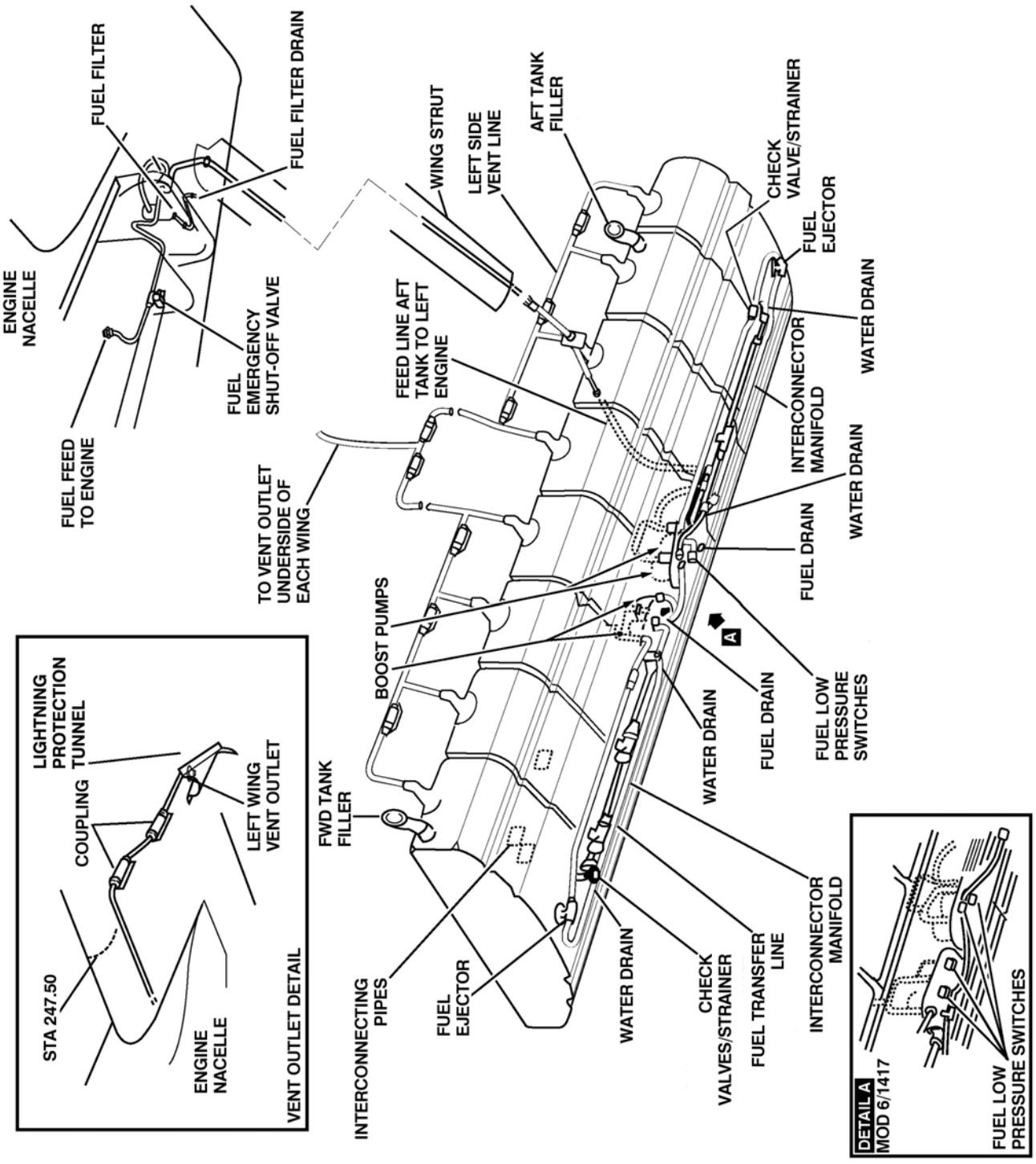
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Fuel (Mod 6/2006) – General Data

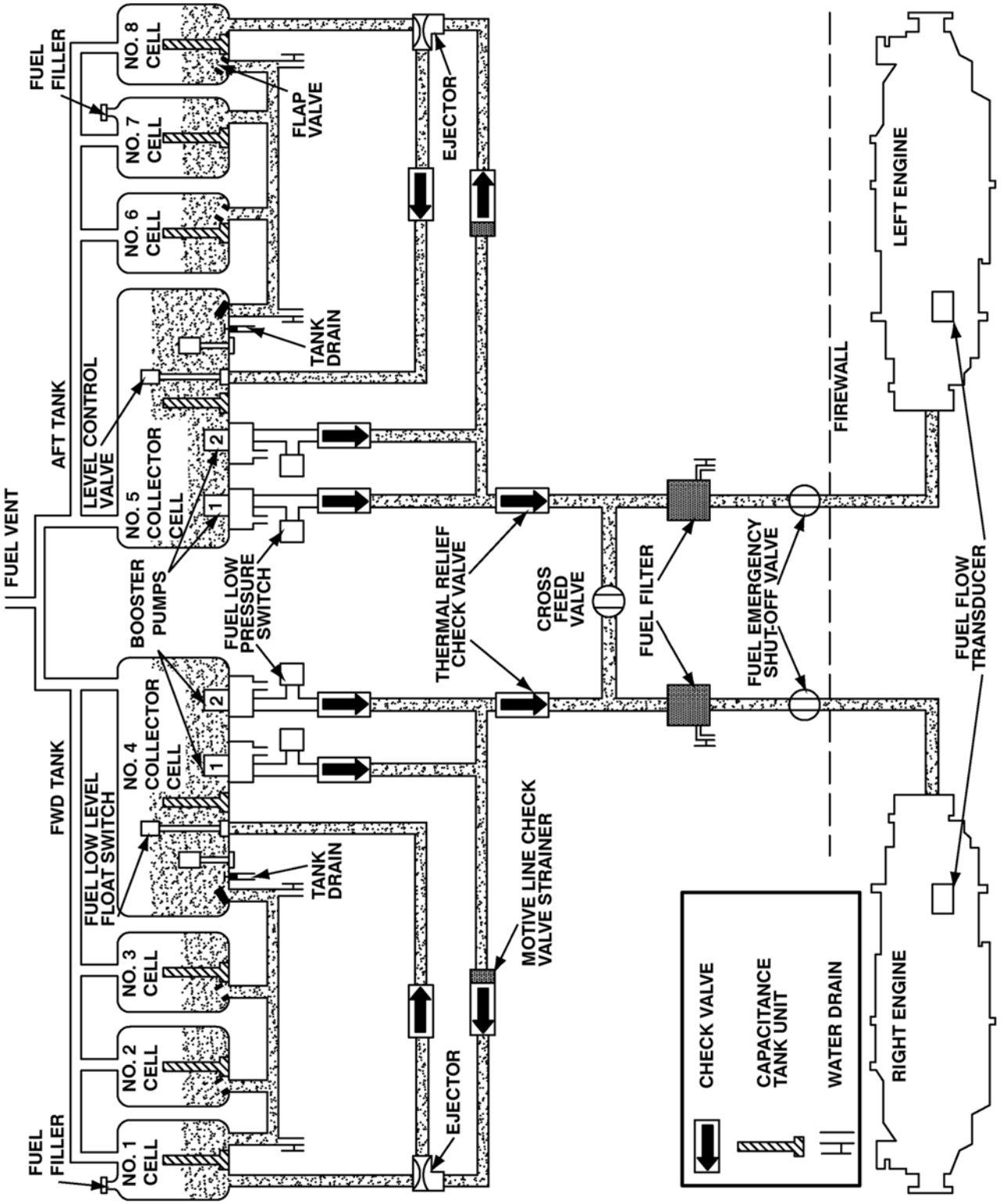
1. General

Refer to Figure 1 and Figure 2 for details.

Fuel is contained in two fuselage fuel tanks, one forward and one aft. The tanks can be gravity refueled through two filler points, one for each tank, located on the left side of the fuselage near the wing strut. The forward tank supplies the right engine and the aft tank the left engine, with the main feed lines to both engines passing through No. 6 cell (aft tank). A crossfeed valve provides a fuel supply to both engines from one tank. Each fuel tank consists of four interconnected fuel cells. Two fuel booster pumps, installed in the collector cell of each tank, deliver fuel under pressure to the engine; this pressure also operates a fuel ejector which transfers fuel from the end cell into the collector cell. From No. 6 cell the main feed line to each engine is routed up the rear side of the wing strut, along the wing and into the appropriate nacelle. Mounted in each nacelle behind the firewall is a fuel filter and a fuel emergency shut-off valve. To permit checks for water accumulation in the fuel system, two water drain valves are provided in each of the forward and aft fuselage tank interconnector manifolds and in each fuel filter. In addition, a fuselage tank fuel drain valve is provided in each sump plate for No. 4 and No. 5 collector cells. A fuel quantity indicating system of the capacitance type is provided for each fuel tank. Fuel low pressure and low level are indicated through Apex on the fuel system window; the upper right 1/6th of the lower Multi-Function Display (MFD). Refer to the Wiring Diagram Manual PSM 1-64-2W for electrical schematic.



Fuel System Arrangement
Figure 1



Fuel System Schematic
Figure 2

2. Adjustment/Test

Note Aircraft must be reasonably level, and in the open air, during all fuel system checks.

A. Electrical Check (No Fuel)

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Ensure that fuel tanks are drained of all usable fuel and, with all relevant circuit breakers engaged, set switches in modes 1 to 5 given in Table 1 — Fuel System Electrical Check – No Fuel. Check that all conditions are produced as shown.

B. Transfer and Amber Block Message Check

1) On the fuel control panel, verify FWD BOOST PUMP 1 switch is in the OFF position and FWD BOOST PUMP 2 switch in the NORMAL position.

2) On the fuel control panel, verify AFT BOOST PUMP 1 switch is in the OFF position and AFT BOOST PUMP 2 switch is in the NORMAL position.

3) Place 30 Imperial (36 U.S.) gallons of fuel in forward fuselage tank and allow fuel level to stabilize for 5 minutes.

4) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.), engage relevant circuit breakers and observe that the forward boost pump amber blocks Pmp1 and Pmp2 are displayed and the Fwd Fuel Qty Low caution message is displayed.

Note Fuel in collector cell is below operating level of low level float switch.

5) On fuel control panel, set FUEL SELECTOR rotary switch to NORMAL and select forward fuselage tank BOOST PUMP 1 switch to the ON position.

6) Check that both forward boost pump amber block Pmp1 and Pmp2 messages are no longer displayed and Fwd Fuel Qty Low caution message is no longer displayed, within 3 minutes.

Note Fuel transfer takes place, raising fuel level in collector cell to operate low level float switch.

7) Switch off booster pump and verify Fwd Fuel Qty Low caution message is no longer displayed.

8) Repeat Para 2.B.3) through Para 2.B.7), substituting aft fuselage tank for forward.

C. Pressure Check

1) Ensure that each fuselage tank contains a minimum of 30 Imperial (36 U.S.) gallons of fuel.

2) Apply DC Power (refer to 12-00-00, Servicing – General Data, Para 2.) and engage relevant circuit breakers.

3) Position fuel levers in overhead console to OFF and FUEL SELECTOR rotary switch on fuel control panel to NORMAL. Verify that the fuel pump amber blocks Pmp1 and Pmp2 for the forward and aft fuel tank are displayed.

4) On fuel control panel, select FWD and AFT BOOST PUMP 1 switches to ON. Verify that the fuel pump amber blocks Pmp1 and Pmp2 for the forward and aft tank on the fuel pane are no longer displayed.

5) Inspect complete fuselage fuel system for leaks. Repair leaks before proceeding with pressure check.

6) Select and hold booster pump switches to TEST. Check that both No. 1 boost pump Pmp1 amber block messages are displayed and both No. 2 boost pump Pmp2 amber block messages are not displayed.

7) Release booster pump switches to OFF and check that all four fuel pump Pmp1 and Pmp2 amber block messages are displayed.

D. Function Check

1) Disconnect each engine fuel supply flexible hose, routed between emergency fuel shut-off valve on firewall and engine oil-to-fuel heater at elbow on heater. Place disconnected end of hose in a suitable container to receive fuel.

2) Check that forward fuselage tank contains approximately 30 Imperial (36 U.S.) gallons of fuel.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.), engage relevant circuit breakers and, using Table 2 — Fuel System Function Check – Fuel In Forward Tank, check fuel system in modes 1 to 5.

4) With switches in mode 2 and fuel flow through engine supply flexible hose stabilized, measure and record time taken for one Imperial (1.2 U.S.) gallon of fuel to be pumped from forward fuselage tank. Maximum time allowed is 30 seconds.

5) With switches in mode 5 and fuel flow through engine supply flexible hose stabilized, measure and record time taken for one Imperial (1.2 U.S.) gallon of fuel to be pumped from forward fuselage tank. Maximum time allowed is 30 seconds.

6) Drain forward fuselage tank and check that aft fuselage tank contains approximately 30 Imperial (36 U.S.) gallons of fuel.

7) Repeat Para 2.D.3) through Para 2.D.5) for aft fuselage tank, substituting AFT for FWD and Table 3 — Fuel System Function Check – Fuel in Aft Tank for Table 2 — Fuel System Function Check – Fuel In Forward Tank.

8) Ensure that all boost pump switches are off and remove external power.

9) Reconnect engine fuel supply flexible hoses to engine oil-to-fuel heaters.

CAUTION **DO NOT ALLOW PUMPS TO RUN MORE THAN 30 SECONDS BEFORE RETURNING THE SWITCHES TO THE OFF POSITION.**

Table 1 Fuel System Electrical Check – No Fuel

SWITCHES	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fuel Selector	NORMAL	BOTH ON FWD	BOTH ON AFT	NORMAL	NORMAL
Fuel Emergency Shut-off – Left	NORMAL	FUEL OFF	FUEL OFF	FUEL OFF	FUEL OFF
Fuel Emergency Shut-off – Right	NORMAL	FUEL OFF	FUEL OFF	FUEL OFF	FUEL OFF
Fwd Boost Pump 1	OFF	ON	ON	TEST	OFF
Aft Boost Pump 1	OFF	ON	ON	TEST	OFF
Left and Right Boost Pump 2	NORMAL	NORMAL	NORMAL	NORMAL	EMERGENCY
PRODUCED CONDITION	↓	↓	↓	↓	↓

Table 1 Fuel System Electrical Check – No Fuel (continued)

SWITCHES	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
No. 1 Boost Pump Fwd	OFF	ON	OFF	OFF	OFF
No. 2 Boost Pump Fwd	OFF	ON	OFF	ON	ON
No. 1 Boost Pump Aft	OFF	OFF	ON	OFF	OFF
No. 2 Boost Pump Aft	OFF	OFF	ON	ON	ON
Crossfeed Valve	CLOSED	OPEN	OPEN	CLOSED	CLOSED
Shut-off Valve – Left	OPEN	CLOSED	CLOSED	CLOSED	CLOSED
Shut-off Valve – Right	OPEN	CLOSED	CLOSED	CLOSED	CLOSED
FUEL PANE AMBER BLOCK MESSAGE STATUS					
Fwd Pmp1	ON	ON	ON	ON	ON
Fwd Pmp2	ON	ON	ON	ON	ON
Aft Pmp1	ON	ON	ON	ON	ON
Aft Pmp2	ON	ON	ON	ON	ON
CAUTION MESSAGE STATUS					
Boost Pump 1 Fwd Press	ON	ON	OFF	ON	ON
Boost Pump 2 Fwd Press	ON	ON	OFF	ON	ON
Boost Pump 1 Aft Press	ON	OFF	ON	ON	ON
Boost Pump 2 Aft Press	ON	OFF	ON	ON	ON
WARNING MESSAGE STATUS					
Fuel Low Both Tanks	ON	ON	ON	ON	ON

CAUTION DO NOT ALLOW PUMPS IN EMPTY TANKS TO RUN MORE THAN 30 SECONDS.

Table 2 Fuel System Function Check – Fuel In Forward Tank

	FORWARD TANK				
SWITCHES	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fuel Selector	BOTH ON FWD	NORMAL	BOTH ON FWD	NORMAL	NORMAL
Fuel Emergency Shut-off – Left	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL

Table 2 Fuel System Function Check – Fuel In Forward Tank (continued)

SWITCHES	FORWARD TANK				
	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fuel Emergency Shut-off – Right	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL
Fwd Boost Pump 1	OFF	ON	ON	TEST	OFF
Aft Boost Pump 1	ON	OFF	OFF	OFF	OFF
Left and Right Boost Pump 2	NORMAL	NORMAL	NORMAL	NORMAL	BOTH TO EMERGENCY
PRODUCED CONDITION	↓	↓	↓	↓	↓
Full Bore Flow At Left Engine	NO FLOW	NO FLOW	YES	NO FLOW	NO FLOW
Full Bore Flow At Right Engine	NO FLOW	YES	YES	NO FLOW	YES
Fuel Quantity Indication – Fwd	NO CHANGE	CHANGE	CHANGE	NO CHANGE	CHANGE
Fuel Quantity Indication – Aft	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE
FUEL PANE AMBER BLOCK MESSAGE STATUS					
Fwd Pmp1	OFF	OFF	OFF	ON	ON
Fwd Pmp2	OFF	OFF	OFF	OFF	OFF
Aft Pmp1	ON	ON	ON	ON	ON
Aft Pmp2	ON	ON	ON	ON	ON
CAUTION MESSAGE STATUS					
Boost Pump 1 Fwd Press	OFF	OFF	OFF	ON	ON
Boost Pump 2 Fwd Press	OFF	OFF	OFF	OFF	OFF
Boost Pump 1 Aft Press	OFF	ON	OFF	ON	ON
Boost Pump 2 Aft Press	OFF	ON	OFF	ON	ON
Fwd Fuel Qty Low	OFF	OFF	OFF	OFF	OFF
Aft Fuel Qty Low	ON	ON	ON	ON	ON

CAUTION DO NOT ALLOW PUMPS IN EMPTY TANKS TO RUN MORE THAN 30 SECONDS.

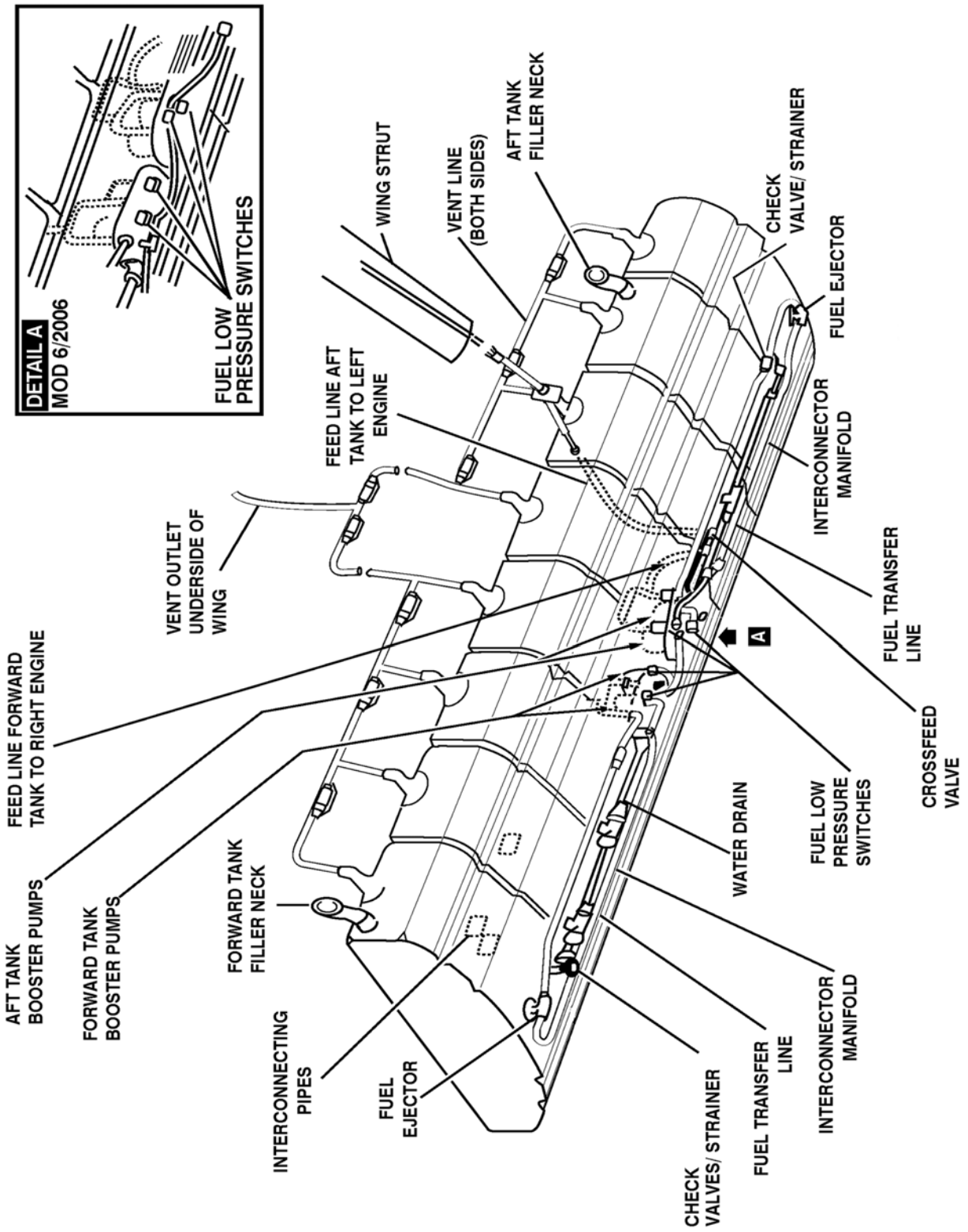
Table 3 Fuel System Function Check – Fuel in Aft Tank

SWITCHES	AFT TANK				
	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fuel Selector	BOTH ON AFT	NORMAL	BOTH ON AFT	NORMAL	NORMAL
Fuel Emergency Shut-off – Left	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL
Fuel Emergency Shut-off – Right	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL
Fwd Boost Pump 1	ON	OFF	OFF	OFF	OFF
Aft Boost Pump 1	OFF	ON	ON	TEST	OFF
Left and Right Boost Pump 2	NORMAL	NORMAL	NORMAL	NORMAL	BOTH ON EMERGENCY
PRODUCED CONDITION	↓	↓	↓	↓	↓
Full Bore Flow At Left Engine	NO FLOW	YES	YES	NO FLOW	YES
Full Bore Flow At Right Engine	NO FLOW	NO FLOW	YES	NO FLOW	NO FLOW
Fuel Quantity Indication – Fwd	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE
Fuel Quantity Indication – Aft	NO CHANGE	CHANGE	CHANGE	NO CHANGE	CHANGE
FUEL PANE AMBER BLOCK MESSAGE STATUS					
Fwd Pmp1	ON	ON	ON	ON	ON
Fwd Pmp2	ON	ON	ON	ON	ON
Aft Pmp1	OFF	OFF	OFF	ON	ON
Aft Pmp2	OFF	OFF	OFF	OFF	OFF
CAUTION MESSAGE STATUS					
Boost Pump 1 Fwd Press	OFF	ON	OFF	ON	ON
Boost Pump 2 Fwd Press	OFF	ON	OFF	ON	ON
Boost Pump 1 Aft Press	OFF	OFF	OFF	ON	ON
Boost Pump 2 Aft Press	OFF	OFF	OFF	OFF	OFF
Fwd Fuel Qty Low	ON	ON	ON	ON	ON
Aft Fuel Qty Low	OFF	OFF	OFF	OFF	OFF

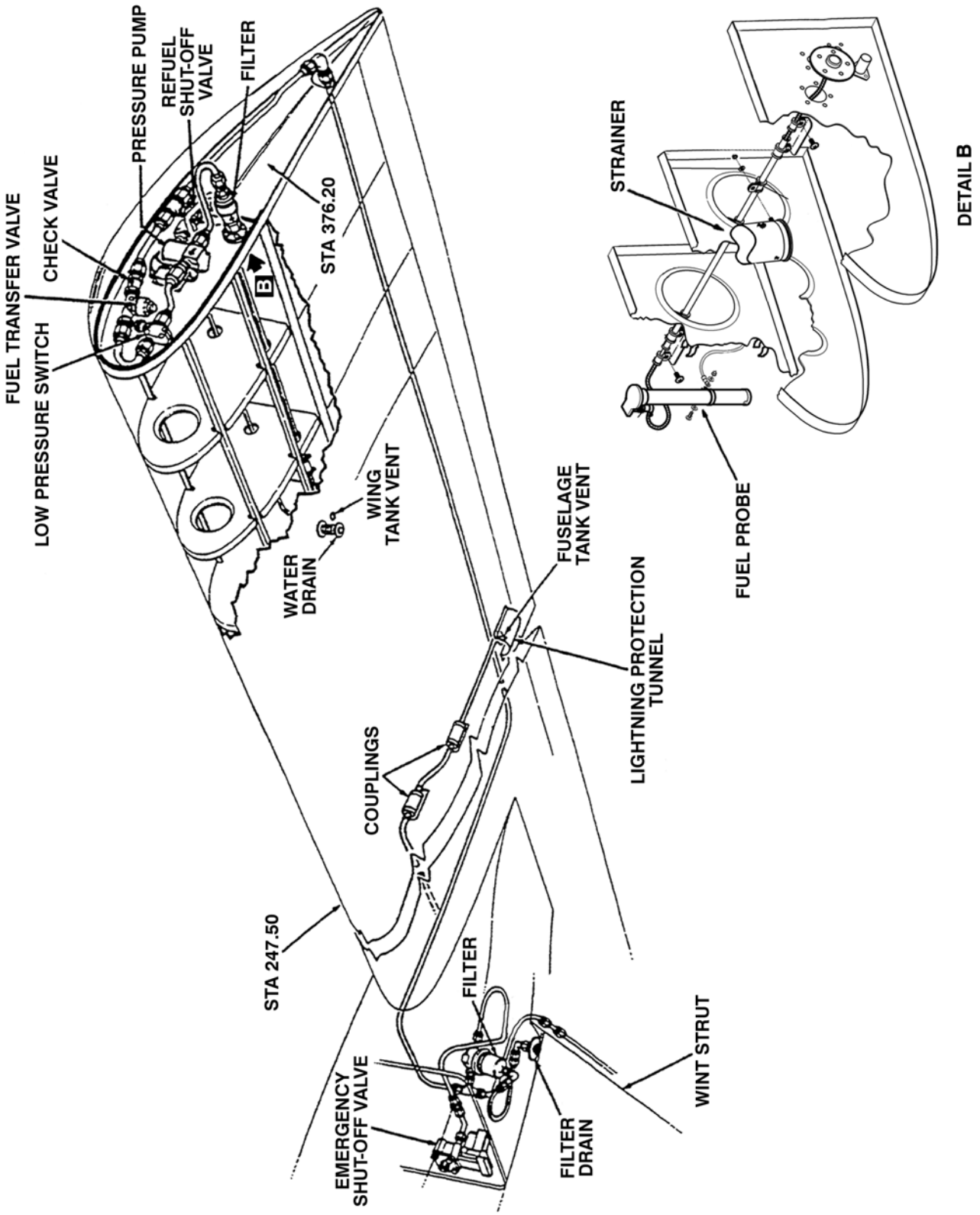
Fuel (With Mod S.O.O. 6247) – General Data

1. General

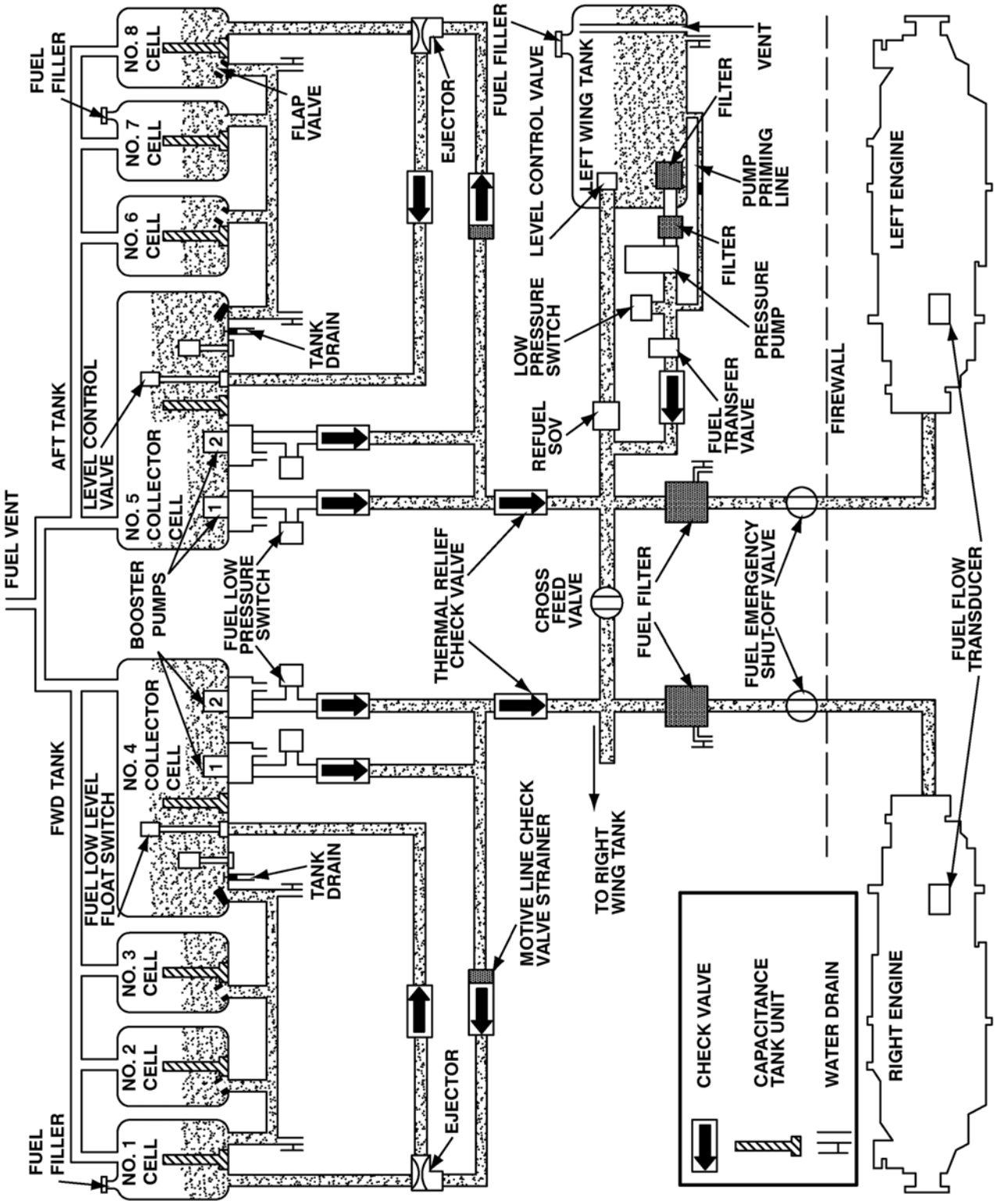
Fuel is contained in four fuel tanks, two in the fuselage and two (Mod S.O.O. 6247 and Mod 6/2046) in the wings. The forward fuselage tank and the right wing tank supply fuel to the right engine, the aft fuselage tank and left wing tank supply the left engine. A crossfeed valve when operated provides a fuel supply to both engines from one fuselage tank and associated wing tank. The fuselage tanks, each consisting of four interconnected fuel cells, deliver fuel under booster pump pressure to a tee union on each nacelle mounted fuel filter. The wing tanks, if installed, form an integral part of each wing, and deliver fuel under pump pressure to the same tee unions as the fuselage tanks. From the tee union on each fuel filter, fuel is routed through a fuel emergency shut-off valve and to the engine. A fuel flow transducer is installed in the fuel line between the FCU and the fuel flow divider. To permit checks for water accumulation in the fuel system, water drain valves are provided in each wing tank, in the forward and aft fuselage tank interconnector manifolds, and in each fuel filter. In addition, a fuselage tank fuel drain valve is provided in each sump plate for No. 4 and No. 5 collector cells. A fuel quantity indicating system and a fuel low pressure indicating system is provided for each tank. A fuel low level indicating system is provided for each fuselage fuel tank. Refer to Figure 1 and Figure 2 for details.



Fuel System Arrangement
Figure 1 (Sheet 1 of 2)



Fuel System Arrangement
Figure 1 (Sheet 2 of 2)



Fuel System Schematic
Figure 2

2. Adjustment/Test

Note Aircraft must be reasonably level, and in the open air, during all fuel system checks. Wing tanks and components are checked separately to permit testing of fuselage tank system if wing tanks are not to be used.

A. Electrical Check (No Fuel)

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Ensure that fuel tanks are drained of all usable fuel and, with all relevant circuit breakers engaged, set switches in modes 1 to 5 given in Table 1 — Fuel System Electrical Check – No Fuel. Check that all conditions are produced as shown.

B. Transfer and Amber Block Message Check

1) On the fuel control panel, verify FWD BOOST PUMP 1 switch is in the OFF position and FWD BOOST PUMP 2 switch in the NORMAL position.

2) On the fuel control panel, verify AFT BOOST PUMP 1 switch is in the OFF position and AFT BOOST PUMP 2 switch is in the NORMAL position.

3) Place 30 Imperial (36 U.S.) gallons of fuel in forward fuselage tank and allow fuel level to stabilize for 5 minutes.

4) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.), engage relevant circuit breakers and observe that the forward boost pump amber blocks Pmp1 and Pmp2 are displayed and the Fwd Fuel Qty Low caution message is displayed.

Note Fuel in collector cell is below operating level of low level float switch.

5) On fuel control panel, set FUEL SELECTOR rotary switch to NORMAL and select forward fuselage tank BOOST PUMP 1 switch to the ON position.

6) Check that both forward boost pump amber block Pmp1 and Pmp2 messages are no longer displayed and Fwd Fuel Qty Low caution message is no longer displayed.

Note Fuel transfer takes place, raising fuel level in collector cell to operate low level float switch.

7) Switch off booster pump and verify Fwd Fuel Qty Low caution message is no longer displayed.

8) Repeat Para 2.B.3) through Para 2.B.7), substituting aft fuselage tank for forward.

C. Pressure Check

1) Ensure that each fuselage tank contains a minimum of 30 Imperial (36 U.S.) gallons of fuel.

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.) and engage relevant circuit breakers.

3) Position fuel levers in overhead console to OFF and FUEL SELECTOR rotary switch on fuel control panel to NORMAL. Verify that the fuel pump amber blocks Pmp1 and Pmp2 for the forward and aft fuel tank are displayed.

4) On fuel control panel, select FWD and AFT BOOST PUMP 1 switches to ON. Verify that the fuel pump amber blocks Pmp1 and Pmp2 for the forward and aft tank on the fuel pane are no longer displayed.

5) Inspect complete fuselage fuel system for leaks. Repair leaks before proceeding with pressure check.

6) Select and hold booster pump switches to TEST. Check that both No.1 boost pump Pmp1 amber block messages are displayed and both No. 2 boost pump Pmp2 amber block messages are not displayed.

7) Release booster pump switches to OFF and check that all four fuel pump Pmp1 and Pmp2 amber block messages are displayed.

D. Function Check

1) Disconnect each engine fuel supply flexible hose, routed between emergency fuel shut-off valve on firewall and engine oil-to-fuel heater at elbow on heater. Place disconnected end of hose in a suitable container to receive fuel.

2) Check that forward fuselage tank contains approximately 30 Imperial (36 U.S.) gallons of fuel.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.), engage relevant circuit breakers and, using Table 2 — Fuel System Function Check – Fuel In Forward Tank, check fuel system in modes 1 to 5.

4) With switches in mode 2 and fuel flow through engine supply flexible hose stabilized, measure and record time taken for one Imperial (1.2 U.S.) gallon of fuel to be pumped from forward fuselage tank. Maximum time allowed is 30 seconds.

5) With switches in mode 5 and fuel flow through engine supply flexible hose stabilized, measure and record time taken for one Imperial (1.2 U.S.) gallon of fuel to be pumped from forward fuselage tank. Maximum time allowed is 30 seconds.

6) Drain forward fuselage tank and check that aft fuselage tank contains approximately 30 Imperial (36 U.S.) gallons of fuel.

7) Repeat Para 2.D.3) through Para 2.D.5) for aft fuselage tank, substituting AFT for FWD and Table 3 — Fuel System Function Check – Fuel in Aft Tank for Table 2 — Fuel System Function Check – Fuel In Forward Tank.

8) Ensure that boost pump switches are off and remove external power.

9) Reconnect engine fuel supply flexible hoses to engine oil-to-fuel heaters.

E. Operational Test (Wing Tanks)

CAUTION AVOID DRY RUNNING OF WING TANK PUMPS.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.) and engage relevant circuit breakers.

2) Fill forward and aft fuselage tanks and ensure that wing tanks are empty.

3) Select the ENGINE switch position on the left wing fuel tank switch on the fuel tank control panel and check that the associated caution message Left Wing Fuel Pump is displayed.

4) Verify the left fuel pump amber block Pump message is displayed.

5) Select OFF at left wing fuel tank switch.

6) Repeat Para 2.E.3) and Para 2.E.5) for right wing fuel tank.

7) Select REFUEL on both wing fuel tank switches and NORMAL at FUEL SELECTOR rotary switch on fuel control panel.

8) Verify that the wing tank advisory CAS messages Left Wing Refuel On and Right Wing Refuel On are displayed.

9) Position fuel levers in overhead console to OFF and select fuselage tank BOOST PUMP 1 switches to ON.

10) Observe wing tank fuel quantity indications and check that tanks start to fill and then stop automatically at FULL. Check that no further spillage occurs at vent holes in lower wing skin.

11) Select OFF on wing fuel tank switches and switch off fuselage tank booster pumps.

12) Select the ENGINE switch position on the left and right wing fuel tank control panel and check that the associated caution messages Left Wing Fuel Pmp and Right Wing Fuel Pmp are not displayed. Verify the left and right wing tank fuel pump blocks Pump are now displayed green. Return both switches to OFF and remove external power supply.

F. Fuel Flow Test (Wing Tanks)

1) Disconnect left wing fuel tank pressure supply line from tee at nacelle-mounted fuel filter. Cap open port in tee.

2) Connect a length of hose to disconnected end of pressure supply line and place end of hose in a suitable container to receive fuel.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.), engage relevant circuit breaker, and select ENGINE at appropriate switch on wing fuel tanks control panel.

4) With fuel flow through hose stabilized, measure and record time taken for 1 Imperial (1.2 U.S.) gallon of fuel to be pumped from left wing tank. Maximum time allowed is 85 seconds.

5) Select left wing fuel tank switch OFF.

6) Remove hose from left wing fuel tank pressure supply line and blanking cap from tee on fuel filter. Reconnect fuel line to tee.

7) Repeat Para 2.F.1) through Para 2.F.6) for right wing fuel tank.

8) Ensure all switches are off and remove external power supply.

CAUTION **DO NOT ALLOW PUMPS TO RUN MORE THAN 30 SECONDS BEFORE RETURNING THE SWITCHES TO THE OFF POSITION.**

Table 1 Fuel System Electrical Check – No Fuel

SWITCHES	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fuel Selector	NORMAL	BOTH ON FWD	BOTH ON AFT	NORMAL	NORMAL
Fuel Emergency Shut-off – Left	NORMAL	FUEL OFF	FUEL OFF	FUEL OFF	FUEL OFF
Fuel Emergency Shut-off – Right	NORMAL	FUEL OFF	FUEL OFF	FUEL OFF	FUEL OFF
Fwd Boost Pump 1	OFF	ON	ON	TEST	OFF
Aft Boost Pump 1	OFF	ON	ON	TEST	OFF
Left and Right Boost Pump 2	NORMAL	NORMAL	NORMAL	NORMAL	EMERGENCY

Table 1 Fuel System Electrical Check – No Fuel (continued)

SWITCHES	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
PRODUCED CONDITION	↓	↓	↓	↓	↓
No. 1 Boost Pump Fwd	OFF	ON	OFF	OFF	OFF
No. 2 Boost Pump Fwd	OFF	ON	OFF	ON	ON
No. 1 Boost Pump Aft	OFF	OFF	ON	OFF	OFF
No. 2 Boost Pump Aft	OFF	OFF	ON	ON	ON
Crossfeed Valve	CLOSED	OPEN	OPEN	CLOSED	CLOSED
Shut-off Valve – Left	OPEN	CLOSED	CLOSED	CLOSED	CLOSED
Shut-off Valve – Right	OPEN	CLOSED	CLOSED	CLOSED	CLOSED
FUEL PANE AMBER BLOCK MESSAGE STATUS					
Fwd Pmp1	ON	ON	ON	ON	ON
Fwd Pmp2	ON	ON	ON	ON	ON
Aft Pmp1	ON	ON	ON	ON	ON
Aft Pmp2	ON	ON	ON	ON	ON
CAUTION MESSAGE STATUS					
Boost Pump 1 Fwd Press	ON	ON	OFF	ON	ON
Boost Pump 2 Fwd Press	ON	ON	OFF	ON	ON
Boost Pump 1 Aft Press	ON	OFF	ON	ON	ON
Boost Pump 2 Aft Press	ON	OFF	ON	ON	ON
WARNING MESSAGE STATUS					
Fuel Low Both Tanks	ON	ON	ON	ON	ON

CAUTION DO NOT ALLOW PUMPS IN EMPTY TANKS TO RUN MORE THAN 30 SECONDS.

Table 2 Fuel System Function Check – Fuel In Forward Tank

SWITCHES	FORWARD TANK				
	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fuel Selector	BOTH ON FWD	NORMAL	BOTH ON FWD	NORMAL	NORMAL
Fuel Emergency Shut-off – Left	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL
Fuel Emergency Shut-off – Right	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL
Fwd Boost Pump 1	OFF	ON	ON	TEST	OFF
Aft Boost Pump 1	ON	OFF	OFF	OFF	OFF
Left and Right Boost Pump 2	NORMAL	NORMAL	NORMAL	NORMAL	BOTH TO EMERGENCY
PRODUCED CONDITION	↓	↓	↓	↓	↓
Full Bore Flow At Left Engine	NO FLOW	NO FLOW	YES	NO FLOW	NO FLOW
Full Bore Flow At Right Engine	NO FLOW	YES	YES	NO FLOW	YES
Fuel Quantity Indication – Fwd	NO CHANGE	CHANGE	CHANGE	NO CHANGE	CHANGE
Fuel Quantity Indication – Aft	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE
FUEL PANE AMBER BLOCK MESSAGE STATUS					
Fwd Pmp1	OFF	OFF	OFF	ON	ON
Fwd Pmp2	OFF	OFF	OFF	OFF	OFF
Aft Pmp1	ON	ON	ON	ON	ON
Aft Pmp2	ON	ON	ON	ON	ON
CAUTION MESSAGE STATUS					
Boost Pump 1 Fwd Press	OFF	OFF	OFF	ON	ON
Boost Pump 2 Fwd Press	OFF	OFF	OFF	OFF	OFF
Boost Pump 1 Aft Press	OFF	ON	OFF	ON	ON
Boost Pump 2 Aft Press	OFF	ON	OFF	ON	ON
Fwd Fuel Qty Low	OFF	OFF	OFF	OFF	OFF
Aft Fuel Qty Low	ON	ON	ON	ON	ON

CAUTION **DO NOT ALLOW PUMPS IN EMPTY TANKS TO RUN MORE THAN 30 SECONDS.**

Table 3 Fuel System Function Check – Fuel in Aft Tank

	AFT TANK				
SWITCHES	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fuel Selector	BOTH ON AFT	NORMAL	BOTH ON AFT	NORMAL	NORMAL
Fuel Emergency Shut-off – Left	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL
Fuel Emergency Shut-off – Right	FUEL OFF	NORMAL	NORMAL	FUEL OFF	NORMAL
Fwd Boost Pump 1	ON	OFF	OFF	OFF	OFF
Aft Boost Pump 1	OFF	ON	ON	TEST	OFF
Left and Right Boost Pump 2	NORMAL	NORMAL	NORMAL	NORMAL	BOTH ON EMERGENCY
PRODUCED CONDITION	↓	↓	↓	↓	↓
Full Bore Flow At Left Engine	NO FLOW	YES	YES	NO FLOW	YES
Full Bore Flow At Right Engine	NO FLOW	NO FLOW	YES	NO FLOW	NO FLOW
Fuel Quantity Indication – Fwd	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE	NO CHANGE
Fuel Quantity Indication – Aft	NO CHANGE	CHANGE	CHANGE	NO CHANGE	CHANGE
FUEL PANE AMBER BLOCK MESSAGE STATUS					
Fwd Pmp1	ON	ON	ON	ON	ON
Fwd Pmp2	ON	ON	ON	ON	ON
Aft Pmp1	OFF	OFF	OFF	ON	ON
Aft Pmp2	OFF	OFF	OFF	OFF	OFF
CAUTION MESSAGE STATUS					
Boost Pump 1 Fwd Press	OFF	ON	OFF	ON	ON
Boost Pump 2 Fwd Press	OFF	ON	OFF	ON	ON
Boost Pump 1 Aft Press	OFF	OFF	OFF	ON	ON
Boost Pump 2 Aft Press	OFF	OFF	OFF	OFF	OFF

Table 3 Fuel System Function Check – Fuel in Aft Tank (continued)

SWITCHES	AFT TANK				
	MODE 1	MODE 2	MODE 3	MODE 4	MODE 5
Fwd Fuel Qty Low	ON	ON	ON	ON	ON
Aft Fuel Qty Low	OFF	OFF	OFF	OFF	OFF

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Storage – General Data

1. General

Fuel storage is accomplished in two fuel tanks located beneath the cabin floor. The tanks are fitted longitudinally and referred to as forward and aft tanks. Each tank consists of four fuel cells, fitted with external and internal transfer lines to provide fuel transfer within each tank. Refer to Figure 1.

A. Tanks

There are two fuel tanks, each consisting of four fuel cells, installed underneath the cabin floor. An external interconnecting manifold, extending between cells 1 and 4 (forward tank) and cells 5 and 8 (aft tank), transfers the fuel within each tank. Contained in each tank are two fuel booster pumps, a level control valve, a fuel low level float switch, four fuel quantity capacitor units, and a filler neck. Two fuel low pressure switches are mounted externally.

B. Fuel Cells

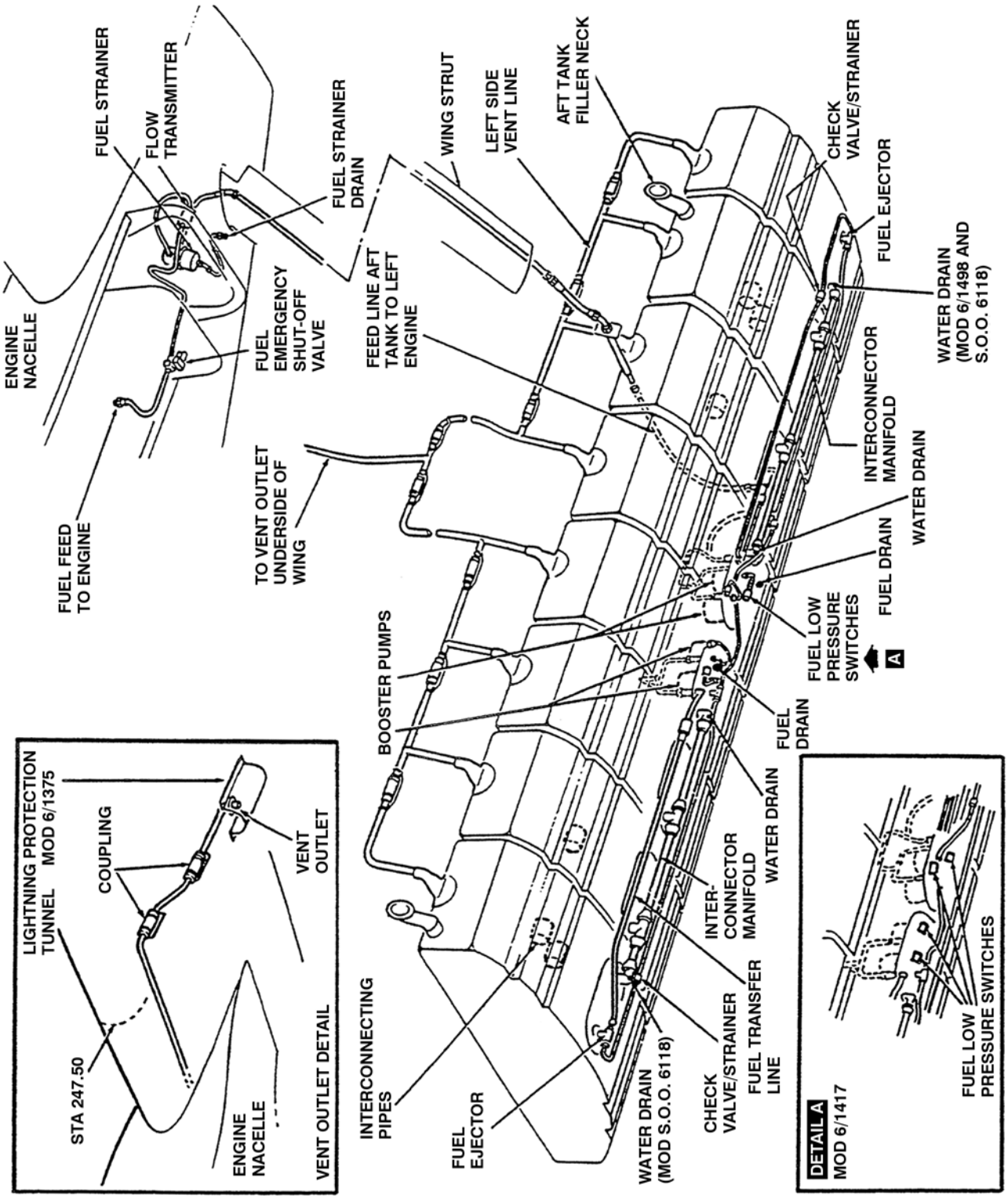
There is a total of eight fuel cells in the fuel system, all of the flexible rubber type. Each cell contains two molded pipes, on each outboard side, which attach to the vent system lines. With the exception of cells 4 and 5, each cell has a molded fitting which accepts the tank internal interconnecting pipes. In addition, cells 1 and 7 are equipped with filler necks, and the main feed lines to both engines pass through cell 6. Each cell is supported by strips of Velcro tape cemented to the top surface, and along the front and rear faces of the fuselage fuel cavities. On installation, the tape strips engage with matching strips on the underside of the cabin floor and on the front and rear faces of the compartment frames. An access opening is provided in the bottom of each cell when the sump plates are removed.

C. Fuel Cell Sump Plates

A sump plate is provided to enclose an access opening in the bottom of each fuel cell. The plate contains a row of sealed anchor nuts equally spaced around the machined edge and has accommodation for various items such as a flap valve, booster pumps, fuel capacitor unit, etc. Only the sump plates in cells 2 and 3, and cells 4 and 5, are interchangeable with each other. Flap valves are fitted to the plates used in cells 2 through 6, and 8. The valves in cells 4 and 5 do not have a hole drilled through the center and are only interchangeable with each other.

D. Venting

Venting of the fuselage fuel tanks employs two separate and similar vent systems. Each vent system provides a positive tank pressure in flight. A series of interconnected lines from the outboard side of each fuel cell joins a single line which extends up the side of the fuselage and along the inside of the wing to a protruding outlet on the under surface, near the inboard hinge bracket of the outboard fore flap; forward of the vent line outlet is a contoured block which acts as an ice deflector during flight to prevent blockage of the vent. A tunnel over the wing fuel vent outlet (Mod 6/1375) to prevent the possibility of vapor ignition due to lightning strikes.



Fuel System Arrangement
Figure 1

Storage (With Mod S.O.O. 6247) – General Data

1. General

Fuel storage is accomplished in four fuel tanks, two located beneath the cabin floor and one in each wing leading edge just inboard of the detachable wing tip. The fuselage tanks are fitted longitudinally and referred to as forward and aft tanks. Each fuselage tank consists of four fuel cells, fitted with external and internal lines to provide transfer within each tank. The wing tanks are fitted one in each wing and referred to as left and right wing tanks. The wing tanks, an integral part of each wing, are connected to the engine feed lines from the fuselage tank system. Refer to Figure 1.

A. Fuselage Tanks

There are two fuselage fuel tanks, each consisting of four fuel cells, installed underneath the cabin floor. An external interconnecting manifold, extending between cells 1 and 4 (forward tank) and cells 5 and 8 (aft tank), transfers the fuel within each tank. Contained in each tank are two fuel booster pumps, a level control valve, a fuel low lever float switch, four fuel quantity capacitor units, and a filler neck. Two fuel low pressure switches are mounted externally.

B. Wing Tanks

There are two integral wing fuel tanks, one in each wing, located in the wing leading edge just inboard of the detachable wing tip. Contained in each tank is a level control valve, a filter, a fuel quantity transmitter, a vent pipe, and a filler cap. Mounted outside each tank on the wing outboard nose rib is a fuel pump, a pressure switch, a fuel transfer valve, a refuel shut-off valve, and a filter.

C. Fuel Cells

There is a total of eight fuel cells in the fuel system, all of the flexible rubber type. Each cell contains two molded pipes, on each outboard side, which attach to the vent system lines. With the exception of cells 4 and 5, each cell has a molded fitting which accepts the tank internal interconnecting pipes. In addition, cells 1 and 7 are equipped with filler necks, and the main feed lines to both engines pass through cell 6. Each cell is supported by strips of Velcro tape cemented to the top surface, and along the front and rear faces. On installation, the tape strips engage with matching strips on the underside of the cabin floor, and on the front and rear faces of the compartment frames. An access opening is provided in the bottom of each cell.

D. Fuel Cell Sump Plates

A sump plate is provided to enclose an access opening in the bottom of each fuel cell. The plate contains a row of sealed anchor nuts equally spaced around the machined edge, and has accommodation for various items such as a flap valve, booster pump, fuel capacitor unit, etc. Only the sump plates in cells 2 and 3, and cells 4 and 5, are interchangeable with each other. Flap valves are fitted to the plates used in cells 2 through 6, and 8. The valves in cells 4 and 5 do not have a hole drilled through the center and are only interchangeable with each other.

E. Fuselage Tank Venting

Venting of the fuselage fuel tanks employs two separate and similar vent systems. Each vent system provides a positive tank pressure in flight. A series of interconnected lines from the outboard side of each fuel cell joins a single line which extends up the side of the fuselage and along the inside of the wing to a protruding outlet on the under surface, near the inboard hinge bracket of the outboard fore flap; forward of the vent line outlet is a contoured block which acts as an ice deflector during flight to prevent blockage of the vent. Aircraft with Mod 6/1375 incorporated, have a tunnel over the wing fuel vent outlet to prevent the possibility of vapor ignition due to lightning strikes.

F. Wing Tank Venting

Each wing fuel tank is vented through a single vent tube installed inside the tank. One end of the vent tube is located near the tank roof at the outboard rear of the tank, and the other end terminates flush with the undersurface of the wing at the inboard rear of the tank.

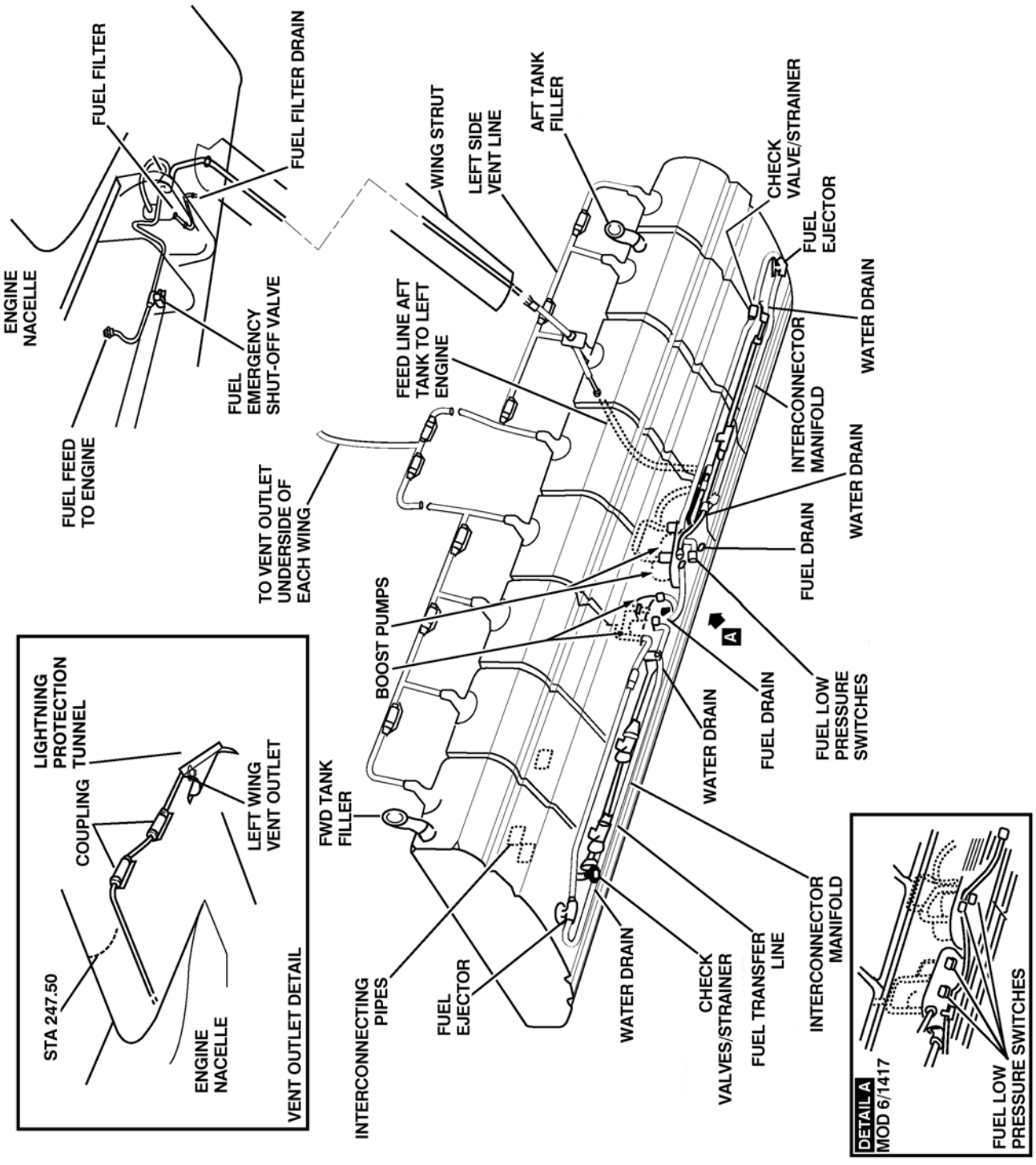
2. Servicing

A. Wing Tank Access Panel – Sealing

1) Check mating surfaces for condition and cleanliness.

2) Check O-ring for condition.

3) Install access panel. Seal gap between access panel edge and wing skin with sealant PR–1776MB (Product Research Co.).



Fuel System Arrangement
Figure 1

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Fuel Cell – Maintenance Practices

1. Removal/Installation

   A. Remove Fuel Cell

      1) Ensure all electrical power is disconnected, crossfeed valve is closed, and wing fuel tank switches (if fitted) are OFF.

      2) Drain fuselage fuel tanks. (Refer to 12-10-10, Fuel System – General Data).

      3) Remove appropriate access panels from underside of fuselage for cell being removed. (Refer to 12-00-00, Servicing – General Data).

      4) Remove relevant sump plate. (Refer to 28-10-21, Fuel Cell Sump Plate – Maintenance Practices).

      5) Working from inside fuel cell, except No. 4 and 5 cells, remove clamp from interconnect pipe molded fitting and pull cell from interconnect pipe.

      6) If either No. 1 or No. 7 fuel cell is being removed, proceed as follows (refer to Figure 1):

         a) Working from inside fuel cell, remove bolts and sealing washers securing cell to fuselage structure around filler neck.

         b) Withdraw retaining ring, gasket, preformed packing, and retaining washer from fuel cell.

      7) If No. 6 fuel cell is being removed, proceed as follows (refer to Figure 1):

         a) Remove main landing gear fairing from each side of fuselage. (Refer to 12-00-00, Servicing – General Data).

         b) Disconnect flexible hose from external end of each main feed line connector.

         c) Remove nut, washer, bolt, and hose clamp securing each main feed line connector to support bracket on fuselage structure.

         d) Working from inside fuel cell, remove bolts and sealing washers securing each main feed line connector and cell to fuselage structure.

         e) Remove both main feed line connectors, with attached flexible hoses, from inside fuel cell.

      8) Remove necessary heating ducts from both sides of cabin to provide access to vent system lines of fuel cell being removed.

      9) Remove hose clamps securing molded vent outlets at each side of cell to left and right vent lines of fuel system. Pull cell vent outlets from fuel system vent lines.

      10) Disengage tape strips along top, front and rear sides of fuel cell.

      11) Ensure that fuel cell is warm enough to be flexible, fold cell carefully, and remove through access opening in lower surface of fuel cell compartment in fuselage structure.

   B. Prepare Fuel Cell Compartment for Installation of Cell

      1) Check that all fuselage structure rivet heads and tails, exposed inside fuel cell compartment, are covered with Scotch foam tape 3M4116 (Minnesota Mining and Manufacturing Co.).

      2) Check that edges of all frame flanges, stringers and skin joints, exposed inside fuel cell compartment, are covered with Scotch foam tape 3M4116.

3) Check that Velcro tape strips are securely bonded to underside of sub-floor structure at all frame faces with adhesive compound EC1099 (Minnesota Mining and Manufacturing Co.).

4) Check that covers at frame stations 218.82, 239.88 and 262.00, in cell compartments 5 through 8, are securely bonded to the fuselage structure with adhesive compound EC1099, or with 3M–5952WF (or 3M–5952) double coated tape.

C. Install Fuel Cell

1) Inspect fuel cell compartment for cleanliness.

2) Observe following fuel cell handling precautions:

a) Do not remove fuel cell from shipping containers until required for installation and then only in immediate vicinity of fuel cell compartment.

b) Carefully examine fuel cell and container for possible damage caused during packing, transportation and unpacking.

c) Immediately after removal from shipping container, place fuel cell on clean, heavy paper.

d) Do not lift, carry or pull fuel cell by hangers or fittings.

e) Check that fuel cell is completely flexible before handling. Temperature should not be lower than 4.5°C (40°F) and preferably at room temperature, approximately 21°C (70°F).

f) Always use original container when moving or shipping fuel cells.

3) Insert fuel cell into appropriate compartment through access opening in lower surface of fuselage structure.

4) Carefully unfold fuel cell and engage Velcro tape strips along top, front and rear sides.

5) Using Petrolatum (Specification VV–P–236) or equivalent, as a lubricant, push molded vent outlet at each side of fuel cell over end of corresponding left and right vent line of fuel system. Secure joints with hose clamps.

6) Reinstall heating ducts, removed for access to vent lines, at each side of cabin.

7) Working from inside fuel cell, except No. 4 and No. 5 cells, push cell interconnect pipe molding over interconnect pipe and secure joint with hose clamp.

8) If No. 6 cell is being installed, proceed as follows (refer to Figure 1):

a) Insert left and right main feed line connectors, with attached flexible hoses, inside fuel cell.

b) Working from inside fuel cell, pass outer end of left main feed line connector through hole in cell and left side of fuselage structure. Repeat operation for right main feed line connector.

c) Align mounting flange on each main feed line connector and secure to structure with bolts and sealing washers. Torque bolts to 35 to 40 pound–inches.

d) Position hose clamp on end of each main feed line connector projecting from fuel cell and secure clamps to support brackets on fuselage structure with bolts, washers and nuts.

e) Connect engine supply flexible hoses to main feed line connectors.

f) Reinstall main landing gear fairings. (Refer to 12-00-00, Servicing – General Data).

9) If either No. 1 or No. 7 fuel cell is being installed, proceed as follows (refer to Figure 1):

a) Position opening in fuel cell over and around filler neck and align bolting holes in cell and fuselage structure.

b) Place retaining washer over filler neck with projecting collar of washer inserted between filler neck and holes in fuel cell and fuselage structure.

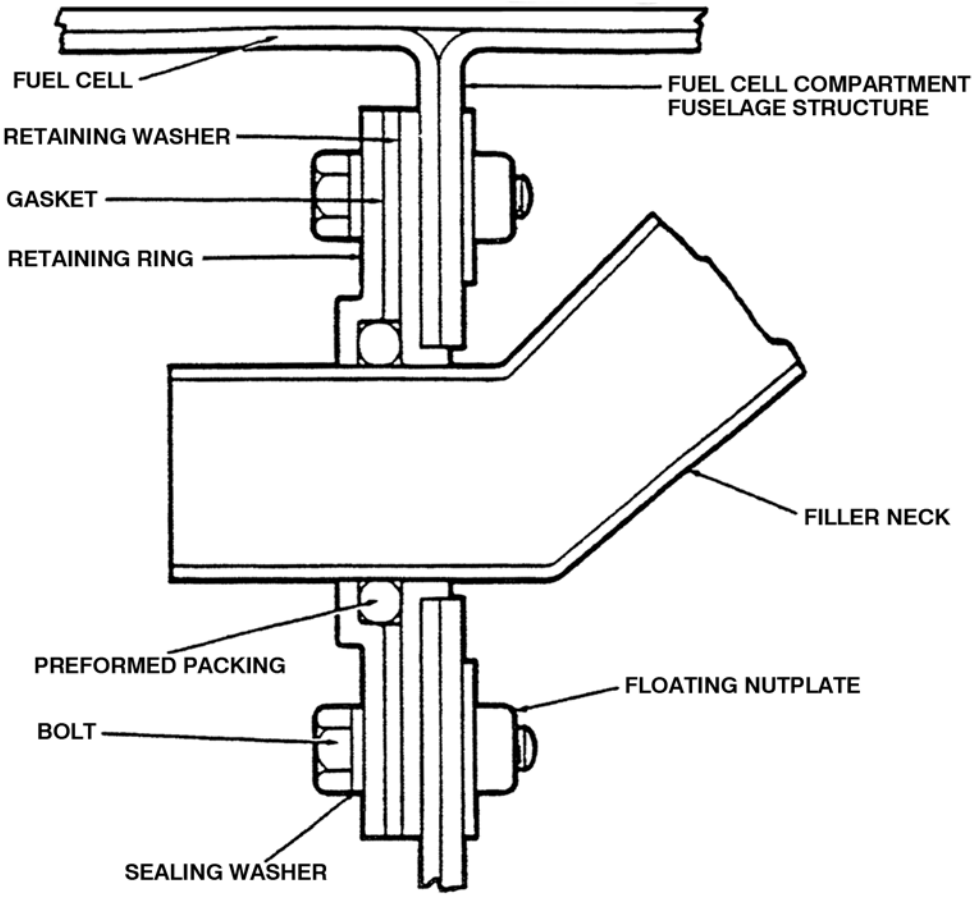
c) Place gasket and preformed packing over filler neck, followed by retaining ring. Ensure that dished center of retaining ring is positioned to enclose preformed packing.

d) Align bolting holes in all parts and secure with bolts and sealing washers. Torque bolts to 35 to 40 pound–inches.

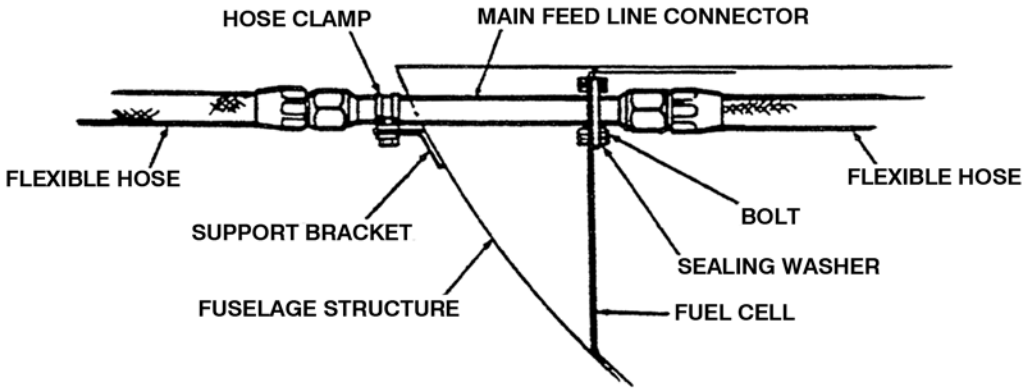
10) Install sump plate. (Refer to 28-10-21, Fuel Cell Sump Plate – Maintenance Practices).

11) Check installed fuel cell for leaks and on cell Nos. 4, 5 and 6, carry out appropriate checks as detailed in 28-00-00, Fuel (Mod 6/2006) – General Data, and 28-00-01, Fuel (With Mod S.O.O. 6247) – General Data.

12) Install access panels on underside of fuselage. (Refer to 12-00-00, Servicing – General Data).



CROSS SECTION THROUGH FILLER NECK
(TYPICAL FOR NO.1 AND NO.7 FUEL CELLS)



CROSS SECTION THROUGH NO.6 FUEL CELL
(TYPICAL BOTH SIDES)

Attachment of Filler Neck and Main Feed Lines in Fuel Cells
Figure 1

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR28-16**

Insert in 28-10-21 before Page 1

FUEL CELL SUMP PLATE

REASON FOR ISSUE:

To clarify fuel cell sump plate installation procedures.

ACTION:

In Chapter 28, Section 28-10-21 ignore existing Page 1 and insert revised Page, as shown on Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

Fuel Cell Sump Plate – Maintenance Practices

1. Removal/Installation

A. Remove Fuel Cell Sump Plate

1) Ensure POWER SOURCE switch is OFF (Pre Mod 6/2314); or BATTERY and EXTERNAL switches are OFF (Post Mod 6/2314), crossfeed valve is closed, and wing fuel tank switches (if fitted) are OFF.

2) Drain fuel from forward tank if No. 1, 2, 3 and/or 4 cell sump plates are to be removed. Drain fuel from aft tanks 5,6,7,8, if cell sump plates are to be removed. Drain both tanks if No. 5 and/or 6 cell sump plates are to be removed. (Refer to 12-10-10, Fuel System – General Data).

3) Remove access panel from underside of fuselage.

4) Disconnect electrical leads from fuel capacitor unit, switches and pumps, where applicable.

5) Disconnect external fuel lines. Remove fuel line from sump plate of No. 4 cell to sump plate of No. 6 cell, if No. 5 cell sump plate is to be removed.

6) Remove bolts securing sump plate, then lift up sump plate, turn and lower from cell. If No. 6 cell sump plate is being removed, remove crossfeed valve (refer to 28-20-21, Crossfeed Valve – Maintenance Practices), lower sump plate sufficiently from cell to disconnect two internal engine feed lines and then remove.

7) Cap open lines.

B. Install Fuel Cell Sump Plate

CAUTION **WHEN SERVICING EQUIPMENT ON SUMP PLATES REMOVED FROM FUEL CELLS NO. 4 AND 5, ENSURE CONNECTIONS TO SUMP PLATE OF THE LEVEL CONTROL VALVE AND OF FLEXIBLE HOSE FROM BOOST PUMPS ARE CORRECTLY LOCATED AND NOT INTERCHANGED. THE FLEXIBLE HOSE FROM BOOST PUMPS MUST MATE WITH THE MACHINED TEE FITTING MOUNTED ON SUMP PLATE LOWER SURFACE TO ENSURE UNINTERRUPTED FUEL SUPPLY TO ENGINES.**

1) If No. 6 cell sump plate is to be installed, connect two internal engine feed lines, and install crossfeed valve. (Refer to 28-20-21, Crossfeed Valve – Maintenance Practices).

2) Insert sump plate through cell aperture, position correctly and secure bolts. Determine run down torque of each bolt and record value at each location. Tighten each bolt to its recorded run down torque plus 45–50 pound inches. Tighten bolts in a crisscross pattern alternately torquing opposite fasteners.

3) Connect external fuel lines.

4) Connect electrical leads to fuel capacitor unit, switches and pumps, if applicable.

5) Check for fuel leaks.

6) Install access panel.

NOTE: Please see the TEMPORARY REVISION that revises this page.

Fuel Cell Sump Plate – Maintenance Practices

1. Removal/Installation

A. Remove Fuel Cell Sump Plate

1) Ensure POWER SOURCE switch is OFF (Pre Mod 6/2314); or BATTERY and EXTERNAL switches are OFF (Post Mod 6/2314), crossfeed valve is closed, and wing fuel tank switches (if fitted) are OFF.

2) Drain fuel from forward tank if No. 1, 2, 3 and/or 4 cell sump plates are to be removed. Drain fuel from aft tanks 5,6,7,8, if cell sump plates are to be removed. Drain both tanks if No. 5 and/or 6 cell sump plates are to be removed. (Refer to 12-10-10, Fuel System – General Data).

3) Remove access panel from underside of fuselage.

4) Disconnect electrical leads from fuel capacitor unit, switches and pumps, where applicable.

5) Disconnect external fuel lines. Remove fuel line from sump plate of No. 4 cell to sump plate of No. 6 cell, if No. 5 cell sump plate is to be removed.

6) Remove bolts securing sump plate, then lift up sump plate, turn and lower from cell. If No. 6 cell sump plate is being removed, remove crossfeed valve (refer to 28-20-21, Crossfeed Valve – Maintenance Practices), lower sump plate sufficiently from cell to disconnect two internal engine feed lines and then remove.

7) Cap open lines.

B. Install Fuel Cell Sump Plate

CAUTION **WHEN SERVICING EQUIPMENT ON SUMP PLATES REMOVED FROM FUEL CELLS NO. 4 AND 5, ENSURE CONNECTIONS TO SUMP PLATE OF THE LEVEL CONTROL VALVE AND OF FLEXIBLE HOSE FROM BOOST PUMPS ARE CORRECTLY LOCATED AND NOT INTERCHANGED. THE FLEXIBLE HOSE FROM BOOST PUMPS MUST MATE WITH THE MACHINED TEE FITTING MOUNTED ON SUMP PLATE LOWER SURFACE TO ENSURE UNINTERRUPTED FUEL SUPPLY TO ENGINES.**

1) If No. 6 cell sump plate is to be installed, connect two internal engine feed lines, and install crossfeed valve. (Refer to 28-20-21, Crossfeed Valve – Maintenance Practices).

2) Insert sump plate through cell aperture, position correctly and secure with bolts. Tighten bolts to torque value of 35 to 40 pound–inches.

3) Connect external fuel lines.

4) Connect electrical leads to fuel capacitor unit, switches and pumps, if applicable.

5) Check for fuel leaks.

6) Install access panel.

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Fuel Gallery Access Panel Seals (Mod 6/2408) – Maintenance Practices

1. Inspection/Check

   A. Access Panels and Seals

      1) Visually inspect the seals for disbond, tears or distortion, replace or repair, as required.

2. Removal/Installations

   A. Remove Access Panel Seal

      1) Remove quick release fasteners from the panel.

      2) Peel off seal completely. If present, remove remaining seal with a plastic scraper.

   B. Install Access Panel Seal

      1) Clean the access panel where the seal is to be bonded, with clean solvent DS108 or equivalent.

      2) Using a suitable bristle brush, apply a thin, even coat of EC–1357N adhesive to both bonding surfaces (on the panel and on the seal), making as few strokes as possible.

      3) If necessary, apply two or more coats of adhesive to the seal surface to ensure that sufficient adhesive remains on the surface. Allow each preceding coat to air dry before the next coat is applied.

      4) Allow the final coat of the adhesive to air dry until it is tacky.

      5) Press the seal to the panel and roll down with a rubber or stitch roller, to ensure equal contact to the full bonding surface.

      6) Create quick release holes on the seal.

      7) Install quick release fasteners on to the panel.

3. Approved Repairs

   A. Splice Seal Repair

      CAUTION DO NOT DAMAGE THE ACCESS PANEL.

      1) Cut through damaged section or defective seal with a sharp knife; cut line to be half inch beyond either side of damaged or defective section. A thin and clean stainless ruler or strap should be inserted between the seal and the panel under the cut line, to prevent potential cutting damage to the panel.

      2) Remove the damaged or defective section of the seal from the panel.

      3) Clean the access panel where the seal is removed with clean solvent DS108 or equivalent.

      4) Obtain a piece of the replacement seal section and cut the length to fit (slightly longer length is necessary to prevent any gap between the existing seal and the replacement seal).

      5) Using a suitable bristle brush, apply a thin, even coat of EC–1357N adhesive to both bonding surfaces (on the panel and on the seal), and the butt surfaces between the existing seal and the replacement seal, making as few strokes as possible.

      6) If necessary, apply two or more coats of adhesive to the seal surface to ensure that sufficient adhesive remains on the surface. Allow each preceding coat to air dry before the next coat is applied.

7) Allow the final coat of adhesive to air dry until it is tacky, but does not transfer to the finger when touched lightly.

8) Press the seal to the panel and roll down with a rubber or stitch roller to ensure equal contact to the full bonding surface.

Distribution – General Data

1. General

Fuel is transferred from No. 1 cell to No. 4 cell and from No. 8 cell to No. 5 cell by corresponding booster pump operated ejectors to maintain maximum fuel supply in each collector cell (4 and 5). The fuel is transferred from cells 2 and 3 into 1, and from cells 6 and 7 into 8 by gravity. Each system consists of a motive line, a fuel ejector, a transfer line and a level control valve. The motive line supplies booster pump pressure to provide the pressure drop across the ejector and there by draw fuel from the cell. Fuel drawn from the cell passes to the collector cell of the tank, through the transfer line and level control valve, together with the motive line fuel. The level control valve prevents over-filling of the collector cell. Check valves are incorporated to prevent siphoning of the fuel into the main cell; the motive line check valves contain fuel filters. The crossfeed check valves have an internal bleed provided to prevent thermal pressure build-up in the engine fuel supply lines. Each engine fuel supply line contains a check valve, a filter and an emergency shut-off valve. Arrows stencilled adjacent to each check valve in the fuel manifold, indicate direction of fuel flow. This ensures correct installation of the check valves when the arrow marked on each check valve coincides with its related stencilled arrow. The right engine obtains its fuel supply from the forward tank booster pumps and the left engine from the aft tank booster pumps, but when crossfeeding is selected. both engines are fed from one tank. The fuel lines are routed from the booster pumps to No. 6 cell, through the cell, up the rear side of the wing struts, along the wings, into the engine nacelles and engines.

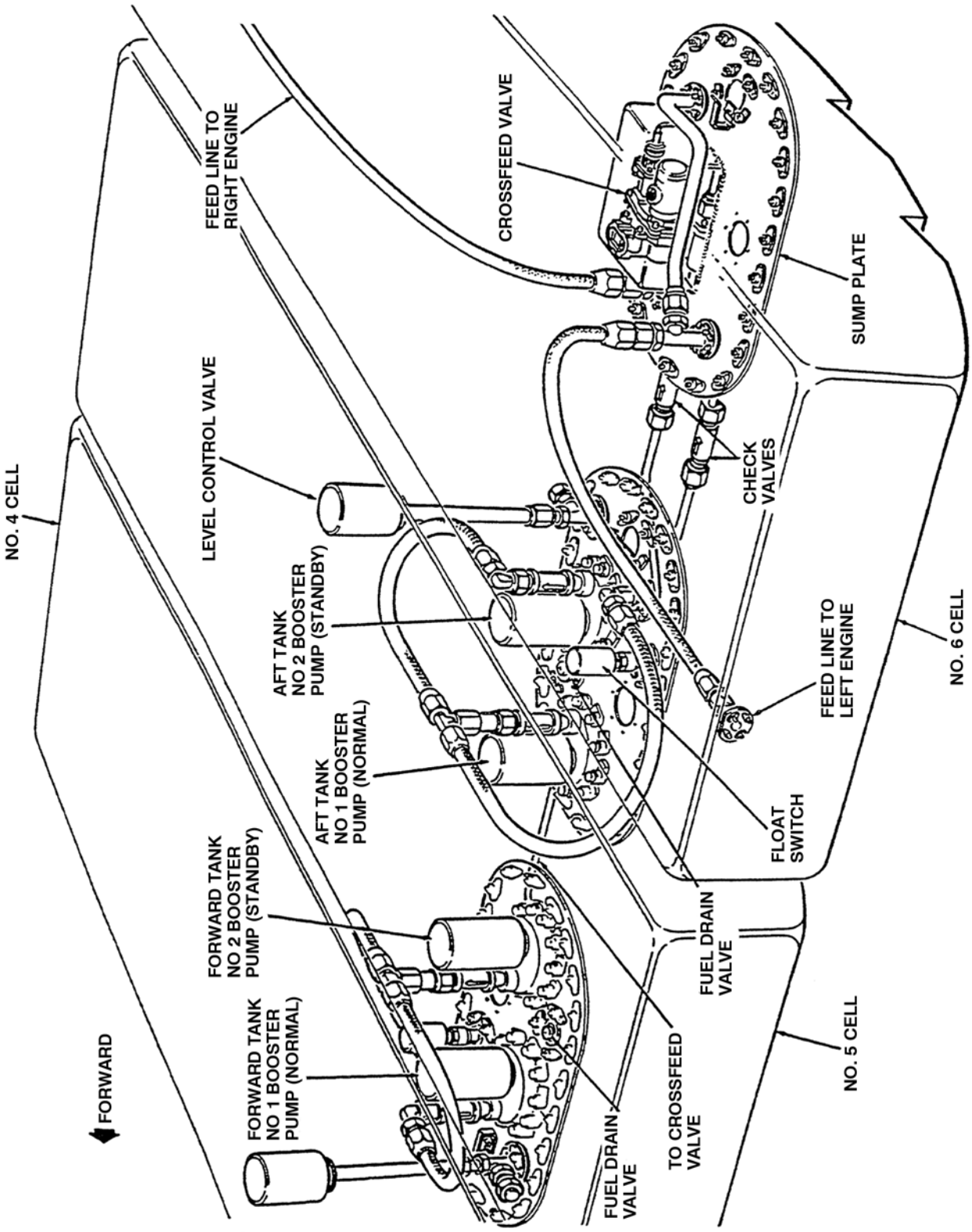
A. Fuel Booster Pumps

Four fuel booster pumps are installed, two in No. 4 cell of the forward tank and two in No. 5 cell of the aft tank. On aircraft with Mod 6/1795 installed, there are two submerged constant speed, centrifugal, electric fuel boost pumps and are located in each fuselage tank collector cell. The pumps deliver low-pressure fuel. They are powered from the left and right DC buses and are protected by circuit breakers labelled BOOST PUMP AFT 1, BOOST PUMP AFT 2, BOOST PUMP FWD 1 and BOOST PUMP FWD 2 on the main circuit breaker panel. Pre Mod 6/2267 pumps contain Captor Corp., part number A–3210 EMI filters installed in the circuit. The filters are mounted to the pump base. Post Mod 6/2267 pumps incorporate Lone Star, part number 122253–10A EMI filters installed in the circuit. The filters are mounted on a sheet metal plate installed on the RH Cabin sidewall at approximately station 135, below the window. On aircraft with Mod 6/2299 installed, improved fuel pumps have a new EMI/RFI electrical noise filter and ground circuit to improve corrosion tolerance of the pump and nearby components. Mod 6/2299 fuel pumps have two electrical fly leads that are connected to the aircraft wiring.

The fuel tank selector is located on the center instrument panel and marked FUEL SELECTOR. It is a rotary knob with three marked positions – BOTH ON FWD, NORMAL and BOTH ON AFT. In the NORMAL position the forward and aft tanks supply the right and left engines, respectively, the crossfeed valve is closed, and the boost pump(s) in each tank is energized. In BOTH ON FWD or AFT position, the crossfeed valve is open, the boost pumps of the non selected tank are automatically deenergized, and both boost pumps in the selected tank are energized (the boost pump switch selection is overridden). Boost pump switches located on the fuel panel are labelled BOOST PUMP 1 with ON, OFF and TEST positions marked. With power ON, circuit breakers set, and the FUEL SELECTOR positioned to NORMAL, the No. 1 boost pump in each collector tank is energized when ON position is selected; the No. 2 boost pumps remain inoperative. Control of the No. 2 boost pump in each tank is by means of an automatic electrical changeover sequence, which is initiated by the failure of the No. 1 boost pump. Refer to Figure 1 and Table 1 — Fuel Booster Pumps – Modification State.

Table 1 Fuel Booster Pumps – Modification State

MODIFICATION NO.	BOOSTER PUMP POSITION		TYPE OF PUMP	EMI FILTER
6/1795	No. 1	No. 2	Airborne (1D2–17)	A–3210 Captor Corp
6/2267	No. 1	No. 2	Airborne (1D2–20)	122253–10A Lone Star
6/2299	No. 1	No. 2	Airborne (1D2–21)	Embedded Filter (Captor Corp)



Booster Pumps and Crossfeed Valve Installation
Figure 1

B. Crossfeed Valve

The crossfeed valve is an electric motor-operated gate valve which is controlled by the FUEL SELECTOR switch on the fuel control panel in the flight compartment. The valve is mounted in the fuel line on the floor of No. 6 fuel cell. To replace a crossfeed valve it will be necessary to first drain all front and rear fuel cells. Crossfeed indication is controlled by a switch located in the crossfeed valve assembly.

C. Fuel Emergency Shut-off Valves

There are two fuel emergency shut-off valves provided to shut-off the fuel supply in case of engine malfunction. Each valve is an electric motor-operated gate valve mounted in the nacelle behind the firewall. A switch is provided on the emergency panel for each valve and is marked FUEL EMERGENCY OFF and NORMAL. To replace an emergency shut-off valve, drain fuel from filter (refer to 28-20-31, Fuel Filter (Mod 6/2006) – Maintenance Practices) then remove valve.

D. Fuel Filter

A fuel filter is provided for each engine and is mounted behind the engine firewall to the rear of the fuel emergency shut-off valve. The fuel filter is an assembly consisting of a housing, filter element and bypass valve. Should the filter element become clogged, the fuel forces the bypass valve to open and the unfiltered fuel passes directly through the center of the filter. A drain valve is installed in the aft nacelle and connected to the filter by a hose.

E. Standby Booster Pump Emergency Switches

Two lever-lock standby booster pump emergency switches, marked BOOST PUMP 2 are located on the fuel control panel. The emergency switches override the automatic change-over system when selected to EMERGENCY to operate the corresponding No. 2 booster pump in the aft and/or forward fuselage tank. The switches are locked in the off position and must be pulled before a selection can be made. Refer to Figure 2.

F. Fuel Ejectors

The fuel ejectors are designed on the venturi principle so that no moving parts are required. Fuel under pressure from the main feed system passing through the ejector draws additional fuel from the tank and moves the total fuel through the transfer line into the main cell. There are two ejectors, one is mounted on the sump plate of No. 1 cell and the other on the sump plate of No. 8 cell.

G. Drain Valves

There are six water drain valves incorporated in the fuel system to provide a check for water accumulation. One is installed on each fuel filter and two in each front and rear fuel manifold. Those on the fuel filters are located in the engine nacelles, the manifold drains are located in the trough on the underside of the fuselage.

H. Fuel Level Control Valves

Fuel level control valves are installed on the sump plates of cells No. 4 (forward tank) and No. 5 (aft tank). The valves control the level of fuel in cells 4 and 5 by opening or closing the fuel transfer system line, and in cells 1 and 7 control the level of fuel in the forward and aft tanks by opening or closing the pressure refuel line.

I. Motive Line Check Valves

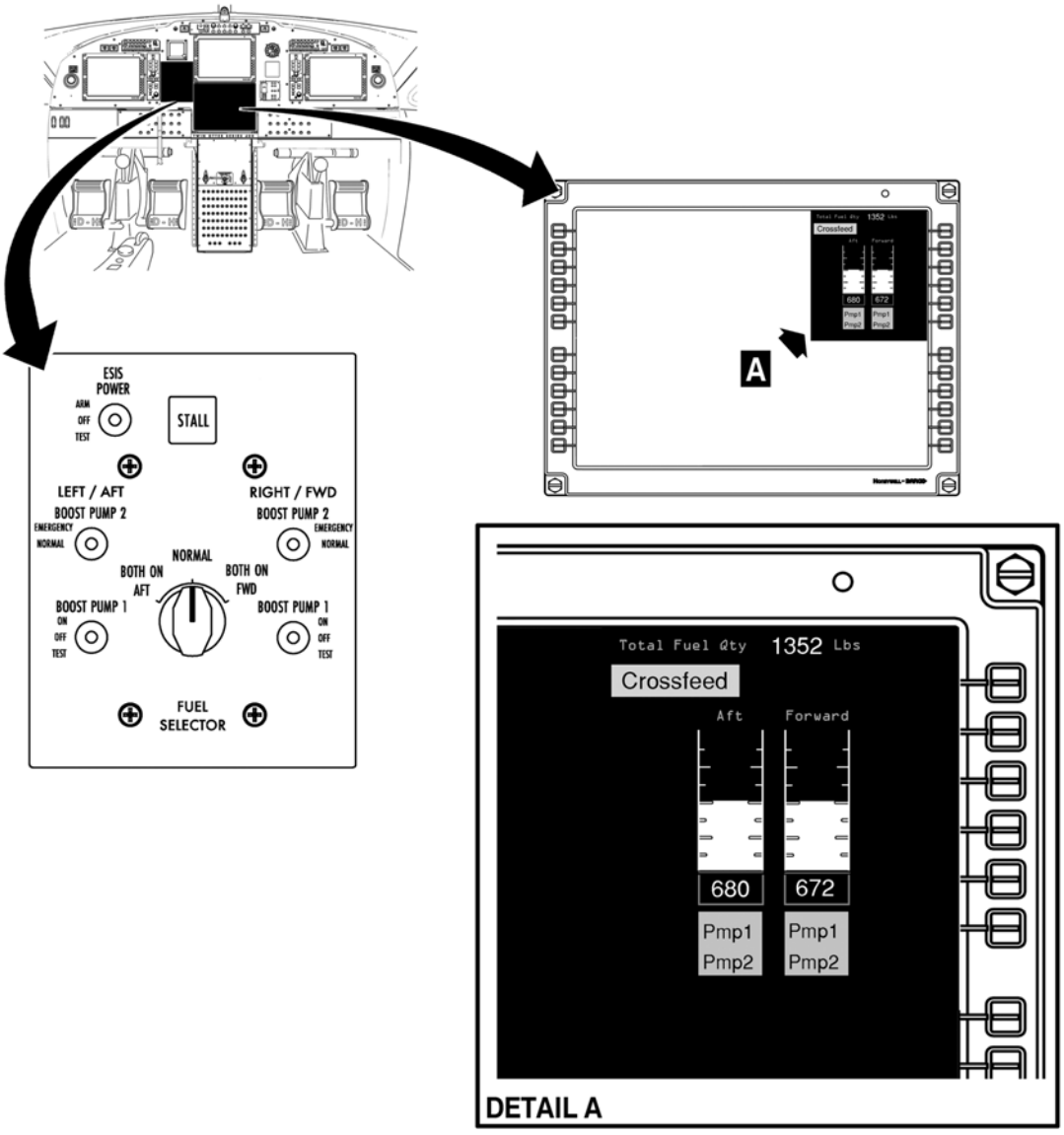
There are two motive line check valves fitted in the pressure lines to the ejectors, one beneath No. 1 cell and one beneath No. 7 cell. Each valve housing contains a fuel filter which can be removed for clearing when valve is disassembled.

J. Thermal Relief Check Valve

Two thermal relief check valves, one for each tank pressure line, are located below the forward edge of No. 6 fuel cell sump plate. The valve consists of a flap type check valve with an internal bleed provided to prevent thermal pressure build-up in the engine fuel supply lines.

K. Fuel Controls

Fuel controls consist of two main booster pump switches (BOOST PUMP 1), two emergency switches (BOOST PUMP 2), and a rotary FUEL SELECTOR switch. Refer to Figure 2.



Fuel Controls
Figure 2

Distribution (With Mod S.O.O. 6247) – General Data

1. General

Fuel distribution provides fuel transfer within the fuselage tanks, and pressure fuel distribution from the fuselage and wing tanks to the engines. The right engine obtains its pressure fuel supply from the forward fuselage and right wing tanks, the left engine from the aft fuselage and left wing tanks. A crossfeed valve when selected allows both engines to be fed from one fuselage tank and its associated wing tank.

2. Description

The fuel distribution system consists of fuel booster pumps, wing fuel pressure pumps, wing tank fuel filter, a crossfeed valve, emergency shut-off valves, fuel transfer valves, fuel filters, control switches, fuel ejectors, drain valves, level control valves, and check valves. Aircraft have arrows stenciled adjacent to each check valve in the fuel manifold, indicating direction of fuel flow. This ensures correct installation of the check valves when the arrow marked on each check valve coincides with its related stenciled arrow. The pressure fuel supply line to each engine is routed from a fuselage booster pump to No. 6 cell, through the cell, up the rear of a wing strut, along the wing into the engine nacelle where it joins the wing tank pressure fuel supply line. From this junction, at the inlet side of the fuel filter, the fuel is routed through the fuel filter element, a fuel emergency shut-off valve, and through the firewall to the engine-driven fuel pump. The electrical power to the system is supplied from the 28 Volt DC bus.

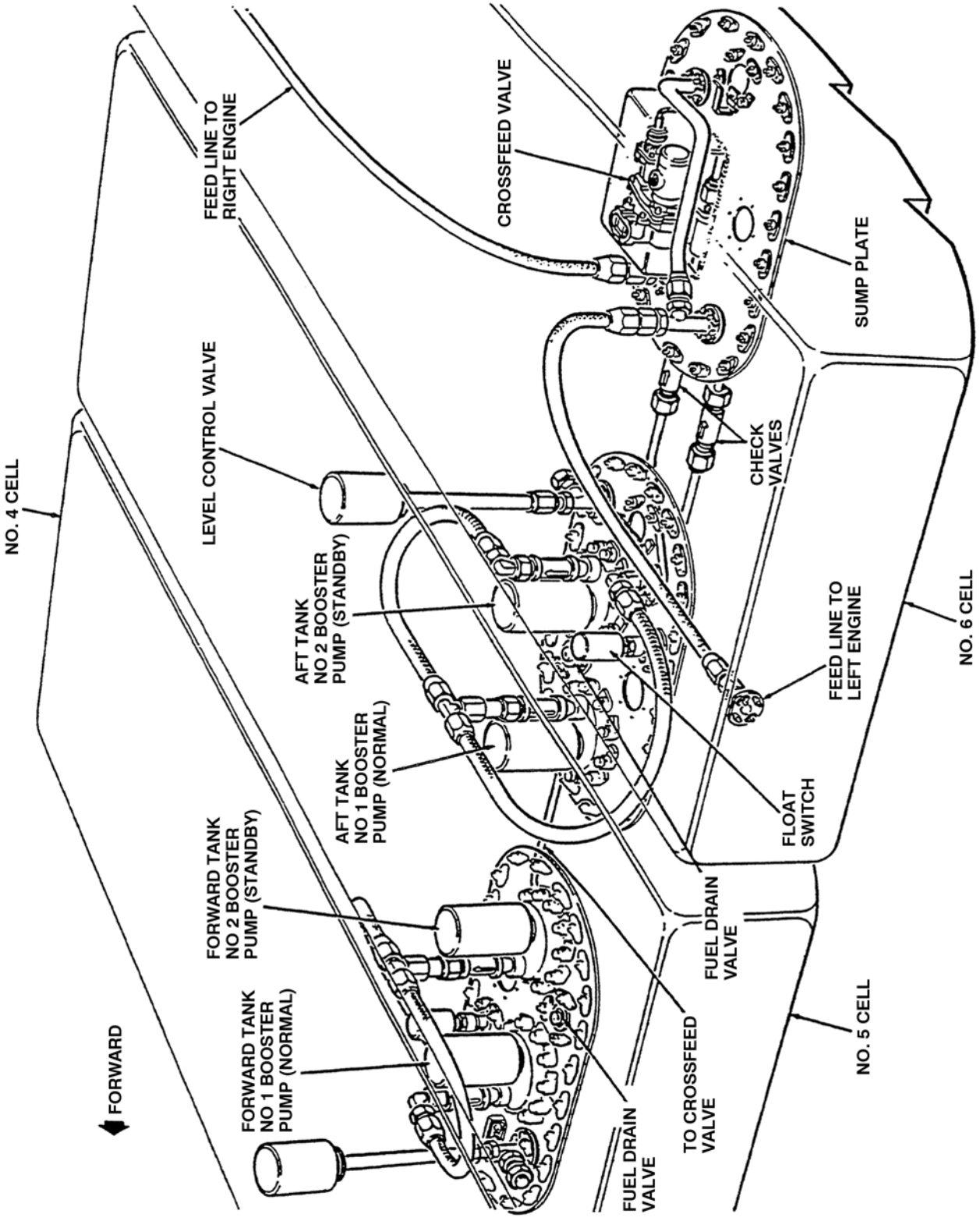
A. Fuel Booster Pumps

Four fuel booster pumps are installed, two in No. 4 cell of the forward tank and two in No. 5 cell of the aft tank. On aircraft with Mod 6/1795 installed, there are two submerged constant speed, centrifugal, electric fuel boost pumps and are located in each fuselage tank collector cell. The pumps deliver low-pressure fuel. They are powered from the left and right DC buses and are protected by circuit breakers labelled BOOST PUMP AFT 1, BOOST PUMP AFT 2, BOOST PUMP FWD 1 and BOOST PUMP FWD 2 on the main circuit breaker panel. Pre Mod 6/2267 pumps contain Captor Corp., part number A–3210 EMI filters installed in the circuit. The filters are mounted to the pump base. Post Mod 6/2267 pumps incorporate Lone Star, part number 122253–10A EMI filters installed in the circuit. The filters are mounted on a sheet metal plate installed on the RH Cabin sidewall at approximately station 135, below the window. On aircraft with Mod 6/2299 installed, improved fuel pumps have a new EMI/RFI electrical noise filter and ground circuit to improve corrosion tolerance of the pump and nearby components. Mod 6/2299 fuel pumps have two electrical fly leads that are connected to the aircraft wiring.

The fuel tank selector is located on the center instrument panel and marked FUEL SELECTOR. It is a rotary knob with three marked positions – BOTH ON FWD, NORMAL and BOTH ON AFT. In the NORMAL position the forward and aft tanks supply the right and left engines, respectively, the crossfeed valve is closed, and the boost pump(s) in each tank is energized . In the BOTH ON FWD or AFT position, the crossfeed valve is open, the boost pumps of the non selected tank are automatically deenergized, and both boost pumps in the selected tank are energized (the boost pump in the selected tank are energized (the boost pump switch selection is overridden). Boost pump switches located on the fuel panel are labelled BOOST PUMP 1 with ON, OFF and TEST positions marked. With power ON, circuit breakers set, and the FUEL SELECTOR positioned to NORMAL, the No. 1 boost pump in each collector tank is energized when ON position is selected; the No. 2 boost pumps remain inoperative. Control of the No. 2 boost pump in each tank is by means of an automatic electrical changeover sequence, which is initiated by the failure of the No. 1 boost pump. Refer to Figure 1 and Table 1 — Fuel Booster Pumps – Modification State.

Table 1 Fuel Booster Pumps – Modification State

MODIFICATION NO.	BOOSTER PUMP POSITION		TYPE OF PUMP	EMI FILTER
6/1795	No. 1	No. 2	Airborne (1D2–17)	A–3210 Captor Corp
6/2267	No. 1	No. 2	Airborne (1D2–20)	122253–10A Lone Star
6/2299	No. 1	No. 2	Airborne (1D2–21)	Embedded Filter (Captor Corp)



Booster Pump and Crossfeed Valve Installation
Figure 1

B. Wing Fuel Pressure Pump

A fuel pressure pump is mounted outside each wing tank on the wing outboard nose rib. The pumps are electric motor-driven and controlled by switches on the fuel control panel. The left wing tank pump is operated when the left switch is selected to left engine; similarly the right switch operates the right wing tank pump. The wing fuel pressure pumps when operating will overcome fuselage booster pump fuel pressure and empty the wing tanks before the fuselage tanks. On aircraft incorporating Mod 6/1374 a radio noise filter is installed in each pump circuit.

C. Crossfeed Valve

The crossfeed valve is an electric motor-operated gate valve which is controlled by the FUEL SELECTOR switch on the fuel control panel in the flight compartment. The valve is mounted in the fuel line on the floor of No. 6 fuel cell. To replace a crossfeed valve it will be necessary to first drain both fuselage fuel tanks.

D. Fuel Emergency Shut-off Valves

There are two fuel emergency shut-off valves provided to shut off the fuel supply in case of engine malfunction. Each valve is an electric motor-operated gate valve mounted in the nacelle behind the firewall. A switch is provided on the emergency panel for each valve and is marked FUEL OFF and NORMAL. To replace an emergency shut-off valve, drain fuel from filter (refer to 28-20-31, Fuel Filter (Mod 6/2006) – Maintenance Practices), then remove valve.

E. Fuel Filter

A fuel filter is provided for each engine and is mounted behind the engine firewall to the rear of the fuel emergency shut-off valve. The fuel filter is an assembly consisting of a housing, filter element and bypass valve. Should the filter element become clogged. The fuel forces the bypass valve to open and the unfiltered fuel passes directly through the center of the filter. A drain valve is installed in the filter cover.

F. Standby Booster Pump Emergency Switches

Two lever-lock type standby booster pump emergency switches, marked, BOOST PUMP 2 are located on the fuel control panel the emergency switches override the automatic change-over system when selected to EMERGENCY to operate the corresponding No. 2 booster pump in the aft and/or forward fuselage tank. The switches are locked in the off position and must be pulled before a selection can be made. Refer to Figure 2 for details.

G. Wing Fuel Tank Switches

Two lever-lock type wing fuel tank switches, marked LEFT WING TANK and RIGHT WING TANK are located on the FUEL SELECTOR panel. There are three selections available on each switch, REFUEL, OFF and ENGINE. The switch lock is in operation in the OFF position and must be pulled before either of the two other selections can be made. When REFUEL is selected, the refuel shut-off valve is opened and allows fuel, from the fuselage booster pumps (if selected), to flow through the level control valve into the wing tank. The ENGINE selection operates the applicable wing tank fuel pressure pump and opens a fuel transfer valve.

H. Fuel Ejectors

The fuel ejectors are designed on the venture principle so that no moving parts are required. Fuel under pressure from the main feed system passing through the ejector draws additional fuel from the tank and moves the total fuel through the transfer line into the main cell. There are two ejectors; one is mounted on the sump plate of No. 1 cell and the other on the sump plate of No. 8 cell.

I. Drain Valves

There are six water drain valves incorporated in the fuel system to provide a check for water accumulation. One is installed on each fuel filter, one in each fuel manifold, and one in each wing tank. Those on the fuel

filters are located in the engine nacelles, the fuel manifold water drains are located in the trough on the underside of the fuselage, and the wing tank water drains protrude from the underside of each wing tank.

J. Fuel Level Control Valves

There are four level control valves incorporated in the fuel system, one in each fuel tank. The fuselage tanks level control valves are mounted on the sump plates of cells No. 4 and 5. Each valve controls the level of fuel in the cell or tank by closing off or admitting fuel from the fuselage transfer system or wing refueling shut-off valve line.

K. Fuel Transfer Valves

A fuel transfer valve is fitted in each wing tank pressure fuel line on the outboard side of the tank. The valve is a solenoid-operated shut-off type valve and is opened when the wing fuel pressure pump is operated by an applicable L or R ENGINE selection on the FUEL SELECTOR wing tank fuel switches.

L. Refuel Shut-off Valve

A refuel shut-off valve is fitted to a tee connection in each wing tank pressure fuel line on the rear outboard side of each tank. When operated, the valve opens the pressure fuel line from the fuselage booster pumps to the level control valve inside the wing fuel tank. This allows booster pump pressure fuel from the fuselage tanks to refuel the wing tanks. A refuel shut-off valve is opened by selecting REFUEL on the wing tank fuel switch.

M. Motive Line Check Valves

There are two motive line check valves fitted in the pressure lines to the ejectors, one beneath No. 1 cell and one beneath No. 7 cell. Each valve housing contains a fuel filter which can be removed for cleaning when the valve is disassembled.

N. Thermal Relief Check Valve

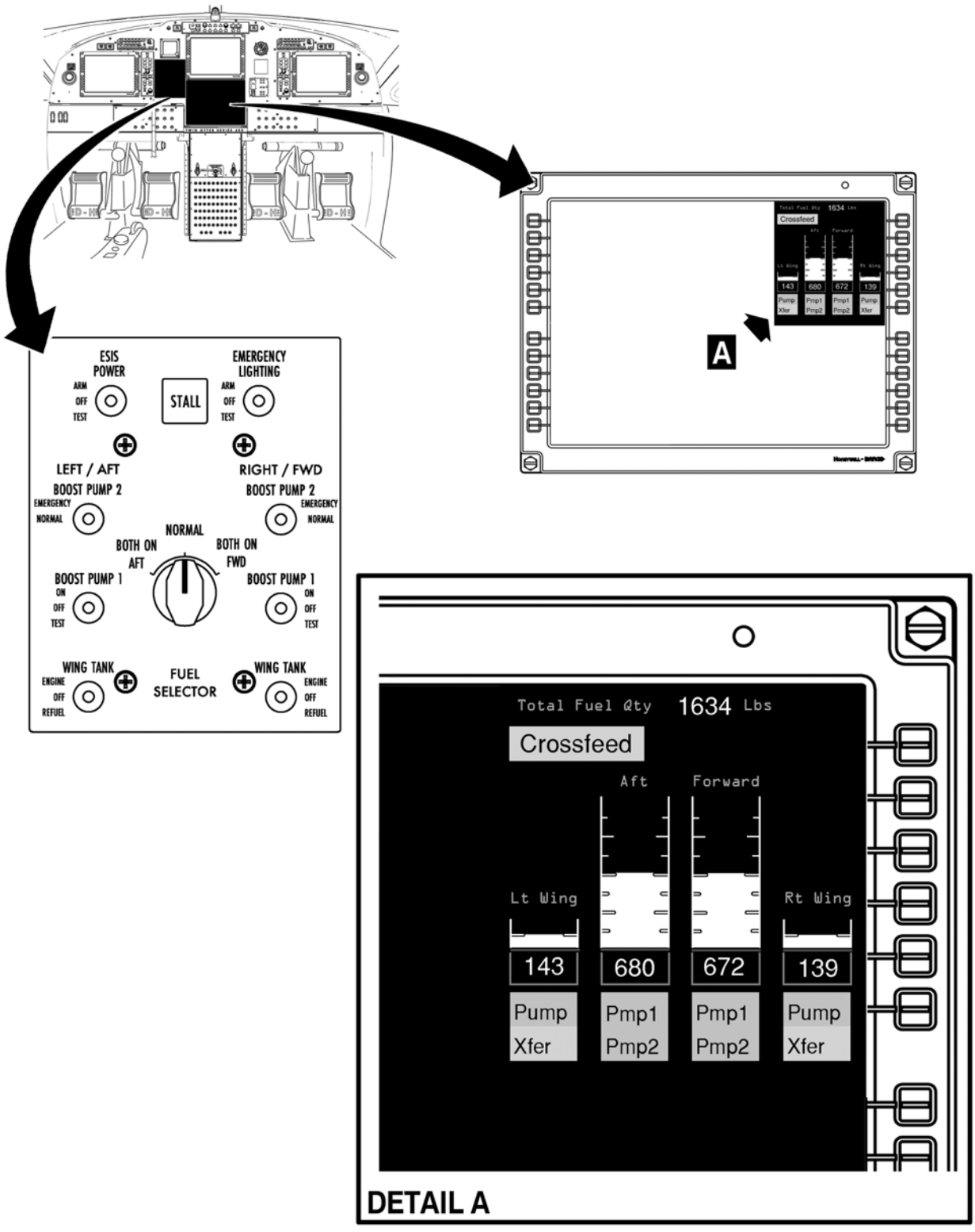
Two thermal relief check valves, one for each tank pressure line, are located below the forward edge of No. 6 fuel cell sump plate. The valve consists of a flap type check valve with an internal bleed provided to prevent thermal pressure buildup in the engine fuel supply line.

O. Fuel Controls

Fuel controls consist of two main booster pump switches (BOOST PUMP 1), two emergency boost pump switches (BOOST PUMP 2), and rotary FUEL SELECTOR switch. The components are all located on the fuel control panel. Fuel controls for the wing tanks consist of two fuel pressure pump switches. The switches are located on the fuel control panel. Indication is displayed on lower 1/6th screen of the fuel window on the MFD. Refer to Figure 2 for details.

P. Wing Tank Filter (Mod 6/1398)

An in-line filter is installed between the wing tank and the wing fuel pressure pump, the filter housing contains a pleated metal filter element which can be removed for cleaning.



Fuel Controls
Figure 2

3. Operation

With the 28 Volt DC bus energized, circuit breakers engaged, FUEL SELECTOR switch at NORMAL, fuel booster pump switches at ON, the wing fuel tank switches at ENGINE and engines running, the system is in the normal operating mode. The wing fuel pressure pumps will supply fuel to the appropriate engine until the wing tanks empty, at which time the No. 1 fuel booster pumps will supply the engines from the fuselage tanks. The wing fuel tank switches should be set to OFF when the wing tanks empty. As the fuel is used from collector cells 4 and 5, fuel is transferred from cells 1 and 8 by booster pump operated ejectors to maintain a maximum fuel level in the collector cells. The fuel level control valves in the collector cells prevent overfilling. As the fuel is transferred from cells 1 and 8, replacement fuel is transferred by gravity from cells 2 and 3 into 1, and from cells 6 and 7 into 8.

When crossfeeding is selected, by moving the FUEL SELECTOR switch to BOTH ON AFT, both engines are supplied with fuel from the aft fuselage and right wing tanks. When BOTH ON FWD is selected, both engines are supplied from the forward fuselage and left wing tanks. If crossfeeding, the pump on the wing tank not in use should be switched off.

Fuel Booster Pump – Maintenance Practices

1. Removal/Installation

A. Remove Booster Pump

1) Ensure battery and/or electrical power supply is disconnected and crossfeed valve closed. Ensure WING TANK FUEL switches (if fitted) are OFF.

2) Drain applicable fuel tank (refer to 12-10-10, Fuel System – General Data).

3) Remove access panel from underside of fuselage (refer to 12-00-00, Servicing – General Data).

4) Disconnect pressure switch line from booster pump base and loosen line to facilitate pump removal.

WARNING **ON AIRCRAFT FITTED WITH E.M.I. FILTER, BEFORE REMOVING BOOSTER PUMP ELECTRICAL CONNECTIONS, LOAD TERMINAL ON MAIN DISTRIBUTION BOX OF ASSOCIATED INTERFERENCE FILTER MUST BE DISCHARGED TO AIRCRAFT GROUND.**

5) Before disconnecting electrical leads from terminal on booster pump, proceed as follows:

a) Pull out associated booster pump circuit breaker.

b) Discharge filter through LOAD terminal either by connecting LOAD terminal to aircraft ground through a 150 ohm, 7.5 watt resistor, for at least 30 seconds, or, connecting a wire to an aircraft ground and discharging by touching other end of wire to LOAD terminal.

Note Do not reengage circuit breaker until leads are reconnected to booster pump terminals.

If leads have to be subsequently removed for further maintenance, discharge procedures in Para 1.A.5) above must be repeated prior to lead disconnection.

6) Disconnect leads from, radio noise filter at pump base, disconnect leads from pressure switch. For Mod 6/2299 fuel pumps, disconnect the electrical fly leads from the quick disconnect splice ends.

7) Remove booster pump retaining bolts and washers, and lower pump until upper end of check valve is exposed.

8) Disconnect flexible line at check valve.

9) Remove booster pump and attached parts.

10) Remove check valve and seal drain tube from booster pump.

11) Cap all openings.

B. Install Booster Pump

1) Install check valve and seal drain tube to booster pump.

2) Position booster pump and gasket below sump plate so that radio noise filter at pump base is parallel to aircraft center line.

3) Connect flexible line to check valve.

4) Lift booster pump into position and install retaining bolts and washers.

5) Connect leads to radio noise filter and connect leads to pressure switch. For Mod 6/2299 fuel pumps, connect the electrical fly leads contacts to the individual quick disconnect splice ends.

Note If leads have to be subsequently removed for further maintenance, discharge procedures in Para 1.A.5) must be repeated prior to lead disconnection.

6) If removed, connect pressure switch line to booster pump and tighten all line connections.

7) Check for fuel leaks (refer to 28-00-00, Fuel (Mod 6/2006) – General Data).

8) Install access panel (refer to 12-00-00, Servicing – General Data).

2. Adjustment/Test

A. Operational Test

1) Ensure each fuselage tank contains a minimum of 30 Imperial (36 U.S.) gallons of fuel.

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Ensure relevant circuit breakers are engaged.

4) Ensure fuel levers are OFF, FUEL SELECTOR switch is set to NORMAL, and wing tank fuel switches (if fitted) are OFF.

5) Set forward and aft BOOST PUMP 1 switches to ON. Check all No. 1 and No. 2 boost pump pressure caution messages go out and verify that the fuel pump amber blocks Pmp1 and Pmp2 for the forward and aft fuel tanks are not displayed.

Note Fuel pressure from the No. 1 booster pump in each tank actuates a pressure switch which renders the Pmp2 amber blocks inoperative.

6) Hold forward and aft BOOST PUMP 1 switches in the TEST position. Check both No. 2 boost pump pressure amber blocks and caution messages disappear.

Note Both No. 2 booster pumps are operating to actuate the No. 2 boost pump amber blocks Pmp2.

7) Release forward and aft BOOST PUMP 1 switches. Check all No. 1 and No. 2 boost pump amber blocks Pmp1 and Pmp2 and associated CAS messages are displayed.

8) With container in position to receive spillage, disconnect right engine fuel supply line at firewall connection.

9) Connect a suitable length of hose to firewall connection and feed into container.

10) Select BOOST PUMP 1 switch ON.

11) When fuel flow is stabilized through hose at engine nacelle, measure flow rate. (Minimum of 1 Imperial gallon in 30 seconds).

12) Hold BOOST PUMP 1 switch in TEST position and repeat Para 2.A.11).

13) Release BOOST PUMP 1 switch, remove hose from firewall connection and reconnect fuel line.

14) Repeat Para 2.A.8) through Para 2.A.13) for aft tank substituting AFT for FWD and left engine for right engine.

15) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Bench Test

Equipment required (As shown in Figure 1):

- Multi-meter
- 27 Volt DC Supply

Note Calibration is done under ambient conditions, using JP–4 Turbine Fuel as the test fluid and Figure 2 to check limits of results.

1) With test setup as shown in Figure 1 and temperature of fluid at 80° ± 10°F, run pump to 460 to 560 pph flow rate for a period of 15 minutes. Ensure max current limit (Figure 2) is not exceeded.

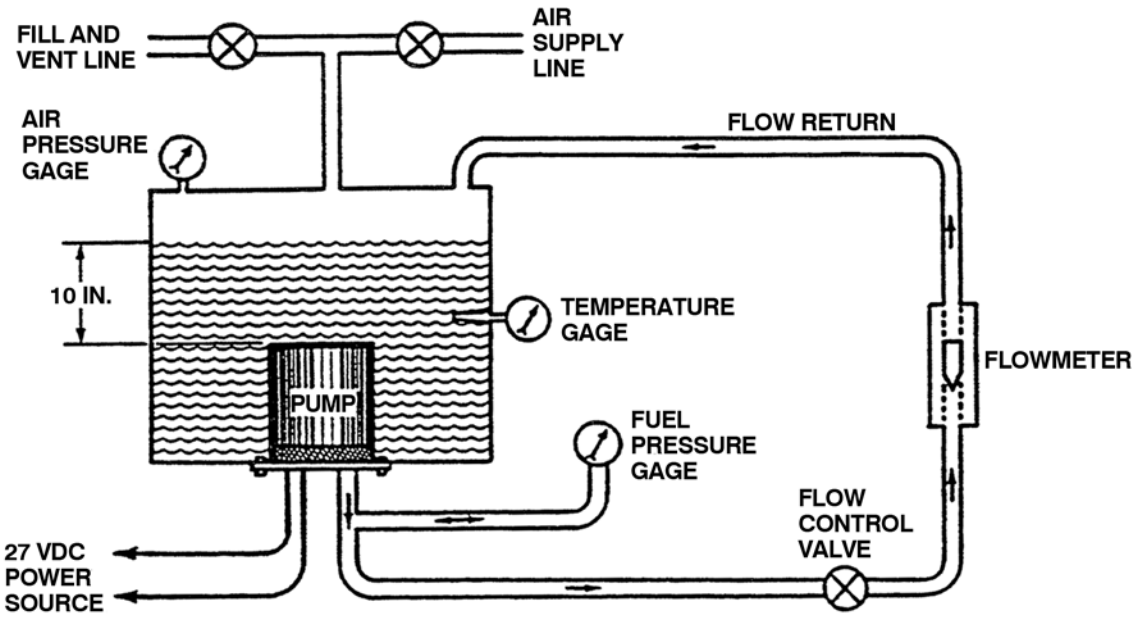
2) Check no leakage occurs through discharge port, tank drain port casting, gaskets, or around motor leads, during Para 2.B.1). A maximum leakage allowance of 5 drops (1/4 cc) in 1 hour is allowed for seal drain port while pump is running.

3) Close flow control valve and check fuel pressure does not exceed 40 psi.

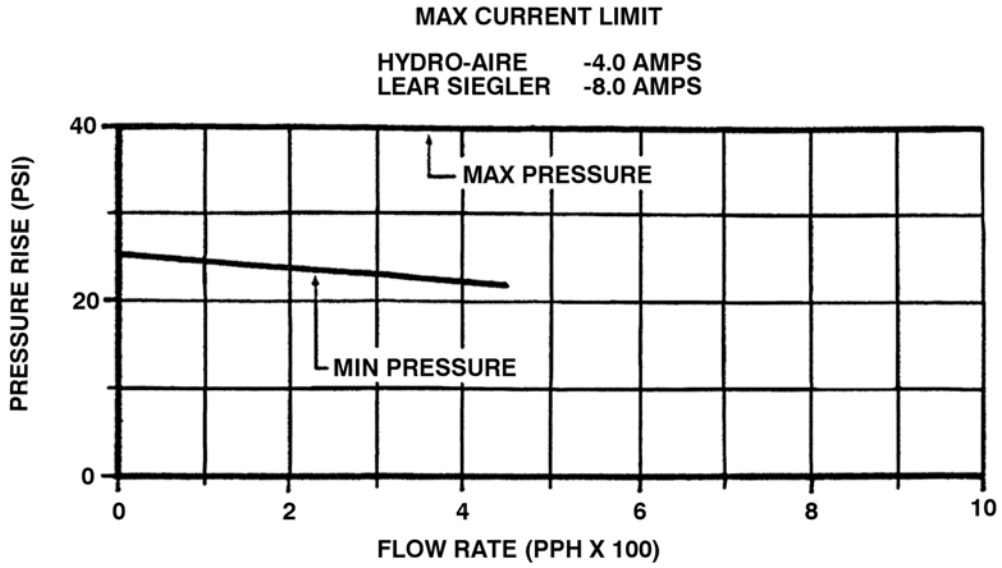
4) Open flow control valve to rate fuel flow at 450 pph check fuel pressure does not fall below 22 psi.

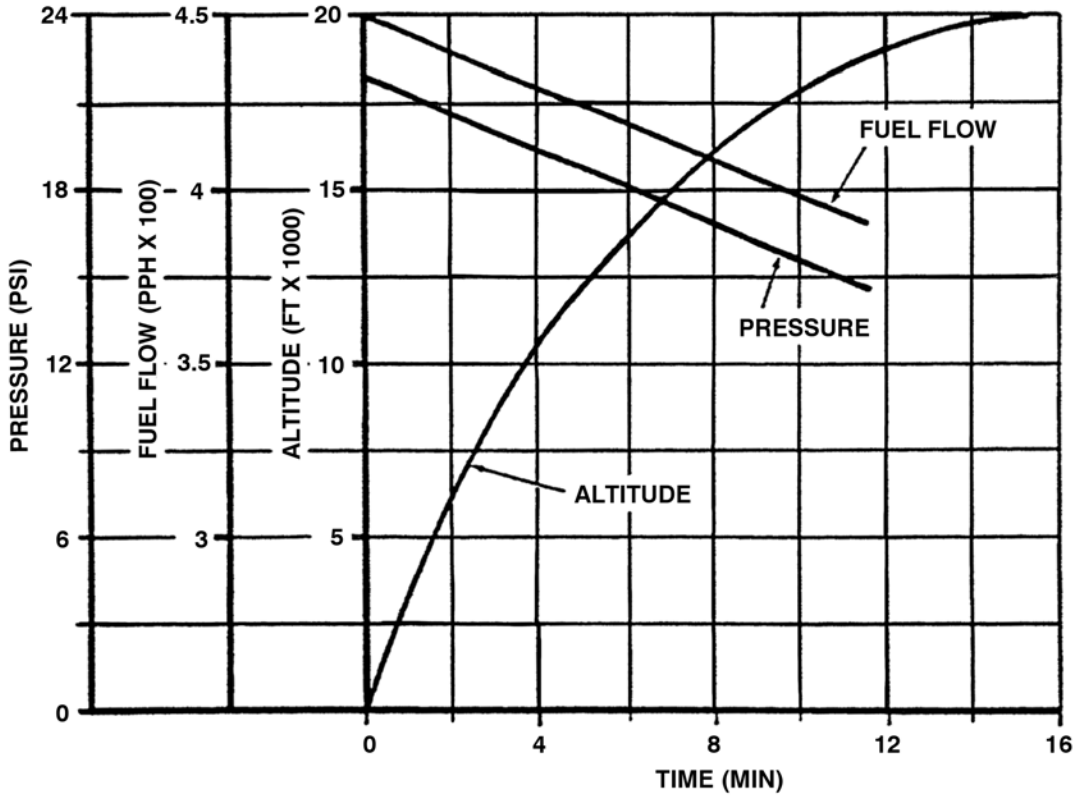
5) Switch off pump.

6) Apply 15 psi air pressure within tank for a period of 5 minutes and check no fuel leaks occur at areas listed in Para 2.B.2).



Pump Test Rig
Figure 1





Bench Test Curves
Figure 2

3. Inspection/Check

A. Leak Check – Seal Drain Port

1) Remove access panel from under side of fuselage (refer to 12-00-00, Servicing – General Data).

2) Ensure each fuselage tank contains a minimum of 30 Imperial (36 U.S.) gallons of fuel.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Ensure relevant circuit breakers are engaged.

5) Ensure fuel levers are OFF, FUEL SELECTOR switch is set to NORMAL, and WING TANK FUEL switches (if fitted) are OFF.

6) Select forward and aft BOOST PUMP 1 switches to ON. Check all No. 1 and No. 2 boost pump amber blocks Pmp1 and Pmp2 and associated CAS messages are not displayed.

7) Check each No. 1 pump seal drain port for fuel leaks. Maximum leakage allowed 5 drops (1/4 cc) in 1 hour while pump is running. No leakage allowed on stationary pump.

8) Switch off No. 1 pumps by selecting booster pump switches to OFF.

9) Select forward and aft BOOST PUMP 2 switches to EMERGENCY. Check both No. 2 boost pump amber blocks Pmp2 are no longer displayed.

10) Repeat Para 3.A.7) for the No. 2 pumps.

11) Switch forward and aft BOOST PUMP 2 switches to OFF.

12) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

13) Install access panel to underside of fuselage (refer to 12-00-00, Servicing – General Data).

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Fuel Booster Pump (Post Mod 6/2267) – Maintenance Practices

1. Removal/Installation

A. Remove Booster Pump

1) Ensure battery and/or electrical power supply is disconnected and crossfeed valve closed. Ensure WING TANK FUEL switches (if fitted) are OFF.

2) Drain applicable fuel tank (refer to 12-10-10, Fuel System – General Data).

3) Remove access panel from underside of fuselage (refer to 12-00-00, Servicing – General Data).

4) Disconnect pressure switch line from booster pump base and loosen line to facilitate pump removal.

WARNING **ON AIRCRAFT FITTED WITH E.M.I. FILTER, BEFORE REMOVING BOOSTER PUMP ELECTRICAL CONNECTIONS, LOAD TERMINAL ON MAIN DISTRIBUTION BOX OF ASSOCIATED INTERFERENCE FILTER MUST BE DISCHARGED TO AIRCRAFT GROUND.**

5) Before disconnecting electrical leads from terminal on booster pump, proceed as follows:

a) Pull out associated booster pump circuit breaker.

b) Discharge EMI filter through +A terminal either by connecting +A terminal to aircraft ground through a 150 ohm, 7.5 watt resistor, for at least 30 seconds, or, connecting a wire to an aircraft ground and discharging by touching other end of wire to +A terminal.

Note Do not reengage circuit breaker until leads are reconnected to booster pump terminals.

If leads have to be subsequently removed for further maintenance, discharge procedures in Para 1.A.5) above must be repeated prior to lead disconnection.

6) Disconnect leads from pump, disconnect leads from pressure switch.

7) Remove booster pump retaining bolts and washers, and lower pump until upper end of check valve is exposed.

8) Disconnect flexible line at check valve.

9) Remove booster pump and attached parts.

10) Remove check valve and seal drain tube from booster pump.

11) Cap all openings.

B. Install Booster Pump

1) Install check valve and seal drain tube to booster pump.

2) Position booster pump and gasket below sump plate so that radio noise filter at pump base is parallel to aircraft center line.

3) Connect flexible line to check valve.

4) Lift booster pump into position and install retaining bolts and washers.

5) Connect leads to pump and connect leads to pressure switch.

Note If leads have to be subsequently removed for further maintenance, discharge procedures in Para 1.A.5) must be repeated prior to lead disconnection.

6) If removed, connect pressure switch line to booster pump and tighten all line connections.

7) Check for fuel leaks (refer to 28-00-00, Fuel (Mod 6/2006) – General Data).

8) Install access panel (refer to 12-00-00, Servicing – General Data).

2. Adjustment/Test

A. Operational Test

1) Ensure each fuselage tank contains a minimum of 30 Imperial (36 U.S.) gallons of fuel.

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Ensure relevant circuit breakers are engaged.

4) Ensure fuel levers are OFF, FUEL SELECTOR switch is set to NORMAL, and wing tank fuel switches (if fitted) are OFF.

5) Set forward and aft BOOST PUMP 1 switches to ON. Check all No. 1 and No. 2 boost pump pressure caution messages go out and verify that the fuel pump amber blocks Pmp1 and Pmp2 for the forward and aft fuel tanks are not displayed.

Note Fuel pressure from the No. 1 booster pump in each tank actuates a pressure switch which renders the Pmp2 amber blocks inoperative.

6) Hold forward and aft BOOST PUMP 1 switches in the TEST position. Check both No. 2 boost pump pressure amber blocks and caution messages disappear.

Note Both No. 2 booster pumps are operating to actuate the No. 2 boost pump amber blocks Pmp2.

7) Release forward and aft BOOST PUMP 1 switches. Check all No. 1 and No. 2 boost pump amber blocks Pmp1 and Pmp2 and associated CAS messages are displayed.

8) With container in position to receive spillage, disconnect right engine fuel supply line at firewall connection.

9) Connect a suitable length of hose to firewall connection and feed into container.

10) Select BOOST PUMP 1 switch ON.

11) When fuel flow is stabilized through hose at engine nacelle, measure flow rate. (Minimum of 1 Imperial gallon in 30 seconds).

12) Hold BOOST PUMP 1 switch in TEST position and repeat Para 2.A.11).

13) Release BOOST PUMP 1 switch, remove hose from firewall connection and reconnect fuel line.

14) Repeat Para 2.A.8) through Para 2.A.13) for aft tank substituting AFT for FWD and left engine for right engine.

15) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Bench Test

Equipment required (As shown in Figure 1):

- Multi-meter
- 27 Volt DC Supply

Note Calibration is done under ambient conditions, using JP–4 Turbine Fuel as the test fluid and Figure 2 to check limits of results.

1) With test setup as shown in Figure 1 and temperature of fluid at 80° ± 10°F, run pump to 460 to 560 pph flow rate for a period of 15 minutes. Ensure max current limit (Figure 2) is not exceeded.

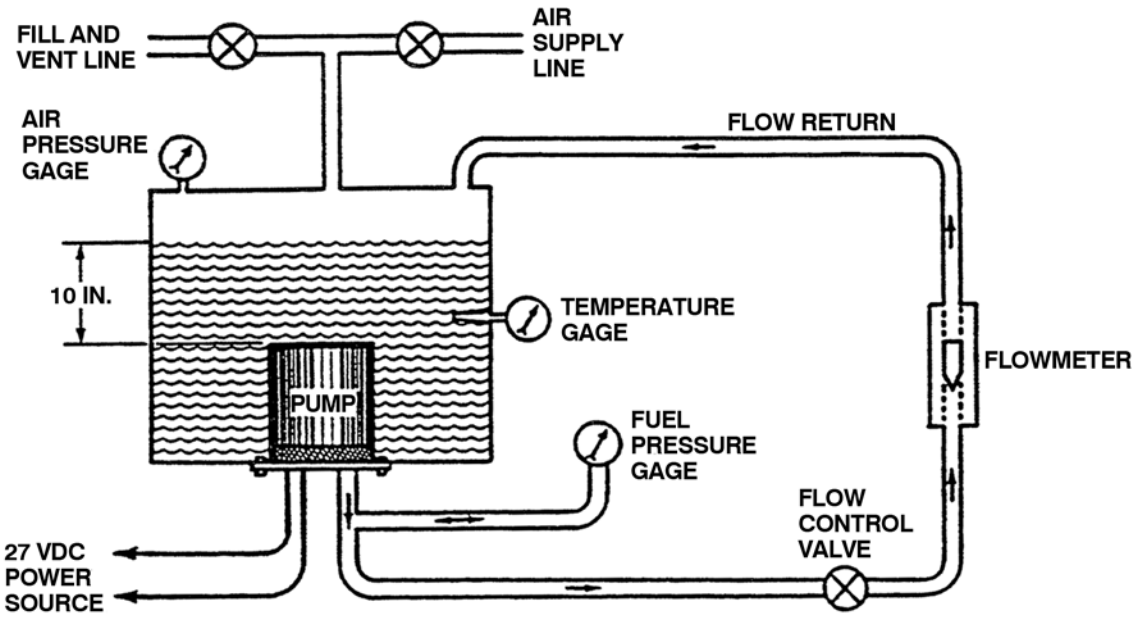
2) Check no leakage occurs through discharge port, tank drain port casting, gaskets, or around motor leads, during Para 2.B.1). A maximum leakage allowance of 5 drops (1/4 cc) in 1 hour is allowed for seal drain port while pump is running.

3) Close flow control valve and check fuel pressure does not exceed 40 psi.

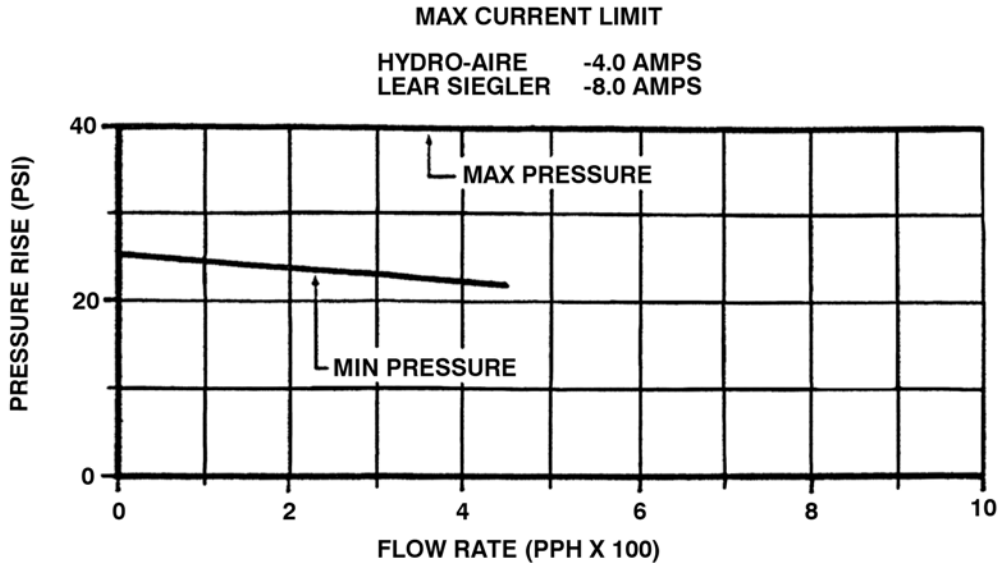
4) Open flow control valve to rate fuel flow at 450 pph check fuel pressure does not fall below 22 psi.

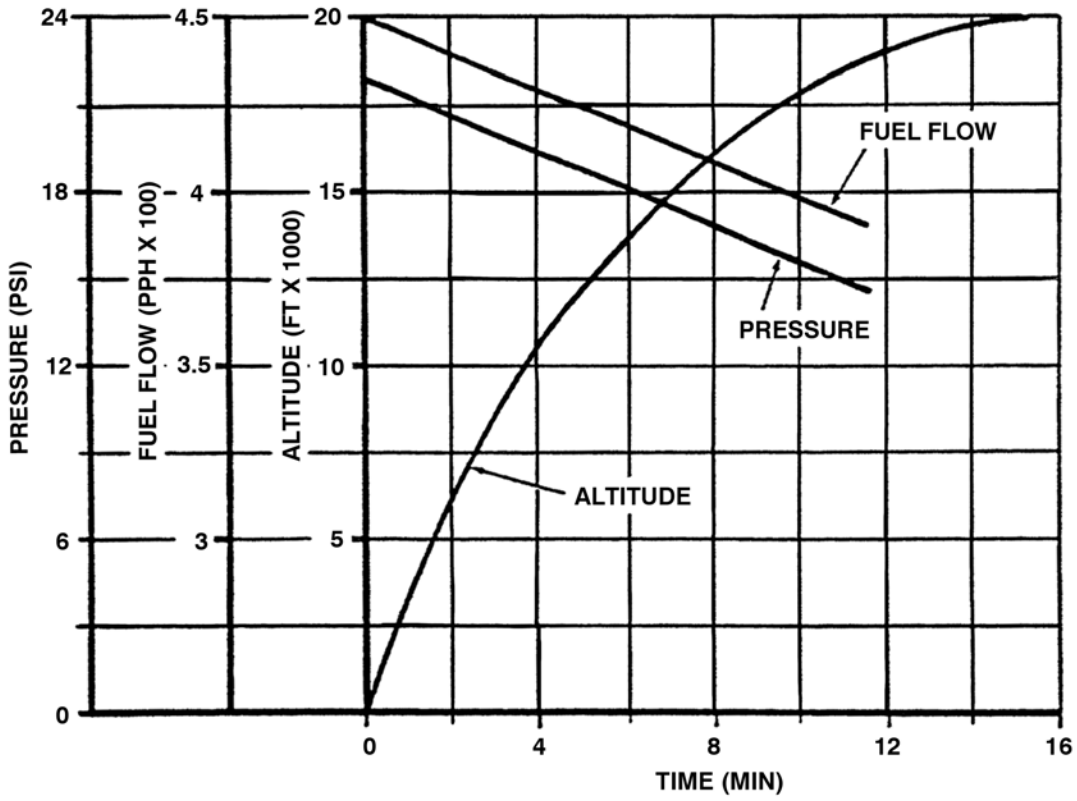
5) Switch off pump.

6) Apply 15 psi air pressure within tank for a period of 5 minutes and check no fuel leaks occur at areas listed in Para 2.B.2).



Pump Test Rig
Figure 1





Bench Test Curves
Figure 2

3. Inspection/Check

A. Leak Check – Seal Drain Port

1) Remove access panel from under side of fuselage (refer to 12-00-00, Servicing – General Data).

2) Ensure each fuselage tank contains a minimum of 30 Imperial (36 U.S.) gallons of fuel.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Ensure relevant circuit breakers are engaged.

5) Ensure fuel levers are OFF, FUEL SELECTOR switch is set to NORMAL, and WING TANK FUEL switches (if fitted) are OFF.

6) Select forward and aft BOOST PUMP 1 switches to ON. Check all No. 1 and No. 2 boost pump amber blocks Pmp1 and Pmp2 and associated CAS messages are not displayed.

7) Check each No. 1 pump seal drain port for fuel leaks. Maximum leakage allowed 5 drops (1/4 cc) in 1 hour while pump is running. No leakage allowed on stationary pump.

8) Switch off No. 1 pumps by selecting booster pump switches to OFF.

9) Select forward and aft BOOST PUMP 2 switches to EMERGENCY. Check both No. 2 boost pump amber blocks Pmp2 are no longer displayed.

10) Repeat Para 3.A.7) for the No. 2 pumps.

11) Switch forward and aft BOOST PUMP 2 switches to OFF.

12) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

13) Install access panel to underside of fuselage (refer to 12-00-00, Servicing – General Data).

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Crossfeed Valve – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Crossfeed Valve

1) Ensure battery and/or DC power (refer to 12-00-00, Servicing – General Data, Para 2.) is disconnected, and wing fuel tank switches (if fitted) are OFF.

2) Drain fuselage fuel tanks (refer to 12-10-10, Fuel System – General Data).

3) Remove access panel from underside of fuselage (refer to 12-00-00, Servicing – General Data).

4) Disconnect electrical connector from crossfeed valve.

5) Remove four retaining bolts from rear of valve, disconnect fuel line at rear end, and remove fuel line assembly.

6) Remove valve front four retaining bolts.

7) Remove crossfeed valve from front pipe flange.

8) Cap all openings.

B. Install Crossfeed Valve

1) Position valve in sump plate recess, install front four retaining bolts and washers.

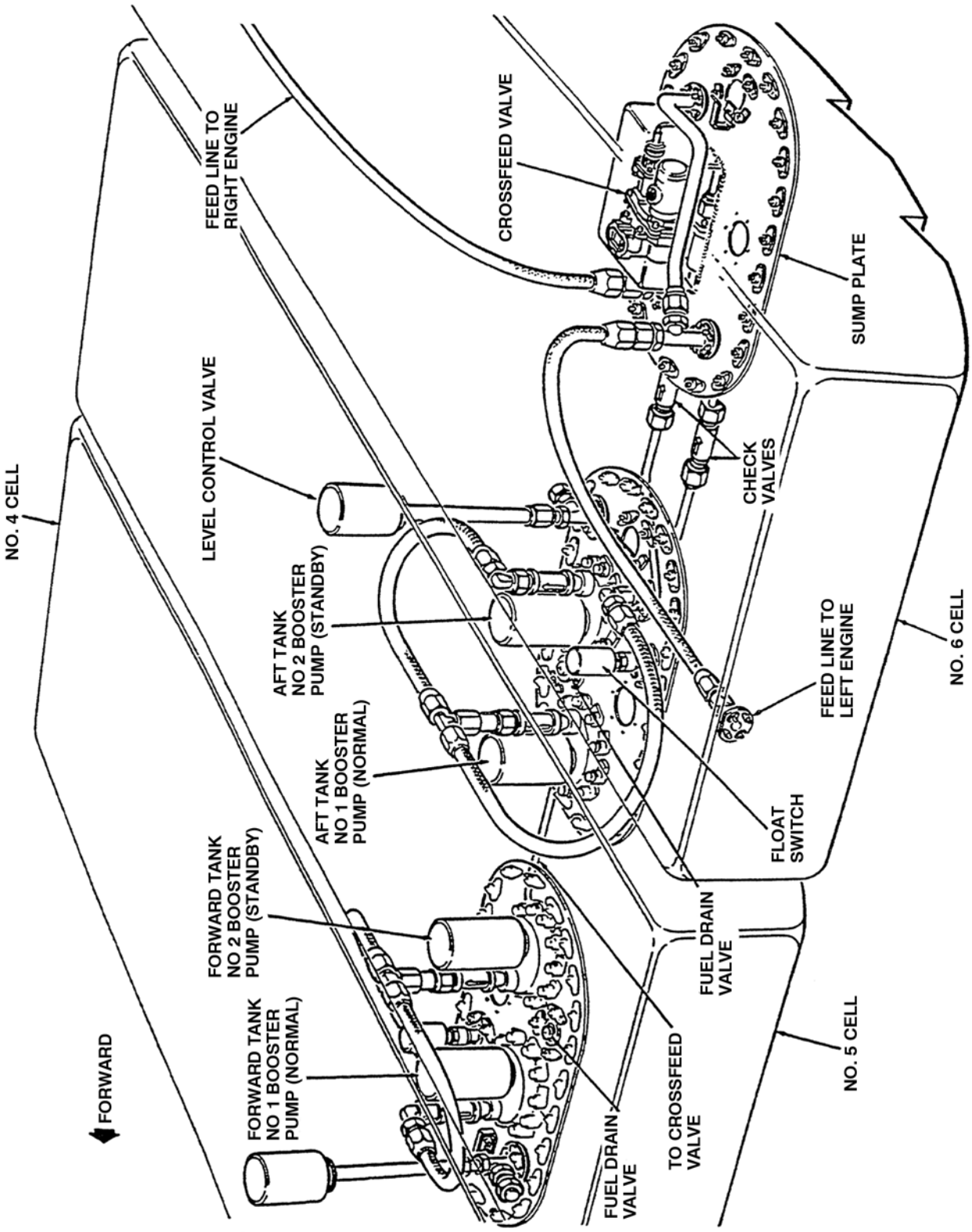
2) Install fuel line assembly, tighten line connection and valve four rear retaining bolts and washers.

3) Check for fuel leaks.

4) Connect electrical connector to valve.

5) Connect battery and/or DC power (refer to 12-00-00, Servicing – General Data, Para 2.), check operation of valve.

6) Install access panel (refer to 12-00-00, Servicing – General Data).



Booster Pump and Crossfeed Valve Installation
Figure 1

2. Adjustment/Test

A. Operational Test

1) Remove access panel from underside of fuselage (refer to 12-00-00, Servicing – General Data).

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Ensure FUEL CROSSFEED circuit breaker is in.

4) Position FUEL SELECTOR switch to BOTH ON (AFT or FWD).

5) Observe crossfeed valve open indication is presented as a green fuel pane text block “Crossfeed” on the fuel indication display pane.

6) Check red indicator arm on crossfeed valve has rotated to the open (down) position.

7) Position FUEL SELECTOR switch to NORMAL.

8) Check red indicator arm on crossfeed valve has rotated to the closed (up) position.

9) Observe crossfeed valve open indication text block “Crossfeed” on the fuel pane display is no longer displayed.

Note A “Fuel Crossfeed Fail” caution message will be displayed if the position of the crossfeed valve does not match the position of the FUEL SELECTOR switch. If the crossfeed valve caution message is active and the position of the FUEL SELECTOR switch is to BOTH ON (AFT or FWD), the text block “Crossfeed” will be displayed in amber text.

10) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

11) Install access panel to underside of fuselage (refer to 12-00-00, Servicing – General Data).

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Emergency Shut-off Valve – Maintenance Practices

1. Adjustment/Test

   A. Operational Test

      1) Release lower cowling and allow to swing down.

      2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      3) Ensure applicable FUEL SHUTOFF (L or R) circuit breaker is in.

      4) Select applicable shut-off switch on the emergency panel to FUEL OFF and observe red indicator arm on valve rotates to the closed (down) position.

      5) Select shut-off switch to NORMAL and observe red indicator arm rotates to the open (up) position.

      6) Raise and secure lower cowling.

      7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Fuel Filter (Mod 6/2006) – Maintenance Practices

1. Servicing

   A. Service Fuel Filter

      1) With electrical power on, close fuel emergency shut-off valve and ensure wing fuel tank switches (if fitted) are OFF.

      2) Switch on fuel booster pumps.

      3) Position container underneath filter and open drain valve.

      4) Ensure fuel filter is clear of water, switch off booster pumps and close drain valve.

      5) Disconnect flexible drain hose at elbow.

      6) Unscrew fuel filter element housing.

      7) Remove filter element.

      8) Inspect filter element and O-ring, discard if necessary.

      9) Clean housing with varsol.

      10) Dry components.

      11) Install O-ring to housing.

      12) Position housing, tighten and lockwire.

      13) Connect flexible drain hose at elbow.

      14) Switch on fuel booster pumps. Check for fuel leaks.

      15) Position container under filter and open drain valve.

      16) When fuel flow is free of air, close drain valve and switch off fuel booster pumps, and open fuel emergency shut-off valve.

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Wing Tank Fuel Filter (Mod 6/1398) – Maintenance Practices

1. Servicing

A. Service Fuel Filter

Refer to Figure 1 for details.

1) Ensure wing tank is empty; drain as necessary.

2) Remove wing tip cover assembly.

3) Unscrew unions and remove tube between pump and filter assembly.

4) Break lockwire at filter housing and unscrew to remove outer part of housing complete with thread reducer.

5) Remove filter element and examine for damage.

6) Clean filter by back-flushing with clean solvent and drying with clean compressed air.

Note Ultrasonic cleaning equipment may be used, if available, to clean filter element.

7) Examine O-ring on housing for serviceability and renew as necessary with new O-ring P/N 50081151000 (Aircraft Appliances and Equipment Ltd.).

8) Ensure that spring is positioned in inner part of housing and install clean filter element.

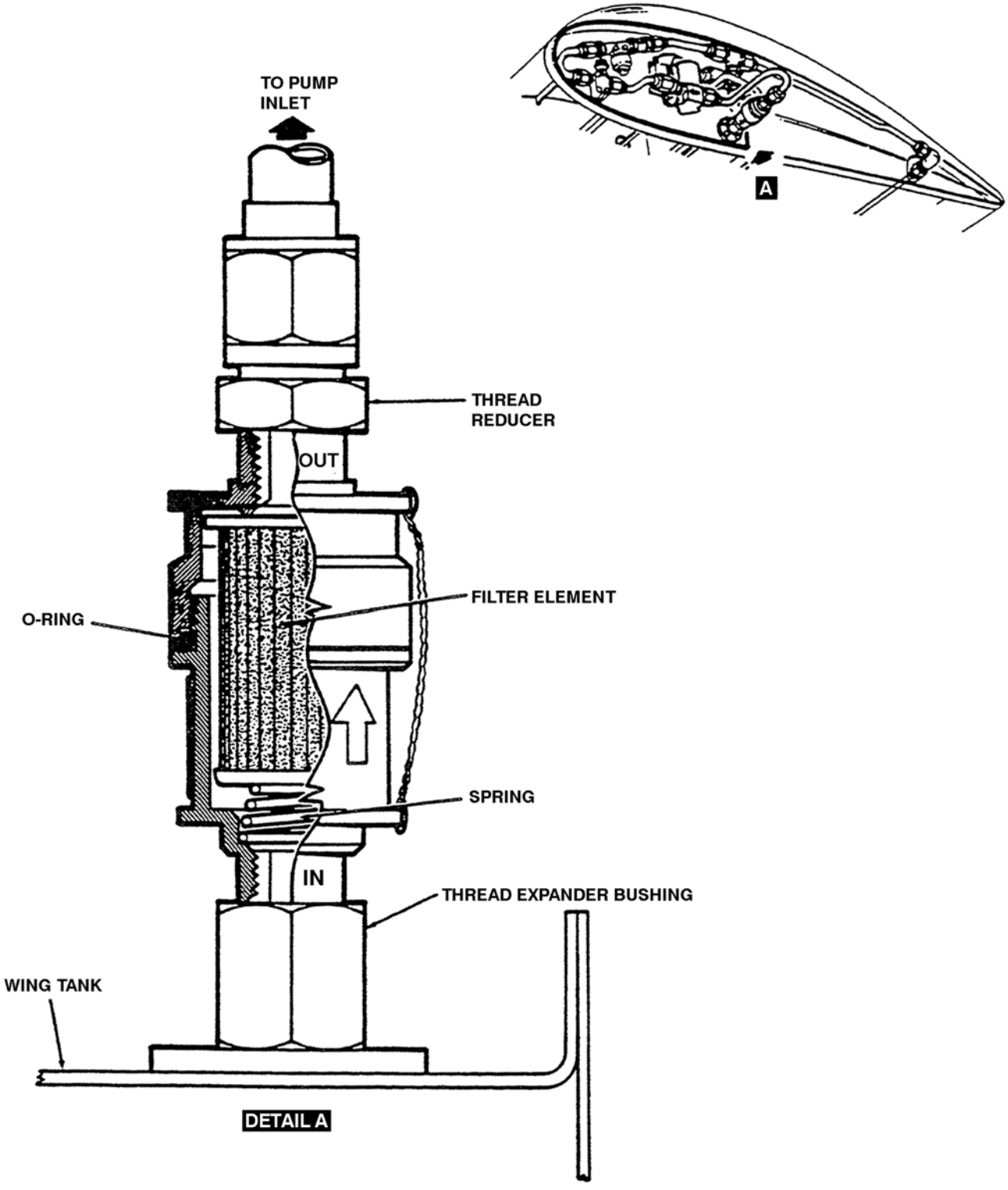
9) Screw on outer part of housing, complete with reducer, and wirelock.

10) Install tube between pump and filter assembly.

11) Refuel wing tank and select wing tank fuel pumps to ENGINE.

12) Check for leaks.

13) Install wing tip.



Wing Tank Fuel Filter
Figure 1

Motive Line Check Valve Filter – Maintenance Practices

1. Servicing

A. Service Filter

1) Ensure crossfeed valve is closed and wing fuel tank switches (if fitted) are OFF. Drain appropriate forward or aft fuselage fuel tank. (Refer to 12-10-10, Fuel System – General Data).

2) Remove access panel from underside of fuselage. (Refer to 12-00-00, Servicing – General Data).

3) Remove check valve from line.

4) Unscrew valve and cap assembly from check valve housing.

5) Remove retaining ring from housing and withdraw filter.

6) Check condition of filter element, clean with varsol and dry with filtered air.

7) Check condition of rubber seal in check valve. If damaged, complete valve and cap assembly must be replaced.

8) Install filter in housing and fit retaining ring. Ensure retaining ring is fully inserted in groove.

9) Install aluminum gasket, valve and cap assembly to check valve housing.

10) Tighten valve and cap assembly to a torque value of 600 pound–inches.

11) Install check valve in line.

12) Select fuel BOOST PUMP 1 to ON.

13) Check for fuel leaks.

14) Install access panel. (Refer to 12-00-00, Servicing – General Data).

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Fuel Level Control Valve – Maintenance Practices

1. Removal/Installation

   A. Remove Fuel Level Control Valve – in Fuselage Tank

      1) Ensure battery and/or DC power (refer to 12-00-00, Servicing – General Data, Para 2.) is disconnected, crossfeed valve closed, and wing fuel tank switches (if fitted) are OFF.

      2) Drain applicable forward or aft fuselage fuel tanks. (Refer to 12-10-00, Fuel System – General Data).

      3) Remove access panel from underside of fuselage. (Refer to 12-00-00, Servicing – General Data).

      4) Remove applicable sump plate. (Refer to 28-10-21, Fuel Cell Slump Plate – Maintenance Practices).

      5) Remove level control valve retaining bolts and washers. Remove valve and gasket from end of tube assembly.

   B. Install Fuel Level Control Valve in Fuselage Tank

      1) Position valve with new gasket on end of tube assembly.

      2) Install four retaining bolts and washers. Lockwire bolts.

      3) When installing valve to sump plates of cells 4 or 5, ensure blanking cap is fitted to tee at base of tube assembly.

      4) Install sump plate. (Refer to 28-10-21, Fuel Cell Slump Plate – Maintenance Practices).

      5) Check for fuel leaks.

      6) Install access panel. (Refer to 12-00-00, Servicing – General Data).

   C. Remove Fuel Level Control Valve in Wing Tank (Mod S.O.O. 6247)

      1) Connect battery and/or DC power (refer to 12-00-00, Servicing – General Data, Para 2.) to aircraft.

      2) Drain applicable wing fuel tank (if necessary). (Refer to 12-10-00, Fuel System – General Data).

      3) Remove wing tip.

      4) Disconnect external fuel line from level control valve elbow at tank outboard face.

      5) Remove tank filler cap and use as access to level control valve.

      6) Support level control valve inside tank and remove retaining bolts and washers from outside tank.

      7) Remove level control valve and attached elbow assembly through filler cap opening. Ensure tank to elbow gasket is removed with elbow assembly.

      8) Cut lockwire and remove elbow to level control valve retaining bolts and washers. Remove level control valve from elbow. Retain elbow.

   D. Install Fuel Level Control Valve – in Wing Tank (Mod S.O.O. 6247)

      1) Position valve with new gasket on elbow.

      2) Install four retaining bolts and washers. Lockwire bolts.

3) Position four retaining bolts and washers to protrude into tank, and working through filler cap, position new gasket over bolts inside tank.

4) Position level control valve and attached elbow inside tank and secure with retaining bolts.

5) Connect external fuel line to level control valve elbow at tank outboard face.

6) Install wing tip.

2. Adjustment/Test

A. Bench Test

Equipment required:

- Controlled air supply 0 to 125 psi
- Controlled test fluid supply (dry cleaning solvent P–D–680) 0 to 125 psi pressure and 0 to 20 gpm flow
- Suitable test fixture

1) Proof Pressure Test Valve

a) Connect air line to valve inlet port.

b) Mount valve upright in suitable test fixture, submerged in test fluid.

c) Apply air to inlet port at increasing pressures up to 125 psi. Check no external leakage occurs.

2) Leak Test Valve

a) With valve in inverted position, apply 60 psi test fluid pressure to inlet port.

b) Check for internal leakage (maximum allowable 31cc per minute).

3) Test Sensing Level of Valve

a) Mount valve upright in suitable test fixture.

b) Apply test fluid at a flow rate of one gpm to valve inlet port.

c) Check valve closes (no further fluid flow) when sensing level of fluid is 3 1/4 ± 3/16 inches above the base of the mounting flange.

4) Pressure Drop Test Valve

a) Mount valve upright on a one-inch diameter calibrated pressure tap.

b) Connect fluid source to inlet and apply test fluid at a flow rate of 20 gpm through valve.

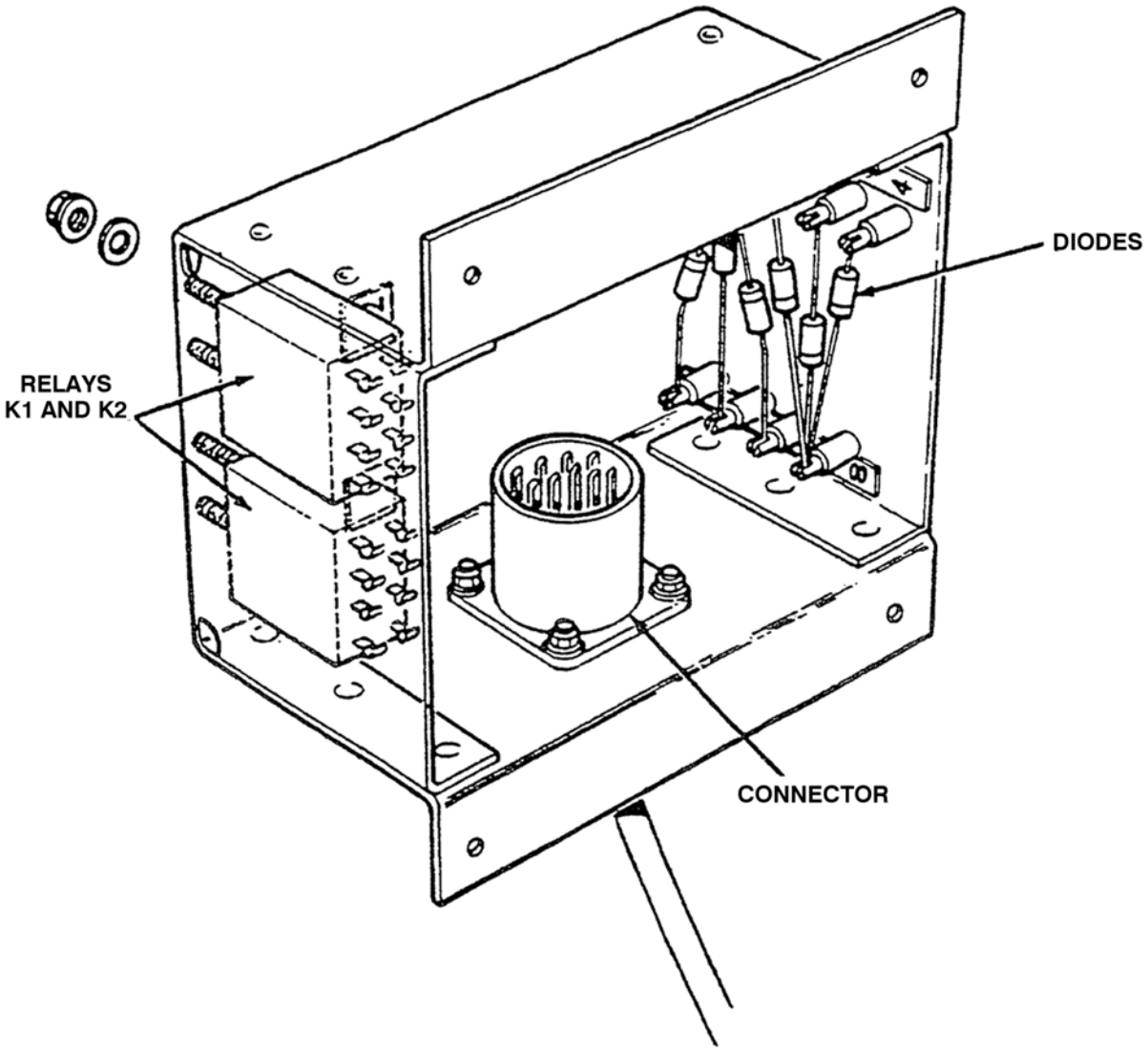
c) Check pressure drop does not exceed 2 psi.

d) Remove valve from test fixture.

Pump Change-over Control Box – General Data

1. Description

The Viking Air Limited Pump Change-Over Control Box. P/N C6NF2232 (Figure 1) forms part of the fuel control system, and consists of a rectangular aluminum alloy box which houses the two booster pump transfer relays (K1, K2) and a number of diodes. Electrical connection to the box is via a multipin connector. The control box is located just forward of the instrument panel.



Pump Change-over Control Box
Figure 1

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Indicating – General Data

1. General

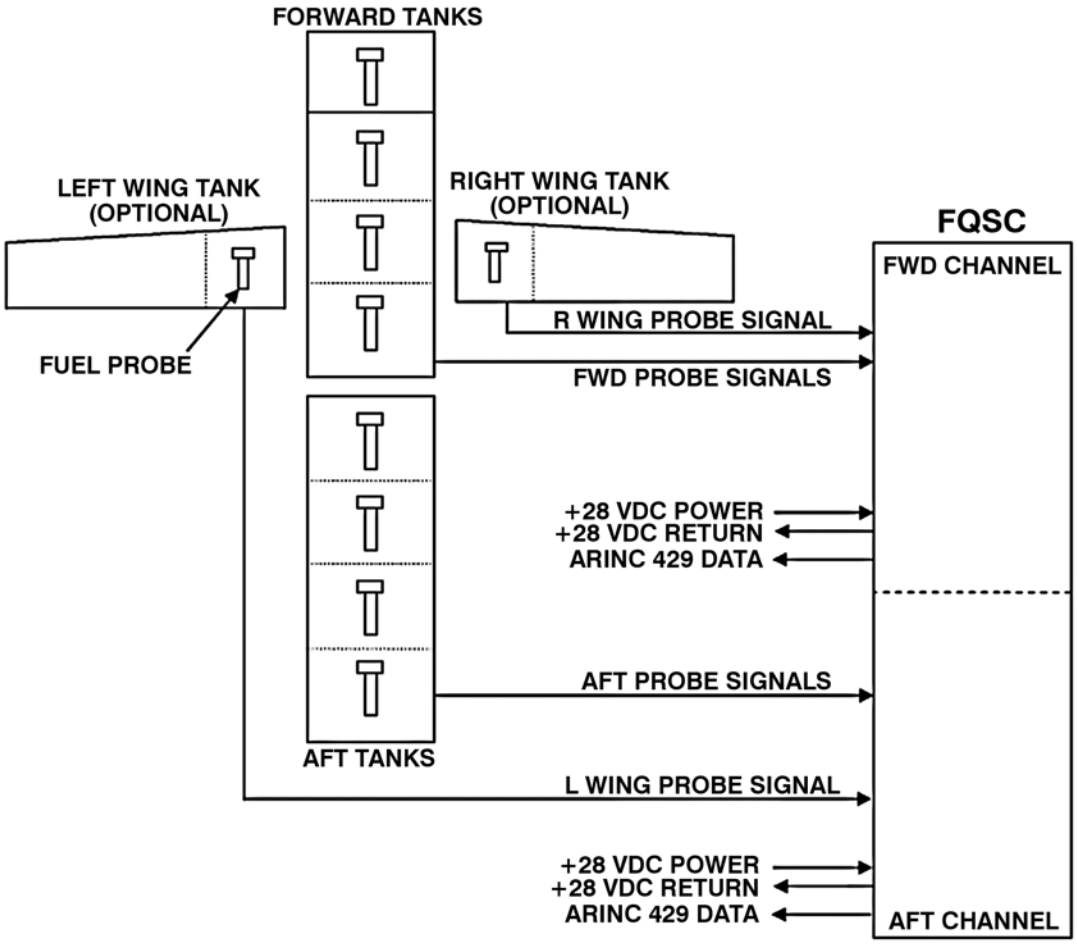
Fuel system indicating consists of four systems; a fuel quantity indicating system, a fuel low level indicating system, a fuel low pressure indicating system, and a crossfeed valve position indicating system.

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Fuel Quantity Indicating System – General Data

1. General

The Fuel Quantity Gauging System consists of 8 fuel probes in the forward and aft fuel cells, 2 optional fuel probes in wing tanks, and one Fuel Quantity Signal Conditioner (FQSC). The FQSC is a two-channel microprocessor-based unit that interfaces with all the fuel probes on the aircraft and computes fuel quantity. The signal conditioner transmits fuel quantity, fault status and diagnostic information to the cockpit display system via independent forward and aft ARINC–429 data buses. A system level block diagram is provided in Figure 1.



Fuel Quantity System
Figure 1

2. Description

The Fuel Quantity Gauging System consists of a Fuel Quantity Signal Conditioner (FQSC), Main Fuel Probes (Qty 8) and Auxiliary Wing Tank Fuel Probes (Qty 2), if installed.

A. Fuel Quantity Signal Conditioner (FQSC)

The FQSC prevents any electrical ignition sources from entering the fuel tank through the fuel system wiring. One FQSC is installed in the system. The FQSC contains protective components which prevent hazardous levels of voltage, power and stored energy from reaching the in-tank wiring and fuel probes. It also contains electrical interface circuits appropriate to produce a signal to an ARINC–429 bus.

B. Fuel Probes

The fuel probes are fluid level sensors with built-in signal conditioning electronics. The probe detects fluid quantity as a change in capacitance between two concentric tubes. A capacitance measurement circuit produces a DC current output proportional to fuel height. The active fuel probe integrates advanced signal processing electronics into the capacitance sensor. The active probe electronics serve to do an accurate measurement of the sensor capacitance and output a robust DC current signal. This signal is protected against electromagnetic interference (EMI) and lightning induced transients. The outer shell of the probe is bonded to airframe ground, making it self-shielding against electromagnetic interference. Each probe consists of two concentric tubes that form the capacitance-sensing element. At one end of the probe is a sealed housing for the electronic circuits. Electrical connections are made via a military spec circular connector. The FQSC acts as the processing unit.

The main probe's electrical connector (Pre Mod 6/2395) consists of a D38999/20WA35PN (mating D38999/26WA35SN). Post Mod 6/2390 have M85049/38S9W backshell with shield termination. The connector backshells are environmentally sealed with a dual wrap kit consisting of polyurethane tape and backfilled with two part polyurethane sealant to reduce moisture ingress.

For Post Mod 6/2395, the main probe's electrical connector consists of a stainless steel connector D38999/21 YA35PN (mating 233–205–G6–Z1–9–35–SN). Post Mod 6/2395 have 620H084BM09A backshell with shield termination. The connector backshells are backfilled with two part polyurethane (self leveling green) sealant to reduce moisture ingress.

The wing probe's electrical connector consists of a MS3449H8C33P per MIL–C–26482 Series 2 (mating connector MS3476L8–33S).

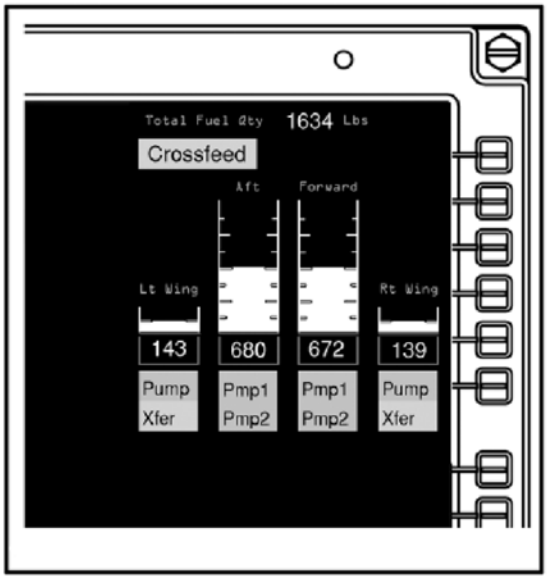
To maintain safety, the current and voltage available via in-tank wiring is limited by components in the FQSC. The probe's signal output circuits incorporate suitable protection against electrical transients and electromagnetic interference.

C. Fuel Quantity

Fuel quantity is displayed on the fuel pane window on Apex. Fuel quantity is provided for each of the aft and forward fuselage tanks, refer to Figure 2. Total fuel quantity of all fuel tanks is also displayed.

D. Wing Tip Tanks Fuel Quantity (Mod S.O.O 6247)

Fuel quantity for the wing tanks is displayed on the fuel pane window on Apex. Fuel quantity is provided for each of the left and right wing tip fuel tanks, refer to Figure 2.



Fuel Indication Display
Figure 2

3. Adjustment/Test

There are no field settable calibration features in the FQSC or the fuel tank probes. The following test is for verification that the resistors in the FQSC are functional and therefore providing the proper level of current limiting to the probes.

A. FQSC Probe Signal Inputs – Current Limit

Temporarily connect an ammeter between each of the FQSC probe signal inputs and the respective return pin, one at a time. Refer to Figure 3 for pin connections. Measure and record the short circuit current for each probe signal input. Current shall be between 5 and 7 mA. Record the findings on tables similar to Table 1 — Fwd Channel (J1) and Table 2 — Aft Channel (J2).

Note Probe 5 for each channel is the associated wing tank probe as per Figure 3. It is not necessary to check this channel on aircraft not equipped with wing tanks.

B. FQSC Probe Signal Input – Short Circuit Current

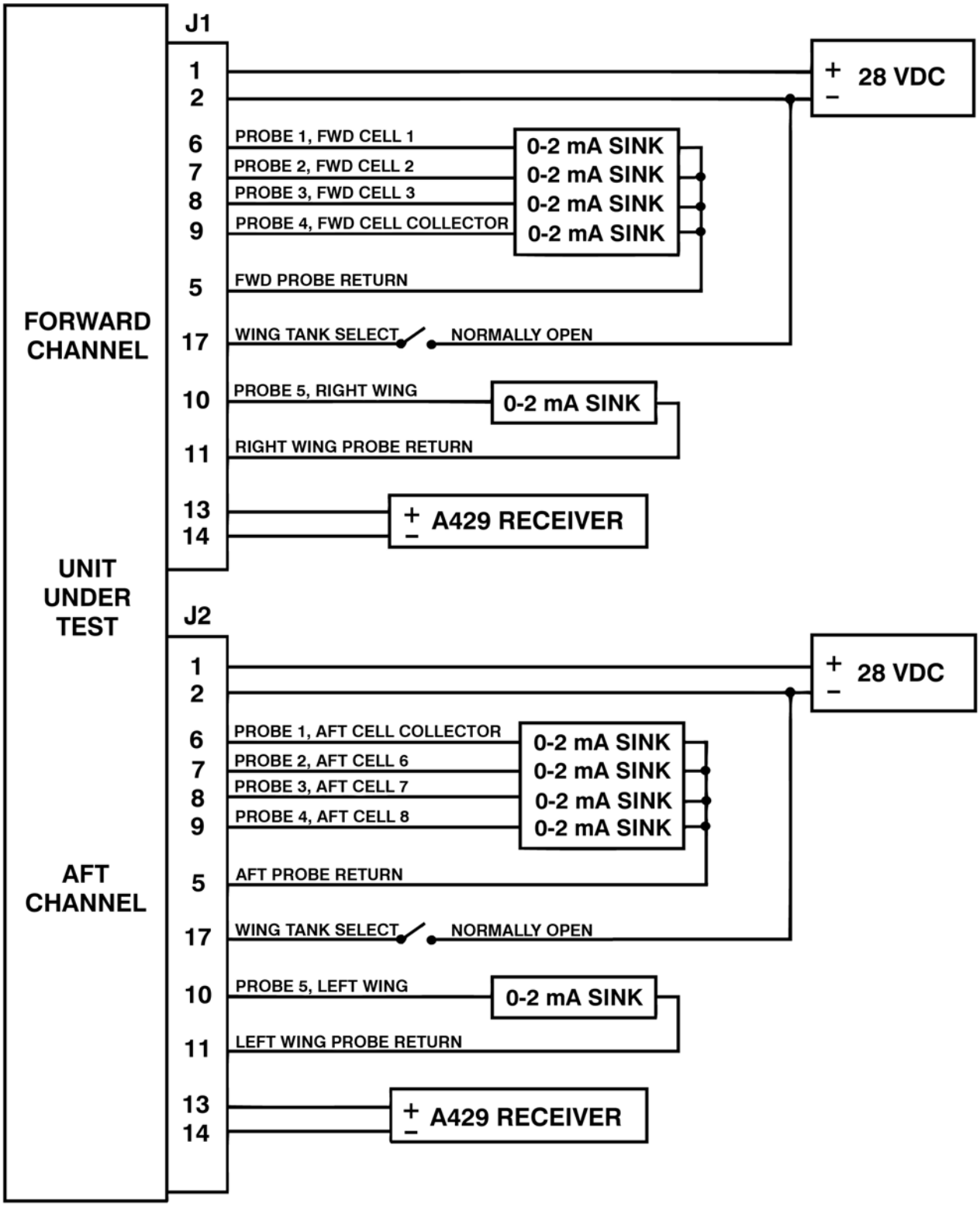
Short circuit current 5.0 < current < 7.0 mA.

Table 1 Fwd Channel (J1)

Probe 1	mA
Probe 2	mA
Probe 3	mA
Probe 4	mA
Probe 5	mA

Table 2 Aft Channel (J2)

Probe 1	mA
Probe 2	mA
Probe 3	mA
Probe 4	mA
Probe 5	mA



Test Connections
Figure 3

C. Calibration Check – Fuselage Tanks (Fuel Tanks Empty)

Note Aircraft must be reasonably level, and in the open air, during all fuel system checks. Wing tanks and components are checked separately to permit testing of fuselage tank system if wing tanks are not to be used.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Drain all usable fuel from the fuselage tanks and ensure all relevant circuit breakers engaged.

3) On the fuel control panel, verify FWD and AFT BOOST PUMP 1 switches are in the OFF position and FWD and AFT BOOST PUMP 2 switches are in the NORMAL position.

4) Fill the fuselage tanks, forward tank first in increments as listed in Table 3 — Fuel Quantity System Calibration.

5) Check indication readings are within limits shown in Table 3 — Fuel Quantity System Calibration.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Table 3 Fuel Quantity System Calibration

FUEL IN TANK (PER TENDER METER)	FUEL QUANTITY INDICATION
0	0 – 15 lbs
30 Imp (36 U.S.) gal	215 – 250 lbs
50 Imp (60 U.S.) gal	370 – 410 lbs
100 Imp (120 U.S.) gal	740 – 820 lbs
150 Imp (180 U.S.) gal	1110 – 1230 lbs
Full	1215 – 1240 lbs – Fwd 1320 – 1345 lbs – Aft

Note For usable capacities and fuel grades refer to Table 4 — Fuel Tank Capacities and Fuel Grades. Table 3 — Fuel Quantity System Calibration is based on Jet A–1 fuel density.

Table 4 Fuel Tank Capacities and Fuel Grades

FUEL	DENSITY (AVER-AGE) LB. PER	FORWARD TANK			AFT TANK		
GRADE	IMP. GAL	Weight-lb	Imp. Gal.	U.S. Gal.	Weight-lb	Imp. Gal.	U.S. Gal.
Jet A	8.18	1235.2	151.0	181.2	1341.5	164.0	197.8
Jet A–1	8.16	1232.2	151.0	181.2	1338.2	164.0	197.8
Jet B	7.80	1177.8	151.0	181.2	1279.2	164.0	197.8
JP–4	7.80	1177.8	151.0	181.2	1279.2	164.0	197.8
JP–5	8.16	1232.2	151.0	181.2	1338.2	164.0	197.8
JP–8	8.07	1218.6	151.0	181.2	1325.5	164.0	197.8

D. Calibration Check – Wing Tanks (Fuel Tanks Empty) (Mod S.O.O. 6247)

Note Aircraft must be reasonably level, and in the open air, during all fuel system checks.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Drain all usable fuel from the wing tanks and ensure all relevant circuit breakers are engaged.

3) On the fuel control panel, verify FWD and AFT BOOST PUMP 1 switches are in the OFF position and FWD and AFT BOOST PUMP 2 switches are in the NORMAL position.

4) Verify the WING TANK fuel switches are in the OFF position.

5) Fill each wing tank in increments as listed in Table 5 — Wing Tanks – Fuel Quantity System Calibration.

6) Check indication readings are within limits shown in Table 5 — Wing Tanks – Fuel Quantity System Calibration.

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Table 5 Wing Tanks – Fuel Quantity System Calibration

FUEL IN TANK (PER TENDER METER)	FUEL QUANTITY INDICATION
0	0 – 10 lbs
11 Imp (13 U.S.) gal	80 – 100 lbs
22 Imp (26 U.S.) gal	165 – 185 lbs
32 Imp (38 U.S.) gal	245 – 265 lbs
Full	285 – 305 lbs

Note For usable capacities and fuel grades refer to Table 4 — Fuel Tank Capacities and Fuel Grades. Table 5 — Wing Tanks – Fuel Quantity System Calibration is based on Jet A–1 fuel density.

4. Maintenance

A. Fuel Quantity Signal Conditioner (FQSC)

Maintenance of the FQSC is ‘on condition' only. On condition maintenance is described as follows: there are no periodic service requirements necessary to maintain continued airworthiness. No maintenance is required until the equipment does not properly do its intended function. If the FQSC is removed or replaced, verify current limiting functionality per procedures of Para 3.A and Para 3.B.

B. Fuel Probes

Maintenance of the main tank fuel probes is ‘on condition' only. On condition maintenance is described as follows: there are no periodic service requirements necessary to maintain continued airworthiness. No maintenance is required until the equipment does not properly do its intended function. For removal and installation of the fuel probes refer to Para 5.If the main tank fuel probes are removed or replaced, do calibration check per procedure of Para 3.C.

C. Wing Tip Tanks Fuel Probes (Mod S.O.O. 6247)

Maintenance of the wing tip fuel tank fuel probes is ‘on condition' only. On condition maintenance is described as follows: there are no periodic service requirements necessary to maintain continued airworthiness. No maintenance is required until the equipment does not properly do its intended function. If the wing tank fuel probes are removed or replaced, do calibration check per procedure of Para 3.D.

5. Removal/Installation

A. Removal of Fuel Probe

1) Ensure POWER SOURCE switch is OFF (Pre Mod 6/2314); or BATTERY and EXTERNAL switches are OFF (Post Mod 6/2314), crossfeed valve is closed, and wing fuel tank switches (if fitted) are OFF.

Note Use appropriate wire stripping and crimping tools for the wire size and contact removal/installation tool.

2) Pull CB151 (FUEL SHUTOFF L), CB153 (FUEL QTY L/AFT), CB 150 (FUEL SHUTOFF R), and CB152 (FUEL QTY R/FWD) circuit breakers on the main circuit breaker panel.

3) Drain fuel from forward tank if No. 1, 2, 3 and/or 4 fuel probes are to be removed. Drain fuel from aft tanks 5, 6, 7 and 8, if fuel probes are to be removed. Drain both tanks if No. 5 and/or 6 cell fuel probes are to be removed. (Refer to 12-10-10, Fuel System – General Data).

4) Gain access to fuel cells under aircraft belly by removing appropriate access panel from underside of fuselage. (Refer to 12-00-00, Servicing – General Data).

5) Disconnect required connectors P113 to P120 from fuel probe, as required.

6) Remove six (6) screws from the fuel probe to the fuel cell sump plate and remove gasket and the fuel probe.

B. Installation of Fuel Probe

1) Visually inspect the environmental sealing kit, if installed on the fuel probe connector. If damaged, replace the environmental sealing kit as per 20-50-03, Environmental Sealing (Dual Wrap Kit) – General Data.

2) Install fuel probe, gasket and secure with screws (6). Safety the screws.

3) Reconnect connectors P113 to P120 to main fuel probe(s).

4) Carry out bonding/resistance measurement between mounting screws and the closet airframe ground or airframe structure to ensure bonding resistance between two points ≤ 2.5 milliohms.

5) Reset CB151 (FUEL SHUTOFF L), CB153 (FUEL QTY L/AFT), CB150 (FUEL SHUTOFF R), and CB152 (FUEL QTY R/FWD) circuit breakers on the main circuit breaker panel.

6) Carry out adjustment test procedures as per Para 3.A to Para 3.C.

7) Re-Install the appropriate access panels from underside of fuselage.

Fuel Low Level Indicating System – General Data

1. General

Low fuel level in the fuselage tanks is indicated by CAS messages displayed on the flight display panel marked Fwd Fuel Qty Low and Aft Fuel Qty Low. Each caution message is operated by a float switch mounted on the sump plate of No. 4 cell for the forward tank and No. 5 cell sump plate for aft tank or by the fuel quantity system total for each tank. The fuel low level indication is activated when either the float switch is triggered or the fuel quantity system for each tank reaches the applicable value. With the aircraft in a level flight position and booster pumps on, the applicable tank fuel low level caution message illuminates when 75 pounds of usable fuel remain in the forward tank collector cell, or 110 pounds of usable fuel remain in the aft tank collector cell. Refer to PSM 1-64-2W for electrical schematic.

2. Adjustment/Test

A. Function Test

Note To test the operation of the fuel low level indicating system, refer to 28-00-00, Fuel (Mod 6/2006) – General Data, Para 2.B.

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Fuel Low Level Float Switch – Maintenance Practices

1. Removal/Installation

   A. Remove Low Level Float Switch

      1) Remove applicable fuel cell sump plate. (Refer to 28-10-21, Fuel Cell Sump Plate – Maintenance Practices).

      2) Loosen locking nut at base of float switch and unscrew float switch from supporting platform.

      3) Remove float switch, attached electrical leads, locking nut and O-ring from platform.

   B. Install Low Level Float Switch

      1) Insert new O-ring into platform recess.

      2) Screw locking nut on float switch base and insert electrical leads and switch into platform.

      3) Install fuel cell sump plate. (Refer to 28-10-21, Fuel Cell Sump Plate – Maintenance Practices).

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Fuel Low Pressure Indicating System – General Data

1. General

Fuel low pressure is indicated by caution messages displayed on the flight display panel, operated by fuel pressure switches mounted beneath the forward and aft fuselage tanks. On aircraft with wing tanks fitted (Mod S.O.O. 6247), additional pressure switches are provided to indicate low fuel pressure conditions in the wing tanks fuel supply lines to the engines.

A. Fuselage Fuel Low Pressure Indicating System

Low fuel pressure in the fuselage tanks is indicated by one of two methods. The first method is the Pmp1 and Pmp2 amber blocks displayed on fuel pane. These amber blocks will disappear during normal system operation, with Pmp2 being suppressed by the pump changeover control box when pump 1 is on and producing pressure. If pump 1 fails, then Pmp1 will become active and Pmp2 will remain off. The second method is through pump pressure CAS messages of Pump 1 Forward Press, Pump 1 Aft Press, Pump 2 Forward Press, and Pump 2 Aft Press. These messages are activated based on switch input that defines which pumps are being commanded on, and the associated pump pressure switch state. Each amber block and CAS message is operated seperately by a fuel pressure switch mounted below either No. 4 cell sump plate (forward tank) or below No. 5 cell sump plate (aft tank). Refer to PSM 1-64-2W for details.

B. Wing Fuel Low Pressure (PUMP FAIL) Indicating System
(Mod S.O.O. 6247)

Indication of pump failure or low fuel pressure in the engine fuel feed lines from the wing tanks is accomplished by caution messages, which can be operated when the applicable ENGINE (L or R) selection is made. Each Left Wing Fuel Pump and Right Wing Fuel Pump message (and the applicable amber fuel pane blocks) is operated by a pressure switch mounted in a four way fitting on the outboard side of the applicable wing tank, downstream from the wing tank fuel pressure pump. Incorporated in each pressure switch mounting is a bleed hole into the tank, which allows the pump to prime if it becomes vapor locked.

Refer to PSM 1-64-2W for electrical schematic for the Wing Fuel Low Pressure Indicating System.

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Fuel Low Pressure Switch (Pre Mod 6/2327) – Maintenance Practices

1. Removal/Installation

   A. Remove Fuselage Tank Fuel Low Pressure Switch

      1) Ensure battery and/or electrical power supply is disconnected and crossfeed valve closed.

      2) Drain applicable fuselage fuel tank (refer to 12-10-10, Fuel System – General Data).

      3) Remove access panel from underside of fuselage (refer to 12-00-00, Servicing – General Data).

      4) Disconnect electrical connection at switch.

      5) Remove switch from pump baseplate and cap opening.

   B. Install Fuselage Tank Fuel Low Pressure Switch

      1) Apply Titeseal #2 or #3 to first three threads of switch.

         CAUTION ENSURE THERE IS NO LUBRICANT IN SWITCH ORIFICE.

      2) Thread switch into pump baseplate.

      3) Check for fuel leaks.

      4) Install access panel. (Refer to 12-00-00, Servicing – General Data).

2. Adjustment/Test

   A. Bench Test

      Equipment required:

      - Manometer (0 to 10 psi)
      - Regulated air pressure supply (0 to 150 psi)
      - Pressure gauge (0 to 150 psi)
      - Megohmmeter
      - Continuity tester

      1) Test Insulation Resistance

         a) Set megohmmeter to 500 Volt DC output potential.

         b) Measure resistance between each terminal of pressure switch and case. Minimum allowable is 20 megohms.

      2) Leak Test Pressure Switch

         a) Connect pressure switch to pressure air supply through the 0 to 150 psi gauge.

         b) Slowly apply increasing air pressure to switch up to 60 psi.

         c) Seal off air pressure source, hold for one minute, and check that no leakage is indicated by the pressure gauge.

         d) Slowly vent off air pressure.

3) Test Operation of Pressure Switch

Refer to Figure 1 for details.

a) Connect pressure switch to pressure air supply through the manometer.

b) Connect continuity tester to terminals. Ensure continuity across terminals.

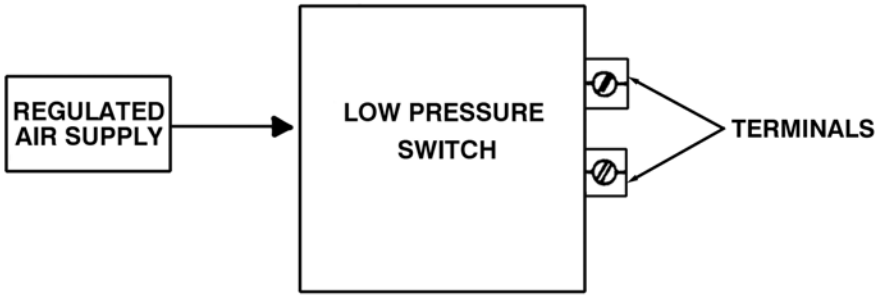
c) Slowly apply air pressure to pressure switch. Check continuity is lost at 4.0 ± 1.0 psi.

d) Continue to raise pressure to 10 psi.

e) Seal off air pressure source.

f) Slowly vent off air pressure. Check that continuity returns as the pressure decreases to 4.0 ± 1.0 psi.

g) Complete venting of air pressure and disconnect pressure switch from bench rig connections.



Low Pressure Switch Testing
Figure 1

Fuel Low Pressure Switch (Post Mod 6/2327) – General Data

1. Removal/Installation

   A. Remove Fuselage Tank Fuel Low Pressure Switch

      1) Ensure battery and/or electrical power supply is disconnected and crossfeed valve closed.

      2) Drain applicable fuselage fuel tank (refer to 12-10-10, Fuel System – General Data).

      3) Remove access panel from underside of fuselage (refer to 12-00-00, Servicing – General Data).

      4) Disconnect electrical connection at switch.

      5) Remove switch from tee (Pre Mod 6/2422) or elbow (Post Mod 6/2422) on pump baseplate and cap opening.

   B. Install Fuselage Tank Fuel Low Pressure Switch

      1) Apply Titeseal #2 or #3 to first three threads of switch.

         CAUTION ENSURE THERE IS NO LUBRICANT IN SWITCH ORIFICE.

      2) Thread switch into tee (Pre Mod 6/2422) or elbow (Post Mod 6/2422) on pump baseplate.

      3) Check for fuel leaks.

      4) Install access panel. (Refer to 12-00-00, Servicing – General Data).

2. Adjustment/Test

   A. Bench Test

      Equipment required:

      - Manometer (0 to 10 psi)
      - Regulated air pressure supply (0 to 150 psi)
      - Pressure gauge (0 to 150 psi)
      - Megohmmeter
      - Continuity tester

      1) Test Insulation Resistance

         a) Set megohmmeter to 500 Volt DC output potential.

         b) Measure resistance between each wire of pressure switch and case. Minimum allowable is 20 megohms.

      2) Leak Test Pressure Switch

         a) Connect pressure switch to pressure air supply through the 0 to 150 psi gauge.

         b) Slowly apply increasing air pressure to switch up to 60 psi.

         c) Seal off air pressure source, hold for one minute, and check that no leakage is indicated by the pressure gauge.

         d) Slowly vent off air pressure.

3) Test Operation of Pressure Switch

Refer to Figure 1for details.

a) Connect pressure switch to pressure air supply through the manometer.

b) Connect continuity tester to switch wires. Ensure continuity across wires.

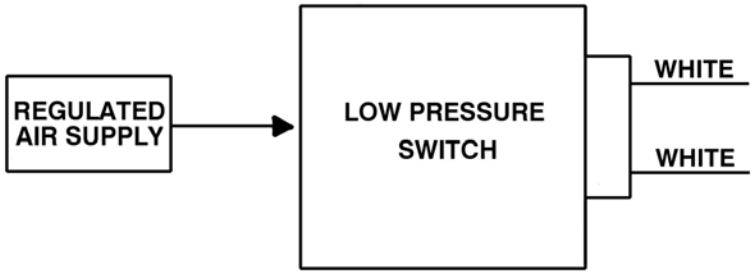
c) Slowly apply air pressure to pressure switch. Check continuity is lost at ≤ 4.5 psi.

d) Continue to raise pressure to 10 psi.

e) Seal off air pressure source.

f) Slowly vent off air pressure. Check that continuity returns as the pressure decreases to 3.5 ± 0.5 psi.

g) Complete venting of air pressure and disconnect pressure switch from bench rig connections.



Low Pressure Switch Testing
Figure 1

Fuel Low Pressure Switch (Mod 6/2006) – Maintenance Practices

1. Removal/Installation

   A. Remove Wing Tank Fuel Low Pressure Switch

      1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      2) Drain applicable wing tank (refer to 12-10-10, Fuel System – General Data).

      3) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      4) Remove wing tip.

      5) Electrically disconnect pressure switch.

      6) Unscrew pressure switch from mounting and cap opening.

   B. Install Wing Tank Fuel Low Pressure Switch

      1) Install pressure switch to mounting.

      2) Electrically connect pressure switch.

      3) Function test system (refer to 28-00-01, Fuel (With Mod S.O.O. 6247) – General Data) and check for leaks.

      4) Install wing tip.

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Crossfeed Valve Position Indicating System (Mod 6/2006) – General Data

1. General

Indication of the position of the crossfeed valve is presented as a green fuel pane text block “Crossfeed” if the valve is open.

A “Fuel Crossfeed Fail” caution message will be displayed if the position of the crossfeed valve does not match the position of the FUEL SELECTOR switch. If the crossfeed valve caution message is active and the position of the FUEL SELECTOR switch is to BOTH ON (AFT or FWD), the text block “Crossfeed” will be displayed in amber text.

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Crossfeed Valve Position Indicator – Maintenance Practices

1. Adjustment/Test

   A. Operational Test

      Note To test the operation of the crossfeed valve position indicator, refer to 28-20-21, Crossfeed Valve – Maintenance Practices.

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CHAPTER 29

HYDRAULIC POWER

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CHAPTER 29 – HYDRAULIC POWER

LIST OF EFFECTIVE PAGES

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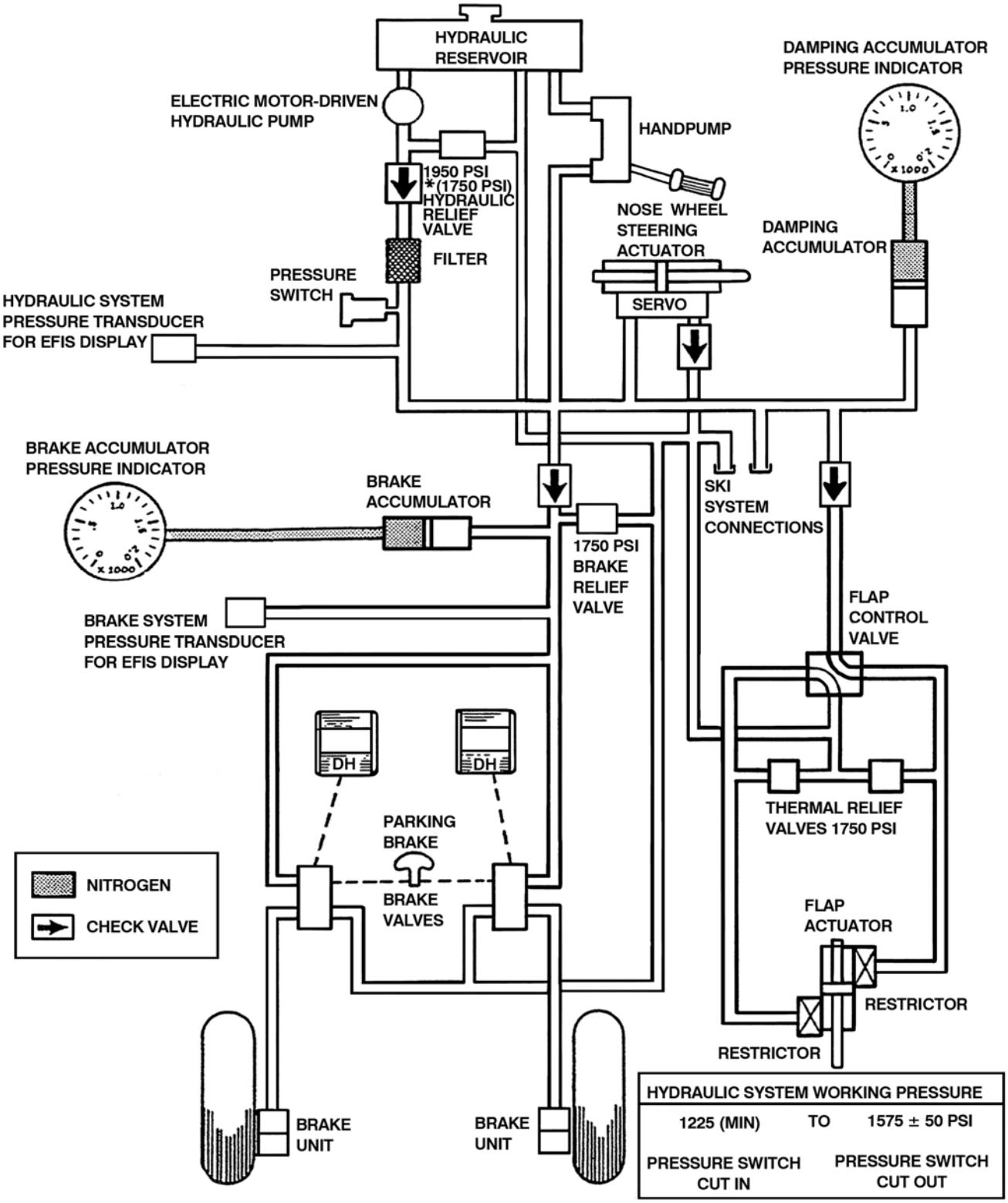
Hydraulic Power – General Data

1. General

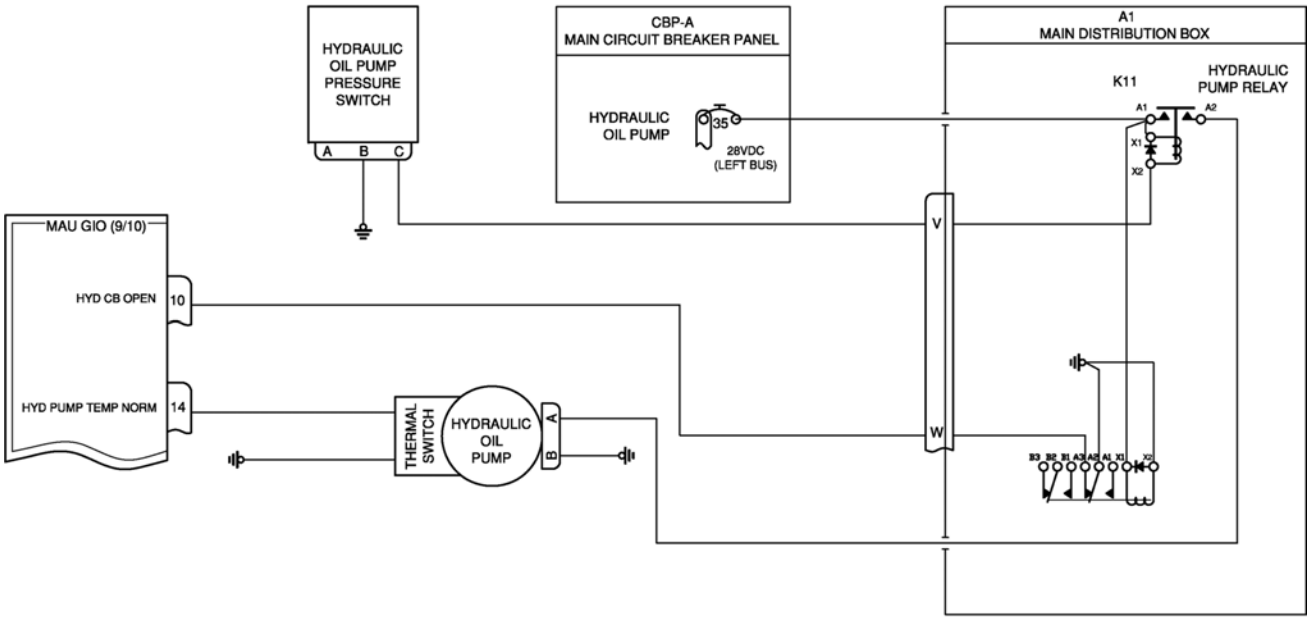
The hydraulic system supplies power to operate the wing flaps, wheel brakes and nose wheel steering. System pressure is supplied by an electric motor-driven hydraulic pump. Secondary sources of hydraulic power are: two hydraulic accumulators, one for airframe system pressure, the other for brakes system, and a hydraulic handpump. Flared pipeline couplings are used throughout the system. Refer to Figure 1 and Figure 2 for details.

Hydraulic system power is supplied by the electric motor-driven hydraulic pump which is supplied with fluid from a reservoir through a suction line, and delivers fluid through a system filter to the various hydraulic circuits. A working pressure of approximately 1575 psi is regulated by a pressure switch, which controls switching the pump motor on and off. An airframe system relief valve relieves any pressure buildup in excess of 1950 psi at full flow, and two thermal relief valves set at 1750 psi cracking pressure protect the flap circuit. A pressure relief valve, also of 1750 psi cracking pressure, is installed in the wheel brakes circuit. Fluid from all circuits is returned to the reservoir through a common return line. The airframe hydraulic system pressure and brake system pressure are indicated on the Engine Indication and Crew Alerting System (CAS) display. The main components of the hydraulic system are mounted together on a hydraulic power package below the flight compartment floor.

Electrical power to operate the electric motor-driven hydraulic pump, is derived from a 35-ampere circuit breaker and the contacts of a hydraulic pump relay, when the relay is energized. Because the relay control coil obtains a ground through the pressure switch contacts, the relay is energized only when the system hydraulic pressure falls below the working pressure value. The circuit breaker, labeled HYDRAULIC OIL PUMP, is located on the main circuit breaker panel, and the relay is mounted in the main distribution box. A hydraulic pump circuit breaker open warning is provided. A relay, mounted in the main electrical distribution box is normally energized when the HYDRAULIC OIL PUMP circuit breaker is engaged. A temperature switch on the pump housing triggers a CAS caution “Hyd Pump Over Temp”.



Hydraulic System Schematic
Figure 1



Hydraulic System Electrical – Schematic
Figure 2

2. Servicing

A. Hydraulic Lines

Removal and installation of hydraulic lines is self-evident, but the following points must be noted:

1) Before removing a pressure line, or a component in the pressure lines, discharge hydraulic pressure by operating wing flaps (for damping accumulator), or pump brake pedals (for brake accumulator).

2) Before removing an accumulator, air pressure lines, indicator or charging valve, discharge all air pressure by loosening air valve approximately one-quarter turn and waiting until all air has escaped.

WARNING **FAILURE TO PROPERLY DISCHARGE ALL AIR PRESSURE FROM AN ACCUMULATOR. MAY RESULT IN SERIOUS INJURY TO PERSONNEL AND/OR DAMAGE TO EQUIPMENT.**

3) Cap all openings on disconnection of lines. Openings must remain blanked until immediately before reconnection.

4) For torque loading procedures of fluid lines and fittings, refer to 20-20-01, Fluid Lines and Fittings – Torquing.

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Hydraulic Power Package (Mod 6/2039) – General Data

1. General

The hydraulic power package supplies hydraulic pressure to operate the wing flaps, wheel brakes, nose wheel steering systems. Refer to Figure 1 for details.

2. Description

The hydraulic power package consists of a tray containing the reservoir, brake and airframe damping accumulators, indicators and charging valves, electric motor-driven hydraulic pump with thermal relief switch, system filter, system pressure relief valve, brake pressure relief valve, system pressure switch, and associated hydraulic lines and check valves. The package is mounted on the fuselage structure beneath the flight compartment floor and is accessible from both sides of the fuselage through access doors.

A. Hydraulic Reservoir

The hydraulic reservoir is a cylindrical tank containing a filler neck, a filter screen, a cap and dipstick assembly, two suction outlets, a return inlet, and a vent. The reservoir is bolted to the power package tray. The filler neck, filter screen, and cap and dipstick assembly are accessible through the left access opening.

B. Airframe System Accumulator

The airframe system accumulator is incorporated in the hydraulic system to dampen out pressure surges in the hydraulic system and provide a secondary source of operating pressure. The accumulator is secured to the power package tray by two clamps. A combined indicator and air charging valve for the airframe system accumulator is mounted on a bracket on the left side of the power package.

C. Wheel Brakes Accumulator

The wheel brakes accumulator provides the wheel brakes with supplemental pressure in an emergency or during limited periods of peak demand, and provides pressure when the hydraulic system is shutdown while parked. The accumulator is secured to the power package tray by two clamps. A combined indicator and air charging valve for the wheel brakes accumulator is mounted on a bracket on the left side of the power package.

D. Electric Motor-Driven Hydraulic Pump

The electric motor-driven hydraulic pump is mounted on the hydraulic power package tray, and is the primary source of power for the hydraulic system. The pump consists of a motor coupled gear type pump. The motor operates on 28 Volt DC power, and is supplied from the left DC bus. A 35-ampere circuit breaker protects the circuit. The motor is equipped with a radio noise suppressor, and a thermal relief switch.

E. Hydraulic System Filter

The hydraulic system filter is a tee type containing a replaceable 10-micron element, and is installed in the hydraulic power package. Provision is made to bypass the element should a blockage or restriction cause a pressure differential of 50 ± 5 psi maximum between inlet and outlet.

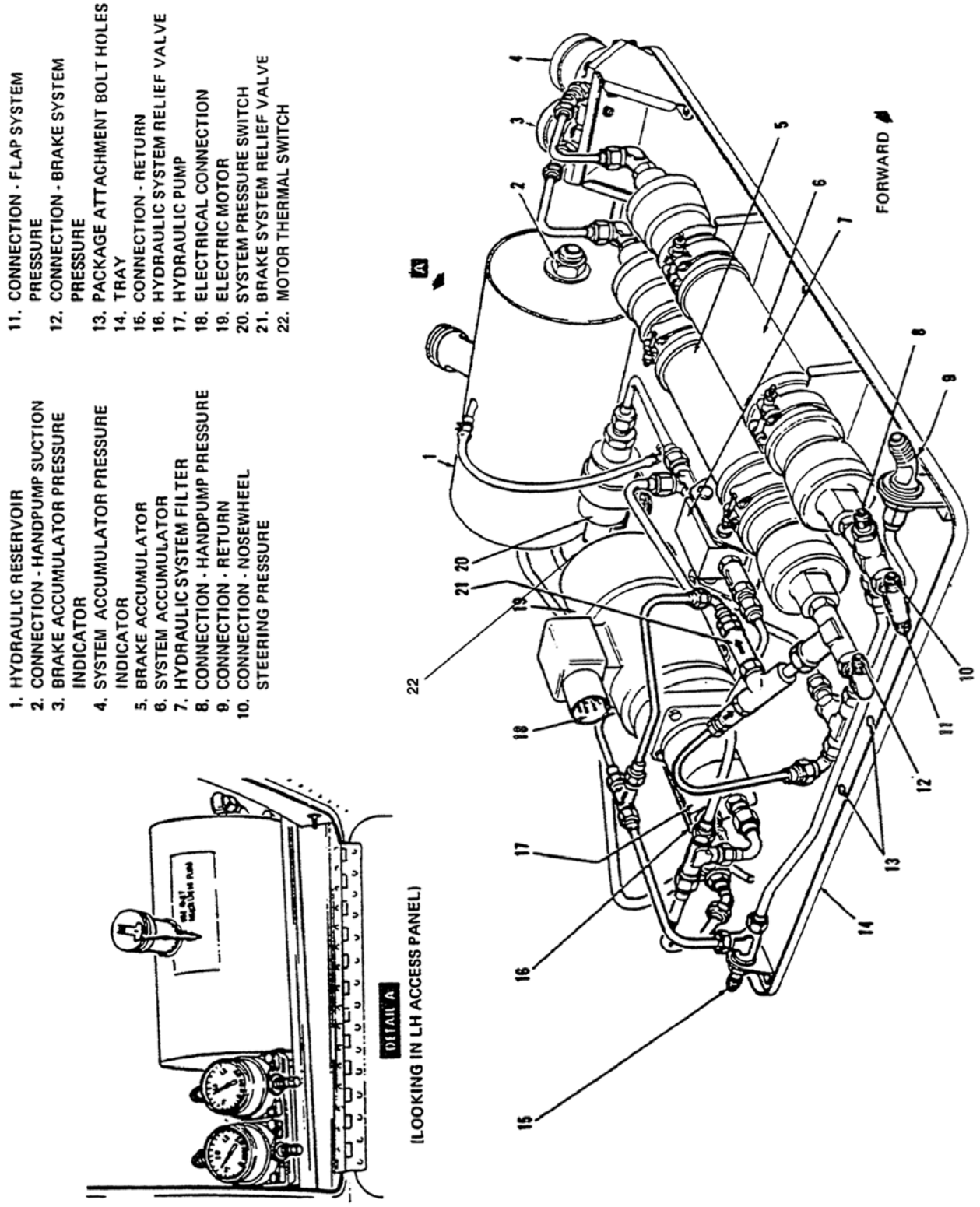
F. Pressure Switch

The pressure switch (Pre Mod 6/2313), mounted on the hydraulic power package tray, electrically controls the motor-driven hydraulic pump. It regulates the working pressure of the hydraulic system by switching off DC power to the motor when the system pressure reaches 1575 ± 50 psi and switches on DC power when the pressure drops to 1225 psi minimum.

The pressure switch (Post Mod 6/2313), mounted on the hydraulic power package tray, electrically controls the motor-driven hydraulic pump. It regulates the working pressure of the hydraulic system by switching off DC power to the motor when the system pressure reaches approximately 1575 psi and switches on DC power when the pressure drops to approximately 1225 psi minimum.

G. Pressure Relief Valve

Two pressure relief valves are installed, one for airframe hydraulic system, and the other for brake system. Both valves are located on the hydraulic power package, and are set to relieve excessive pressure buildup. The airframe hydraulic system relief valve relieves pressure buildup in excess of 1950 psi at full flow. The brake system relief valve is set to crack at a pressure of 1750 psi.



Hydraulic Power Package
Figure 1

3. Removal/Installation

Refer to Figure 2 for details.

A. Remove Hydraulic Power Package

1) Discharge airframe system accumulator hydraulic pressure by operating wing flaps, and wheel brakes accumulator hydraulic pressure by pumping brake pedals.

2) Open left and right-hand access panels on underside of nose section (refer to 12-00-00, Servicing – General Data).

3) Ensure electric power supply is OFF. Disconnect electrical connectors at hydraulic pump motor and pressure switch.

4) Discharge all air pressure from airframe system accumulator and wheel brakes accumulator by loosening each air valve approximately one-quarter turn, and wait until all air has escaped. Check indicators of both accumulators read zero.

WARNING FAILURE TO PROPERLY DISCHARGE ALL AIR PRESSURE FROM AN ACCUMULATOR MAY RESULT IN INJURY TO PERSONNEL AND/OR DAMAGE TO EQUIPMENT.

5) Place container to receive spillage. Disconnect return lines at elbow (on forward bracket) and tee (on aft bracket) on right-hand side of hydraulic power package tray.

6) Disconnect pressure lines to nose wheel steering, wing flaps, hand pump, and wheel brakes.

7) Disconnect hand pump suction line at reservoir.

8) Remove seven screws and washers attaching tray to structure, and remove power package through left-hand access panel.

9) Cap all open lines or fittings as required.

B. Install Hydraulic Power Package

1) Position power package and secure tray to aircraft structure with seven screws and washers.

2) Connect handpump suction line at reservoir.

3) Connect wheel brake, handpump, wing flaps, and nose wheel steering pressure lines.

4) Connect return lines at elbow (on forward bracket) and tee (on aft bracket) on right-hand side of power package tray.

5) Connect hydraulic pump motor and pressure switch electrical connectors.

6) Refill reservoir with hydraulic fluid, Specification MIL–H–5606.

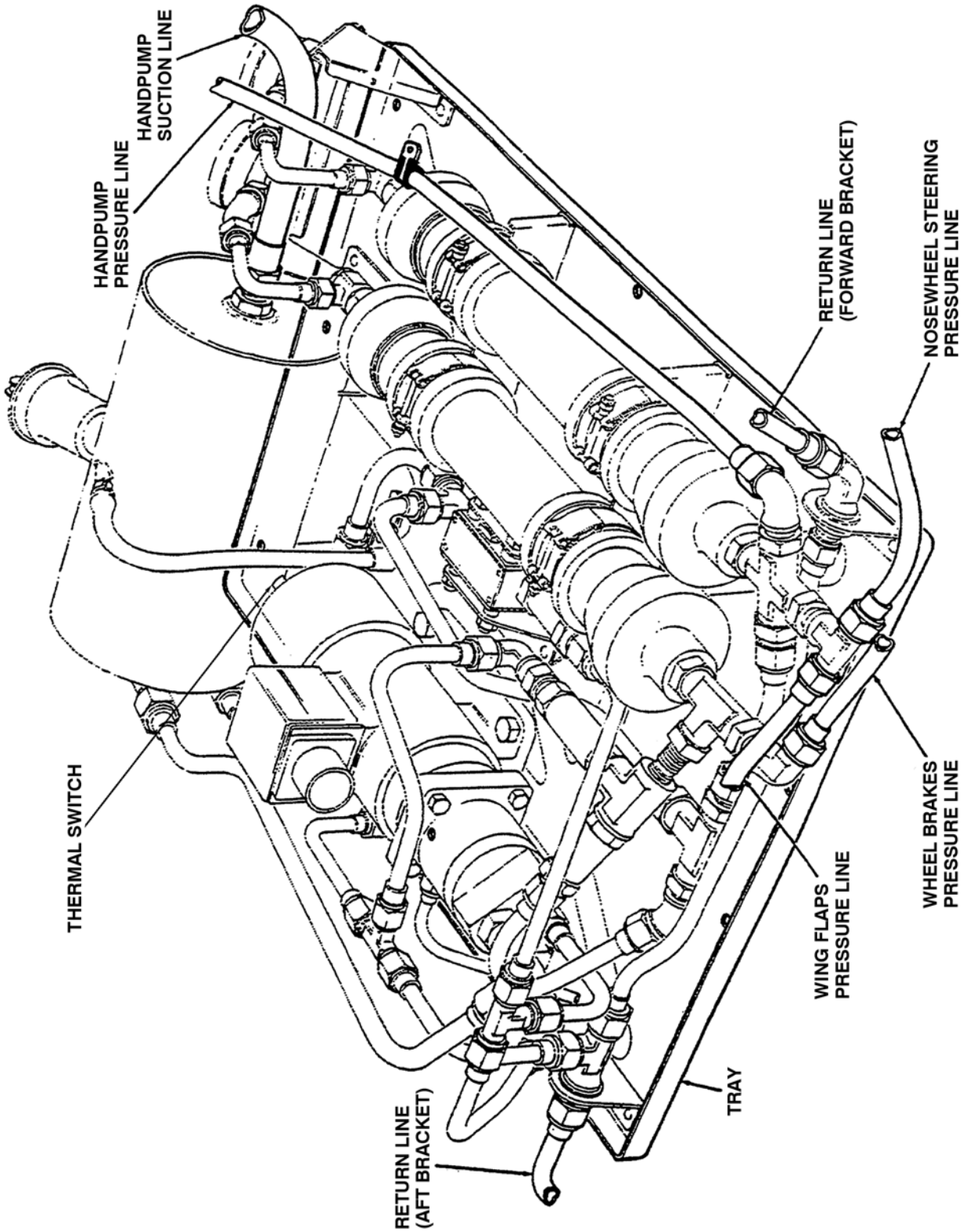
7) Recharge airframe system accumulator and wheel brake accumulator with dry nitrogen to 750 ± 50 psi.

8) Carry out function test of hydraulic power package (refer to Para 4.B). Bleed handpump (refer to 29-20-11, Hydraulic Handpump (Mod 6/2093) – Maintenance Practices).

9) Bleed nose wheel steering, wing flaps, and wheel brakes systems (refer to 27-50-11, Flap Actuator – Maintenance Practices, 32-40-00, Wheels and Brakes (Mod 6/2002) – General Data and 32-50-00, Nose Wheel Steering System – General Data).

10) Check all joints in hydraulic power package for leaks. Retighten as necessary.

11) Close left and right-hand access panels (refer to 12-00-00, Servicing – General Data).



Hydraulic Power Package Installation
Figure 2

4. Adjustment/Test

A. Function Test Hydraulic Power Package (Pre Mod 6/2313)

1) Open left and right-hand access panels on underside of nose section (refer to 12-00-00, Servicing – General Data).

2) Discharge airframe hydraulic system pressure by operating wing flaps, and brake system hydraulic pressure by pumping brake pedals. Check indicators on both accumulators read 750 ± 50 psi. Recharge accumulators with dry nitrogen if necessary.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Check nose wheel steering lever corresponds with position of nose wheel, wing flaps selector with position of wing flaps, and set parking brake handle to park.

4) Check that electric motor-driven hydraulic pump charges airframe system accumulator and wheel brakes accumulator, and that motor cuts out when pressure reading on indicator is 1575 ± 50 psi.

5) Operate wing flaps. Check that motor cuts in when reading on airframe hydraulic system pressure indicator drops to 150 – 300 psi below initial setting.

6) Pump brake pedals. Check that motor cuts in when reading on brake system pressure indicator drops to 150 – 300 psi below initial setting. Reset parking brake handle to park.

7) Check all joints on hydraulic power package for leaks. Retighten connections if necessary.

8) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

9) Close left and right-hand access panels (refer to 12-00-00, Servicing – General Data).

B. Function Test Hydraulic Power Package (Post Mod 6/2313)

1) Open left and right-hand access panels on underside of nose section (refer to 12-00-00, Servicing – General Data).

2) Discharge airframe hydraulic system pressure by operating wing flaps, and brake system hydraulic pressure by pumping brake pedals. Check indicators on both accumulators read 750 ± 50 psi. Recharge accumulators with dry nitrogen if necessary.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Check nose wheel steering lever corresponds with position of nose wheel, wing flaps selector with position of wing flaps, and set parking brake handle to park.

4) Check that electric motor-driven hydraulic pump charges airframe system accumulator and wheel brakes accumulator, and that motor cuts out when pressure reading on indicator reaches <1700 psi.

5) Operate wing flaps. Check that motor cuts in when reading on airframe hydraulic system pressure indicator drops to 150 – 350 psi below initial setting and >1225 psi.

6) Pump brake pedals. Check that motor cuts in when reading on brake system pressure indicator drops to 150 – 350 psi below initial setting and >1225 psi. Reset parking brake handle to park.

7) Check all joints on hydraulic power package for leaks. Retighten connections if necessary.

8) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

9) Close left and right-hand access panels (refer to 12-00-00, Servicing – General Data).

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Hydraulic System Filter – Maintenance Practices

1. Servicing

A. Replace Filter Element

1) Discharge airframe system accumulator pressure by operating wing flaps.

2) Open access panel on underside of nose section. (Refer to 12-00-00, Servicing – General Data).

3) Cut lockwire, unscrew and remove filter bowl complete with element from underside of power package tray.

4) Drain filter bowl, wash with a cleaning solvent, and dry with a filtered air supply.

5) Check gasket and backup ring, and replace gasket if flattened or chipped.

6) Fill filter bowl with clean hydraulic fluid, Specification MIL–H–5606, and install bowl complete with new element, serviceable gasket and backup ring.

7) Lockwire filter bowl to tray.

8) Close access panel. (Refer to 12-00-00, Servicing – General Data).

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Damping Accumulator – Maintenance Practices

1. Removal/Installation

   A. Remove Airframe System Accumulator

      1) Discharge airframe system accumulator hydraulic pressure by operating wing flaps.

      2) Open left and right-hand access panels on underside of nose section. (Refer to 12-00-00, Servicing – General Data).

      3) Discharge all air pressure from airframe system accumulator by loosening air valve approximately one-quarter turn and waiting until all air has escaped. Check indicator on accumulator reads zero.

         WARNING FAILURE TO PROPERLY DISCHARGE ALL AIR PRESSURE FROM AN ACCUMULATOR MAY RESULT IN INJURY TO PERSONNEL AND/OR DAMAGE TO EQUIPMENT.

      4) Disconnect air line at head of accumulator. Cap and plug open connection and line.

      5) Place container to receive spillage. Remove nuts and release two clamps, and unscrew accumulator with union from crosspiece.

   B. Install Airframe System Accumulator

      1) If new accumulator is to be installed, transfer fitting from unserviceable accumulator and use new O-rings.

      2) Install new O-ring at crosspiece, and screw accumulator union into crosspiece.

      3) Attach accumulator to tray with two clamps and nuts.

         CAUTION OVERTIGHTENING OF ACCUMULATOR MOUNTING CLAMPS COULD RESULT IN DAMAGE TO ACCUMULATOR.

      4) Overcome run down torque of clamp nuts then add 15 – 20 pound–inches torque maximum to seat nut.

      5) Connect air line at head of accumulator.

      6) Charge accumulator with dry nitrogen to 750 ± 50 psi.

      7) Check reservoir fluid level. Function test hydraulic power package. (Refer to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data).

      8) Bleed hydraulic handpump. (Refer to 29-20-11, Hydraulic Handpump (Mod 6/2093) – Maintenance Practices).

      9) Bleed nose wheel steering and wing flaps systems. (Refer to 32-50-00, Nose Wheel Steering System – General Data and 27-50-00, Wing Flaps System – General Data).

      10) Close left and right-hand access panels. (Refer to 12-00-00, Servicing – General Data).

2. Adjustment/Test

   Equipment required:

   - Manometer or U tube
   - Suitable hydraulic test rig
   - Suitable water container

A. Bench Test (Non Destructive) Airframe System Accumulator

1) Remove accumulator from hydraulic power package (refer to Para 1.A).

2) Remove fittings from hydraulic and gas end caps.

3) Inspect accumulator for obvious damage.

4) Do a fluorescent penetrant inspection on hydraulic and gas end caps. Check for imperfections.

5) Do a magnetic particle inspection on shell. Check for imperfections.

6) Operate piston two times in each direction using hydraulic fluid, nitrogen gas or dry air, to a maximum pressure of 50 psi.

7) Check that piston moves through entire length of shell without seizing or binding.

Note Do not operate piston manually.

8) With the piston in the balanced position (at least one inch from each end cap), fill hydraulic end and air end with fluid and plug one end. Attach a suitable pressure indicator to other end.

9) Ensure all air is removed from accumulator and lines.

10) Apply 2250 psi hydraulic pressure to the accumulator and hold for three minutes. Check that no external leakage or damage exists.

11) Release pressure and remove fluid from accumulator.

12) Connect a nitrogen or dry air source to gas end cap port. Slowly apply pressure until piston bottoms lightly on hydraulic end cap.

13) Connect port of hydraulic end cap to manometer or one limb of U tube.

14) Apply gas pressure of 2000 ± 100 psi to gas end cap port, and hold for minimum of three minutes. Internal air leakage must not exceed 0.10 milliliters.

15) Reduce pressure to 200 ± 25 psi, and hold for three minutes. Internal air leakage must not exceed 0.10 milliliters. Release pressure.

16) Connect nitrogen or dry air source to both gas and hydraulic end cap ports.

17) Prepare to submerge accumulator in water.

CAUTION **EXERCISE CARE TO PREVENT WATER FROM ENTERING AND MAKING CONTACT WITH INTERNAL SURFACES OF ACCUMULATOR BY APPLYING GAS PRESSURE PRIOR TO SUBMERSION, AND MAINTAINING UNTIL AFTER WITHDRAWAL FROM WATER.**

18) Apply gas pressure of 3000 ± 100 psi to hydraulic and gas end cap ports simultaneously and submerge accumulator in water. Hold pressure for five minutes.

19) Check that no external leakage exists.

20) Reduce pressure to 200 ± 25 psi and hold for five minutes.

21) Check that no external leakage exists.

22) Remove accumulator from water and slowly release pressure.

23) If required for service, dry, fill with hydraulic fluid (MIL–H–5606) and reinstall in hydraulic power package. If not required for immediate use, fill with hydraulic fluid (MIL–H–6083 Type 1), and cap ports.

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Wheel Brake Accumulator – Maintenance Practices

1. Adjustment/Test

A. Bench Test (Non Destructive) Wheel Brakes Accumulator

Carry out test as shown in 29-10-21, Damping Accumulator – Maintenance Practices, Para 2.A.

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Hydraulic Pressure Switch – Maintenance Practices

1. Removal/Installation

   A. Remove Hydraulic Pressure Switch

      1) Remove hydraulic power package (refer to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data).

      2) Disconnect hydraulic line from pressure switch.

      3) Remove and retain nut, bolt, washer and spacer securing pressure switch clamp to hydraulic package tray. Remove pressure switch.

      4) Cap open hydraulic line and pressure switch inlet.

   B. Install Hydraulic Pressure Switch

      1) Check replacement pressure switch is of the correct type.

      2) Examine pressure switch for signs of damage and corrosion.

      3) Install pressure switch on hydraulic package tray and secure with clamp, bolt, spacer, washer and nut.

      4) Remove blanking caps, connect and secure hydraulic line to pressure switch.

      5) Install hydraulic power package (refer to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data).

2. Adjustment/Test

   A. Operational Test Hydraulic Pressure Switch (Pre Mod 6/2313)

      1) With the HYDRAULIC OIL PUMP circuit breaker engaged, apply external power (refer to 12-00-00, Servicing – General Data).

      2) Check hydraulic pump operates, and when hydraulic pressure reaches 1575 ± 50 psi (as registered on hydraulic system indicator), that pump stops.

      3) Operate brake pedals and check hydraulic pump cuts in when hydraulic pressure falls to 150 – 300 psi below the cut out pressure obtained in Para 2.A.2).

      4) Repeat Para 2.A.2) and Para 2.A.3) to recheck cut in and cut out pressures.

   B. Operational Test Hydraulic Pressure Switch (Post Mod 6/2313)

      1) With the HYDRAULIC OIL PUMP circuit breaker engaged, apply external power (refer to 12-00-00, Servicing – General Data).

      2) Check hydraulic pump operates, and when hydraulic pressure reaches <1700 psi (as registered on hydraulic system indicator), that pump stops.

      3) Operate brake pedals and check hydraulic pump cuts in when hydraulic pressure falls to 150 – 350 psi below the cut out pressure obtained in Para 2.B.2) and pump restarts at >1225 psi.

      4) Repeat Para 2.B.2) and Para 2.B.3) to recheck cut in and cut out pressures.

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Electric Motor-Driven Hydraulic Pump – Maintenance Practices

1. Adjustment/Test

Equipment required:

- Suitable Test Rig with components as shown in Figure 1 filled with clean fluid (MIL–H–5606 or MIL–H–6083, Type 1)

A. Bench Test Electric Motor-Driven Hydraulic Pump

Refer to Figure 1 for details.

1) Remove hydraulic power package. (Refer to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data).

2) Remove hydraulic pump with motor from hydraulic power package. (Refer to Para 2.A).

3) Check for obvious damage.

4) Check electric motor brush wear (3/16 inch minimum allowable remaining brush length).

5) Connect pump into circuit of test rig. Secure to test bench.

6) Connect 18 to 28 Volt DC power supply and switch power on.

7) Bleed all air from pump.

8) Close valve B slowly until 1500 psi pressure is obtained on the discharge pressure indicator. Regulate voltage supply to 27 ± 0.5 volts.

9) Run pump for 6 minutes. Check the following:

a) Leakage from seal drain port does not exceed 0.5 cc.

b) Current draw does not exceed 33 amperes.

c) Discharge flow is 0.42 Imp (0.50 U.S.) gpm minimum.

10) Open valve B until pressure reads 1000 psi.

11) Slowly close valve A until suction line depression reads – 1.5 inches Hg. Regulate supply voltage to 18 + 0.5 or – 0 volts. Switch off electrical supply.

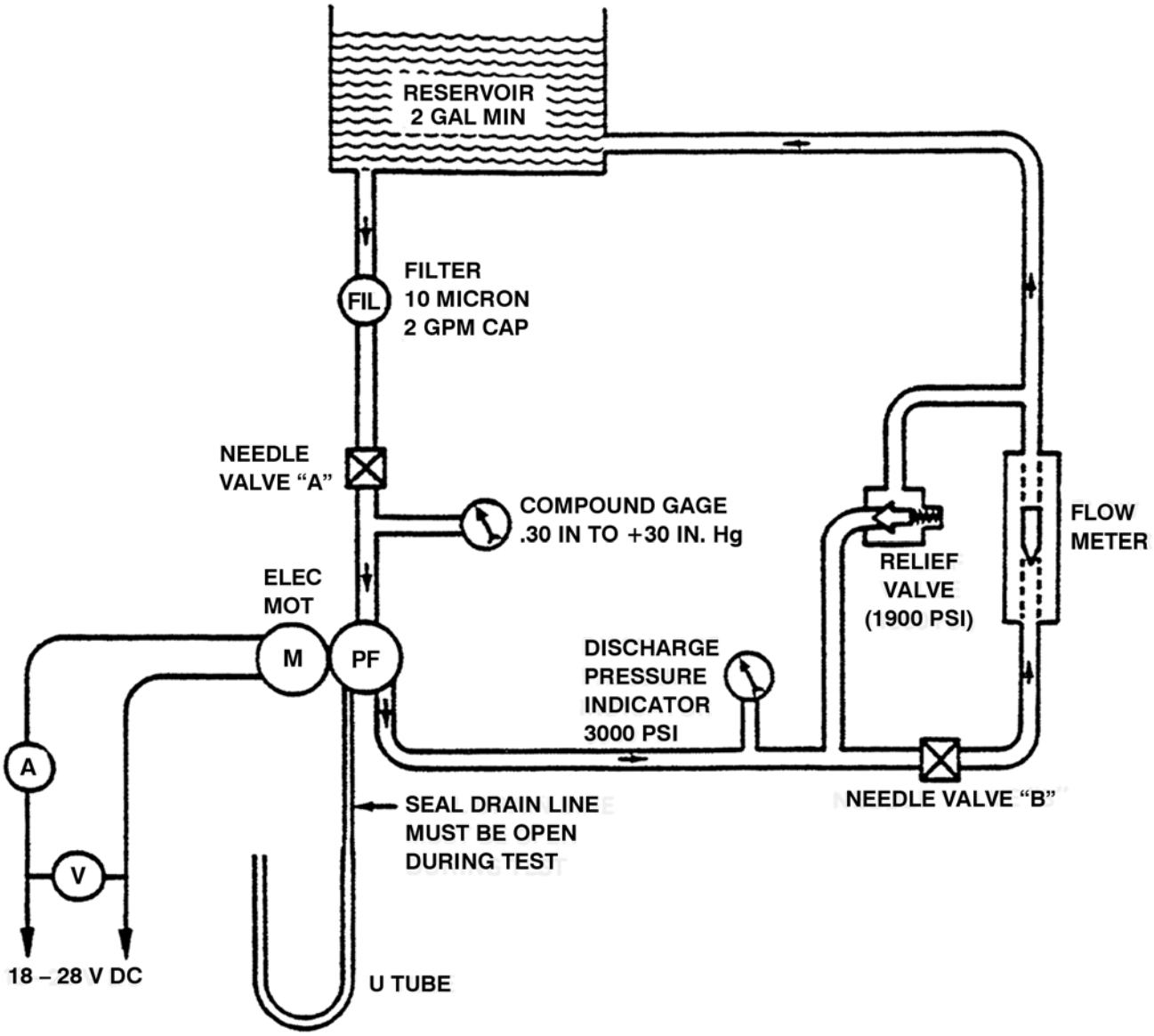
12) Allow pump to stand for 2 minutes.

13) Without altering any settings, switch on electrical supply. Check that pump discharge pressure rapidly builds up to 1000 psi.

14) Switch off electrical supply. Open valve B and release pressure.

15) Remove pump from test rig.

16) If required for service, reinstall pump on hydraulic power package. If to be placed in storage, fill pump with inhibiting fluid (MIL–H–6083 Type 1), and fit dust plugs.



Suggested Testing Rig
Figure 1

2. Removal/Installation

A. Removal of Hydraulic Pump and Motor

Refer to 29-00-00, Hydraulic Power – General Data.

1) Remove both inlet and outlet hydraulic lines from pump.

2) Remove the four AN4 bolts.

3) Lift pump motor clear of power package.

4) Clean any excess hydraulic fluid off the power package pallet.

5) Disconnect the Pressure Switch on the power package.

6) Disconnect the MS power connector from the noise filter of motor.

7) Disconnect the Thermal Switch connector.

B. Installation of Hydraulic Pump and Motor

Instruction to install the hydraulic pump and motor and connection of Thermal Switch:

1) Set the hydraulic pump and motor in place on the power package pallet.

2) Install the four AN4 bolts, washer and nuts, finger tight.

3) Connect both inlet and outlet lines to pump.

4) Torque the four AN4 bolts to 35 – 40 in–lbs.

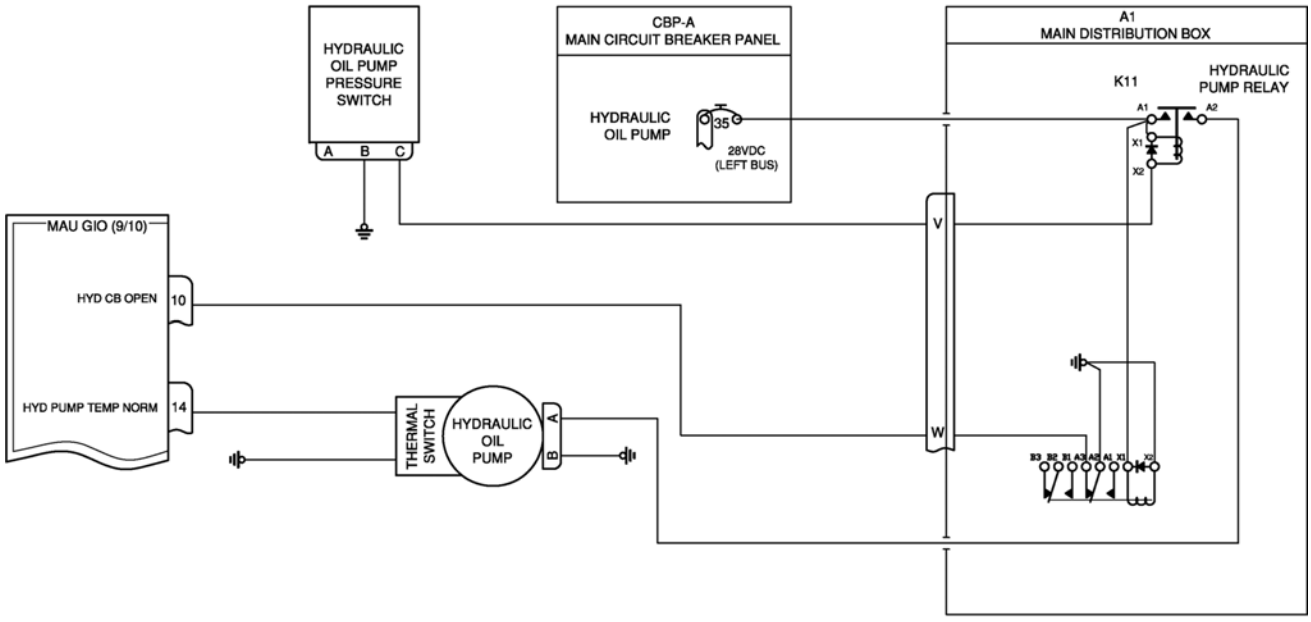
5) Connect Pressure Switch on the power package.

6) Connect the MS power connector to the noise filter of motor.

7) Connect the Thermal Switch connector.

8) Refer to Figure 2 for wiring reference.

The Thermal Switch is mounted to the motor near the shaft bearing area and senses the temperature of the motor cap end. The Thermal Switch is a normally closed switch. When the temperature reaches 180°F ± 5°F, the switch will open and provide an indication in CAS. When the temperature reaches 165°F ± 5°F, the switch will close and the indication will be extinguished.



Hydraulic System Electrical
Figure 2

C. Troubleshooting

1) Do a function test according to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data.

2) If the motor fails to run during the test:

   a) Remove the motor from the power package and test per Para 1.A.

   b) If the motor runs on the above test then the motor is functioning properly.

   c) If the motor fails to run on above test, remove and replace.

3) If the motor runs and the pump does not produce the required pressure:

   a) Remove the pump and motor assembly and test per Para 1.A.

   b) If the pump produces the required pressure and flow, the pump is working properly.

   c) If the pump does not produce the required pressure and flow, remove and replace.

4) The thermal switch is tested by reading the resistance across terminals C of the two MS connectors. If the reading is more than 1.0 ohms remove and replace.

Hydraulic System Pressure Relief Valve – Maintenance Practices

1. Adjustment/Test

Equipment required:

- Hydraulic test rig using hydraulic fluid (MIL–H–5606) suitably fitted with electric motor-driven hydraulic pump
- Shut-off and metering valves
- Gauge
- Flowmeter
- Graduated cylinder which, in conjunction with the unit being tested, will reproduce all conditions required during the test

A. Bench Test Airframe Hydraulic System Pressure Relief Valve

1) Relieve airframe hydraulic system pressure, by cycling flaps.

2) Remove hydraulic system relief valve from power pack.

3) Install airframe hydraulic relief valve in test rig.

4) Apply 1950 ± 25 psi hydraulic pressure to valve inlet and check that rated flow through valve is 2.10 Imp (2.5 U.S.) gal/min.

5) Reduce inlet pressure to 1680 psi. Check that leakage through valve does not exceed 4.5 cc/min.

Note Leakage to be checked during third minute of a three minute waiting period.

6) Slowly reduce inlet pressure. At pressures of 1460 psi and 975 psi, check that leakage rate through valve does not exceed 1.5 cc/min. Refer to Note in Para 1.A.5).

7) Slowly increase inlet pressure to 1460 psi. Repeat leakage check given in Para 1.A.6).

8) Increase inlet pressure to 1680 psi. Repeat leakage check given in Para 1.A.5).

9) Reduce inlet pressure to zero and remove valve from test rig. If satisfactory and required for service, install in aircraft.

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Brake Pressure Relief Valve – Maintenance Practices

1. Adjustment/Test

Equipment required:

- Suitable hydraulic test rig

A. Bench Test Brake Pressure Relief Valve

1) Gain access to hydraulic power package, under flight compartment floor, from right hand access panel. (Refer to 12-00-00, Servicing – General Data).

2) Remove brake pressure relief valve from hydraulic power package.

3) Check for obvious damage.

4) Connect relief valve into circuit of suitable test rig.

5) With outlet port to return, slowly apply hydraulic pressure to inlet port.

6) Check that valve cracks open at 1750 + 50 or – 0 psi pressure.

7) Slowly release pressure. Check that valve reseats at 1600 psi minimum pressure.

8) Remove valve from test rig.

9) If required for service, reinstall relief valve in hydraulic power package.

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Flap System Thermal Relief Valve – Maintenance Practices

1. Adjustment/Test

Equipment required:

- Suitable hydraulic test rig

A. Bench Test Thermal Relief Valve

1) Remove appropriate cabin ceiling panels to gain access to thermal relief valve.

2) Relieve hydraulic system pressure by cycling flaps.

3) Remove thermal relief valve from aircraft.

4) Check for obvious damage.

5) Connect thermal relief valve into circuit of test rig.

6) With outlet port to return, slowly apply hydraulic pressure to inlet port.

7) Check that valve cracks open at 1750 + 50 or – 0 psi pressure.

8) Slowly release pressure. Check that valve reseats at 1575 psi minimum pressure.

9) Remove valve from test rig.

10) If required for service, reinstall relief valve in aircraft.

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Auxiliary System – General Data

1. General

The auxiliary system consists of a hydraulic handpump with a suction line from the reservoir and a pressure line connected into the power system on the hydraulic power package.

A. Hydraulic Handpump (Mod 6/2093)

The hydraulic handpump is of the double acting type, displacing an equal amount of fluid on each stroke. The handpump is installed beneath the flight compartment floor immediately to the left of the control column base. A hinged door provides access to the handpump and the handle is stowed on the sub-floor, right-hand side.

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Hydraulic Handpump (Mod 6/2093) – Maintenance Practices

1. Servicing

   A. Bleed Hydraulic Handpump

      1) Ensure electrical power is OFF.

      2) Discharge damping accumulator by operating wing flaps and wheel brake accumulator by pumping brake pedals.

      3) Open access panels on left and right-hand underside of nose section. (Refer to 12-00-00, Servicing – General Data).

      4) Slacken off handpump pressure line at tee of forward accumulator on hydraulic power package. Operate handpump until air has been expelled from line and fluid emerges in solid stream. Retighten handpump pressure line at tee. Stow handpump handle.

      5) Clean up hydraulic fluid spillage. Check reservoir fluid level.

      6) Close access panels. (Refer to 12-00-00, Servicing – General Data).

2. Removal/Installation

   A. Remove Handpump

      1) Discharge damping accumulator by operating wing flaps and wheel brakes accumulator by pumping brake pedals.

      2) Open access panel on left-hand underside of nose section. (Refer to 12-00-00, Servicing – General Data).

      3) Place container to receive spillage. Disconnect and cap return line and pressure line at elbows on handpump.

      4) Remove two bolts attaching handpump to structure and remove handpump.

   B. Install Handpump

      1) Position handpump and attach to structure with two bolts, washers and nuts, ensuring that longer bolt is used at upper attachment point.

      2) Connect return line and pressure line to elbows on handpump.

      3) Check reservoir fluid level. Bleed hydraulic handpump system. Refer to Para 1.A.

      4) Function test handpump. Refer to Para 3.A.

3. Adjustment/Test

   A. Function Test Handpump

      1) Ensure electrical power is switched OFF.

      2) Open access panel on left-hand underside of nose section. (Refer to 12-00-00, Servicing – General Data).

      3) Discharge airframe accumulator by operating wing flaps and wheel brakes accumulator by pumping brake pedals.

      4) Operate handpump and recharge both accumulators. Check that flight compartment hydraulic pressure indicators show same reading as accumulator pressure indicators within tolerance range.

5) Operate wing flaps until damping accumulator pressure is discharged. Check that wheel brakes accumulator pressure remains fully charged.

6) Operate handpump and select wing flaps up and down at random, ending with wing flaps fully up.

7) Recharge airframe accumulator using handpump. Stow handpump handle.

8) Close access panels. (Refer to 12-00-00, Servicing – General Data).

Indicating – General Data

1. General

The hydraulic power indicating system comprises of two transducers, two hydraulic/nitrogen accumulator pressure indicators, and a thermal overheat switch.

A. Hydraulic Pressure Indicator

Two hydraulic pressure transducers are mounted in the lower cockpit floor inside the center pedestal area.

One transducer senses airframe hydraulic pressure, and one transducer senses brake hydraulic pressure.

These airframe hydraulic and brake pressure transducers send a signal to display the amount of hydraulic pressure on the cockpit multi-function display screen.

B. Hydraulic/Nitrogen Accumulator Pressure Indicators

Two hydraulic/nitrogen pressure indicators (each combined with a charging valve) are installed, one for damping airframe system accumulator, and the other for wheel brakes system accumulator. Both indicators with charging valves are mounted on the hydraulic power package, and are accessible through the left-hand access panel on the underside of the nose section. Each indicator is direct-reading bourdon tube type, with dials marked X 1000, and scales marked from 0 to 2.0 in increments of 0.1.

C. Pump Thermal Overheat

A thermal switch on the pump end bell triggers a CAS message. When the temperature reaches 180°F ± 5°F, the switch will open and turn on the indication. When the temperature reaches 165°F ± 5°F, the switch will close and extinguish the indication.

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Hydraulic Accumulator Air Charging Valve and Indicator Assembly – Maintenance Practices

1. Adjustment/Test

Equipment required:

- Pressure indicator of proven accuracy

A. Bench Test Pressure Indicator

Note This test applies to both damping and wheel brake accumulators.

1) Discharge system hydraulic pressure by operating wing flaps and brakes.

2) Release accumulator air pressure.

3) Remove air charging valve and indicator assembly from aircraft. Block outlet port.

4) Connect source of nitrogen or dry air and standard indicator of proven accuracy to air charging valve.

5) Apply pressure and increase gradually. At each test pressure point noted in Table 1 — Calibration Tolerances maintain pressure and compare assembly indicator with standard indicator or proven accuracy. Tap indicator gently to obtain true reading.

6) Slowly reduce pressure and check accuracy of indicator at test pressures shown in Table 1 — Calibration Tolerances.

7) Remove block from outlet port, and source of pressure. Cap port.

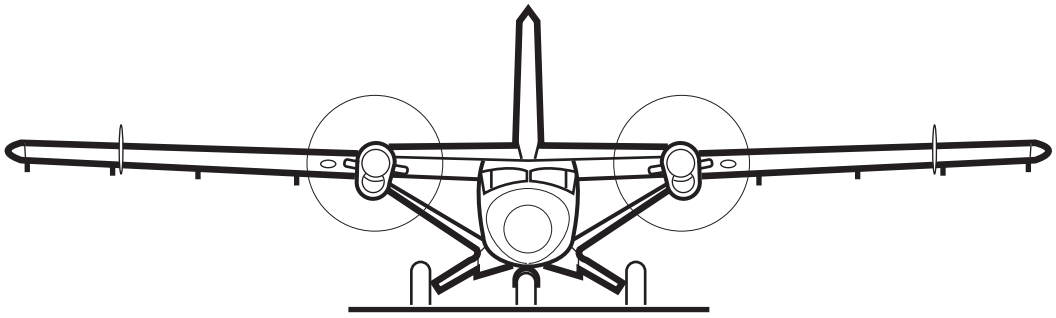
8) If required for service, install assembly in aircraft.

Table 1 Calibration Tolerances

TEST PRESSURE PSI	TOLERANCE ± PSI
0	30
400	40
800	50
1200	50
1600	50
2000	50

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PSM 1-64-2



DHC-6 TWIN OTTER
SERIES 400

MAINTENANCE MANUAL
VOLUME 2

VIKING

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Revision 8 - Sep. 28/18

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January 31, 2006

CHAPTER 30

ICE AND RAIN PROTECTION

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CHAPTER 30 – ICE AND RAIN PROTECTION

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Ice and Rain Protection – General Data

1. General

Customer options for the de-icing and anti-icing systems for the aircraft wings, tail, propeller, and windshield include the following:

A. Wing and tail de-icing, (S.O.O. 6004 Mod 6/2042) is accomplished by using pneumatically-operated boots. Includes timer systems, automatic activation with flap extension, wing inspection lights and annunciation of stabilizer boot inflation.

B. Propeller De-Icing, (S.O.O. 6202) is accomplished by the use of electrically heated boots. Requires the fuselage ice guards (S.O.O. 6168) to be installed.

C. Heated Windshield (S.O.O. 6187 Mod 6/2045) is accomplished by electrically heated elements integral to the windscreen.

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Wing and Tail De-icing System (Mod S.O.O. 6004 and Mod 6/2042) – General Data

1. General

The electrically controlled wing and tail de-icing system utilizes bleed air tapped from the engine compressors to inflate and deflate rubber de-icing boots installed on the leading edges of both wings and the horizontal stabilizer. The inflation and deflation pulses flex both wing boot inner sections, both wing boot outer sections, and the left and right boots on the horizontal stabilizer, in sequence, to break up any ice formation. Two wing inspection lights may be installed which can be used during darkness in icing conditions, to determine the thickness of ice accumulation on the wing leading edges (refer to 33-40-00, Exterior Lights (Mod 6/2058) – General Data).

For aircraft incorporating S.O.O. 6367 (Wing De-ice Provisions Removal), the wing de-icing tube assembly is not installed.

2. Description

The system consists of three distributor valves, an ejector, an electronic timer, three water separators, control switches, and rubber boots bonded to the leading edge of each wing and horizontal stabilizer. The regulated air supply is provided from a pneumatic system package (refer to 36-00-00, Pneumatic – General Data), located in the cabin roof. Electrical power for the wing and tail de-icing system is derived from the left and right DC bus bars through two circuit breakers marked AIRFRAME DEICE AUTO, and AIRFRAME DEICE MANUAL, located on the circuit breaker panel. If sufficient pressure exists at the pressure sensing switch for each horizontal stabilizer boot, the message “R STAB” or “L STAB” shall appear in green inverse video in the de-ice portion of the electrical system pane. An electrical heater jacket is installed around each distributor valve (Mod 6/1440). Refer to Figure 1 for details.

Included in the pneumatic system package is a low-pressure switch, the pressure switch consists of a single switch that activates when pneumatic pressure has decreased to 13 – 15 psi. A CAS message is displayed to the crew when pneumatic pressure falls below this level and the surface de-ice system has been selected ON. This message indicates that there is insufficient pressure to satisfactorily operate the de-ice boots.

A. Distributor Valves

The distributor valves are mounted one in each wing at station 260.00, and one in the rear section of the fuselage at station 486.00 approximately. Each distributor valve incorporates a pressure inlet port, a suction outlet port, two ports A and B to the de-icer boots, an exhaust port piped overboard to a low-pressure area, and two control solenoids, A and B.

B. Ejector

The ejector, located in the rear section of the fuselage, aft of the horizontal stabilizer distributor valve, is a venturi-type device which creates a vacuum to deflate the de-icing boots. The ejector is in continuous operation whenever the pneumatic supply is switched on.

C. Electronic Timer

The electronic timer, mounted in the cabin roof, is an electronic device which provides a cycling electrical output to the de-icing distributor valve solenoids. Fast or slow cycling of the timer may be selected by the operation of a switch, mounted on the de-icing control panel on the right hand switch panel.

D. Water Separators

Three water separators, one installed in each wing pneumatic line, and a third installed in the pneumatic line to the horizontal stabilizer distributor valve, are of the self-draining type. Water separators are located adjacent to the distributor valves (Mod 6/1440) and each separator contains a disposable 2 micron filter.

E. De-icing Boots

Inflatable de-icing boots are bonded to the leading edges of each wing, from a point just outboard of the engine nacelles, to the wing tips. Similar type boots are bonded to the entire leading edge of the horizontal stabilizer. Each boot is connected by hoses to the A and B ports of its relevant distributor valve: inner wing to A, outer wing to B, left stabilizer to A and right stabilizer to B.

F. Control Switches

The control switches are located on the de-icing panel located in the right pilot's sub-panel, under the label SURFACE DE-ICE. They consist of the following:

1) A BOOTS AUTO switch for the selection of automatic speeds or manual, with the positions marked FAST, SLOW and OFF/MANUAL.

2) A manual mode switch to momentarily inflate the wing de-icer boots. Alternate positions are marked INNER and OUTER.

3) A second manual STABILIZER switch to momentarily inflate the horizontal stabilizer boots. Alternate positions are marked LEFT and RIGHT.

4) A switch marked VALVE HEAT (Mod 6/1440), to energize the three distributor valve heaters.

G. Horizontal Stabilizer De-icing Boot Indication

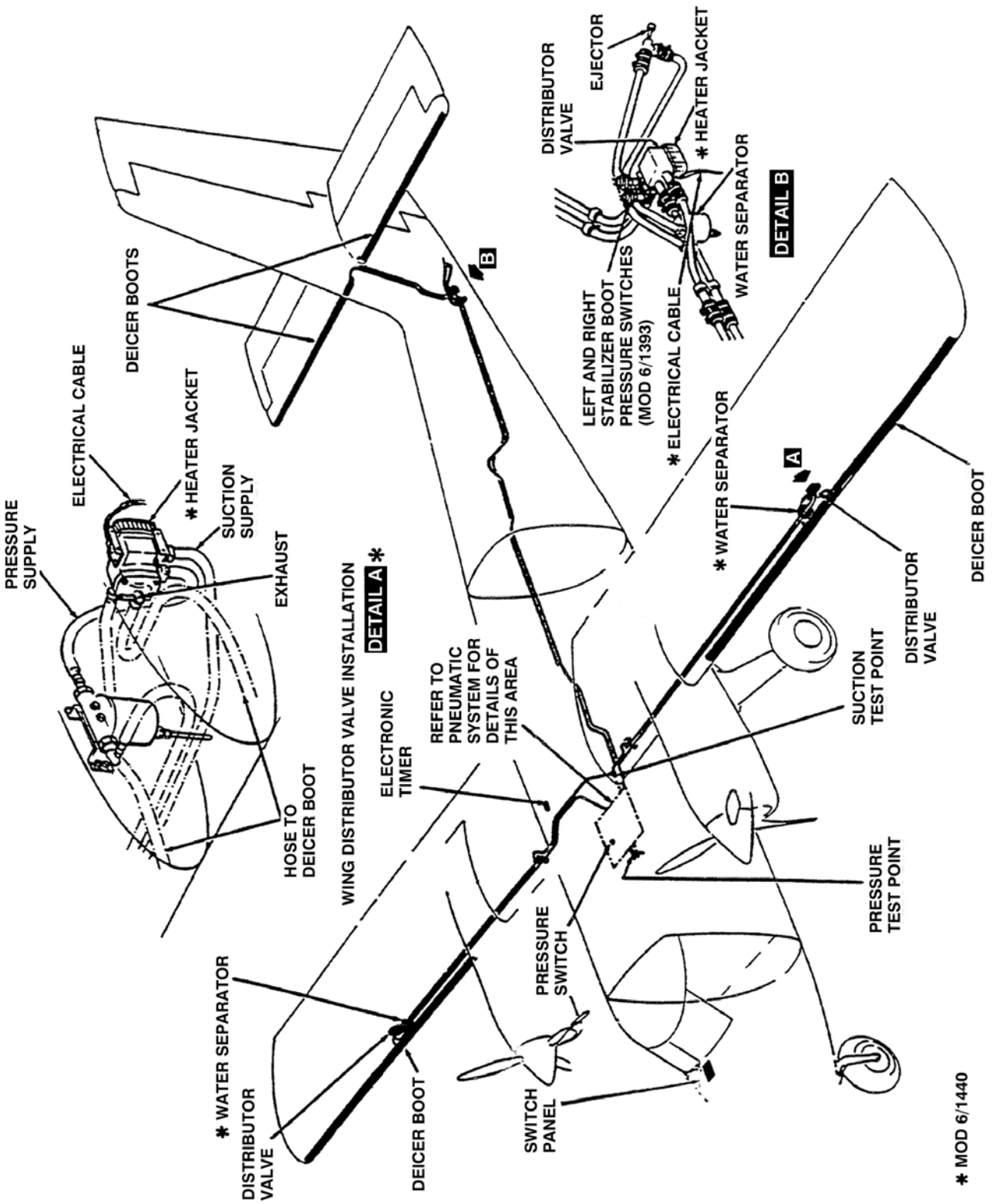
The Horizontal Stabilizer De-icing Boot Indicator is generated by a pressure switch in the air pressure line to the related tail plane de-icer boot. The message is displayed in the Fuel System pane: “L STAB” and “R STAB” is illuminated for the period of time that air at operating pressure is metered to each boot in each inflation cycle, in either automatic or manual mode.

H. Horizontal Stabilizer De-icing Pressure Switches (Mod 6/1393)

The two pressure switches are installed in the rear fuselage at the distributor valve pressure outlets A and B. The normally open contacts of the pressure switches close at 15 psi rising pressure.

I. Distributor Valve Heater Jacket (Mod 6/1440)

An electrical heater jacket is wrapped around the solenoid body of each distributor valve. The heater jacket is secured to the distributor valve by two springs. All three heater jackets are powered through the AIRFRAME DEICE MANUAL circuit breaker and VALVE HEAT switch.



Wing and Tail De-icing Systems (Mod S.O.O. 6004)
Figure 1

3. Operation

With the system circuit breakers closed and the BOOTS AUTO switch set to FAST or SLOW, the regulated pneumatic supply (refer to 36-10-00, Distribution (Mods S.O.O. 6004) – General Data) is directed to the distributor valves and the ejector. The venturi action of the ejector provides a suction at each of the distributor valves. The distributor valve solenoids are energized in a sequence determined by the operation of the electronic timer, at a fast or slow rate depending on the selection of the BOOTS AUTO, FAST or SLOW switch. As each solenoid is energized, a servo valve is operated to reverse suction and pressure conditions across a diaphragm within the distributor valve. The diaphragm mechanically positions a valve which shuts off the suction and admits pressure into the boot port, thereby allowing a section of the boot to inflate for a predetermined time. When the solenoid is deenergized, air flow through the valve on that side is cut off, and air discharges out of that particular boot section through a check valve until the pressure drops to approximately 1 inch Hg, at which time, that boot section is ported to the suction system and the remaining air is evacuated. The boot is held down until another cycle starts. When the system is selected OFF, only suction is applied to the de-icer boots to deflate and hold them flush to the wing profile. In the event of a failure of the electronic timer, the selector switch can be selected to OFF/MANUAL, and the operation of the distributor valves controlled by momentarily operating the WING INNER–OUTER or STABILIZER LEFT–RIGHT switches. A L STAB or R STAB indication comes on whenever the associated pressure switch contacts close at 15 psi rising pressure. On aircraft incorporating Mod 6/1440, when the VALVE HEAT switch is selected ON, the heater jacket of each distributor valve is energized. Refer to Figure 1 and PSM 1-64-2W for electrical schematic.

4. Adjustment/Test

Refer to Figure 1 for details.

A. Function Test Wing and Tail De-icing System

Equipment required:

- Pressure rig and gauge required for testing the de-icing system, is as specified for the pneumatic and air conditioning systems (refer to 21-40-00, Heating System – General Data and 36-10-11, Pressure Regulator Valve – Maintenance Practices).
- Suction gauge capable of measuring 4 to 10 inches Hg, with the necessary adapter to connect gauge to the suction test point.

1) Remove cabin roof upholstery in location of station 177.00 to provide access to heat exchanger.

2) Disconnect bleed air line at inlet to heat exchanger and connect pressure rig.

3) Remove cabin roof upholstery as required and connect suction gauge to the tee fitting in suction line forward of station 200.36.

4) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

5) With de-icer circuit breakers (AIRFRAME DEICE AUTO and AIRFRAME DEICE MANUAL) engaged, check that the low pressure advisory message is displayed.

6) Start pressure rig and pressurize system to maintain 18 ± 2 psi on rig pressure gauge, and 4 to 10 inches Hg suction. Check that low pressure advisory message is no longer displayed.

Note If a leak in the system is apparent, leak test the system in accordance with 36-10-11, Pressure Regulator Valve – Maintenance Practices.

7) Select system selector BOOTS AUTO switch to FAST. A temporary loss of pressure and suction is allowed.

8) Check that sequence of boot inflation, correct inflation time and dwell time are in accordance with Table 1 — De-icer Boots Timing Sequence. Check the left and right horizontal stabilizer boot indicators each come on for 3 seconds in each 60 second cycle (Mod 6/1393). Ensure that the indicators do not, at any time, come on together.

9) Ensure that each boot section deflates fully in 20 seconds.

10) Select cycle selector switch to BOOTS AUTO and SLOW and check that sequence of inflation, correct inflation time and dwell time are in accordance with Table 1 — De-icer Boots Timing Sequence.

11) Select BOOTS AUTO switch to OFF/MANUAL.

12) Select each distributor valve switch, in turn, to WING INNER, WING OUTER, LEFT STABILIZER, RIGHT STABILIZER. The appropriate boot section must inflate in time specified in Table 1 — De-icer Boots Timing Sequence, remain inflated as long as switch is held in position, and deflate when switch is released.

13) Check that when LEFT is selected the L STAB indicator comes on within 2 seconds, and that when RIGHT is selected the R STAB indicator comes on within 2 seconds. Ensure that the indications do not, at any time, come on together.

14) Release pressure and note that Pneumatic Press Low advisory message is displayed at 15 ± 2 psi. Ensure that both L STAB and R STAB indications are OFF.

15) Select VALVE HEAT (Mod 6/1440), switch ON and check that heater jackets on wing and fuselage distributor valves are warm. Set VALVE HEAT switch to OFF.

16) Set BOOTS AUTO switch to OFF/MANUAL.

17) Disconnect test rig and suction gauge. Reconnect bleed air line at inlet to heat exchanger. Reinstall cabin roof upholstery where necessary.

18) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Table 1 De-icer Boots Timing Sequence

SEQUENCE OF INFLATION	TIME TAKEN FOR INFLATION	DWELL TIME AT INFLATION SELECTION	
		FAST	SLOW
Wing Inner	5 seconds	44 ± 10 seconds	2 min 44 ± 20 seconds
Wing Outer	5 seconds		
Left Stabilizer	3 seconds		
Right Stabilizer	3 seconds		

B. Operational Test (Pre Mod 6/2314)

1) Set BUS TIE switch to NORMAL and POWER SOURCE switch to BATTERY.

2) Select MASTER switch to ON.

3) Momentarily select VALVE HEAT switch to ON and check that Battery Amps Indication shows a discharge, then set switch to OFF.

4) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.) and set POWER SOURCE switch to EXTERNAL.

5) Start engine. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

6) Ensure AIRFRAME DEICE AUTO and AIRFRAME DEICE MANUAL circuit breakers are engaged and Pneumatic Press Low advisory message is extinguished.

7) Select BOOTS AUTO switch to FAST.

8) Have second operator outside aircraft to observe boot inflation sequence as listed in Table 1 — De-icer Boots Timing Sequence. Check horizontal stabilizer de-icing boots indicator operation as in Para 4.A.8).

9) Select BOOTS AUTO switch to SLOW and repeat Para 4.B.8).

10) Select switch BOOTS AUTO to OFF/MANUAL.

11) Select each distributor valve switch in turn, to WING INNER, WING OUTER, LEFT STABILIZER, RIGHT STABILIZER. The appropriate boot section must inflate in time specified in Table 1 — De-icer Boots Timing Sequence, remain inflated as long as switch is held in position, and deflate when switch is selected to OFF or another position. Check horizontal stabilizer de-icing boots indicator operation as in Para 4.A.13).

12) Select BOOTS AUTO switch to OFF/MANUAL.

13) Shutdown engine (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data) and check Pneumatic Press Low caution CAS message is displayed before switching MASTER switch to OFF.

14) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Operational Test (Post Mod 6/2314)

1) Set BUS TIE switch to NORMAL and BATTERY switch to ON.

2) Momentarily select VALVE HEAT switch to ON and check that Battery Amps Indication shows a discharge, then set switch to OFF.

3) Set BATTERY switch to OFF.

4) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.) and set EXTERNAL switch to ON.

5) Start engine. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

6) Ensure AIRFRAME DEICE AUTO and AIRFRAME DEICE MANUAL circuit breakers are engaged and Pneumatic Press Low advisory message is extinguished.

7) Select BOOTS AUTO switch to FAST.

8) Have second operator outside aircraft to observe boot inflation sequence as listed in Table 1 — De-icer Boots Timing Sequence. Check horizontal stabilizer de-icing boots indicator operation as in Para 4.A.8).

9) Select BOOTS AUTO switch to SLOW and repeat Para 4.C.8).

10) Select switch BOOTS AUTO to OFF/MANUAL.

11) Select each distributor valve switch in turn, to WING INNER, WING OUTER, LEFT STABILIZER, RIGHT STABILIZER. The appropriate boot section must inflate in time specified in Table 1 — De-icer Boots Timing Sequence, remain inflated as long as switch is held in position, and deflate when switch is selected to OFF or another position. Check horizontal stabilizer de-icing boots indicator operation as in Para 4.A.13).

12) Select BOOTS AUTO switch to OFF/MANUAL.

13) Shutdown engine (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data) and check Pneumatic Press Low caution CAS message is displayed before switching EXTERNAL switch to OFF.

14) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Wing and Tail De-icing Boots (S.O.O. 6004 and Mod 6/2045) – Maintenance Practices

1. Removal/Installation

Refer to PSM 1-6-6A – Component Maintenance Manual.

A. Remove Wing or Horizontal Stabilizer De-icer Boots

WARNING ADEQUATE VENTILATION MUST BE PROVIDED AND THE USE OF PROTECTIVE CLOTHING STRICTLY ADHERED TO AT ALL TIMES DUE TO THE TOXIC NATURE OF THE SOLVENTS.

Note To facilitate installation of boot, accurately mark position of boot centerline on adjacent wing or horizontal stabilizer at each end of boot. Also, accurately locate position of stall bars on wings, span and chordwise, by measuring to fixed datum points.

1) Loosen clamps and disconnect relevant de-icer hoses. Blank open ends and temporarily stow hoses.

2) Remove bolts securing wing fences, and release bond between wing fence seals and de-icer boots by continuous application of toluol at seam line when removing fences.

3) If the de-icer boot is damaged, check the repair limits according to CMM. If the boot is beyond repair limits, remove it according to PSM 1-6-6A – Component Maintenance Manual. If it is within limits, repair according to PSM 1-6-6A – Component Maintenance Manual.

B. Install Wing or Horizontal Stabilizer De-icer Boots

WARNING ADEQUATE VENTILATION MUST BE PROVIDED AND THE USE OF PROTECTIVE CLOTHING STRICTLY ADHERED TO AT ALL TIMES DUE TO TOXIC NATURE OF THE SOLVENTS.

Refer to PSM 1-6-6A – Component Maintenance Manual to install boots.

1) Horizontal stabilizer de-icer boot trimming:

a) On lower inboard end of boots, use tool SD5197 or a sharp knife to trim edge to accommodate installation of vortex generators.

2) Wing de-icer boot trimming:

a) Trim inboard end of wing boot to landing lamp and outboard end to wing tip joint.

b) On left-hand wing boot, remove with tool SD5197 or a sharp knife, the portion of the boot covering cut out for lift detector installation. Cut out boot so as to leave a gap of approximately 1/32–inch between boot and lift detector mounting plate.

3) Ensure bond is intact in areas that have been trimmed or cut out.

4) Seal gap between left-hand wing de-icer boot and lift detector mounting plate.

5) Install stall bar to wings as follows:

a) Clean as per PSM 1-6-6A – Component Maintenance Manual instructions.

b) Allow to dry.

c) Locate and mark stall bar position as noted during removal, refer to Para 1.A.

d) Apply adhesive to both stall bar and the surface of boot in accordance with PSM 1-6-6A – Component Maintenance Manual, boot bonding procedure.

e) Reactivate cement of bar and boot as per PSM 1-6-6A – Component Maintenance Manual instructions.

f) Firmly press stall bar on to boot and use a rubber roller on leading edge of bar to seat it.

g) Clean up any excess adhesive around edge of stall bar as per PSM 1-6-6A – Component Maintenance Manual instructions.

6) Remove blanks from de-icer boot air connections and end of hoses. Connect hoses and secure using clamps.

7) Operate de-icer system and check de-icer boots for bubbles.

8) Install wing fences as follows:

a) Clean fence seal and area of boot contacted by seal with a toluol-soaked cloth, and wipe dry with a clean cloth before solvent evaporates.

b) Apply by brush, a thin, even coat of Bostik 1142 to both bonding surfaces, making as few strokes as possible. Stir adhesive thoroughly immediately before use.

c) Allow adhesive to dry until tacky.

d) Position wing fence on wing and insert attachment bolts. Press seal in contact with boot using fingers or rubber roller and tighten attachment bolts.

9) Function test wing and tail de-icing system. (Refer to 30-10-00, Wing and Tail De-icing System (Mod S.O.O. 6187 and Mod 6/2045) – General Data).

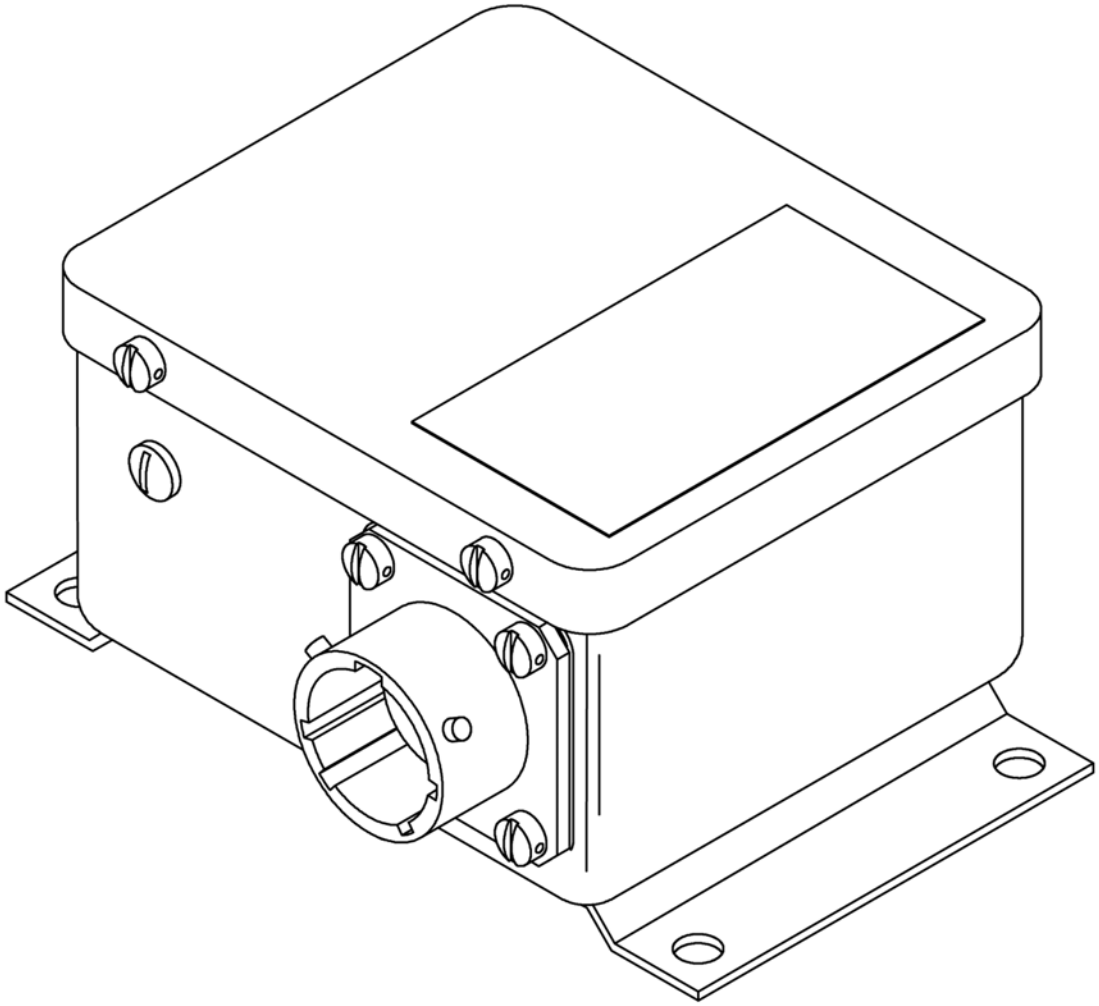
10) Verify a conductive path exits between the de-ice boot, the conductive edge sealer and the skin. Using a Resistivity Meter, measure the conductivity, acceptable range is 10^3 to 10^5 ohms.

Note Resistivity Meter (Monroe Electronic Model 291 or equivalent) contains two parallel bars on the underside, one is placed on the boot and the other is placed on the edge sealer.

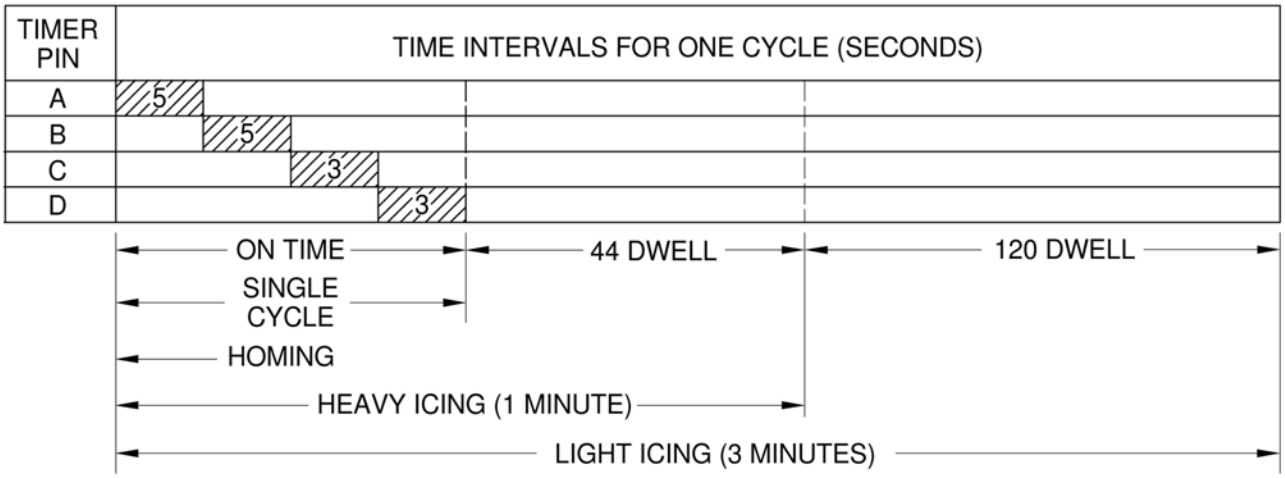
Electronic Timer (Mod S.O.O. 6004) – General Data

1. General

The Goodrich pneumatic timer (P/N 3D2485–27), provides the energizing voltage to the wing and tail de-icing distributor valves, in the system introduced by Mod S.O.O. 6004. The timer, located in the left side of the cabin roof between fuselage station frames 200.36 and 219.32, provides a five second “on” period for the two solenoids in each wing distributor valve, and a three second “on” period for both solenoids in the tail distributor valve. Cycling can be controlled at a fast or slow rate depending on the selection of BOOTS AUTO – FAST – SLOW switch. The total cycling time (including dwell time) is 60 seconds when selected to FAST, and 180 seconds at SLOW. The timer (Figure 1), consists of a metal box with an external connector receptacle, which houses a circuit selector, component board and the necessary electrical wiring. The box has a cover which is sealed with silicon rubber and secured with four screws, it also has two flanges which are drilled to facilitate mounting the complete assembly.



Pneumatic De-icing Timer
Figure 1



NOTES:

1. SHADED AREAS REPRESENT CONTROL INTERVAL TIMES.
2. CYCLING IS CONTINUOUS AND STARTS AT THE BEGINNING OF THE CONTROL INTERVALS.
3. TIME INTERVALS DO NOT OVERLAP.

Timing Chart
Figure 2

2. Servicing

Since the pneumatic de-icing timer is an electronic assembly the servicing excludes any repair according to the manufacturer. Any repair and replacement of component parts must be done only through authorized repair and overhaul facilities.

3. Adjustment/Test

A. Test Equipment

In addition to the required test equipment listed in Table 1 — Test Equipment, a bench test circuit must be available to conduct an operational check of the timer. Individual components of the bench test circuit are listed in Table 3 — Bench Test Circuit Components. The bench test circuit is illustrated in Figure 3.

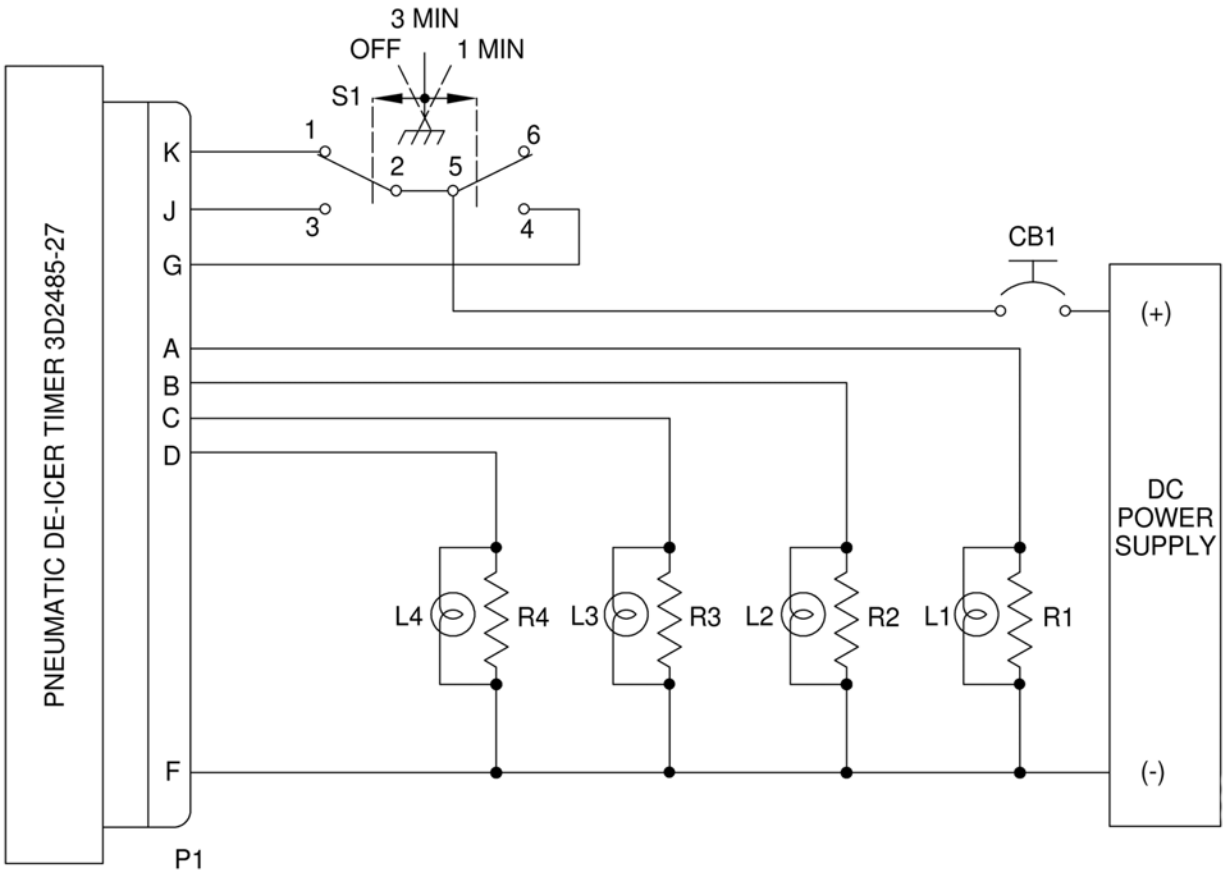
Note Equivalent substitutes may be used.

Table 1 Test Equipment

Item	Identification	Source (CAGE Code)
Bench test circuit		Local fabrication. Refer to Table 3 — Bench Test Circuit Components for individual components
DC power supply; variable, regulated	0 to 32 Volt DC, 5 amps	Local purchase
Insulation resistance tester	Model 225 (500 Volt DC)	Slaughter Co.; 8011 Hailey St.; Ardmore, OK 73401 (V05611)

Table 1 Test Equipment (continued)

Item	Identification	Source (CAGE Code)
Mating electrical connector (with terminating test leads)	MS3116E12–10S	Local purchase
Stopwatch		Local purchase



Timer Test Circuit
Figure 3

4. Inspection/Check

A. Procedure

1) Visually examine the timer for:

a) Signs of corrosion.

b) Loose or missing sealant.

c) Nicks, cracks and dents.

d) Distortion.

e) Legible label.

2) Shake the timer and check for the sound of loose parts inside the case.

3) Examine black sealant for a continuous fillet around the case-to-lid seam and around the connector. If required, reapply sealant to damaged areas.

4) Visually examine electrical connector for:

a) Loose, broken, or shorted lead wire connectors.

b) Electrical arcing paths.

c) Loose or defective attaching parts.

d) Corroded contacts and terminals.

e) Bent and damaged pins.

f) Damage caused by excessive heat.

B. Insulation Resistance Test

WARNING **THE INSULATION RESISTANCE TESTER CAN OUTPUT POTENTIALLY LETHAL VOLTAGES. AVOID CONTACT WITH THE CONNECTION POINT ON TESTER AND PROBE LEAD WIRES. FAILURE TO COMPLY WITH THIS WARNING CAN RESULT IN SEVERE ELECTRICAL SHOCK OR DEATH.**

Do the insulation resistance check before doing the operational tests so that any circuit damage due to this check can be detected during the functional testing that follows.

1) Using a mating electrical connector with terminating test leads, connect test leads from all electrical connector pins together, forming a wire bundle.

2) Ensure that insulation resistance tester is OFF.

3) Connect one lead of insulation resistance tester to wire bundle.

4) Connect other lead of insulation resistance tester to external screw on timer.

5) Apply power to insulation resistance tester.

6) Apply 500 (±25) Volt DC between wire bundle and external screw on timer.

7) Observe that resistance is greater than 100 megohms.

8) Turn insulation resistance tester OFF.

9) Disconnect insulation resistance tester from wire bundle and external screw on timer.

10) Remove mating electrical connector from timer.

C. Operational Tests

1) Operational Test Setup

a) With circuit breaker CB1 and switch S1 in OFF position, connect timer to bench test circuit as shown in Figure 3.

b) Apply power to power supply, and adjust power supply output to 18 (±1) Volt DC.

c) Close circuit breaker CB1.

2) Time Output Checks

a) Single-Cycle Check

1 Momentarily set switch S1 to either the 1 MIN or 3 MIN position and then back to the OFF position, and time output intervals with stopwatch.

2 Observe that lamp operate as shown in the following chart.

Lamp	Time (seconds)			
L1	5			
L2		5		
L3			3	
L4				3

3 After 16 seconds, observe that timer rapidly “homes” in a series of audible clicks, and then stops.

b) One Minute Mode Check

1 Set switch S1 to 1 MIN position, and time output intervals with stopwatch.

2 Observe that lamps operate as shown in the following chart.

Lamp	Time (seconds)				
L1	5				
L2		5			
L3			3		
L4				3	
	60				

c) Three Minute Mode Check

1 Set switch S1 to 3 MIN position, and time output intervals with stopwatch.

2 Observe that lamps operate as shown in the following chart.

Lamp	Time (seconds)				
L1	5				
L2		5			
L3			3		
L4				3	
	180				

D. Special Tools Fixtures and Equipment

The following items are required for testing of the timer. Standard shop tools and consumables required for these functions are not listed.

In addition to the test equipment listed in Table 2 — Special Tools, Fixtures, and Equipment, a bench test circuit must be available to conduct an operational check of the timer. Individual components of the bench test circuit are listed in Table 3 — Bench Test Circuit Components. The bench test circuit is illustrated in Figure 3.

Note Equivalent subtitles may be used for listed items.

Table 2 Special Tools, Fixtures, and Equipment

Item	Identification	Source (CAGE Code)
Bench test circuit		Local Fabrication. Refer to Table 3 — Bench Test Circuit Components for individual components
DC power supply; variable, regulated	0 to 32 Volt DC, 5 amps	Local purchase
Insulation resistance tester	Model 225 (500 Volt DC)	Slaughter Co.; 801 Hailey St.; Ardmore, OK 73401 (V05611)
Mating electrical connector (with terminating test leads)	MS3116E12–10S	Local purchase
Stopwatch		Local purchase

Table 3 Bench Test Circuit Components

Reference Designator (Refer to Figure 3)	Component	Description/Identification	Source (CAGE Code)
CB1	Circuit breaker	5 amp	Local purchase
L1 through L4	Lamp	28 Volt DC , MS25041–6–327	Local purchase
P1	Plug	MS3116E12–10S	Local purchase
R1 through R4	Resistor	50 watt, 20 ohm	Local purchase
S1	Switch, toggle	Double-pole, Double-throw, MS27407–4	Local purchase
–	Wiring	20 gauge	Local purchase

Water Separator and Filter (Mod 6/1440) – Maintenance Practices

1. Removal/Installation

A. Remove Filter Element and Drain Screen

1) Gain access to water separator as follows:

a) Through access panel on underside of wing, inboard of wing station 260.00, for a water separator in the wing.

b) Through rear baggage compartment for water separator in the rear fuselage.

2) Cut and remove lockwire from water separator bowl retaining bolt and drain nipple.

3) Unscrew water separator bowl retaining bolt and withdraw water separator bowl.

4) Remove cotter pin and filter retainer from end of retaining bolt and remove filter element.

5) If fitted, remove drain pipe from water separator bowl.

6) Remove drain nipple, drain screen and washer from water separator bowl drain fitting.

B. Install Filter Element and Drain Screen

1) Check drain screen is clean and not damaged.

2) Position washer and drain screen on water separator bowl drain fitting and install drain nipple. Torque drain nipple to a value of 20 to 25 pound–inches.

3) If removed, connect drain pipe to drain nipple.

4) Check that gaskets (O-rings) in head of water separator are not damaged. Sparingly lubricate each O-ring with lubricant FS–1265 (Dow Corning Corp.), or equivalent.

5) Lubricate O-ring seal on filter element with lubricant FS–1265, or equivalent.

6) Install filter element in water separator bowl. Ensure that spring retainer and spring are in position under filter element.

7) Place filter retainer over retaining bolt and install new cotter pin.

8) Press separator bowl firmly into separator head and engage retaining bolt.

CAUTION **ENSURE THAT BOWL IS FULLY SEATED IN HEAD BEFORE TIGHTENING RETAINING BOLT. DO NOT FORCE BOWL INTO HEAD BY TURNING RETAINING BOLT.**

Note If water separator is in wing, position bowl to align drain pipe with grommet in wing access panel.

9) Torque retaining bolt to a value of 9 to 11 pound–inches above drag torque.

10) Lockwire retaining bolt to drain nipple using MS20995C20.

11) Install wing access panel (if removed).

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Pitot Heating (Mod 6/2033) – General Data

1. General

The pitot heads, have integral electrically heated elements which may be switched on to prevent ice formation which could block the pitot head.

2. Description

The aircraft is fitted with two pitot heads, one is installed on a mast projecting from the left side of the nose section, and a second is installed in an identical position on the right side of the nose section (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data). Pitot heat is controlled by a switch labelled PITOT HEAT, located on the LH sub panel. Each heating circuit is monitored by current sensing relays which provide grounds to the MAU when the current flow is interrupted.

28 Volt DC power is provided from the left and right buses through a 7 1/2 amp PITOT HEAT L (A6) circuit breaker and a 7 1/2 amp PITOT HEAT R (B6) circuit breaker on the main circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

Two configurations are available for displaying the Left and/or Right Probe Fail, depending on aircraft operational requirements. The configuration is selected by changing the DAU configuration plug (refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data). The configurations are outlined below:

A. Configuration 1 – A CAS message is displayed whenever the current flow to a pitot head is interrupted. This includes when the PITOT HEAT switch is selected to the OFF position.

B. Configuration 2 – A CAS message is only displayed when the PITOT HEAT switch is selected to the ON position and current flow to the pitot head is interrupted.

3. Adjustment/Test

A. Operational Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Ensure PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers are closed.

3) Select PITOT HEAT switch to ON and check that pitot head warms up.

CAUTION DO NOT LEAVE PITOT HEATER SWITCHED ON FOR MORE THAN A MINUTE.

4) Confirm no Left and/or Right Probe Fail CAS messages are present.

5) Pull the PITOT HEAT L (A6) circuit breaker and confirm the Left Probe Fail caution CAS appears.

6) Pull the PITOT HEAT R (B6) circuit breaker and confirm the Right Probe Fail caution CAS appears.

7) Reset both circuit breakers and confirm the CAS messages clear.

8) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal and Installation

A. Pitot Heads

Removal/Installation of the pitot heads is detailed in 34-11-00, Pitot-Static System (Mod 6/2033) – General Data.

B. Pitot Heat

1) Removal of Pitot Heat Relay

a) Ensure the aircraft electrical system is OFF.

b) Open the PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers and fit safety clips.

c) Gain access to the fwd bulkhead by removing the MFDs as required (refer to 31-13-00, Apex Controllers – General data).

d) Remove (4) electrical terminals. Protect terminals from short circuits to airframe or each other.

e) While supporting the relay, remove (2) mounting screws.

f) Remove unit from bulkhead.

2) Installation of Pitot Heat Relay

a) Ensure the aircraft electrical system is OFF.

b) Set relay in position and install (2) mounting screws.

c) Remove protection and install (4) electrical terminals. Ensure terminals are tight.

d) Install MFDs as required (refer to 31-13-00, Apex Controllers – General data).

e) Remove safety clips and close the PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers.

5. Troubleshooting

A. Pitot Heat

1) Ensure PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers are closed.

2) Visually inspect the pitot heat connectors for damaged wiring or contacts. Ensure electrical connectors are fully seated.

3) Visually inspect the electrical terminals on the pitot fail relays for damaged wiring. Ensure the nuts holding the electrical terminals are tight.

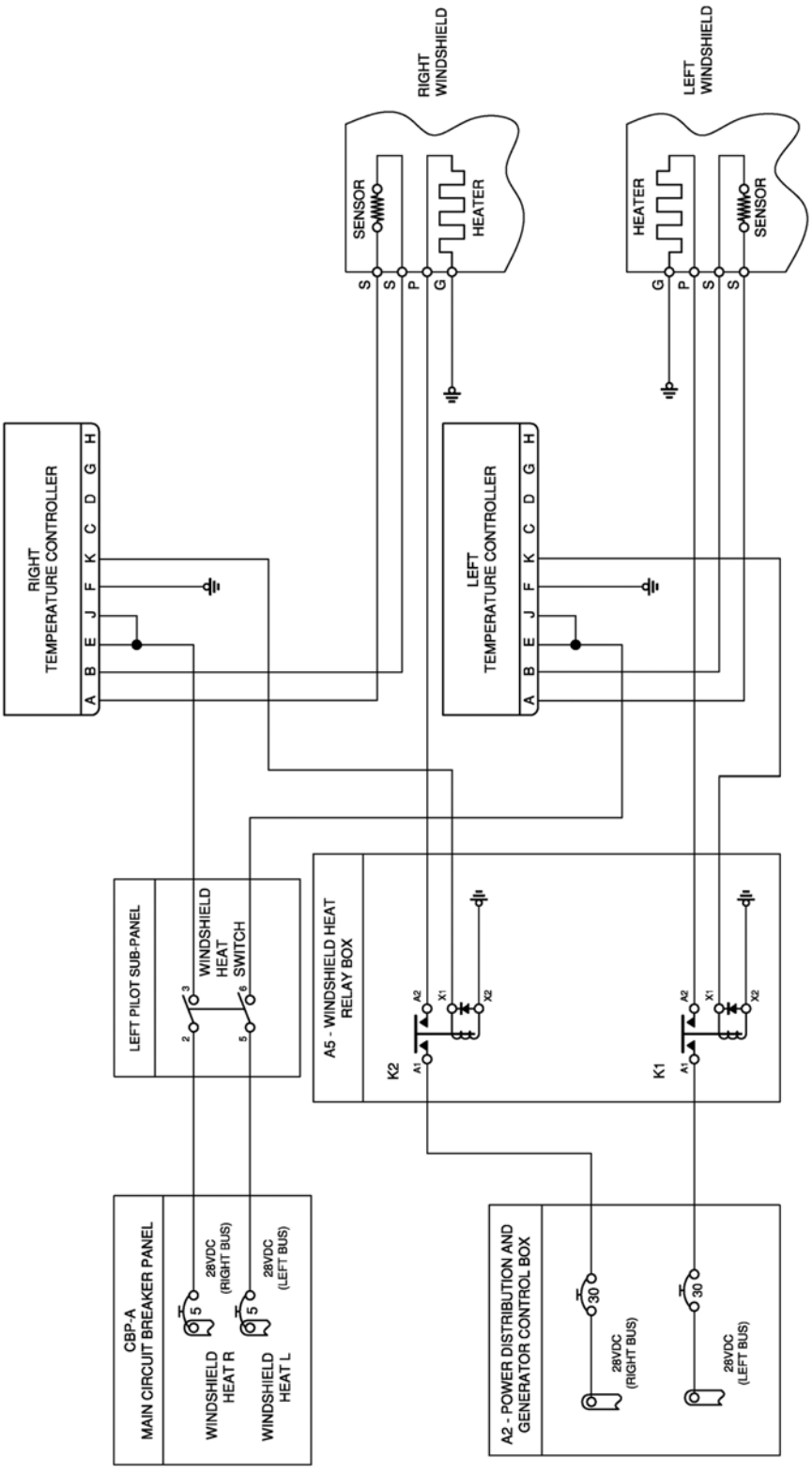
4) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

5) If repair or service is required, send to appropriately rated avionics repair facility.

Windshield Heating System (Mod S.O.O. 6187 and Mod 6/2045) – General Data

1. General

The electrically heated windshield system in effect consists of two systems, one for each windshield, both of which are controlled by a single two-pole switch, labelled OFF and ON, located on the W/S HEAT section of the left hand pilot's sub-panel. Each system comprises of a relay, a temperature controller, two circuit breakers, and a windshield manufactured from laminations of glass and vinyl. A fine wire heating element and a sensing element are incorporated in the laminations during the windshield manufacture. The temperature controllers and the relays are installed in the cabin roof under the upholstery. When the windshield heating switch is selected to ON, electrical power from the 28 Volt DC bus bar is routed through the WINDSHIELD HEAT L and R 5-ampere circuit breakers (located on the circuit breaker panel), and the switch, to the left and right temperature controllers. Each temperature controller in conjunction with its related windshield sensing element, provides an output to energize its associated relay coil when the windshield temperature is low. With the relay energized, a circuit from the left (or right) 28 Volt DC bus bar is completed to its associated windshield heating element, through the CB2 (or CB1) 30-ampere circuit breaker located on the power distribution and generator control box. When the windshield reaches its maximum temperature, or when the heating switch is selected OFF, the temperature controller output ceases, the relay (or relays) are deenergized and power is disconnected from the windshield heating element.



Windshield Heating – Electrical Schematic
Figure 1

Note For the removal and installation of electrically heated windshields refer to 56-10-00, Flight Compartment Windshields – General Data.

2. Adjustment/Test

A. Function Test Windshield Heating System

1) Disconnect electrical wiring at each windshield.

2) Check for continuity between terminals P and G (heating elements) and between terminals S (sensing elements).

Note Resistance between terminals P and G is 1.0 to 1.3 ohms.

3) Reconnect electrical wiring at each windshield.

4) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Engage circuit breaker CB2 in power distribution box and circuit breaker WINDSHIELD HEAT L in overhead console.

5) Connect voltmeter across terminals P and G (G is ground) of pilot's windshield.

6) Select windshield heat switch to ON. Check that voltage exists at terminals until temperature of windshield rises and sensing control de-energizes the power relay.

7) When temperature of windshield drops sufficiently to cause the sensing control to reenergize the power relay, check that power is restored to the windshield.

8) Disengage circuit breakers engaged in Para 2.A.4), and engage circuit breaker CB1 in power distribution box and circuit breaker WINDSHIELD HEAT R in left hand pilot's sub-panel. Repeat Para 2.A.5) through Para 2.A.7) for co-pilot's windshield.

9) Return windshield heat switch to OFF and disengage circuit breakers engaged in Para 2.A.8).

10) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Temperature Controller – Maintenance Practices

1. Adjustment/Test

A. Function Test Temperature Controller

Equipment required:

- DC supply – Adjustable 15 – 30 volts
- Multimeter
- Decade Resistance Box 0 to 400 ohms in 0.1 ohm steps (10 ma rating)

1) Check temperature operating point:

a) Ensure aircraft power is OFF.

b) Disconnect electrical connector from controller.

c) Connect multimeter (set to ohms range) between pins J and K of receptacle on controller.

d) Apply 28 Volt DC (positive to pin E, negative (ground) to pin F).

e) Connect decade resistance box between pins A and B. Set decade to 330 ohms.

f) Turn temperature adjustment control (R14) set screw on top cover of controller until relay K1 picks up (indicated when meter reads zero ohms).

g) Vary decade resistance to determine “dead band”. Check that the difference between the pickup and dropout resistance is 7 +2 or –3 ohms.

h) Set decade resistance to 300 ohms. Vary R14 to check that relay K1 picks up.

i) Set decade resistance to 370 ohms. Vary R14 to check that relay K1 picks up.

j) Reset resistor R14 to pick up relay K1 with decade resistance setting of 330 +5 or –4 ohms.

k) Adjust input voltage to 22 Volt DC. Check that relay K1 pickup and dropout points do not differ by more than 1.6 ohms from values obtained in Para 1.A.1).j).

l) Check that relay K1 does not chatter when decade resistance is varied, and that the “dead band” is within ± 0.5 ohm of value obtained in Para 1.A.1).g).

2) Fail-safe test:

a) Adjust input voltage to 28 volts, and set decade resistance to pick up relay K1.

b) Remove connection from pin A or pin B. Check that relay K1 immediately drops out (indicated by meter reading infinity).

c) Set decade resistance to 1 ohm greater than value required for relay K1 to pick up.

d) Apply ground at pin A (jumper pin A to pin F) and check that relay K1 drops out. Repeat by applying ground at pin B.

e) Disconnect meter, decade resistance and power supply. Reconnect electrical connector to temperature controller.

3) Continuity test:

With all power disconnected from controller, check pin to pin resistance as follows using multimeter:

a) Between connector pins J, K, I, C, L, N, H, D, M and G to all other pins. Check for infinity.

b) Pin A to pin E; 250 ohms ± 5%.

c) Pin A to pin F; 250 ohms ± 5%.

4) Zener diode test:

a) Apply 28 Volt DC (positive to pin E, ground to pin F) through a 0 to 0.5 ampere DC ammeter.

b) Check that ammeter indicates 175 to 250 milliamperes if R12 is 47 ohms and 248 to 350 milliamperes if R12 is 33 ohms.

Note In certain units a 33 ohm resistor is installed in location R12 instead of a 47 ohm resistor. When CR1 is changed, check that R12 is 47 ohms and replace if necessary.

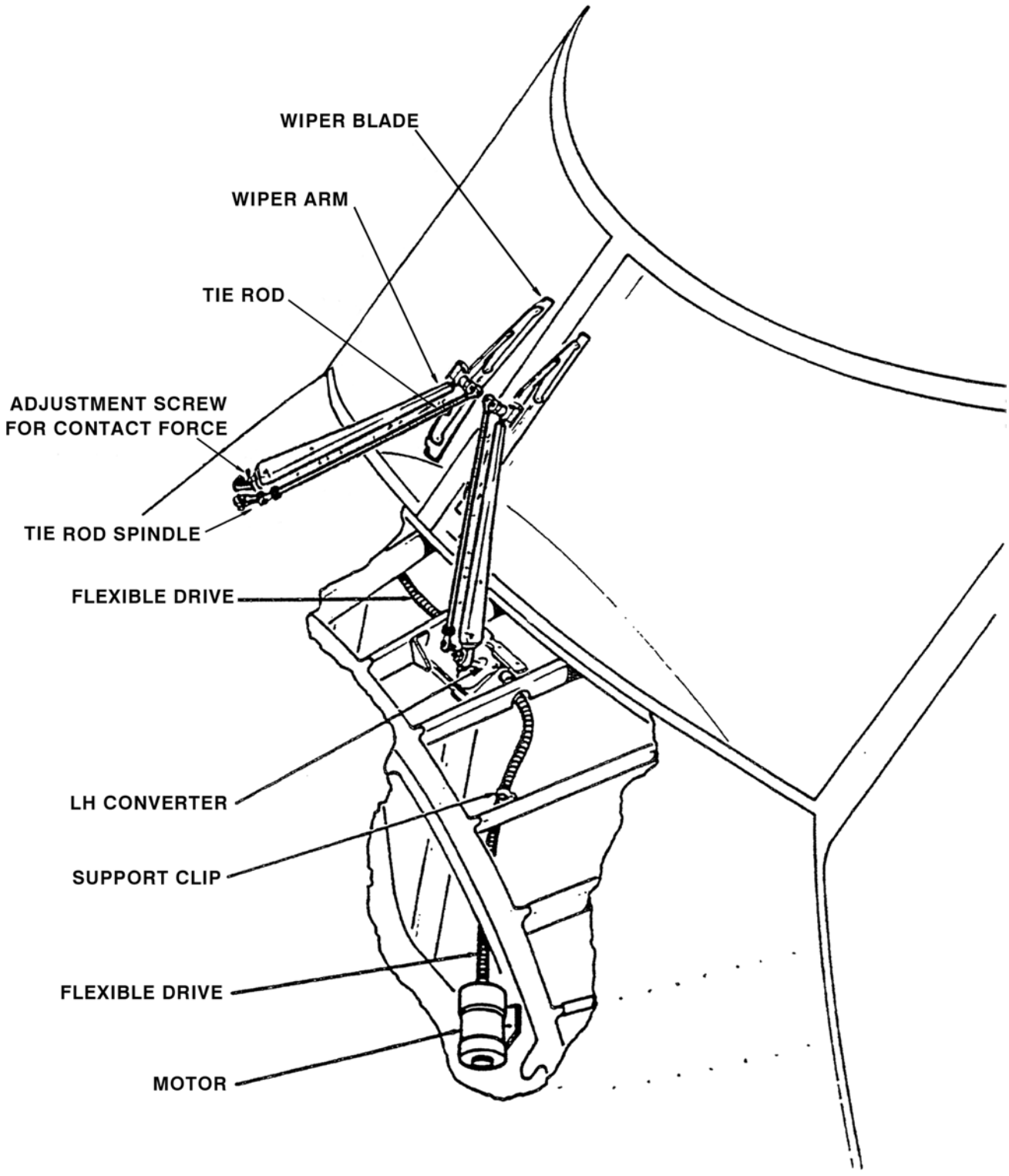
c) Adjust input supply to 22 Volt DC. Check that ammeter current is 47 to 123 milliamperes if R12 is 47 ohms, and 66 to 176 milliamperes if R12 is 33 ohms.

d) Adjust input supply to 29 Volt DC. Check that ammeter current is 195 to 270 milliamperes if R12 is 47 ohms, and 279 to 388 milliamperes if R12 is 33 ohms.

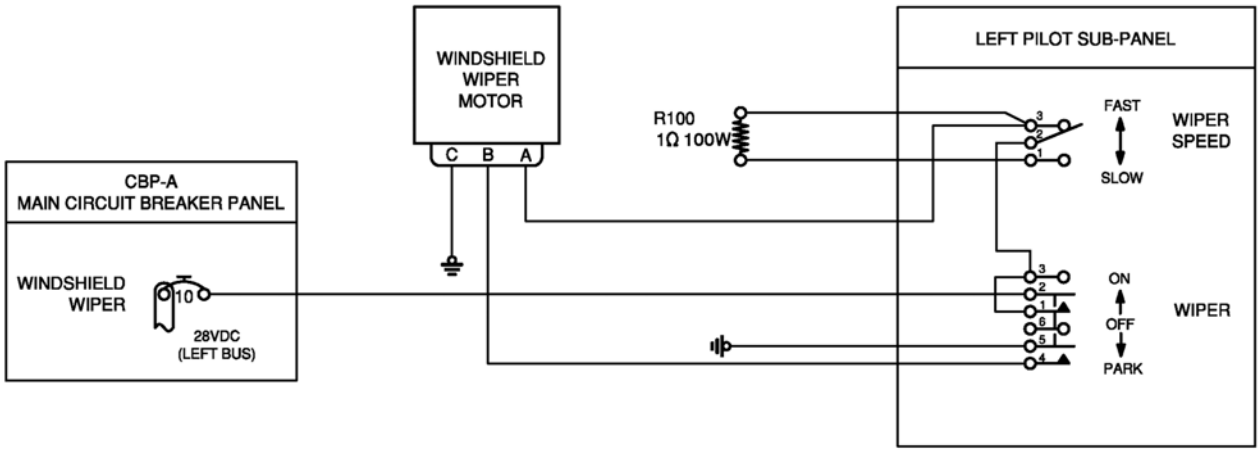
Windshield Wiper System (Mod 6/2088) – General Data

1. General

The windshield wiper system consists of two converters, one for each window panel, each having a driving arm and a 12–inch wiper blade. The converters are driven through flexible drive shafts by a single 28 Volt DC motor. The system is controlled by two switches labeled WIPER SPEED and WIPER located on the pilot's subpanel. The WIPER SPEED selects the operating speed, has two marked positions SLOW and FAST. The WIPER switch has 3 positions ON–OFF–PARK. The circuit is protected by a WINDSHIELD WIPER circuit breaker on the circuit breaker panel. When selected to PARK, both wipers should stop adjacent to, and parallel with, the windshield center post. Refer to Figure 1 and Figure 2 for details.



Windshield Wiper System Installation
Figure 1



Windshield Wiper Systems – Electrical Schematic
Figure 2

2. Adjustment/Test

A. Operational Test Windshield Wiper System

Note During the following test, maintain an adequate supply of water to the windshields, to prevent the blades operating on a dry surface.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Engage the WINDSHIELD WIPER circuit breaker and select the WIPER switch to ON.

3) Ensure that each blade wipes the windshield clean of water, that wiper arms traverse 90°, and that both blades operate to between 1.8 and 2.0 inches from the center of the windshield center post.

4) Select the WIPER SPEED switch between FAST and SLOW and check that speed of wipers varies accordingly.

5) Select the WIPER switch to PARK then release to OFF position. Ensure left wiper blade parks 1.8 to 2.0 inches from center of the windshield center post.

6) Disengage WINDSHIELD WIPER circuit breaker.

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

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Windshield Wiper Arm and Blade (Mod 6/2088) – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Windshield Wiper Arm and Blade

1) Remove nut and washers attaching tie rod to spindle.

2) Remove bolt and washer securing arm to converter shaft.

3) Remove arm complete with tie rod and blade.

B. Install Windshield Wiper Arm and Blade

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Engage WINDSHIELD WIPER circuit breaker, and select the WIPER switch to PARK.

3) Install left or right wiper arm and blade assembly as follows:

a) Left wiper arm and blade:

Align arm splines with splines on left converter shaft, so that distance between center of windshield center post and the wiper blade pivot point is between 1.8 and 2.0 inches. Install washer and bolt to secure arm to converter shaft; lockwire bolt.

b) Right wiper arm and blade:

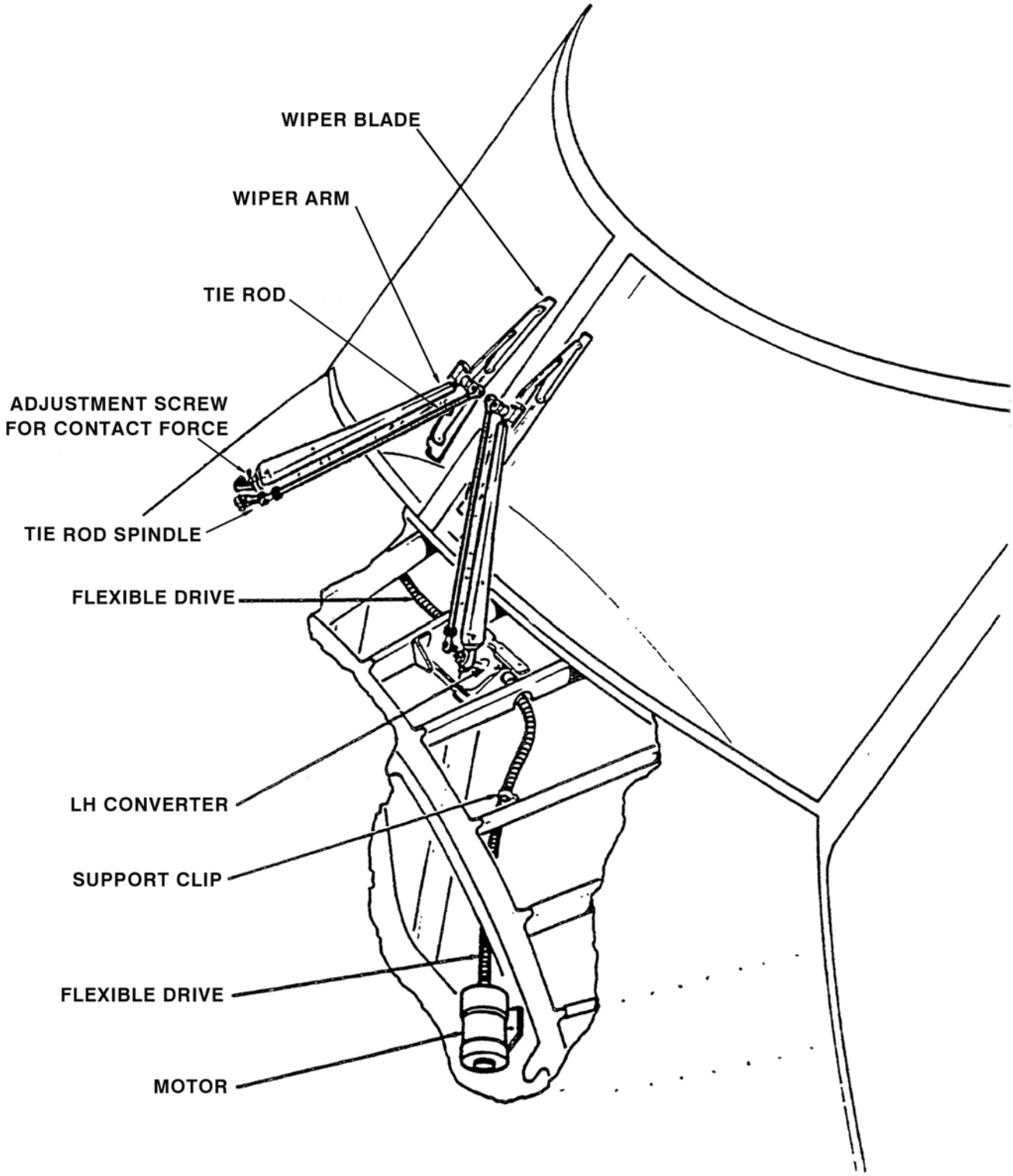
Align arm splines with splines on right converter, so that distance between center of windshield center post and the wiper blade pivot point is between 1.8 and 2.0 inches. Install washer and bolt to secure and to converter shaft; lockwire bolt.

4) Adjust tie rod until wiper blade is parallel with windshield center post. Secure tie rod to spindle with washers and nut.

5) Using special tool XW20509 (Alco Controls Div. Emerson Electric Co., St. Louis. Mo.), adjust wiper arm spring tension by adjusting screw at lower end of wiper arm, until a blade contact force of 4.5 pounds is obtained.

Note Measure contact force by attaching spring scale to wiper blade pivot, and by pulling at a right angle to and away from windshield.

6) Test windshield wiper system for operation. (Refer to 30-45-00, Windshield Wiper System (Mod 6/2088) – General Data).



Windshield Wiper System Installation
Figure 1

Windshield Wiper Motor (Mod 6/2088) – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Windshield Wiper Motor

1) Gain access to forward face of bulkhead at station 60.00.

2) Disconnect electrical wiring from windshield wiper motor.

3) Disconnect flexible drive from motor.

4) Remove four bolts and washers securing mounting plate to the bulkhead bracket and remove mounting plate with wiper motor attached.

5) Remove four screws securing wiper motor to mounting plate and remove motor.

B. Install Windshield Wiper Motor

1) Position wiper motor on mounting plate and secure with four screws.

2) Position mounting plate on bulkhead bracket and secure with four bolts and washers.

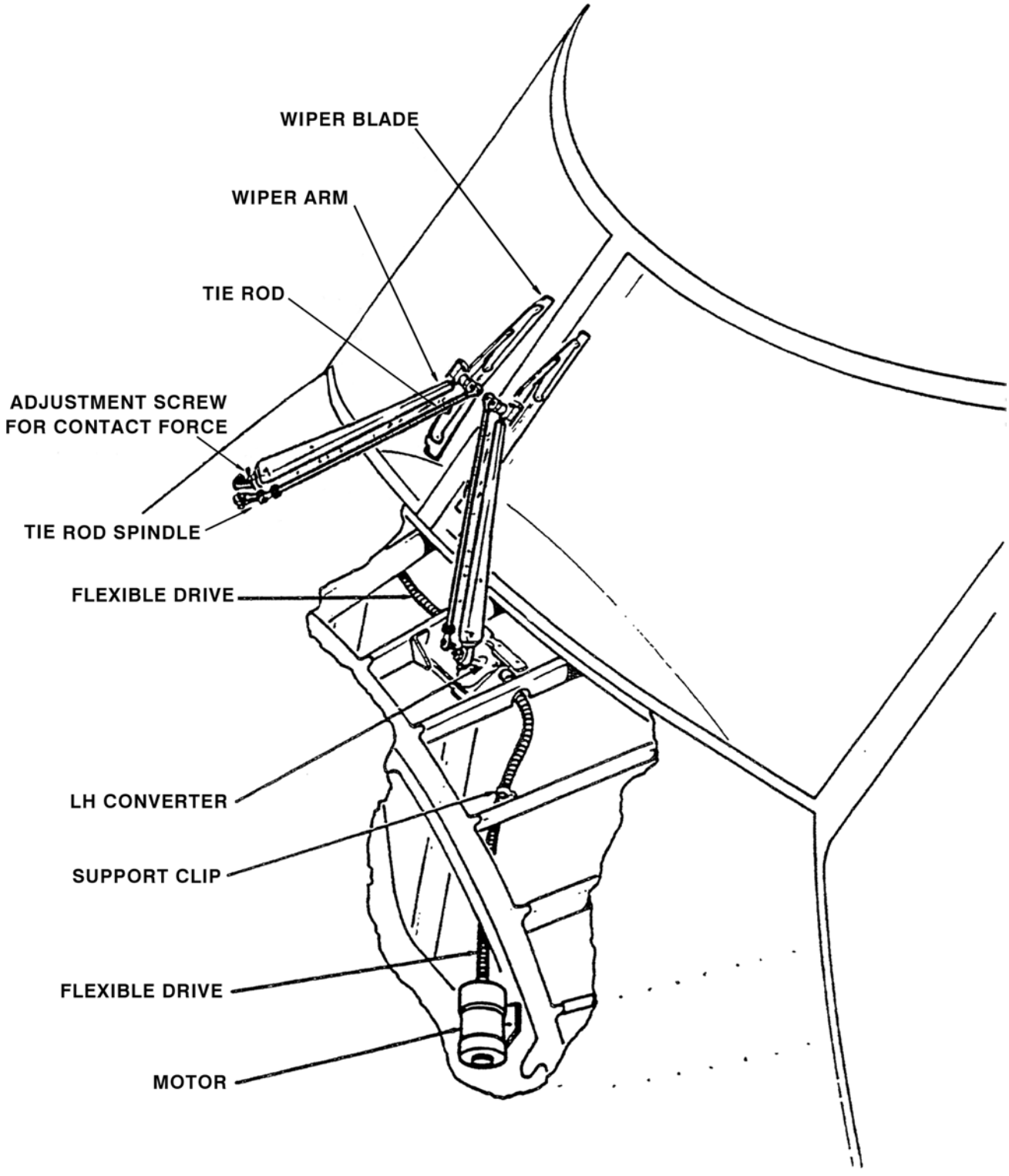
3) Ensure that flexible drive shaft is correctly engaged in left converter.

4) Connect flexible drive shaft to motor. Lockwire drive shaft connecting nut.

Note Position support clip (if fitted), to ensure flexible drive is a minimum of 0.10 inch clear of artificial horizon and does not contact other systems.

5) Connect electrical wiring to motor.

6) Test windshield wiper system for operation. (Refer to 30-45-00, Windshield Wiper System (Mod 6/2088) – General Data).



Windshield Wiper System Installation
Figure 1

Windshield Wiper Converters (Mod 6/2088) – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Windshield Wiper Converter

1) Remove relevant windshield wiper arm and blade. (Refer to 30-45-12, Windshield Wiper Arm and Blade (Mod 6/2088) – Maintenance Practices).

2) Disconnect flexible drive(s) from appropriate converter.

3) Remove and retain the three bolts and washers, securing converter to mounting plate.

4) Withdraw converter from mounting plate.

B. Install Windshield Wiper Converter

1) Locate converter drive shaft through skin and secure to mounting plate using bolts and washers retained during removal.

2) For installation of left converter, (pilot's side) proceed as follows:

a) Connect input drive shalt.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Engage WINDSHIELD WIPER circuit breaker and select the WIPER switch to PARK.

d) Ensure that left converter wiper arm drive shaft is at limit of counterclockwise movement, (looking aft at front of windshield.)

e) Ensure that right converter wiper arm drive shaft is also at limit of clockwise movement, looking aft at front of Windshield adjusting manually as required.

f) Ensure that flexible drive shaft is correctly engaged in right converter.

g) Connect interconnecting flexible drive to left converter and lockwire drive shaft connector.

Note Position support clip (if fitted), to ensure flexible drive is a minimum of 0.10 inch clear of artificial horizon and does not contact other systems.

3) For installation of right converter, (co-pilot's side) proceed as follows:

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Engage WINDSHIELD WIPER circuit breaker and select the WIPER switch to PARK.

c) Ensure that left converter wiper arm drive shaft is at limit of counterclockwise movement, looking aft at front of windshield.

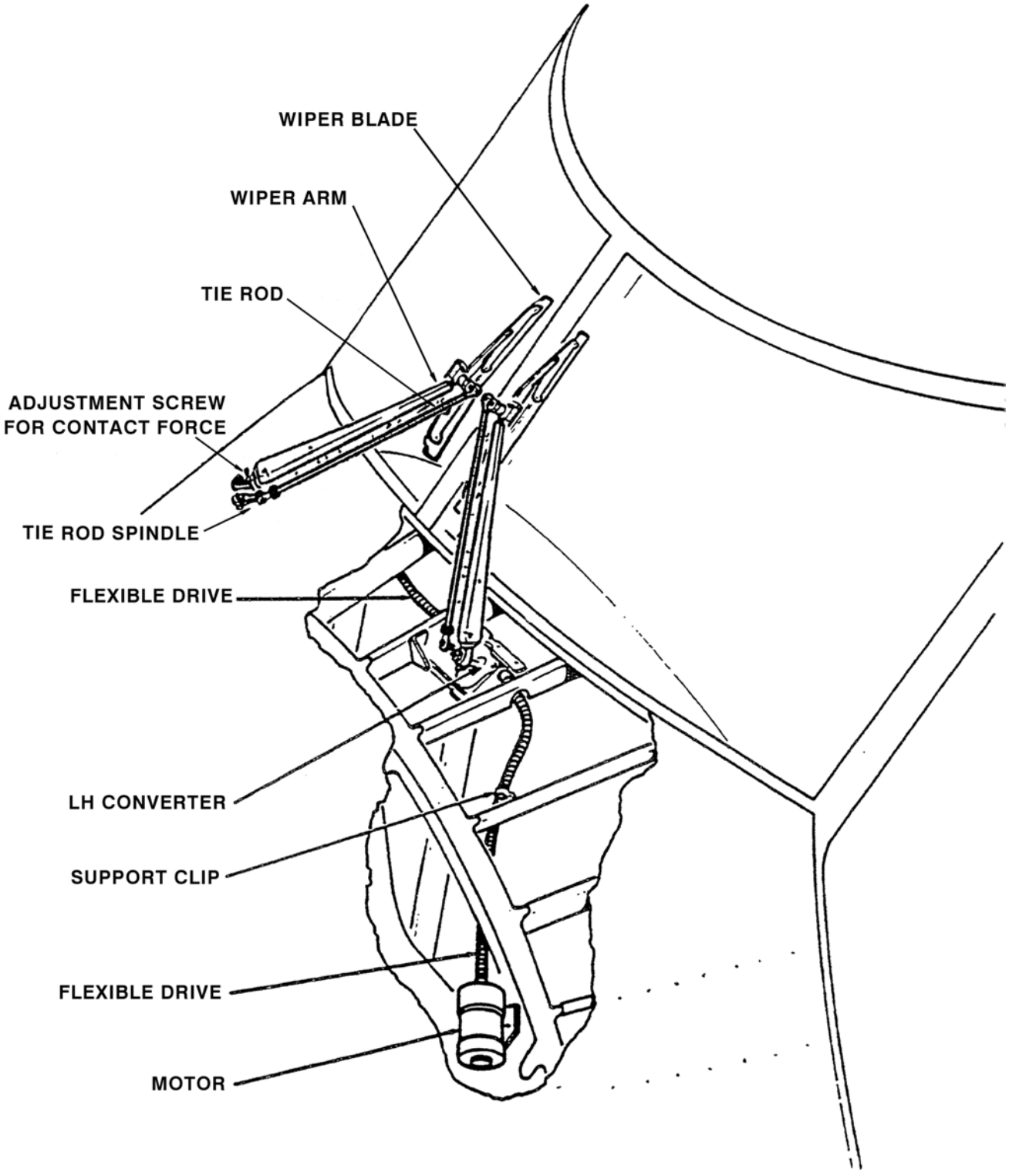
d) Ensure that right converter wiper arm drive shaft is also at limit of clockwise movement, looking aft at front windshield adjusting manually as required.

e) Ensure that flexible drive shaft is correctly engaged in left converter.

f) Connect interconnecting flexible drive to left converter and lockwire drive shaft connector.

4) Seal between fuselage skin and converter drive outer sleeve using Pro-Seal 735. (Refer to 20-50-02, Sealant and Adhesives – General Data).

5) Install relevant windshield wiper arm and blade. (Refer to 30-45-12, Windshield Wiper Arm and Blade (Mod 6/2088) – Maintenance Practices).



Windshield Wiper System Installation
Figure 1

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Propeller De-icing System (Mod S.O.O. 6202) – General Data

1. General

Propeller de-icing is effected by providing a cycled electrical power supply to inner and outer heating elements, contained in de-icer boots, installed on the leading edge of each propeller blade.

2. Description

The propeller de-icing system comprises the boots and slip ring assemblies which are a part of the two propeller rotating assemblies (refer to 61-10-11, Propeller De-icing (S.O.O. 6202) – General Data), a brush block assembly and oil pressure switch installed on each engine, two relays, a timer, and a control switch. Electrical power to the system is supplied from the 28 Volt DC bus, through a 20-ampere circuit breaker, identified PROPELLER DEICE, located on the circuit breaker panel. A brush module assembly replaces the brush block assembly (Mod 6/1674). Refer to Figure 1 for details.

A. Brush Module Assembly – (Mod 6/1674)

A brush module assembly is installed on a bracket secured to the front of each engine reduction gearbox. Each assembly houses three brushes which make contact with three slip rings on the rotating slip ring assembly (refer to 61-10-11, Propeller De-icing (S.O.O. 6202) – General Data). Each brush module assembly consists of a bracket and a brush retainer assembly comprised of three brush modules wired to a connector. Each brush module consists of a plastic housing with an integral brush and spring. The modules each have a wire internally connected to the brush with the three wires terminating in a single connector for connection to the propeller de-icing system.

B. Oil Pressure Switches

An oil pressure switch, which is also connected into the low oil pressure caution warning circuit (refer to 79-36-00, Low Oil Pressure Indicating System – General Data), is mounted on each engine accessory gearcase. When an engine is running, the oil pressure switch contacts are closed to provide a ground for the relevant propeller de-icing relay operating coil circuit, to allow the relay to energize when the system is selected on.

C. Relays

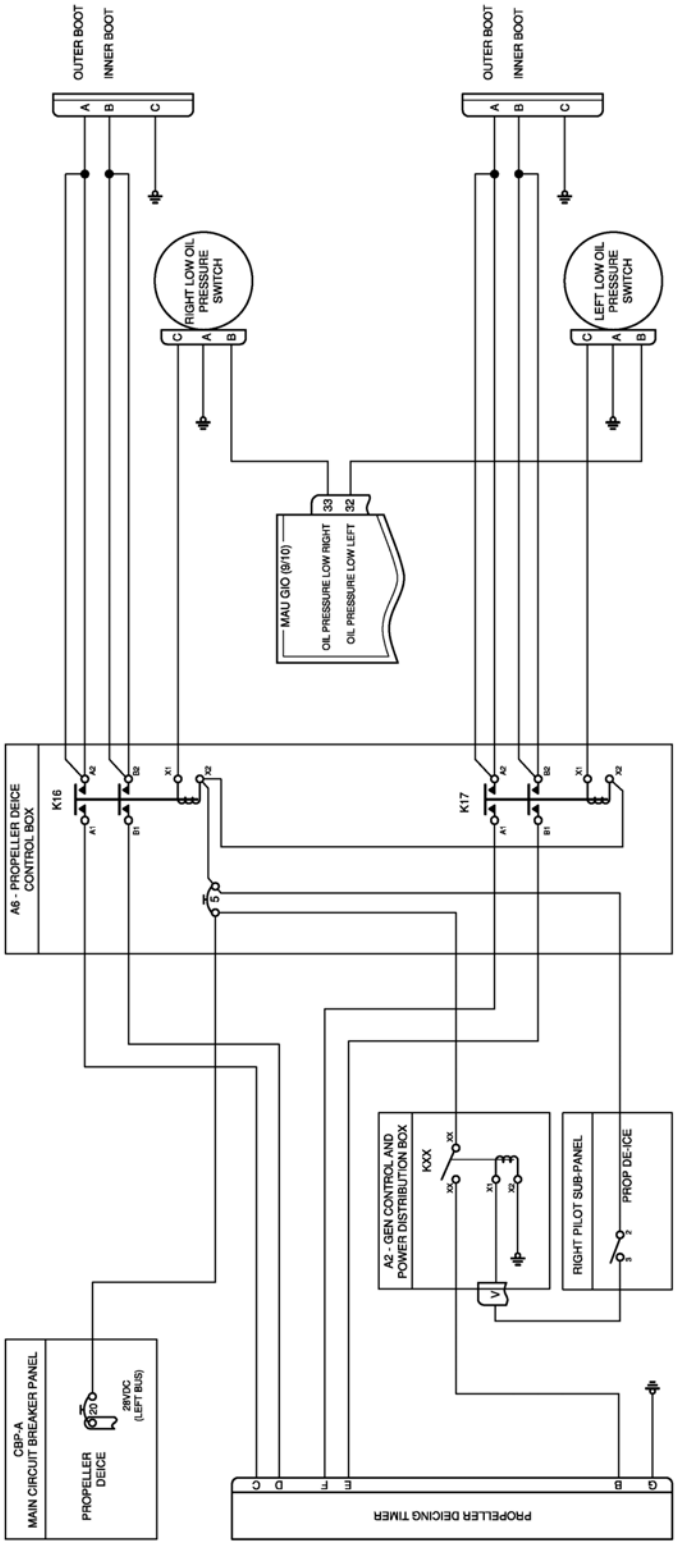
Two propeller de-icing relays, each controlling a single propeller system, are installed in the propeller de-icing control box located in the cabin roof. Each relay, when energized, completes two circuits, one from the timer to the outer heater elements in the blade boots of the associated propeller, and the other to the inner elements of the boots of the same propeller. A manual reset 5-ampere circuit breaker, connected in series with both relay control coils, is installed on the de-icing control box to protect the relay coils.

D. Timer

The timer, mounted in the cabin roof, is an electronic device which provides four sequenced power outputs, two to each propeller system.

E. Control Switch

A control switch identified PROP DE-ICE is located on the right pilot sub-panel. The switch completes the circuit to provide electrical power to the timer and each propeller de-icing relay control coil.



Propeller De-icing System – Electrical Schematic
Figure 1

3. Operation

With both engines running, the PROPELLER DEICE circuit breaker engaged, and the PROP DE-ICE control switch selected ON, electrical power is provided to the timer, and the relays are energized resulting in a sequenced power output being provided to the inner and outer elements of both propellers in turn. In the event of a propeller being feathered, or an engine shutting down, the oil pressure switch contacts open (due to low or loss of oil pressure), and the appropriate propeller de-icing relay de-energizes to open the circuits to the heating elements of the affected engine's propeller.

4. Adjustment/Test

A. Operational Test Propeller De-icing System (with engines running)

Note It is essential that the propeller de-icer brushes are properly run in on their slip rings before carrying out the test. After installation, whether new or replacement brushes are used, allow a run-in period of at least one hour.

1) Engage PROPELLER DEICE circuit breaker.

2) Start engines, turn on one generator and turn off electrical equipment not required. Observe limitations and procedures in accordance with 71-00-00, Power Plant (Mod 6/2012) – General Data.

3) Select PROP DE-ICE switch on right pilot's sub-panel to ON.

4) Check for correct functioning of propeller heating circuits as shown by an initial increase in generator load of approximately 6 amps.

Note Observe that the current remains approximately as above for 2 minutes and 30 seconds to verify that the system cycles through all four heating circuits.

5) Switch PROP DE-ICE switch to OFF.

6) Shutdown engines in accordance with 71-00-00, Power Plant (Mod 6/2012) – General Data.

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Propeller De-icing Brush Module Assembly (Mod 6/1674) – Maintenance Practices

1. Servicing

Refer to Figure 1 for details.

A. Replacement of Brush Retainer Assembly (B. F. Goodrich P/N 3E2114–1)

CAUTION **BRUSH MODULES ARE PACKED WITH A PROTECTIVE CARDBOARD COVER AND ELASTIC BAND WHICH RETAINS THE BRUSHES UNDER SPRING TENSION. FAILURE TO RESTRAIN BRUSHES WHEN REMOVING PROTECTIVE COVER OR ALLOWING BRUSHES TO SPRING FORWARD COULD RESULT IN BRUSH DAMAGE.**

1) Remove brush module assembly from engine mounting bracket, refer to Para 2.A.

2) Remove and retain four screws, nuts and washers securing connector to bracket. Note orientation of connector relative to bracket, so that new or replacement retainer assembly connector will be replaced in exact same position.

3) Remove and retain two screws, flat washers, lock washers and nuts securing brush retainer assembly to bracket.

4) Discard brush retainer assembly.

Note Refer to Figure 1 and check that brush modules are stacked in correct sequence as follows:

a) Observe connector pin identification and physically check wiring to determine related module A, B, C.

b) Ensure that modules are stacked in A, B, C sequence with module A positioned against bracket.

5) Secure brush retainer assembly to bracket with two screws, flat washers, lock washers and nuts.

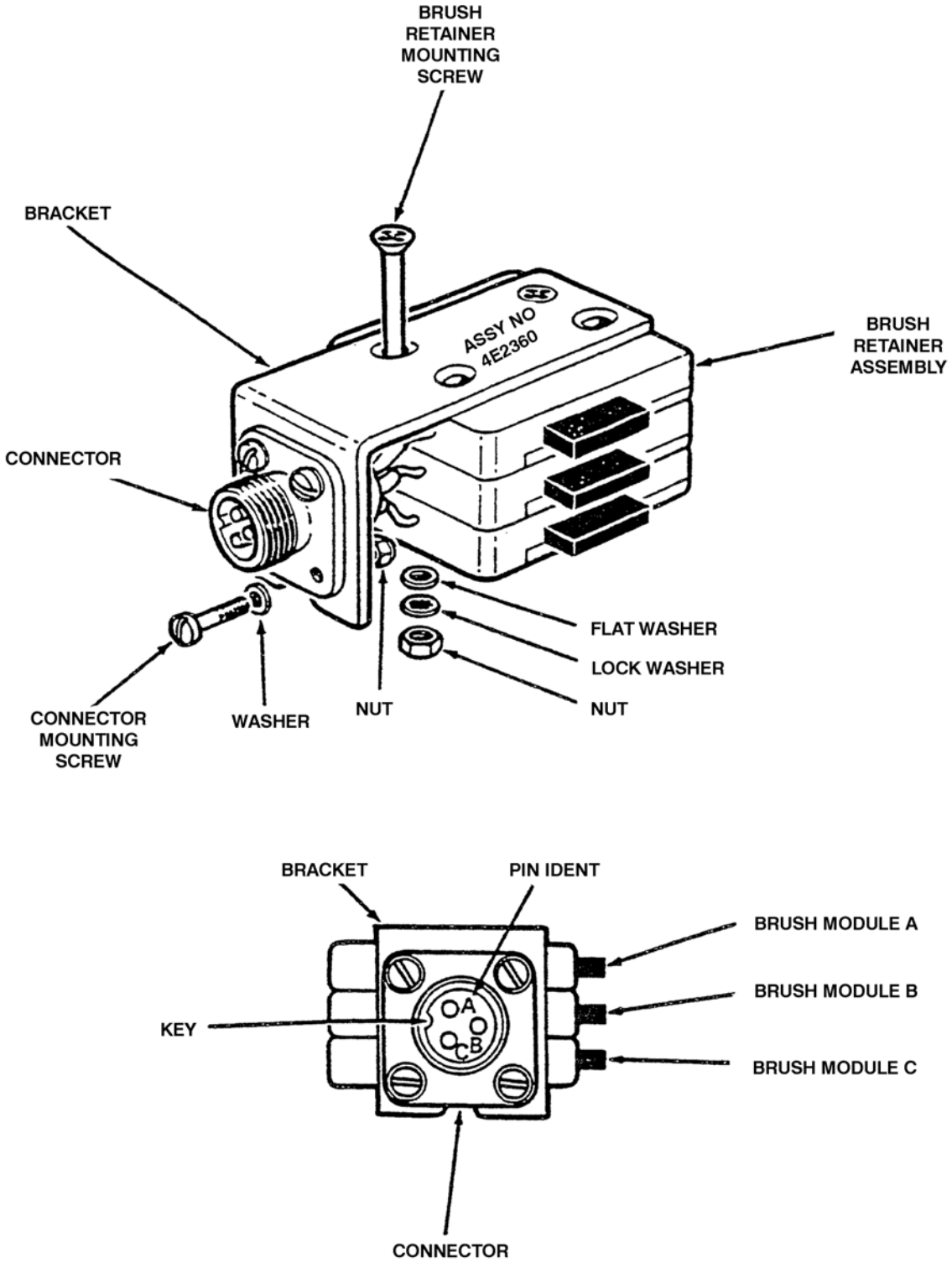
6) Ensure brush modules are square to one another then tighten screws.

7) Check brushes for freedom of movement.

8) Ensure that connector orientation to bracket is as noted in Para 1.A.2).

9) Secure connector to bracket with four screws, washers and nuts.

10) Install brush module assembly on engine mounting bracket, refer to Para 2.B.



Brush Repair Assembly – Removal/Installation
Figure 1

2. Removal/Installation

Refer to Figure 2 for details.

A. Remove Brush Module Assembly

1) Ensure that PROPELLER DEICE circuit breaker is disengaged.

2) Disconnect electrical connector from brush module assembly.

3) Remove and retain two nuts, four washers and two bolts then remove brush module and shim (if shim is installed). Retain shim.

B. Install Brush Module Assembly

CAUTION **FAILURE TO EXERCISE EXTREME CARE WHEN HANDLING, OR IMPOSING SIDE LOADS ON BRUSHES DURING INSTALLATION COULD RESULT IN BRUSH DAMAGE.**

1) Position brush module assembly on engine mounting bracket and insert bolts with washers under bolt heads through module and bracket. Do not install nuts and washers at this time.

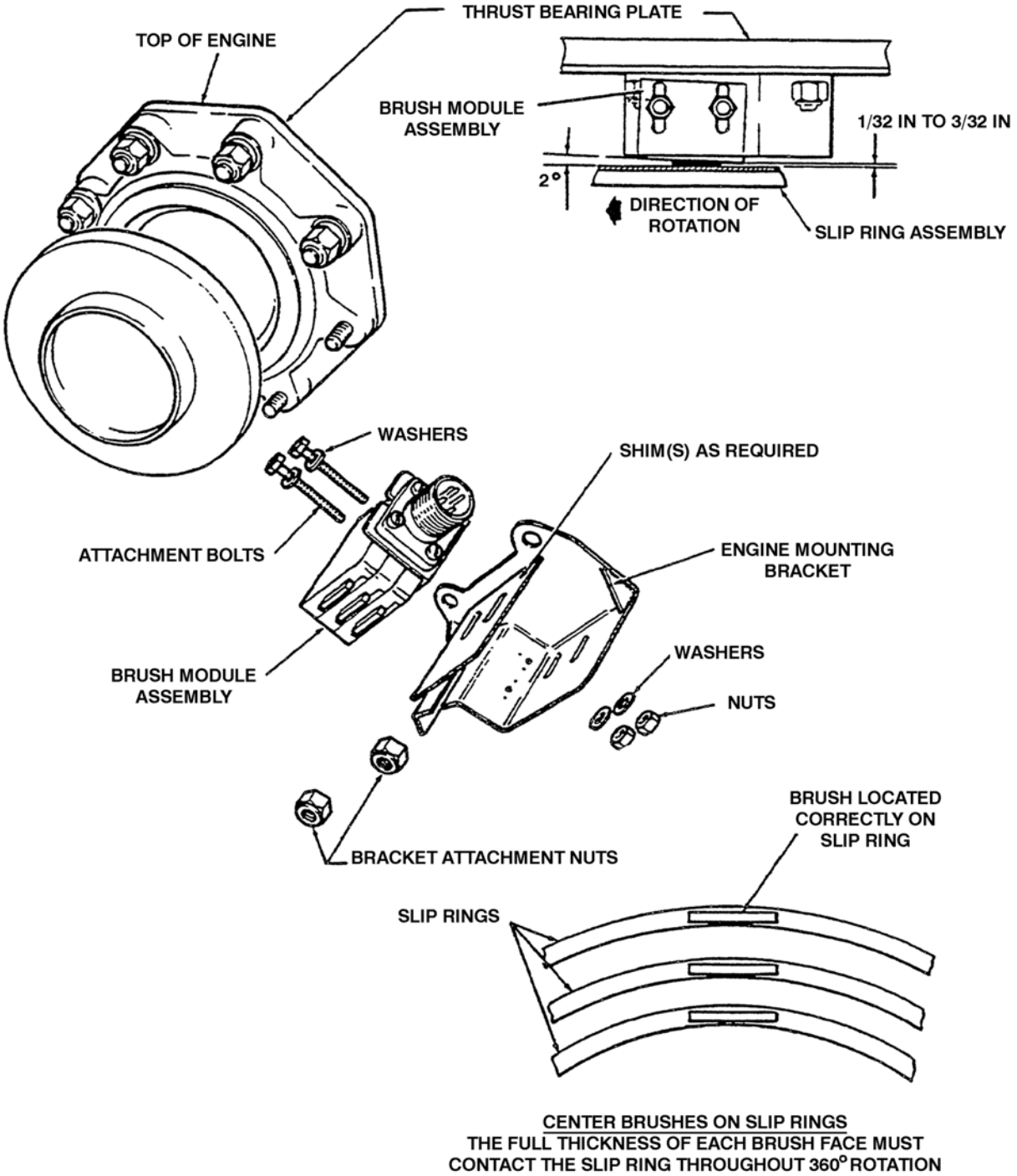
2) Check that brushes are aligned with propeller slip rings so that entire brush face is in contact with copper rings. Check alignment through entire 360 degrees of slip ring rotation. If not aligned install one or more shims as required to align brushes to approximate center of slip rings.

3) Install nuts and washers, and tighten to secure brush module assembly to engine mounting bracket. A minimum distance of 1/32–inch and a maximum of 3/32–inch must be maintained between brush module assembly and slip ring surface.

4) Brush module assemblies must be positioned so that brushes contact slip rings at an angle of approximately 2°, to prevent chattering that could cause brushes to break up (refer to Figure 2). The angle is measured toward the direction of rotation of slip rings.

Note On all installations, whether new or replacement, allow a run-in period of at least one hour before operating the propeller de-icing system.

5) Connect electrical connector to brush module assembly.



Brush Module Assembly – Removal/Installation
Figure 2

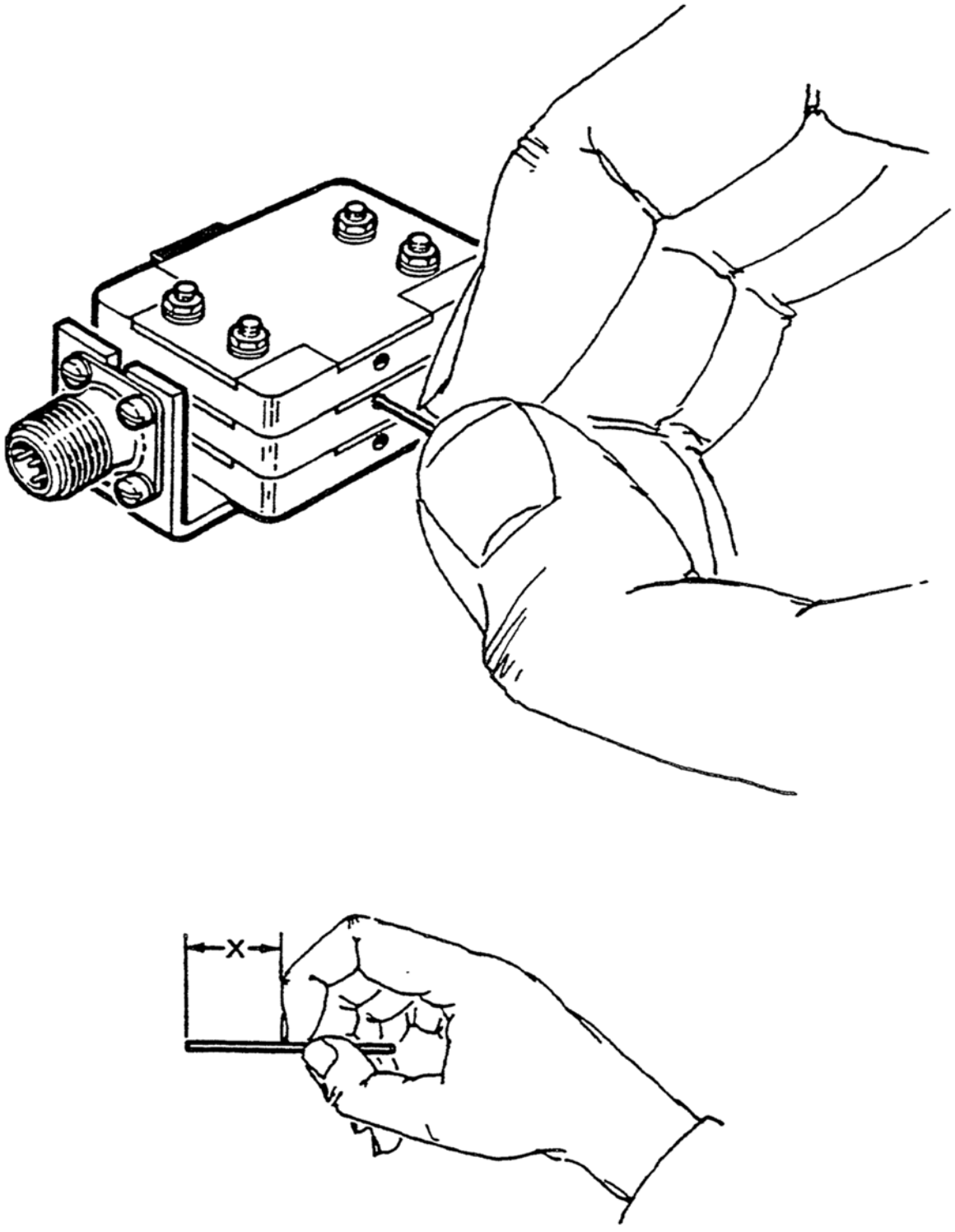
3. Inspection/Check

Refer to Figure 3 for details.

A. Check Brush Module Assembly, Brush Wear

1) Measure brush wear as shown in Figure 3. Replace brush retainer assembly when dimension X of any module is up to 23/64 inch maximum (Mod 6/1771).

Note During measurement allow only 1/16 inch of brush to protrude from module. This corresponds with normal brush extension when brush module assembly is installed on aircraft.

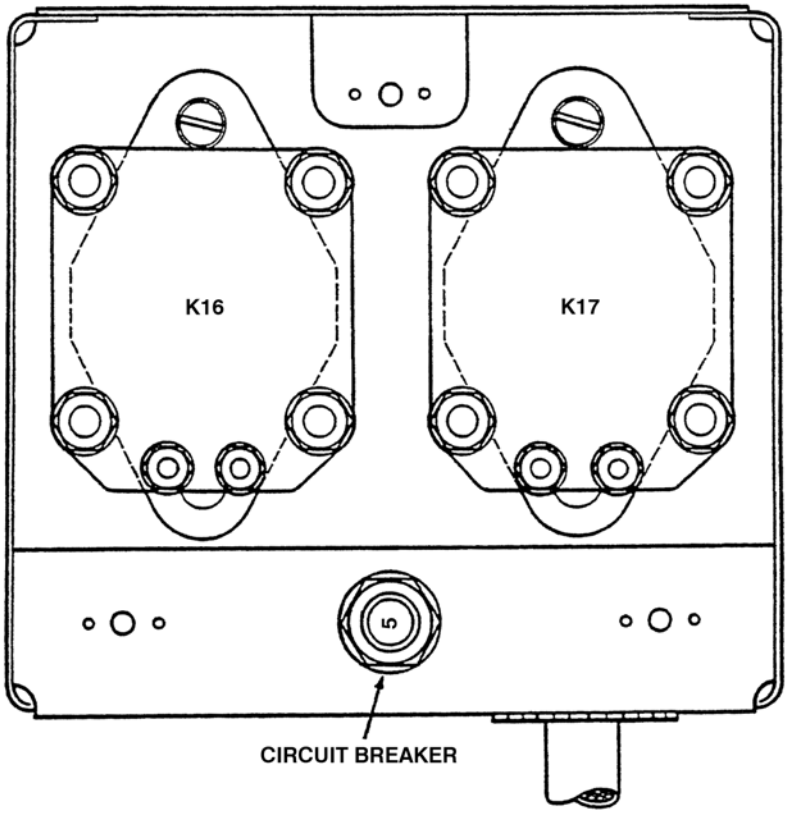


Brush Module Assembly – Checking Brush Wear
Figure 3

Propeller De-icing Control Box (Mod S.O.O. 6202) – General Data

1. Description

The Propeller De-icing Control Box, P/N C6NF1015 (Figure 1) consists of a rectangular aluminum alloy box which houses two control relays (K16 and K17) and a 5–ampere circuit breaker. The box is closed by a cover secured by three screws and washers, and all electrical leads are routed out of the box through a hole fitted with a grommet. The control box is located in the cabin roof.



Propeller De-icing Control Box
Figure 1

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Propeller De-icing Timer – Maintenance Practices

1. Adjustment/Test

A. Functional Test Propeller De-icing Timer

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Disconnect connector plug from timer.

3) Connect a jumper wire from pin B of timer receptacle to terminal B of the connector plug, and from pin G of the timer receptacle to ground.

4) Select PROP DE-ICE switch ON.

5) Using a voltmeter, check DC voltage to ground from pin B of the timer. This should be approximately 24 Volt DC.

6) Check DC volts to ground from timer pins C, D, E and F.

Note Pins C, D, E and F are the points at which the system voltage is impressed in sequence to cycle power to the propeller de-icers.Table 1 — Timer Cycling Action shows the cycling action of the timer.

7) Check pins C, D, E and F until a voltage reading of approximately 24 Volt DC is obtained. Hold the voltmeter probe on the pin until the voltage drops to zero. Move the probe to next pin in sequence and check DC volts at each pin. 24 volts should be measured at each pin.

8) Check to ensure that while one pin is energized, that other three pins have zero voltage on them.

9) Select PROP DE-ICE switch OFF.

10) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

11) Remove jumper wire, connect and secure plug to timer.

Table 1 Timer Cycling Action

TIMING SEQUENCE	APPROXIMATE TIME ON*	AREA OF PROPELLER DE-ICERS HEATED
Pin C	34 sec	Right engine propeller, outboard halves
Pin D	34 sec	Right engine propeller, inboard halves
Pin E	34 sec	Left engine propeller, outboard halves
Pin F	34 sec	Left engine propeller, inboard halves
* On time is approximately 34 seconds when operating on normal electrical system voltage. When operating on battery voltage, as in this test procedure, on time may be longer.		

Note The timer does not reposition itself to start at pin C when the system is switched off, but will begin to cycle at the next position from where it was switched off.

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CHAPTER 31

INDICATING /RECORDING SYSTEMS

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CHAPTER 31 – INDICATING/RECORDING SYSTEMS

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

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	* 11	Sep. 28/18		* 3	Sep. 28/18
	* 12	Sep. 28/18		* 4	Sep. 28/18
31-44-00	* 1	Sep. 28/18	31-60-00	* 1	Sep. 28/18
	* 2	Sep. 28/18		* 2	Sep. 28/18

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CHAPTER 31 – INDICATING/RECORDING SYSTEMS

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Instruments – General Data

1. General

The aircraft indicating and recording systems comprises of the flight and navigation system information, the Data Recording System, the Central Computing System, the Master Warning and Caution System, the instrument and control panels, the overhead panel and the independent instruments.

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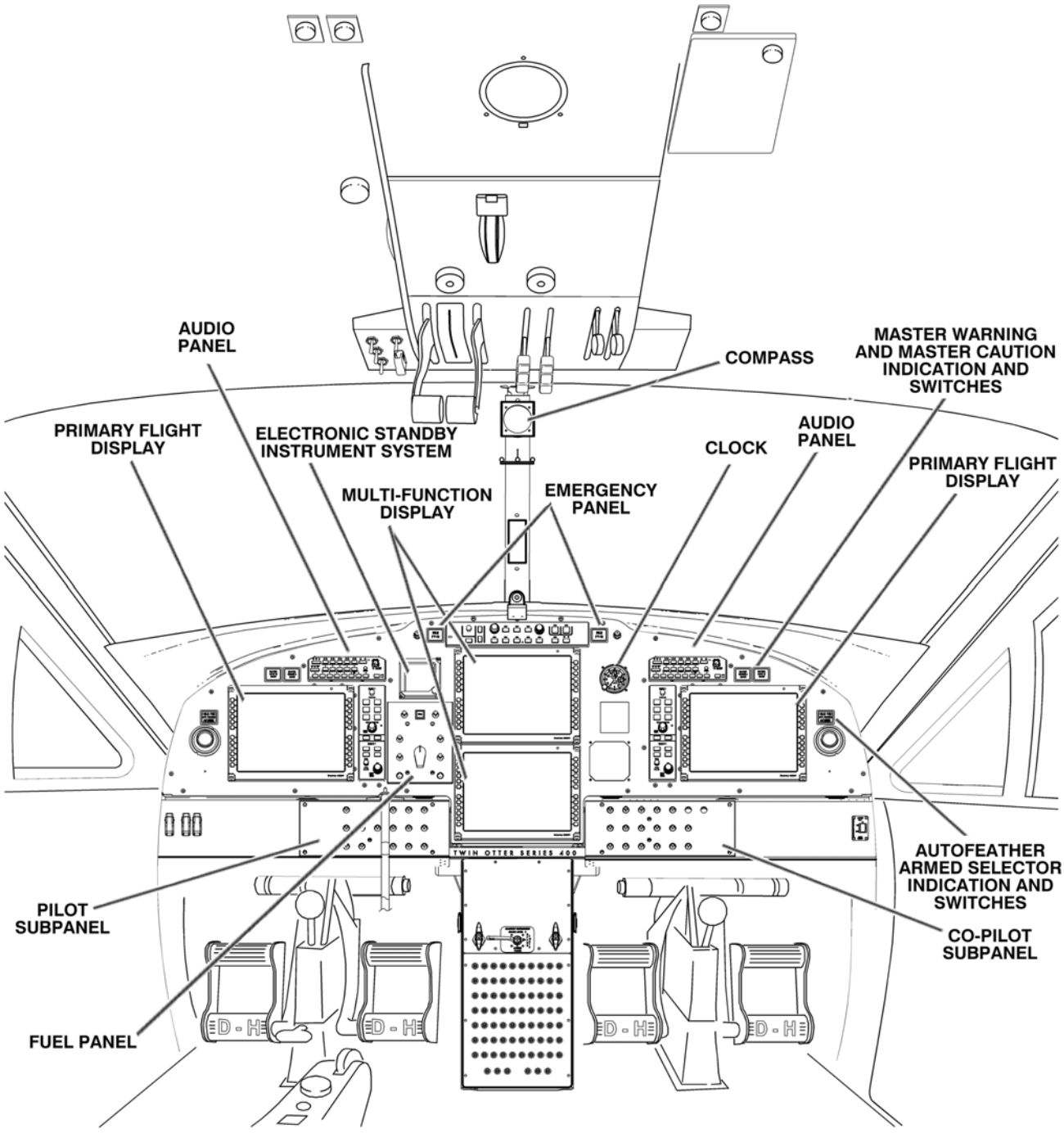
Instrument and Control Panels – General Data

1. General

Refer to Figure 1 for details.

The instrument and control panels are comprised of the Pilot's and Co-pilot's Primary Flight Displays, the Electronic Standby Instrument System Display, the fuel panel, stall caution light, clock, master warning/caution indication, the propeller autofeather control switch and indication, Pilot's and Co-pilot's sub panels, the emergency panel, the upper and lower Multi-Function Displays.

The instrument panel is located across the front of the flight compartment below the windshield and secured to the A frame which is shock mounted to the aircraft. The area between the multi-function display panels and the Co-pilot's Primary Flight panel is reserved for customer standard option radio equipment. Refer to 23-00-00, Communications – General Data.



Instrument Panel
Figure 1

2. Description

A. Primary Flight Display (PFD)

The Pilot's and Co-pilot's Primary Flight Display shows all of the essential flight and navigation information, the engine instruments and radio tuning settings. Each Primary Flight Display screen presents, under normal operations, four predefined windows.

The Flight Display Window is comprised of the following displays:

- Attitude Director Indicator (ADI) Display
- Horizontal Situation Indicator (HSI) Display
- Engine Instruments Display
- Radio Tuning Display

28 Volt DC power is protected by a 5 Amp PILOT PFD (H1) (Left Bus) and COPILOT PFD (H2) (Right Bus) circuit breaker, located on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

1) Attitude Director Indicator (ADI) Display

Refer to Figure 2 for details.

The Attitude Director Indicator (ADI) Display contains the essential flight information such as aircraft Pitch/Roll Attitude, Slip/Skid, Airspeed, Altitude, Vertical speed, Altimeter barometric setting, Rate of Turn indication, Composite lateral and vertical deviations.

Airspeed information displayed is:

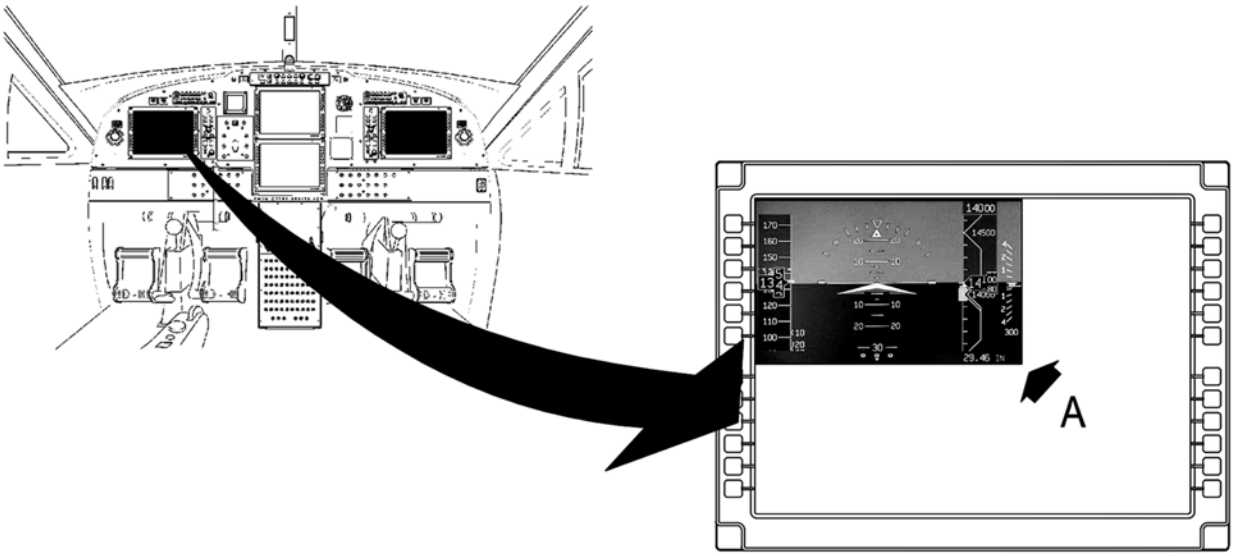
- Selected speed reference digital readout and bug
- Indicated airspeed tape and rolling digits
- Airspeed trend vector
- Maximum Operating Speed Indication
- V_{SPEED} bugs
- Flaps and gear speed bugs
- Tape markers for V_{YSE} and V_{MC}
- Low speed awareness bar and Stall annunciation
- Airspeed target bug

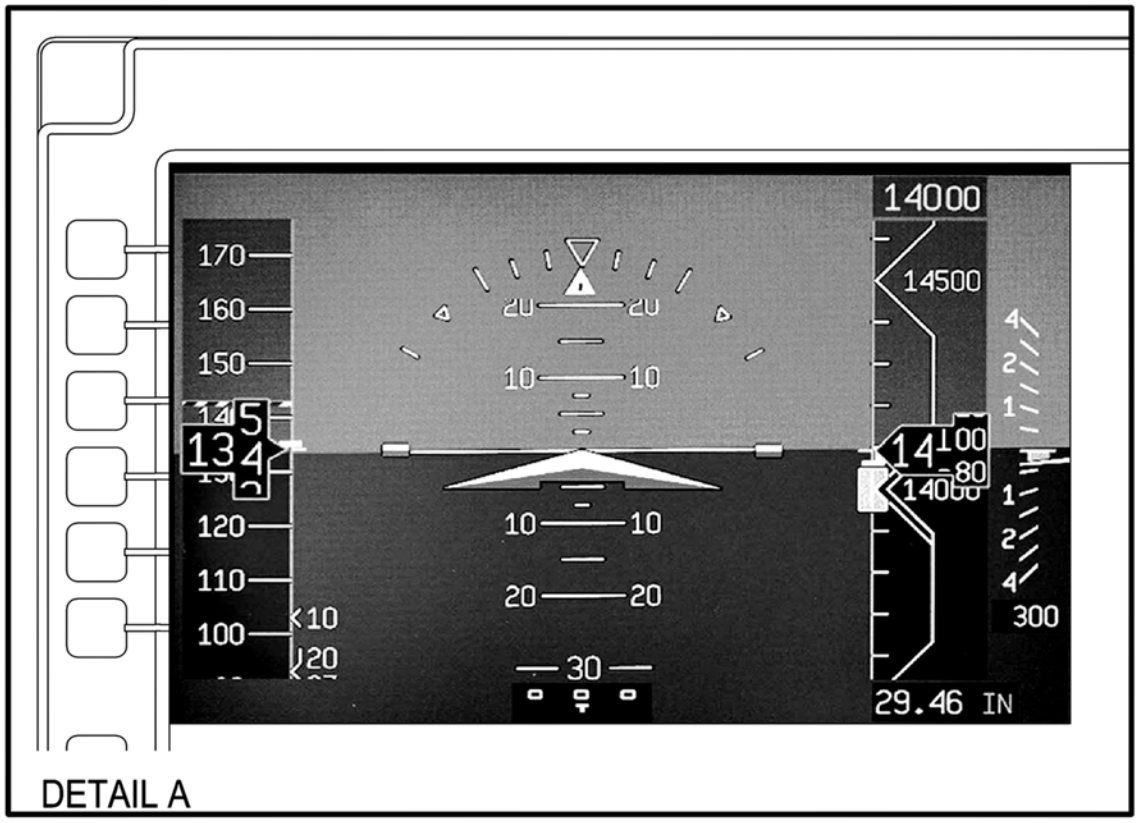
Altitude information displayed is:

- Altitude pitch tape
- Altitude rolling digits
- Metric altitude (when selected)
- Altitude trend vector
- Altitude selection and bug
- Radio attitude digital readout
- Barometric Altimeter Correction
- Minimum descent attitude and bug

Vertical Speed information displayed is:

- Vertical Speed Scale and pointer
- Vertical Speed digital readout
- Vertical Speed Target and bug





Attitude Director Indicator (Pre Mod 6/2303)
Figure 2

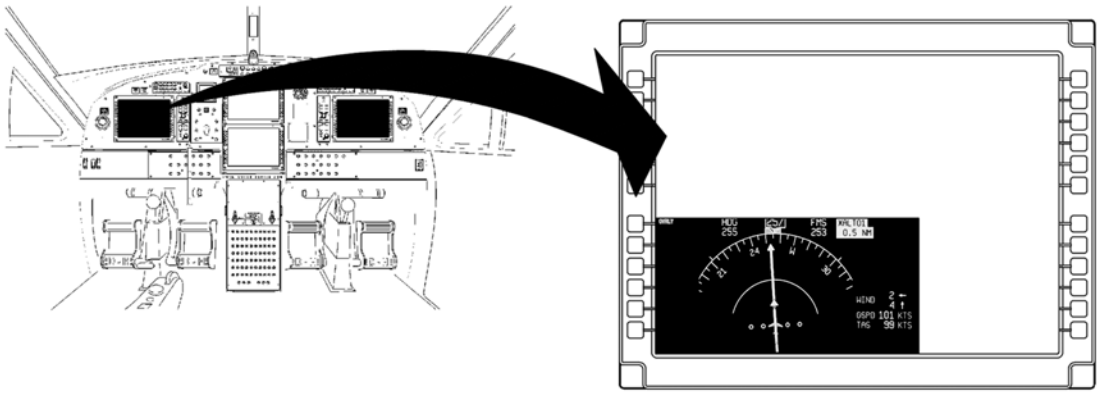
2) Horizontal Situation Indicator (HSI) Display

Refer to Figure 3 for details.

The Horizontal Situation Indicator (HSI) Display contains the navigational information required by the pilot or co-pilot. The information displayed is wind speed, ground and true airspeed, full and partial compass heading with selected course track and digital readout, lateral deviation indication and navigation system generated annunciations, refer to Monitor Warning System, 31-50-00, Monitor Warning System – General Data. Information also displayed as an HSI overlay includes TAWS generated alerts, TCAS, and Weather Radar. The navigational information is generated from several sources; the VOR/LOC, ADF, or FMS.

Information that may be overlaid on the HSI includes:

- Weather Overlay
- Terrain Awareness Display Overlay
- LX (lightening) Overlay
- TCAS



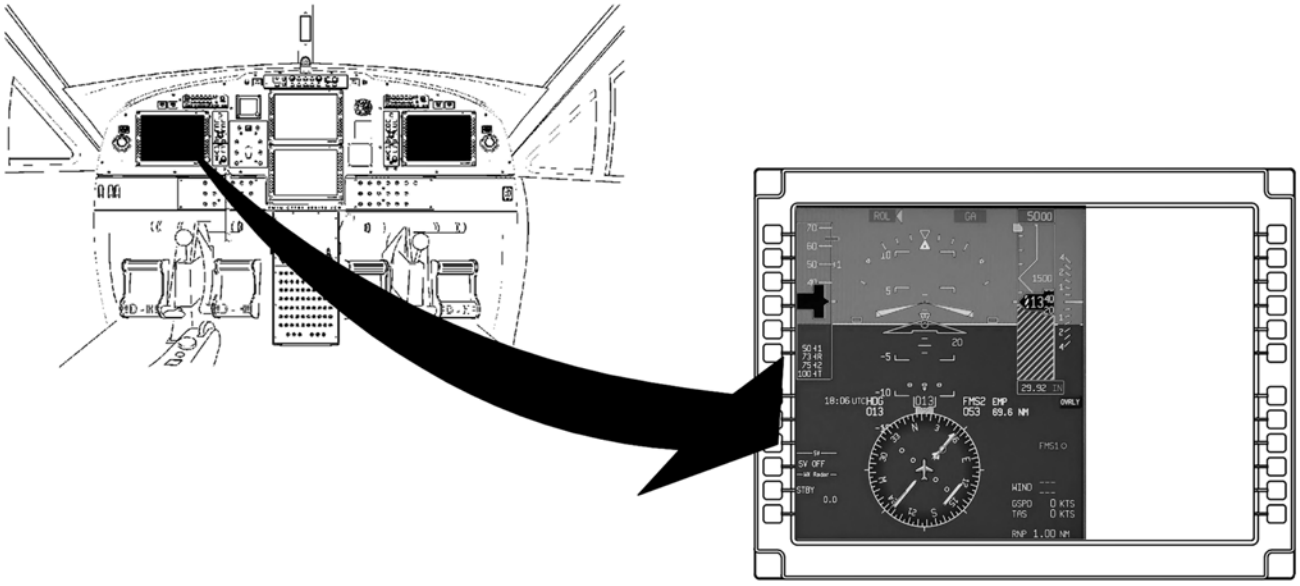


Horizontal Situation Indicator (Pre Mod 6/2303)
Figure 3

3) Integrated Primary Flight Display (IPFD) (Mod 6/2303)

Refer to Figure 4 for details.

Aircraft incorporating Mod 6/2303, have an Integrated Primary Flight Display (IPFD) which enhances the images presented on both PFD's, to provide a more representative aspect ratio from the pilot's eye point blue over brown representation. The IPFD combines traditional ADI and HSI displays into one window, allowing the attitude display to continue behind the HSI. The aspect ratio of the IPFD presents a pitch range of +15 to –20 degrees, relative to the aircraft's current attitude.



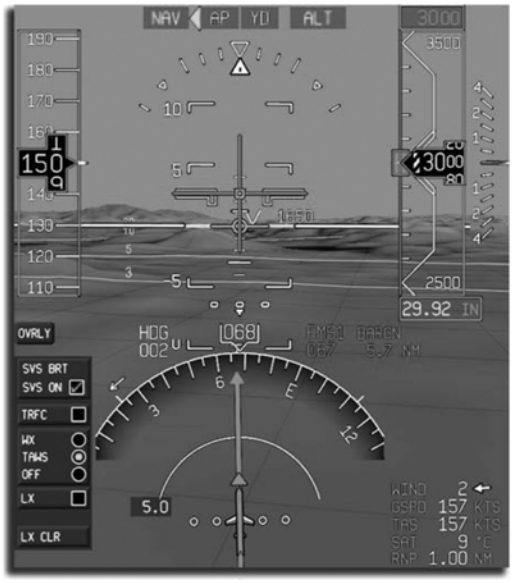
Attitude Director and Horizontal Situation Indicators (Post Mod 6/2303)
Figure 4

4) SmartView Synthetic Vision System (S.O.O. 6222)

Refer to Figure 5 for details.

Provisions have been made for SmartView Synthetic Vision System (SVS) to be enabled through the customer standard option S.O.O. 6222.

The SVS is automatically activated and displayed when the aircraft is powered on. The SVS is turned on and off by selecting or deselecting the SVS ON checkbox from the OVRLY menu. When the SVS ON checkbox is deselected, the standard IPFD standard blue over brown representation replaces the terrain on the PFD. SVS provides a computer generated three dimensional view of the terrain forward and below the aircraft's flight path, obstacles and runways, in addition to standard flight and navigation data. Aspect ratio of the SVS display matches that of the IPFD and more correctly represents the view of the outside world to enhance pilot situational awareness.



SmartView Synthetic Vision System (SVS) (S.O.O. 6222)
Figure 5

5) Engine Instrument Window

Refer to Figure 6 for details.

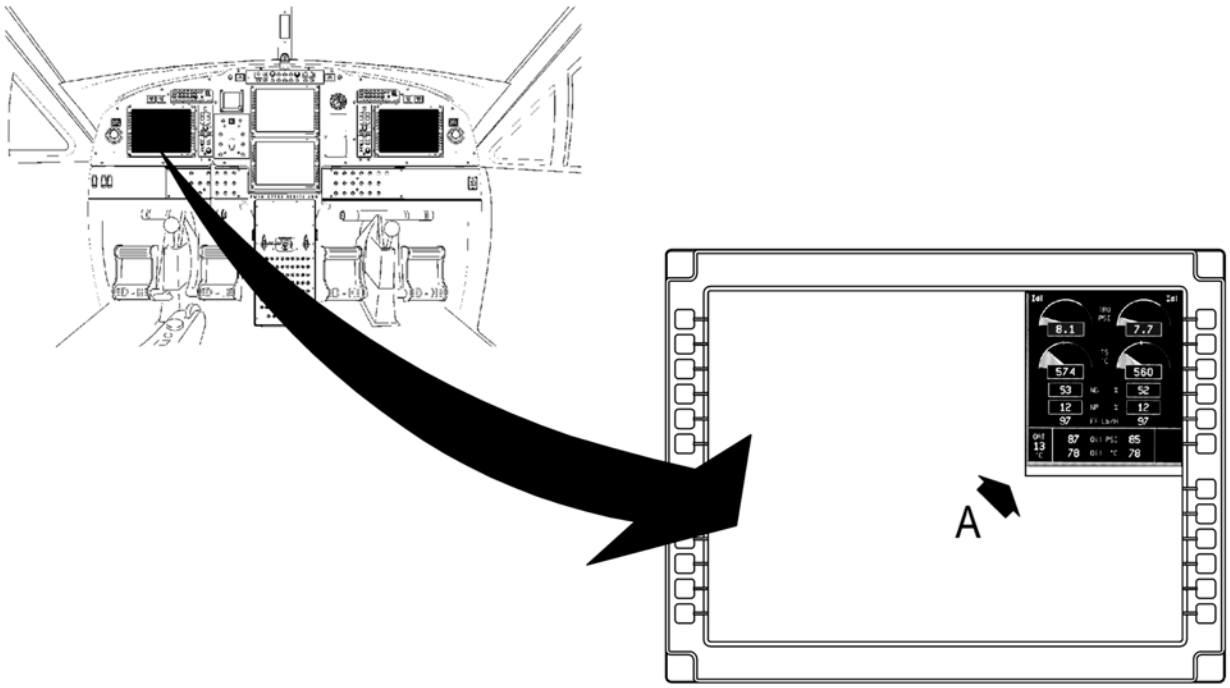
The engine instrument displays two sets of engine operating data, one for each engine.

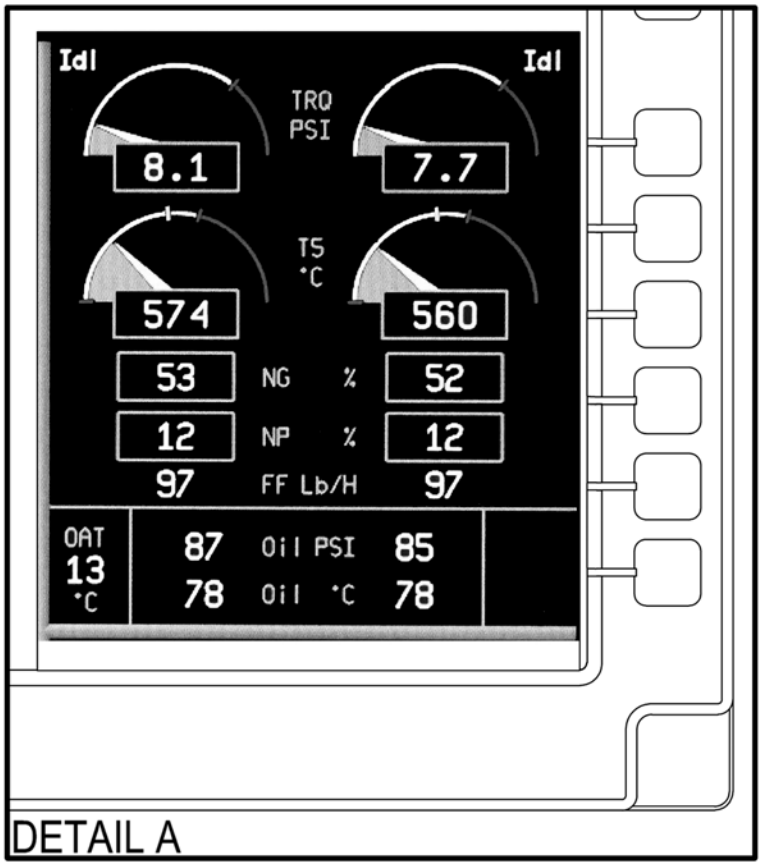
The engine instrument window information displayed is:

- Torque pressure indication
- Propeller tachometer indication
- Gas generator indication
- Fuel flow indication
- Turbine temperature indication
- Oil pressure indication
- Oil temperature indication
- Intake deflector indication
- Outside air temperature
- Engine Limits annunciation
- Start annunciation
- Anti-Ice annunciation (if installed)
- Propeller beta range annunciation
- Ignition annunciation

The torque pressure indication and the engine temperature indication are depicted by a pointer type display and integer values.

For aircraft incorporating S.O.O. 6280, FAA Engine Display Colours, the normal engine operating range parameters are displayed with a green arc or green line.





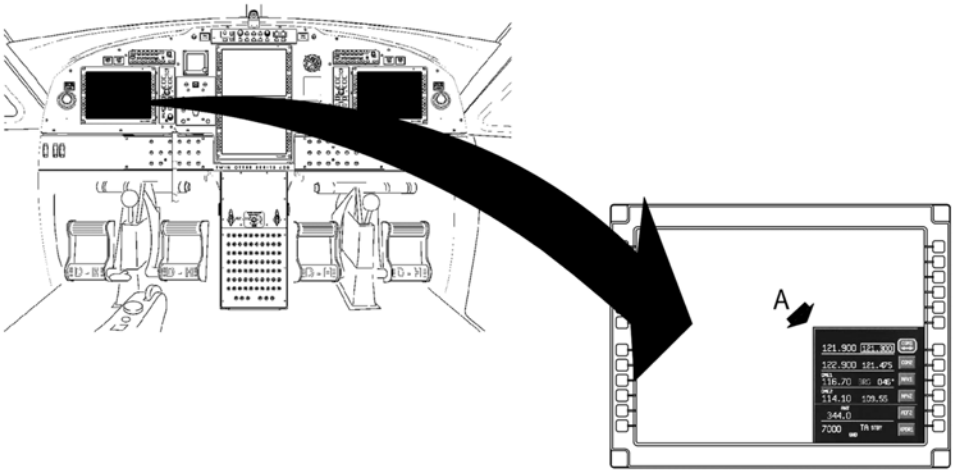
Engine Instrument Window
Figure 6

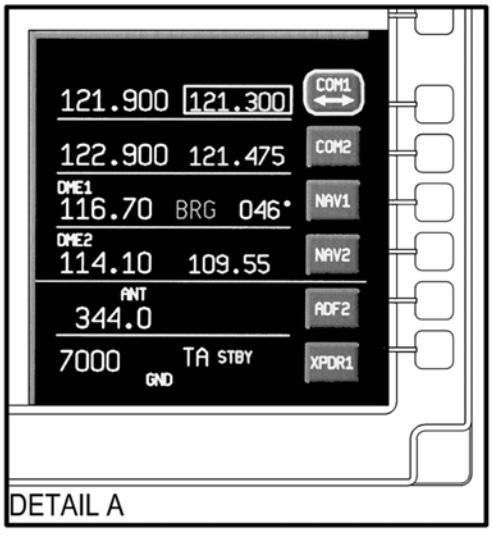
6) Radio Tuning Window

Refer to Figure 7 for details.

The radio tuning window is an interactive window which displays the selected operation frequencies for the following systems:

- COM 1
- COM 2
- NAV 1
- NAV 2
- Selected ADF
- Selected XPDR



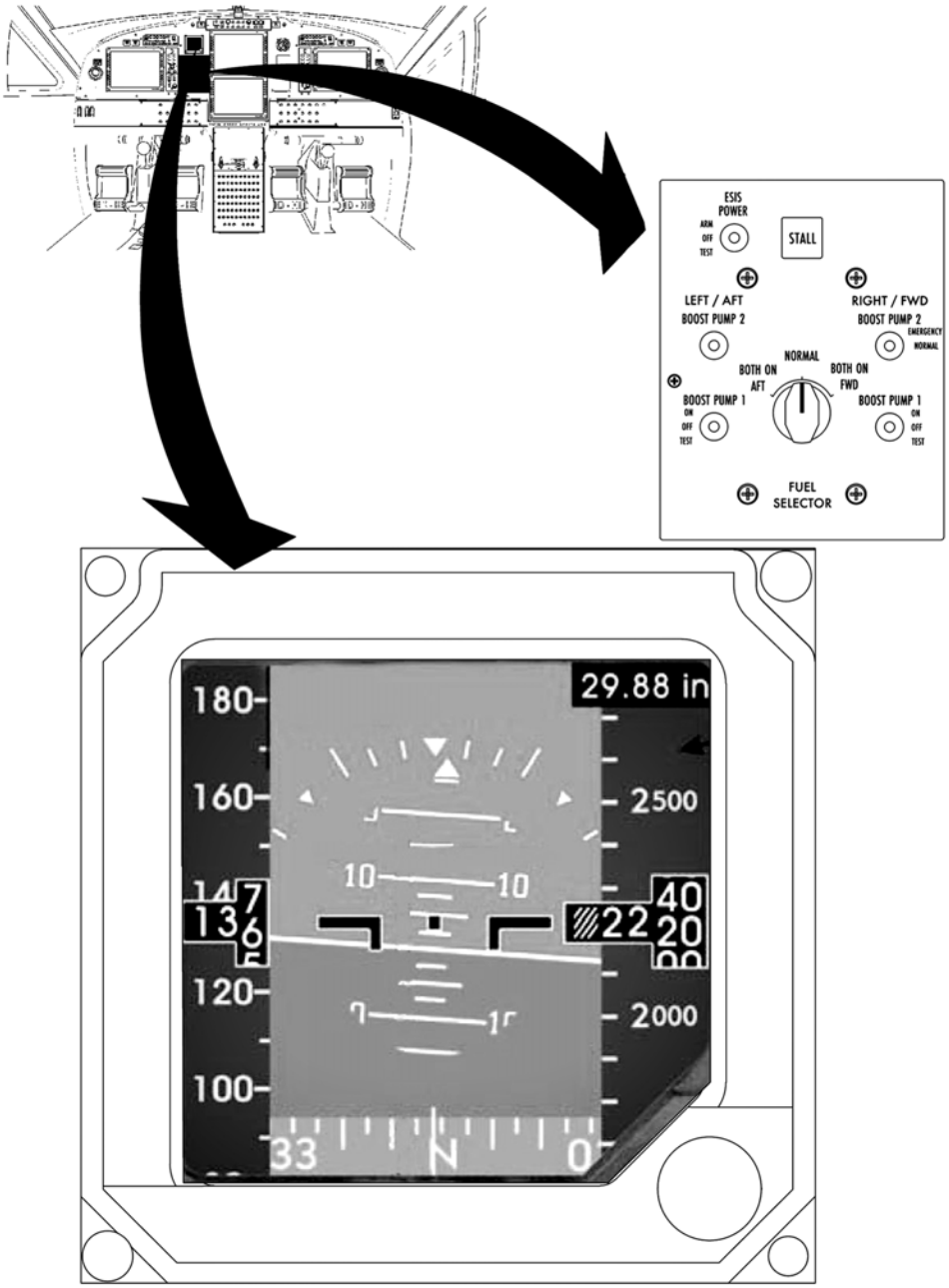


Radio Tuning Window
Figure 7

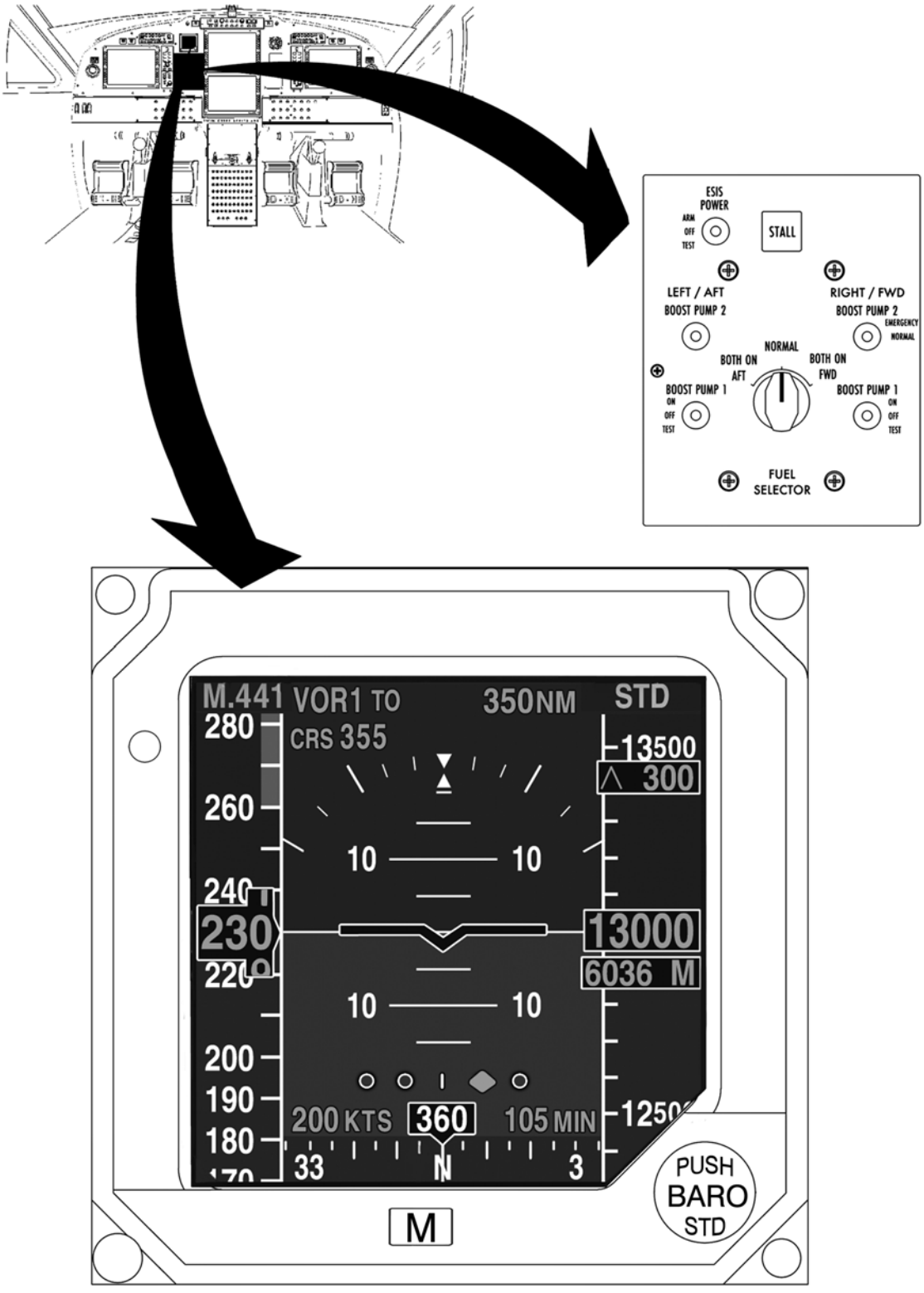
B. Electronic Standby Instrument System (ESIS)

Refer to Figure 8 and Figure 9 for details.

The ESIS instrument is powered by its own dedicated backup battery, and is entirely independent of the Apex system. When fully charged, the backup battery has sufficient capacity to operate the ESIS for at least 6 hours. Refer to 31-20-21, Electronic Standby Instrument System (ESIS) (Mod S.O.O. 6209, Mod 6/2049 and Mod 6/2170) – General Data.



L3 GH-3100 Electronic Standby Instrument System (Pre Mod 6/2170)
Figure 8



L3 GH-3900.2 Electronic Standby Instrument System (Post Mod 6/2170)
Figure 9

C. Multi-Function Display (MFD)

1) The upper multi-function display, presents the situational awareness information, under normal operation.

The display is divided into 3 predefined windows:

- Way point list
- Flight plan data
- Flight plan map

Provisions have been made for Vertical Situation Display (VSD) to be enabled through the incorporation of customer standard option S.O.O. 6312. When selected, VSD depicts a profile view of the aircraft's predicted flight path in relation to the terrain along the planned route or track in the lower 1/4 of the INAV Map window. VSD function is selected via the soft key on the Map window. The VSD selection is saved (retained) through an aircraft power cycle.

28 Volt DC power is protected by a 5 Amp UPPER MFD (J1) (Left Bus) and LOWER MFD (J2) (Right Bus) circuit breaker, located on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

2) The lower multi-function display presents the system information under Normal operation, divided into predefined windows:

a) Crew Alerting System Window

The Crew Alerting System window displays the visual system warnings that are generated from system malfunctions or abnormal operating conditions. The visual warning may be accompanied by an aural warning. Refer to 31-50-00, Monitor Warning System – General Data.

b) Flight Control and Hydraulic Window

Flight Control and Hydraulic Window displays:

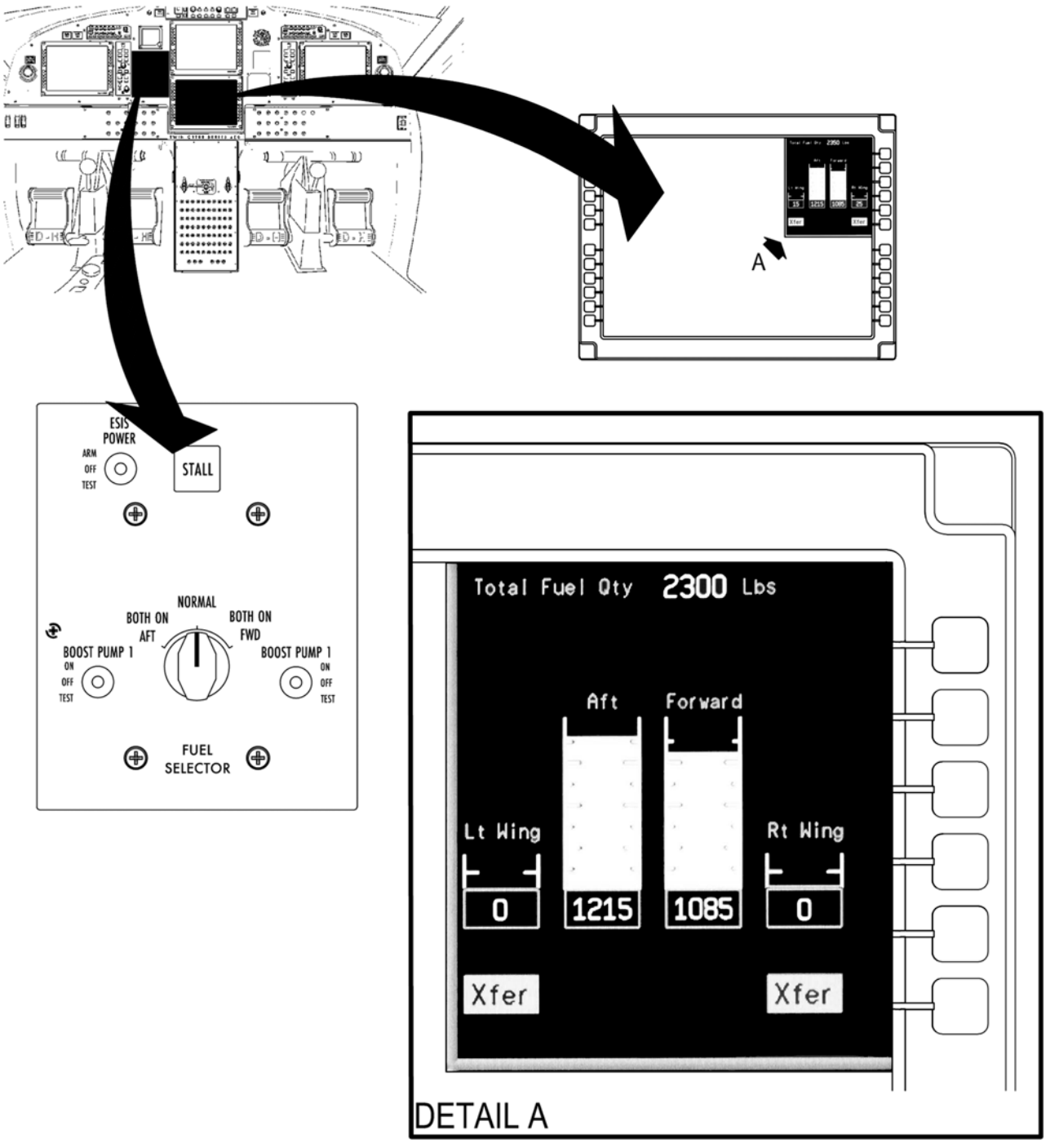
- Flap position Indication
- Rudder Trim Indication
- Aileron Trim Indication
- Pitch Trim Indication
- Hydraulic Pressure Indication
- Brake Pressure Indication

c) Fuel Quantity Window

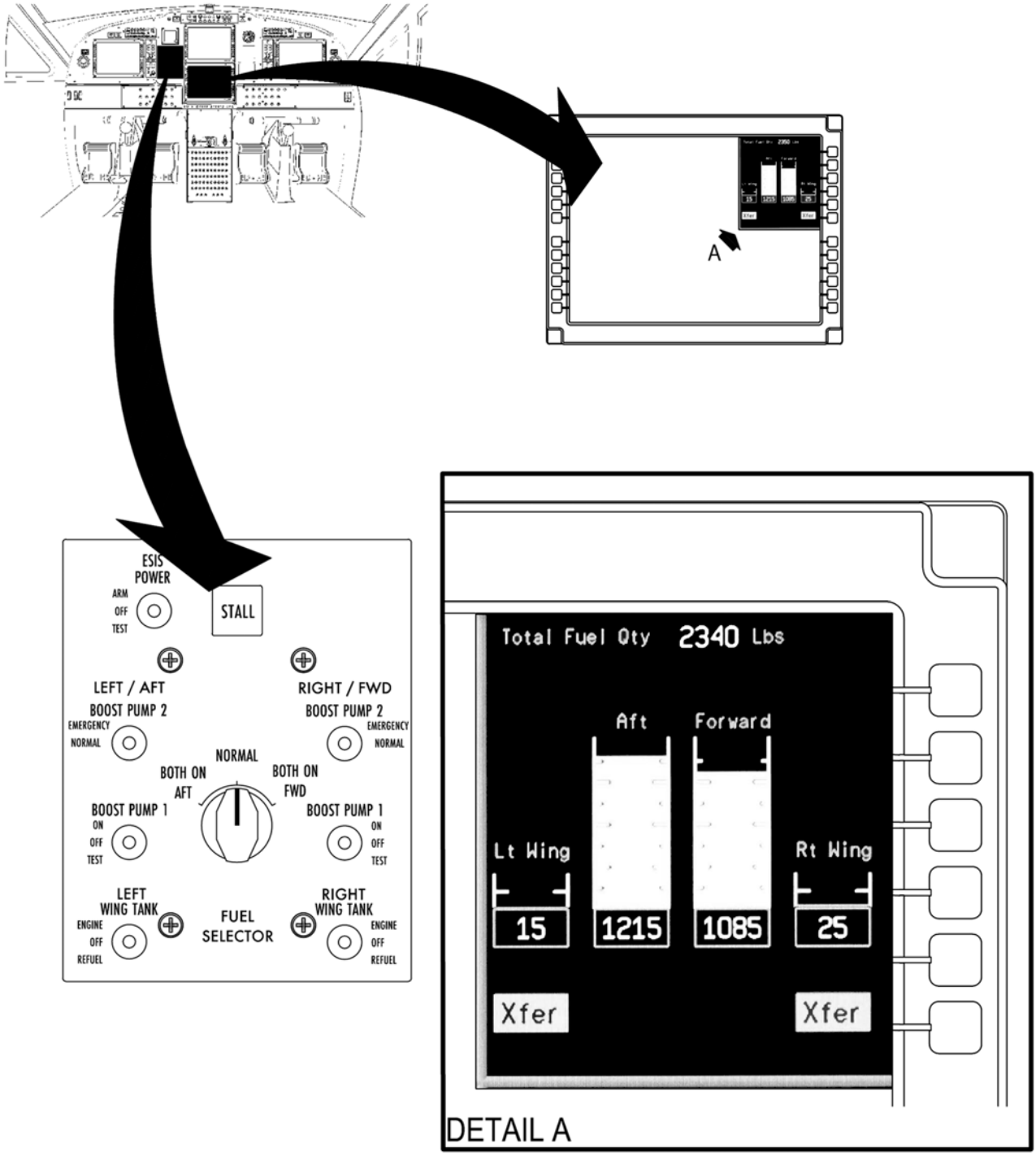
Refer to Figure 10 for details.

Fuel Quantity Window displays:

- Total Fuel Quantity in LBS
- Cross Feed Valve Status
- Fuel Tank Level Status for the Aft, Forward, Left and Right wing, (if installed) Tank status
- Boost Pump Status
- Wing Pump Status (if installed) (refer to Figure 11)



Fuel Quantity Window
Figure 10



Wing Pump Status (S.O.O. 6247)
Figure 11

d) Environmental Display

Environmental Window displays:

- Cockpit temperature
- Cabin temperature

e) Miscellaneous Window

Miscellaneous Window displays:

- Electrical System Indication
- Ice Protection status

f) Electronic Checklist Window

The electronic checklist window displays the predefined checklist but also can display the following windows:

- Navigation Sensor Window
- Weather/Lightning/Terrain Awareness and Warning System Window
- Avionics Window
- Software configuration and data loading window

D. Fuel Panel

The fuel panel contains the fuel control switches, the standby booster pump switches and the Stall Warning Light. The panel is internally lit with LED lights. (Refer to 33-00-00, Lights – General Data).

E. Clock

The clock is mounted on the instrument panel right of the multi-function displays. Refer to 31-20-11, Clock (Mod 6/2049) – General Data.

F. Master Warning and Master Caution Indication and Switches

Two 1 inch by 1.2 inch LED switchlights, internally labelled MASTER CAUTION and MASTER WARNING, are provided directly above each pilot's primary flight display, refer to 31-50-00, Monitor Warning System – General Data.

G. Autofeather Switch/Indicators

Two identical propeller autofeather switches are provided, located outboard of each pilot's primary flight display. When activated (accomplished by pressing either switch, selected autofeather armed) electrical power is supplied to the autofeather system as indicated by the illumination of SELECT on two switches, refer to 61-22-02, Propeller Autofeather System (Mod 6/2022) – General Data.

H. Pilot's Subpanel

The Pilot's Subpanel contains the System Test switches, the control switches for the Ice & Rain, External Lighting and Internal Lighting. The panel is backlit with LED lights. (Refer to 33-00-00, Lights – General Data).

I. Co-pilot's Subpanel

The Co-pilot's Subpanel contains the control switches for Environmental, Surface De-Ice, Emergency and the Artex ELT Arming switch. The panel is backlit with LED lights. (Refer to 33-00-00, Lights – General Data).

J. Emergency Panel

The emergency panel is a designated area at the top of the instrument panel where the left and right fuel emergency shut-off switches, the integral fire warning light/extinguisher push buttons are located. (Refer to 26-10-00, Fire Detecting System – General Data).

3. Removal/Installation

A. Removal of Flight Instrument Panel

1) Ensure all aircraft electrical power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Remove all instruments. (Refer to Instrument Chapters).

3) Disconnect electrical connectors, bonding wire and pressure lines. Cap connectors.

4) Support panel and remove mounting screws.

5) Remove panel from aircraft.

B. Installation of Flight Instrument Panel

1) Install panel in aircraft.

2) Support panel and install mounting screws.

3) Uncap and connect electrical connectors, bonding wire and pressure lines.

4) Install all instruments. (Refer to Instrument Chapters).

C. Removal of the Lower Shockmounts

1) Remove instrument panel, refer to Para 3.A.

2) Remove rivets securing shockmount to structure by carefully drilling out manufactured rivet head and tap out shank with suitable punch.

D. Installation of the Lower Shockmount

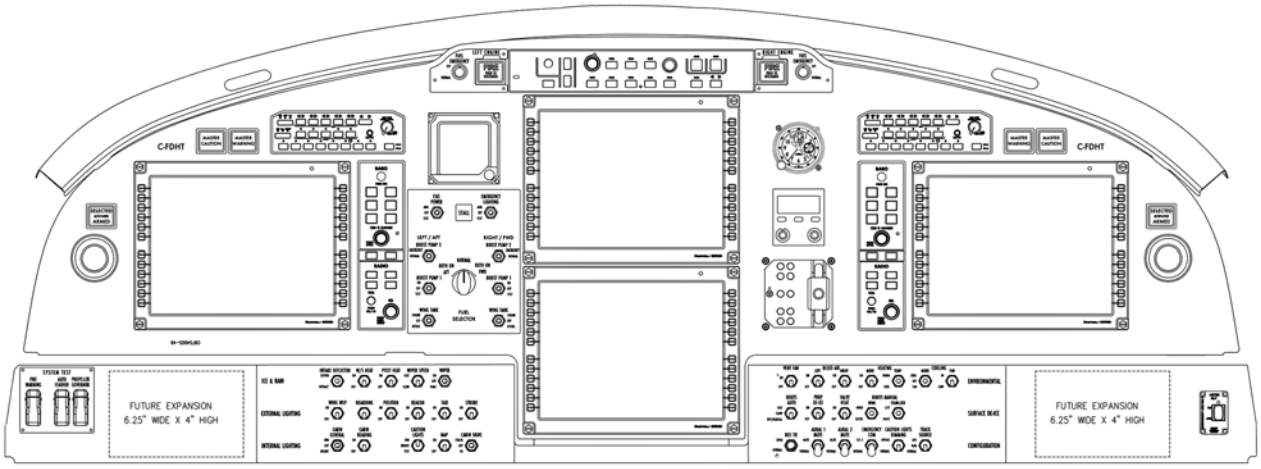
1) Position shockmount on structure mounting point as the removed one and secure with four rivets (MS20470–AD5).

2) Install instrument panel, refer to Para 3.B.

Display Units – General Data

1. General

The Apex Avionics suite is based on a four Display Unit (DU) layout arranged in a T configuration, refer to Figure 1.



Primus Apex Display Units (T–Configuration)
Figure 1

The displays are manufactured by Barco for Honeywell. Replacement displays must be designated Honeywell KDU 1080 display units. For alternative screen model and part numbers suitable for use with the DHC-6 Series 400 installation, contact Viking Air Limited.

Additional descriptive material and operating instructions are provided in the DHC-6 Series 400 Pilot Operating Handbook (POH), PSM 1-64-POH, Section 7 and in the Honeywell Apex Pilot's Guide.

2. Description

A. Display Units (Honeywell Model KDU 1080)

There are four interchangeable displays units installed on the instrument panel:

DU 1 – Pilot's Primary Flight Display, PFD 1

DU 2 – Upper Multi-Function Display, MFD 1

DU 3 – Lower Multi-Function Display, MFD 2

DU 4 – Co-pilot's Primary Flight Display, PFD 2

The DU is a 10 x 8 inch (10.4 inch diagonal display area) LCD display screen with integral LED backlighting. The display unit does not contain any system software. The display units are driven by the Apex AGMs installed in the MAU (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data). The display screen as currently installed on the aircraft does not contain integral heaters. Optional installations may be available in the future that approve this option.

The display screen is divided into 1/6 or 2/3 size sections referred to as windows. Each window has a default display and functionality defined. Each window functions independently from the other windows. This allows multiple system operation and functions to be displayed on one DU at the same time, and also to maintain independence and a high degree of software reliability in each window.

Each type of display unit has a designated controller which allows the operator to navigate within the windows on the screen and set the format for the system information to be displayed. For the PFDs, the associated controllers are Primary Flight Display controllers, and the two MFDs are controlled through the Multi-Function Controller.

Bezel lighting on each DU is controlled by dimming potentiometers labelled INSTRUMENT in the centre pedestal controller mounted on the control column near the MFD Controller.

Display brightness is controlled by the Display Reversion Control Panel. The brightness of each DU may be controlled independently of the other DUs.

B. Primary Flight Displays (PFDs)

The Pilot's PFD is installed on the left side of the instrument panel. 28 Volt DC power is provided from the left bus through a 5 ampere PILOT PFD (H1) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The Co-pilot's PFD is installed on the right hand side of the instrument panel. 28 Volt DC power is provided from the right bus through a 5 ampere COPILOT PFD (H2) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

C. Multi-Function Display (MFD)

The upper MFD is installed in the center of the instrument panel. The Upper MFD default format is the Situational Awareness Display.

28 Volt DC power is provided from the left bus through a 5 ampere UPPER MFD (J1) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The lower MFD is installed below the upper MFD in the centre of the instrument panel. The Lower MFD default format is the Systems Display.

28 Volt DC power is provided from the right bus through a 5 ampere LOWER (MFD) (J2) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

3. Adjustment/Test

A. Power-up

1) Power On

a) Connect external power to the aircraft (refer to 12-00-00, Servicing – General Data).

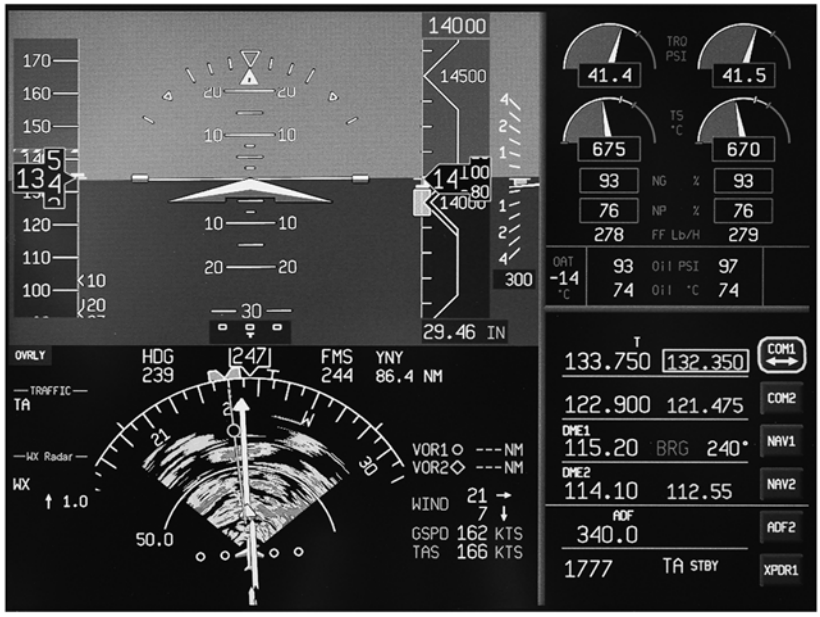
b) Ensure that the switches on the overhead panel are set as follows:

1 Select POWER SOURCE switch to EXTERNAL and MASTER switch to ON (Pre Mod 6/2314); or EXTERNAL switch to ON (Post Mod 6/2314).

c) Ensure that PFD 1, MFD 1, MFD 2 and PFD 2 come on.

Note Boot-up of the Apex system may take up to 24 seconds.

B. Left Primary Flight Display (PFD 1) (Pilot's)



Primary Flight Display
Figure 2

1) Ensure that the pilot's (left) PFD is on and shows:

   - ADI and HSI
   - Engine window
   - Radio window

2) Ensure that left PFD controller is working and can do its control functions on the PFD. Test functions such as navigation source select (including FMS), tuning VHF communication and navigation systems such as COM 1, COM 2, NAV 1, NAV 2, and tuning other systems such as ILS, ADF 1, ADF 2, DME 1, DME 2, Transponder 1, and Transponder 2. (Refer to Figure 2).

   Note Not all aircraft will have all of this equipment installed.

3) Using Reversion Control Panel, confirm pilot's PFD can be dimmed and that it can be turned off.

4) Using Reversion Control Panel, revert the pilot PFD to AGM 2.

5) Ensure the pilot PFD display matches the co-pilot PFD, and the co-pilot PFD reverts to the pilot PFD display format (engine and radio window on the RH side).

C. Upper Multi-Function Display (Upper MFD)

1) Ensure that the upper MFD is on and shows:

   a) Situational Awareness Display

2) Ensure that the MF controller is working and can focus on various displays on the upper MFD. Ensure bezel buttons, where enabled, are functioning.

3) Using Reversion Control Panel, confirm upper MFD can be dimmed and that it can be turned off.



Situational Awareness Display
Figure 3

D. Lower Multi-Function Display (Lower MFD)



Systems Display
Figure 4

1) Ensure that the lower MFD display is on and shows:

- The systems windows
- Fuel system window including left wing fuel quantity, aft fuselage belly tank fuel quantity, forward fuselage belly tank fuel quantity, and right wing fuel tank quantity (wing tanks are optional)
- Left and right generator voltages
- Left and right generator currents

- Hydraulic system pressure
- Brake system pressure
- Electronic checklist (ECL) (if option installed)
- CAS message window
- Ice protection system annunciations (if ice protection system installed)
- Cabin and cockpit temperatures
- Battery voltage and current

2) Ensure that the MF controller is working and can focus on various displays on the lower MFD.

3) Using Reversion Control Panel, confirm lower MFD can be dimmed and that it can be turned off.

E. Right Primary Flight Display (PFD 2) (Co-Pilot's)

Note The co-pilot's PFD is the mirror image, in terms of screen arrangement, from the pilot's (left) PFD (refer to Figure 2).

1) Ensure that the co-pilot's PFD is on and shows:

- ADI and HSI
- Engine window
- Radio window

2) Ensure that: right PFD controller is working and can do its control functions on the PFD. Test functions such as navigation source select (including FMS), tuning VHF communication and navigation systems such as COM 1, COM 2, NAV 1, NAV 2, and tuning other systems such as ILS, ADF 1, ADF 2, DME 1, DME 2, Transponder 1, and Transponder 2.

Note Not all aircraft will have all of this equipment installed.

3) Using Reversion Control Panel, confirm co-pilot's PFD can be dimmed and that it can be turned off.

4) Using Reversion Control Panel, revert the co-pilot PFD to AGM 1.

5) Ensure the co-pilot PFD display matches the pilot PFD (engine and radio window on the right side).

F. Miscellaneous Checks

1) Window Navigation and Focus

a) On the Multi-Function Controller (MF Controller), press the left arrowhead button.

b) Ensure that window focus (a cyan coloured box border) moves to the next available interactive window left of the current window having focus.

c) On the MF Controller, press the down arrowhead button.

d) Ensure that window focus moves to the next available interactive window below the current window having focus.

e) On the MF Controller, press the up arrowhead button.

f) Ensure that window focus moves to the next available interactive window above the current window having focus.

g) On the MF Controller, press the right arrowhead button.

h) Ensure that window focus moves to the next available interactive window to the right of the current window having focus.

i) Wait approximately thirty seconds. The window focus is removed after 30 seconds.

2) Cursor Display Check

a) On the MF Controller, press the COM button to move the window focus to the PFD 1 radio window.

b) Ensure that the cursor focus defaults to COM 1.

c) Use the MF Controller joystick and ensure that you can move the cursor focus to:

- COM 1
- COM 2

d) Put the cursor focus on XPDR and wait for approximately twenty seconds.

e) Ensure that the cursor focus moves to the first field that can be modified in the current window.

f) Use the PFD bezel buttons to select the different functions in the radio window. Ensure that you can select the different functions.

3) Detail Window Display

a) Ensure that the PFD radio window has window focus, and ensure that COM 1 (or COM 2) has cursor focus.

b) Press the DETAIL button on the PFD 1 Controller. Ensure that the COM details page opens. Press the ENT button on the MF Controller, or the SEL knob on the PFD Controller and ensure that the entered value is selected/activated.

c) Press the DETAIL button again and ensure that the:

- Details page closes
- Radio window opens
- Cursor focus is on the radio selected on the details page

G. Close Up

1) On the overhead panel, switch OFF:

a) Select MASTER and POWER SOURCE switch to OFF (Pre Mod 6/2314); or EXTERNAL switch to OFF (Post Mod 6/2314).

2) Disconnect external power source from aircraft (refer to 12-00-00, Servicing – General Data).

4. Removal and Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – PANEL/PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

Note The Display Units have additional chassis grounding requirements, refer to the Note in Para 4.A.3).

A. Primary Flight Display (PFD)

1) General

These procedures apply to the pilot and co-pilot's PFD.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open (5 ampere PILOT PFD (H1) for the left PFD or 5 ampere COPILOT PFD (H2) circuit breaker for the right PFD) circuit breaker and fit with safety clip.

c) Remove the screws that hold the PFD in the instrument panel.

d) Carefully pull the PFD out of the instrument panel to get access to the electrical connectors (refer to PSM 1-64-2W) on the PFD.

e) Disconnect the electrical connectors from the PFD.

f) Remove the PFD from the aircraft.

Note Stow connectors as required.

3) Installation

Note Instrument panel surface area under the display unit is used for electrical bonding of the display unit. Ensure bonding surface between display and instrument panel is clean and not contaminated. Clean if required to ensure bonding.

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the (5 ampere PILOT PFD (H1) for the left PFD or 5 ampere COPILOT PFD (H2) circuit breaker for the right PFD) circuit breaker is opened and fitted with a safety clip and the connectors are stowed as required.

c) Hold the PFD (refer to Figure 1 and Figure 2) near the instrument panel and connect the electrical connectors.

d) Put the PFD into the instrument panel.

e) Insert the screws that hold the PFD in the instrument panel. If screws are damaged, replace.

f) Tighten the screws.

g) Remove safety clips and close (5 ampere PILOT PFD (H1) for the left PFD or 5 ampere COPILOT PFD (H2) circuit breaker for the right PFD) circuit breaker on the avionics circuit breaker panel.

h) Complete the applicable adjustment/test procedure, refer to Para 3.

B. Multi-Function Display (MFD)

1) General

These procedures apply to the upper and lower MFD.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) On the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data), open and safety the affected circuit breakers (5 ampere UPPER MFD (J1) for the upper MFD or 5 ampere LOWER MFD (J2) circuit breaker for the lower MFD).

c) Remove the screws (refer to Figure 1 and Figure 3 for the upper MFD or Figure 4 for the lower MFD) that hold the MFD in the instrument panel.

d) Carefully pull the MFD out of the instrument panel to get access to the electrical connectors (refer to PSM 1-64-2W) on the MFD.

e) Disconnect the electrical connectors from the MFD.

Note Stow connectors as required.

f) Remove the MFD from the aircraft.

Note Ensure that the MFD screen and connectors are protected during the removal process.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the (5 ampere UPPER MFD (J1) for the upper MFD or 5 ampere LOWER MFD (J2) circuit breaker for the lower MFD) circuit breaker is opened and fitted with a safety clip and the connectors are stowed as required.

c) Hold the MFD (refer to Figure 1 and Figure 3 for the upper MFD or Figure 4 for the lower MFD) near the instrument panel and connect the electrical connectors.

d) Put the MFD into the instrument panel.

e) Insert the screws (refer to Figure 3 for the upper MFD or Figure 4 for the lower MFD) that hold the MFD in the instrument panel. If screws are damaged, replace.

f) Tighten the screws.

g) Remove safety clips and close (5 ampere UPPER MFD (J1) for the upper MFD or 5 ampere LOWER MFD (J2) circuit breaker for the lower MFD) circuit breaker on the avionics circuit breaker panel.

h) Complete the applicable adjustment/test procedure, refer to Para 3.

5. Troubleshooting

A. Primary Flight Display (PFD)

Note A fault with the Display Reversion Control Panel may cause the symptoms of a faulty display. Refer to 31-15-00, Display Reversion Control Panel – General data, Para 5. for Display Reversion Control Panel troubleshooting.

1) Ensure PILOT PFD (H1) & COPILOT PFD (H2) circuit breakers are closed.

2) If presented with any of the following CAS messages:

- Check DU 1 (Caution)
- Check DU 4 (Caution)
- Check Pilot PFD (Caution)
- Check Copilot PFD (Caution)

a) Use the associated DU brightness knob on the reversion panel to turn the affected display completely off (past the final detent).

b) Wait several seconds, then turn the display back on.

c) If CAS message persists, the associated PFD may require service (refer to Para 5.A.7)).

3) If display is blank:

a) Ensure associated DU brightness knob is not in the OFF position.

b) Do an AGM reversion for the affected display.

c) If display remains blank, it may require service (refer toPara 5.A.7)).

d) If display mirrors the opposite PFD, the issue may be in the wiring or the associated AGM.

4) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

5) Swap PFD with a known good unit. If issue clears, the removed PFD requires service (refer to Para 5.A.7)).

6) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

7) If repair or service is required, remove and send to an appropriately rated avionics repair facility.

B. Multi-Function Display (MFD)

Note A fault with the Display Reversion Control Panel may cause the symptoms of a faulty display. Refer to 31-15-00, Display Reversion Control Panel – General data, Para 5. for Display Reversion Control Panel troubleshooting.

1) Ensure UPPER MFD (J1) & LOWER MFD (J2) circuit breakers are closed.

2) If presented with any of the following CAS messages:

- Check DU 2 (Caution)

- Check DU 3 (Caution)

a) Use the associated DU brightness knob on the reversion panel to turn the affected display completely off (past the final detent).

b) Wait several seconds, then turn the display back on.

c) If CAS message persists, the associated MFD may require service (refer to Para 5.B.7)).

3) If display is blank:

a) Ensure associated DU brightness knob is not in the OFF position.

b) Do an AGM reversion for the affected display (DU1 outer knob for DU 2, DU 4 outer knob for DU 3).

c) If display remains blank, it may require service (refer to Para 5.B.7)).

d) If display shows a red X, the issue may be in the wiring or the associated AGM.

4) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

5) Swap MFD with a known good unit. If issue clears, the removed MFD requires service (refer to Para 5.B.7)).

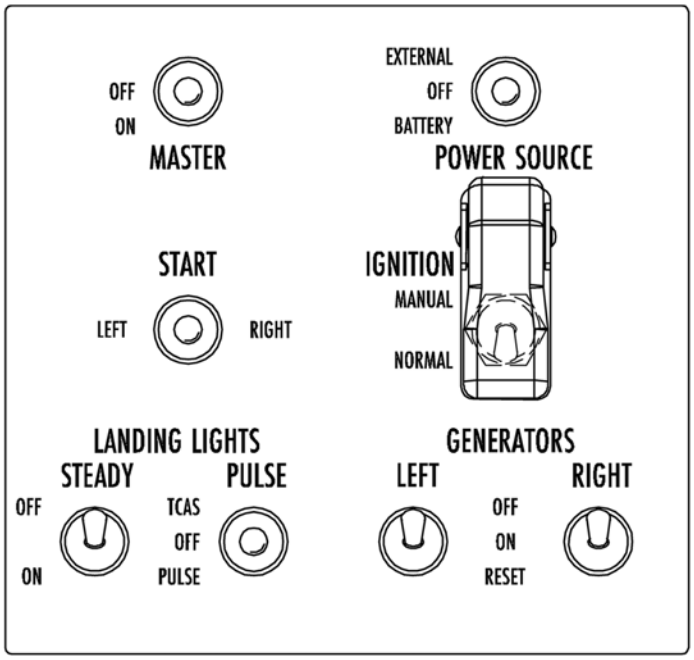
6) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

7) If repair or service is required, remove and send to an appropriately rated avionics repair facility.

Overhead Panel – General Data

1. General

The overhead control panel contains the electrical master switch, the power source selection switch, the start and ignition switches, landing light and generator control switches (refer to the specific chapters for the controls installed). (Refer to Figure 1).

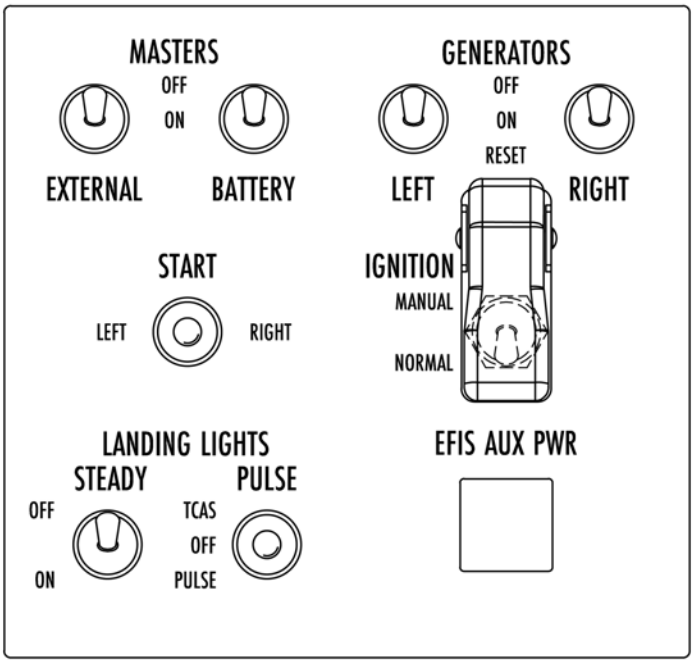


Starter Switch Panel (Pre Mod 6/2314)
Figure 1

The panel is internally lit with LED lights. (Refer to 33-00-00, Lights – General Data).

The overhead panel is located on the flight compartment roof on the pilot's left hand side.

Aircraft incorporating Mod 6/2314, the overhead control panel contains the electrical master switches for the EXTERNAL and BATTERY power sources, the start and ignition switches, landing light and generator control switches (refer to the specific chapters for the controls installed). (Refer to Figure 2).



Starter Switch Panel (Post Mod 6/2314)
Figure 2

Apex Controllers – General Data

1. General

The Primary Flight Display (PFD) controller, Multi-Function (MF) controller, and a flight controller which provides pilot interface to the Apex system.

The DHC-6 Series 400 avionics suite is developed around a four-screen Honeywell Primus Apex® Electronic Flight Instrument System (EFIS). The suite fully integrates primary flight display, communication, navigation, engine instrument displays, electrical and other systems displays, and crew alerting system (CAS) messages through Apex. Options are provided for some Apex and federated equipment items and safety systems including but not restricted to electronic checklists, electronic charts, TCAS I (optional TCAS II), EGPWS, and flight data recorder, cockpit voice recorder and emergency locator transmitter systems. Airframe signals are interfaced to Apex through two Data Acquisition Units (DAUs).

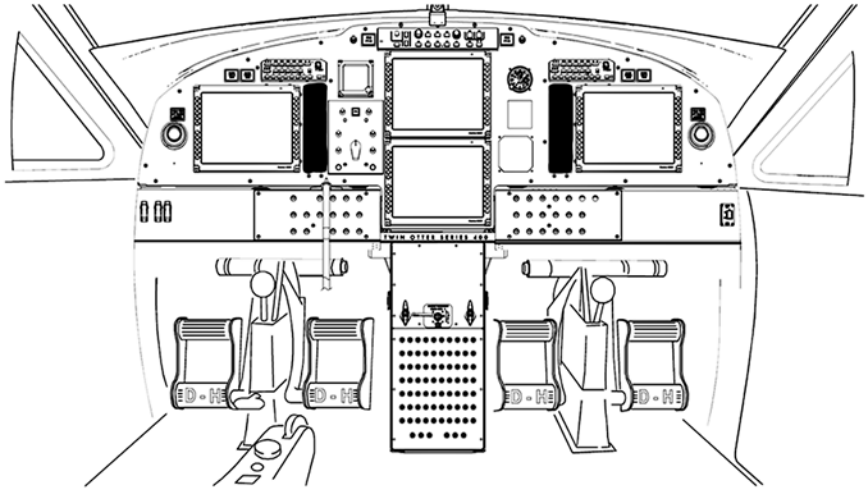
2. Description

A. PFD Controller (KMC 2210)

During normal, non-reversion operation, the PFD controller provides control of window navigation on the lower 1/6th window of the PFD for radio operation. Key functions are illustrated in Figure 1. The left and right PFD controllers each have an Air Data and solid-state Attitude/Heading Reference System (ADHRS) button, which can be used to change the PFD ADAHRS source channel. In the event of failure or degradation of the signal from the channel normally used, the signal from the opposite-side channel may be selected. ADAHRS source annunciations will be shown in amber in the lower left region of PFD attitude director indicator (ADI) window when the same source has been selected on both left and right PFDs, otherwise, the source annunciation is displayed in white.

The PFD Controller for the Pilot's PFD is installed on the instrument panel to the right of the Pilot's PFD. It allows the Pilot to navigate within the windows on the Pilot's PFD screen and set the format for the system information to be displayed. The Pilot's PFD Controller also acts as a backup to the Co-pilot's PFD controller in the event of a failure. 28 Volt DC power is provided from the left bus through a 3 amp PILOT PFD CTRL (Q1) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data). Bezel key and face lighting is controlled by the INSTRUMENT PANEL dim potentiometer.

The PFD Controller for the Co-pilot's PFD is installed on the instrument panel to the left of the Co-pilot's PFD. It allows the Co-pilot to navigate within the windows on the Co-pilot's PFD screen and set the format for the system information to be displayed. The Co-pilot's PFD controller also acts as a backup to the Pilot's PFD Controller in the event of a failure. Bezel key and face lighting is controlled by the INSTRUMENT PANEL dim potentiometer. 28 Volt DC power is provided from the right bus through a 3 amp COPILOT PFD CTRL (Q2) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).





BARO rotary click knob for the setting of the current barometric pressure valve for display on the PFD altitude window. Clockwise rotation increments and counter clockwise decrements the barometric correction valve.

PUSH STD push button to set the current barometric pressure value to standard pressure.

NAV SEL push button to cycle through the navigation sources shown on the HSI display.

NAV PREVIEW push button to activate and cycle through available navigation sensors when FMS is the active sensor.

O (circle) push button to cycle through the No. 1 sources of navigation bearing to be displayed on HSI as a circle pointer (single pointer).

◊ **(diamond)** push button to cycle through the No. 2 source of navigation bearing to be displayed on HSI as a diamond pointer (double pointer).

HSI push button to alternate HSI display between compass and arc formats.

ET push button to activate and control an elapsed timer displayed on PFD.

CRS/RNG dual rotary click knob, inner for control of the desired VOR/LOC course to be flown and the selected navigation sensor shown on the HSI. Outer for control of the range display on the HSI.

PUSH SYNC push button to cause a synchronization of the selected course to the current VOR bearing if a VOR is the selected navigation sensor.

The pilot may configure the PFD to display altitude in meters as well as feet. The pilot may also configure the system for entering the altimeter setting in inches of mecury or millibars.

PFD Controller
Figure 1

B. MF Controller (KMC 2220)

The MF controller is mounted on the aft face of the central control yoke in the cockpit. The MF controller provides the operator key access to menus, functions and formats for the information within each defined windows displayed on the MFDs, as well as the radio window of the PFDs. Refer to Figure 2.

Window focus is obtained using the MF controller. Pressing a bezel button does not bring focus to a window. The Map window in the upper MFD is the default window for focus, except in composite mode. Only one window can be in focus across the displays. When focus is obtained, a cyan border will be shown around the window. After a period of 60 seconds of inactivity, the window focus border will be removed and focus will be returned to the MAP window. When there is no window focus present, a press of the arrow keys on the MF controller will highlight the focus border on the MAP window. When window focus is brought to a window that has data entry fields, a cyan colour cursor will be placed on the first data entry field. The MF Controller joystick can then be used to position the cursor onto the desired data entry field. The MF controller joystick can be used to move focus between different windows on the same display unit, but cannot be used to move the focus from one display unit to another.

Page operation is accomplished by pressing the PAGE button on the MF controller when in an active window. A menu listing the available pages for the window will be displayed.

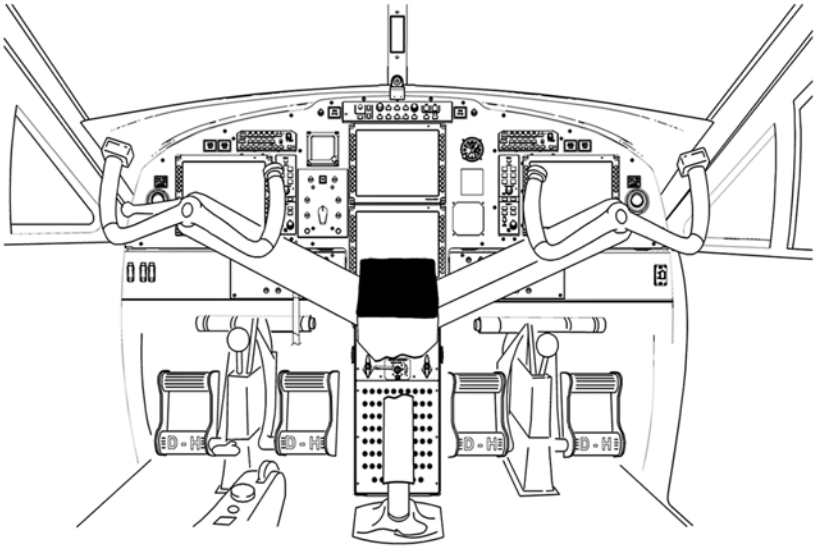
There are two types of menus:

1) Page menus – to access pages of functions contained in the same window.

2) Functionality menus – to show selection headings that remain the same regardless of the current mode of operation.

   All menus once selected have cursor snapping, whereby the cursor automatically moves to the first item in the menu. The joystick or arrow keys can then be used to move to other items in the menu. When a keyboard or an MF controller entry is made, the cursor is caged inside the data field until the entry is completed either by pressing the ENT key or by clearing (cancelling) the entry with the CLR or DEL keys. If an MF controller short cut key is pressed or the time out period is reached, the entry is considered not finished and reverts to the previous value.

   28 Volt DC power is provided from the left bus through a 3 amp MFD CTRL (Q3) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data). Bezel key and face lighting is controlled by the INSTRUMENT PANEL dim potentiometer.





ARROW keys (up, down, left and right) move window focus (i.e. select the window to be made active) in the direction indicated to the next interactive window.

JOYSTICK & KNOB: These control cursor focus within a window, allowing operation control from selectable item to item. Depending on the data field, the concentric knob can control the item that has cursor focus, or, a value can be entered via the alphanumeric keypad.

ENT confirms and executes the entry of data.

PAGE displays a menu of possible pages that can be selected.

MFD SWAP switches the display of the Situation Awareness and Systems MFDs between the upper and lower screens.

DETAIL calls up a secondary window related to the current active window, providing additional details related to the selected item.

KEYPAD alpha numeric push buttons provide direct data entry into an active data field.

DEL deletes the last character entered on the active data field.

CLR erases the entire value of a data entry field on the active data field.

CKLST calls up the electronic checklist.

CHARTS/VIDEO displays charts on the right 2/3rd window of the System MFD.

Secure Digital (SD) data card slot, used to download new FMS navigation database (e.g. Jeppesen) from the SD card, and also to upload maintenance records to the SD card for review by the maintenance technician.

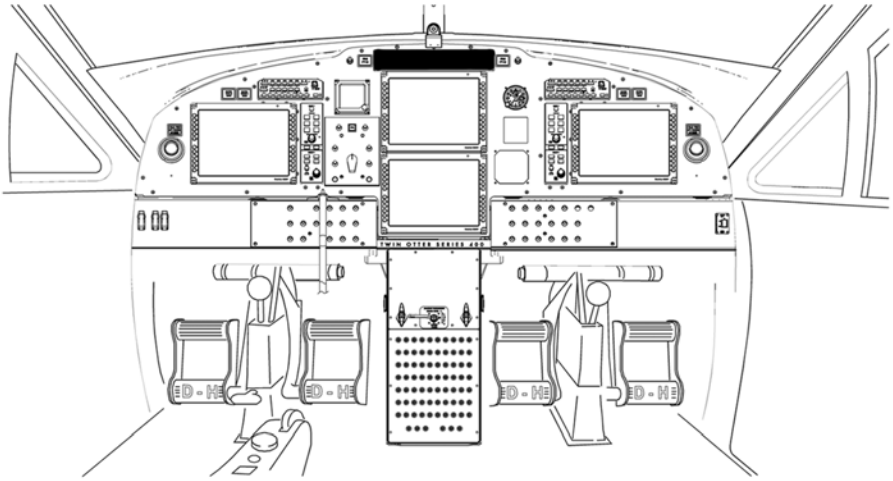
Multi-Function Controller
Figure 2

C. Flight Controller (KMC 9200)

The KMC 9200 Flight Controller (refer to Figure 3) is used to enter selected minimums, set heading or track, and enter selected altitude. All other controls on the flight controller are inoperative and will produce a Function Unavailable CAS message.

For aircraft with Mod 6/2303 incorporated, when the Flight Director (FD) mode is engaged, a green annunciation above the FD button is illuminated. (Refer to 22-00-00, Auto Flight – General Data).

For aircraft with S.O.O. 6221 incorporated, the Flight Controller (FC) (also referred to as the Flight Control Panel [FCP]) provides selections for all AFCS FD and Autopilot (AP) modes except, Go-Around (GA) mode, Touch Control Steering (TCS) and Quick Disconnect (AP DISC). (Refer to 22-10-00, Automatic Flight Control System (AFCS) (Mod 6/2303 and S.O.O. 6221) – General Data).





Flight Controller (KMC 9200)
Figure 3

28 Volt DC power is provided from the left bus through a 3 amp FLT CTRL 1A (M1) circuit breaker, and from the right bus through a 3 amp FLT CTRL 1B (M2) circuit breaker on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data). Bezel key and face lighting is controlled by the INSTRUMENT PANEL dim potentiometer.

3. Adjustment/Test

A. PFD Controller (KMC 2210)

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Rotate the BARO knob. Verify barometric pressure on the PFD increases with clockwise rotation and decreases with counterclockwise rotation.

4) Push the BARO knob (PUSH STD). Verify the current barometric value changes to standard pressure (29.92).

5) Select NAV SEL. Verify navigation sources are displayed on the HSI.

6) Select NAV PREVIEW. Verify the ability to cycle through available navigation sources.

7) Select O (circle). Verify the ability to cycle through the No.1 navigation sources which are displayed on the HSI as a circle pointer (single pointer).

8) Select ◊ (diamond). Verify the ability to cycle through the No.2 navigation sources which are displayed on the HSI as a diamond pointer (double pointer).

9) Select HSI. Verify the HSI display changes between compass and arc formats.

10) Select ET. Verify activation and control of a elapsed timer displayed on the PFD.

11) Rotate the inner CRS/RNG knob. Verify VOR/ILS course to be flown can be selected to different navigation sensor on the HSI.

12) Rotate the outer CRS/RNG knob. Verify the change of range display on the HSI.

13) Select ADHRS. Verify ADAHRS source changes from No.1 to No.2 source or No.2 to No.1 source.

14) Select PFD. Verify control of the RH PFD when the LH PFD controller key is selected and control on the LH PFD when the RH PFD controller key is selected.

15) Turn the INSTRUMENT PANEL dim potentiometer on the aircraft dimming control panel. Verify the PFD dims and brightens accordingly.

16) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Multi-Function (MF) Controller (KMC 2220)

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Initialize the FMS with current position, if available, or with a known waypoint (eg. CYYJ) (Pre Mod 6/2303). Ensure FMS has initialized (Post Mod 6/2303).

4) Select COM. Verify selection of COM 1 window on the pilot PFD. Using the keypad, enter frequency 123.95 MHz and press the ENT button adjacent to the keypad. Ensure the entered frequency appears on pilot's and co-pilot's PFD.

5) Select FREQ SWAP. Ensure COM 1 active and standby frequencies swap positions.

6) Select NAV. Verify selection of NAV 1 window on the pilot PFD. Using the keypad, enter frequency 116.85 MHz and press the ENT button adjacent to the joystick. Ensure the entered frequency appears on the pilot's and co-pilot's PFD.

7) Select XPDR. Verify selection of XPDR window on the pilot PFD. Using the keypad, enter code 1470 and press ENT. Ensure the entered code appears on pilot's and co-pilot's PFD.

8) While focus remains on the XPDR radio window, select DETAIL. Ensure the XPDR detail page opens in the pilot's PFD radio window. Select DETAIL a second time to close XPDR detail page.

9) Select TAWS overlay for display on PFD.

10) Select FLAP OVRD. Ensure FLAP OVRD annunciator appears beside the HSI. Select FLAP OVRD a second time to clear annunciator.

11) Select TERR INHIB. Ensure TERR INHB annunciator appears beside the HSI. Select TERR INHIB a second time to clear annunciator.

12) Select CHART VIDEO. Ensure "Function Unavailable" Status CAS message appears (Pre S.O.O. 6214). Ensure charts window opens on MFD (S.O.O. 6214).

13) Select EVENT. Ensure "Event" Status CAS message is displayed (Post Mod 6/2303).

14) Select CKLST. Ensure "Function Unavailable" Status CAS message appears (Pre S.O.O. 6213). Ensure checklist window becomes active (S.O.O. 6213).

15) Select INFO. Ensure "Show Object" dialog box opens on the INAV map.

16) Move the joystick in all directions. Ensure the cursor moves around the display appropriately.

17) Close the "Show Object" dialog box.

18) With focus remaining on the INAV map, rotate the ring surrounding the joystick in both directions. Ensure map range changes appropriately. Leave MFD range at 50nm.

19) Select DIRECT TO (). Ensure direct-to dialog box appears in the waypoint window.

20) Using the keypad enter an airport code (eg. CYVR), do not press ENT.

21) Select DEL. Ensure the last character entered is deleted.

22) Hold CLR for half a second or more. Ensure the remaining three characters are deleted.

23) Close the direct to dialog box.

24) Select each ARROW key. Ensure window focus moves within the displays in the direction selected (Pre Mod 6/2303). Ensure cursor focus moves to the selected display (Left – DU 1, Up – DU 2, Down – DU 3, Right – DU 4) (Post Mod 6/2303).

25) Bring focus to the System Configuration Window in the bottom left corner of the Systems MFD. Select PAGE. Ensure page menu opens.

26) Select MFD SWAP (

). Ensure Situational Awareness Display (default DU 2) and Systems Display (default DU 3) swap MFD positions.

27) Select WX radar for display on the INAV map. Select WX RADAR outer knob to TEST. Ensure WX test pattern is displayed on MFD.

28) Rotate the WX RADAR inner knob in both directions. Ensure WX tilt indication on the MFD changes appropriately.

29) Press the WX RADAR knob. Rotate the WX RADAR inner knob. Ensure the TILT adjustment has changed to a GAIN adjustment.

30) Select WX RADAR outer knob to OFF.

31) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Flight Controller (KMC 9200) (Pre Mod 6/2303)

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Rotate MINIMUMS knob. Ensure RA minimums indication appears in the attitude indicator area of the PFD and increases and decreases as the knob is rotated.

4) Press MINIMUMS knob. Ensure RA minimums indication clears from PFD.

5) Rotate MINIMUMS knob. Ensure BARO minimums indication appears in the attitude indicator area of the PFD and increases and decreases as the knob is rotated.

6) Set HDG TRK outer knob to TRK. Ensure the indication on HSI switches to TRK.

7) Set HDG TRK outer knob to HDG. Ensure the indication on HSI switches to HDG.

8) Rotate inner knob. Ensure the HDG bug moves around the HSI.

9) Press the HDG TRK knob. Ensure the HDG bug moves to the current aircraft heading.

10) Rotate ALT outer knob. Ensure the selected altitude indication appears above the altitude tape and increments and decrements by 1000 ft per detent.

11) Rotate ALT inner knob. Ensure the selected altitude indication increments and decrements by 100 ft per detent.

12) Select the remaining buttons on flight controller. Ensure “Function Unavailable” Status CAS message appears.

13) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

D. Flight Controller (KMC 9200) (Post Mod 6/2303)

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Rotate MINIMUMS knob. Ensure BARO minimums indication appears in the attitude indicator area of the PFD and increases and decreases as the knob is rotated.

4) Press MINIMUMS knob. Ensure BARO minimums indication clears from the PFD.

5) Rotate MINIMUMS knob. Ensure RA minimums indication appears in the attitude indicator area of the PFD and increases and decreases as the knob is rotated.

6) Set HDG TRK outer knob to TRK. Ensure indication on HSI switches to TRK.

7) Set HDG TRK outer knob to HDG. Ensure indication on HSI switches to HDG.

8) Rotate inner knob. Ensure HDG bug moves around the HSI.

9) Press the HDG TRK knob. Ensure the HDG bug moves to the current aircraft heading.

10) Rotate ALT outer knob. Ensure the selected altitude indication appears above the altitude tape and increments and decrements by 1000 ft per detent.

11) Rotate ALT inner knob. Ensure the selected altitude indication increments and decrements by 100 ft per detent.

12) Select L/R. Ensure FD arrow lights and indications on the PFD toggle between the left and right.

13) Select FD. Ensure FD light illuminates and ROL and PIT indications appear on the PFD.

14) Select ALT. Ensure ALT light illuminates, and ALT indication appears on the PFD.

15) Select VS. Ensure VS light illuminates and VS indication appears on the PFD.

16) Select NOSE DN. Ensure VS bug moves down with each press.

17) Select NOSE UP. Ensure VS bug moves up with each press.

18) Select SPD. Ensure SPD light illuminates and SPD indication appears on the PFD.

19) If required, select NAV SEL on PFD controller to set NAV source to FMS.

20) Select VNAV. Ensure VNAV light illuminates and VNAV indication appears on the PFD.

21) Select HDG/T. Ensure HDG/T light illuminates and HDG indication appears on the PFD.

22) Select BL. Ensure BL light illuminates and bank limits are displayed in magenta on the roll indicator.

23) Select NAV. Ensure NAV light illuminates and NAV indication appears on the PFD.

24) Select NAV SEL on PFD controller to set NAV source to VOR or LOC.

25) Select APR. Ensure APR light illuminates and APR indication appears on the PFD.

26) Select FD. Ensure all FD mode lights extinguish.

27) Connect air data test set to the aircraft pitot heads (S.O.O. 6221) (refer to 34-11-00, Connecting Air Data Test Set, Para 4.A.).

28) Adjust air data test set to provide an airspeed of 60 knots (S.O.O. 6221).

29) Select AP. Ensure “Function Unavailable” Status CAS message appears (Pre S.O.O. 6221). Ensure AP and YD lights illuminate and AP and YD indications appear on the PFD (S.O.O. 6221).

30) Select YD. Ensure “Function Unavailable” Status CAS message appears (Pre S.O.O. 6221). Ensure YD light extinguishes and YD indication clears from the PFD (S.O.O. 6221).

31) Vent the air data test set to ambient conditions (S.O.O. 6221).

32) Disconnect the air data test set (S.O.O. 6221) (refer to 34-11-00, Disconnecting Air Data Test Set, Para 4.B.).

33) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal and Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – PANEL / PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

A. PFD Controller (KMC 2210)

1) Removal of PFD Controller

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING; UNIT CAN BE DAMAGED.**

b) Using a 3/32” hexagonal-head wrench, engage it into the PFD controller locking screw.

c) Turn the locking screw counterclockwise while gently pulling and shifting the controller, until the controller disengages from the instrument panel.

d) Carefully pull on the edges of the controller until removed from its mount in the instrument panel.

e) Disconnect the electrical connector.

f) Remove unit from aircraft.

2) Installation of PFD Controller

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Connect the electrical connector to the PFD controller.

c) Push carefully on the face of the controller until the locking screw is engaged.

d) Slide the controller into its mount in the instrument panel.

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING; UNIT CAN BE DAMAGED.**

e) Using a 3/32” hexagonal-head wrench, while gently pushing and shifting the controller, engage it into the PFD controller locking screw.

f) Carefully apply a light pressure to the edges of the unit, while gently pushing and shifting the controller, turning the 3/32” wrench clockwise to tighten the locking screw.

g) Do a function test, refer to Para 3.A.

B. MF Controller (KMC 2220)

1) Removal of MF Controller

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING; UNIT CAN BE DAMAGED.**

b) Using a slotted screwdriver, engage it into the MF controller screw.

c) Turn the locking screw counterclockwise until the controller disengages from its mount.

d) Pull on the edges of the controller until it is removed from its mount.

e) Disconnect connectors.

f) Remove unit from aircraft.

Note Stow connectors as required.

2) Installation of MF Controller

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Connect connectors.

c) Slide the MF controller into mount on the control yoke.

d) Push carefully on the face of the MF controller until locking screw is engaged.

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING; UNIT CAN BE DAMAGED.**

e) Using a slotted screwdriver, engage it into the MF controller locking screw.

f) Apply light pressure to the edge of unit, while turning the screwdriver clockwise 1/4 turn.

g) Do a function test, refer to Para 3.B.

C. Flight Controller (KMC 9200)

1) Removal of Flight Controller

a) Ensure the aircraft electrical system is OFF. (Refer 24-00-00, Electrical Power – General Data).

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING; UNIT CAN BE DAMAGED.**

b) Using a 3/32” hexagonal-head wrench, engage it into the Flight controller locking screw.

c) Turn the locking screw counterclockwise while gently pulling and shifting, until the controller disengages from the instrument panel.

d) Carefully pull on the edges of the controller until removed from the instrument panel.

e) Disconnect the electrical connector.

f) Remove unit from aircraft.

2) Installation of Flight Controller

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Connect the electrical connector to the Flight controller.

c) Carefully slide the controller into the instrument panel.

CAUTION **CARE MUST BE TAKEN TO ENSURE THE WRENCH IS SEATED CORRECTLY ON THE SCREW BEFORE ROTATING; UNIT CAN BE DAMAGED.**

d) Using a 3/32" hexagonal-head wrench, while gently shifting and pushing, engage it into the Flight Controller locking screw.

e) Turn the locking screw counterclockwise until the controller disengages from the instrument panel.

f) Do a function test, refer to Para 3.C (Pre Mod 6/2303) or Para 3.D (Post Mod 6/2303).

5. Troubleshooting

A. PFD Controller (KMC 2210)

1) Ensure PILOT PFD CTRL (Q1) & COPILOT PFD CTRL (Q2) circuit breakers are closed.

2) If presented with any of the following CAS messages, the associated PFD controller(s) requires servie (refer to Para 5.A.6)):

- LH PFD CTRL Fail (Caution)
- RH PFD CTRL Fail (Caution)
- LH PFD CTRL Fail (Advisory)
- RH PFD CTRL Fail (Advisory)
- LH+RH PFD CTRL Fail (Advisory)

3) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

4) Swap PFD controller with a known good unit. If issue clears, removed PFD controller requires service (refer to Para 5.A.6)).

5) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

6) If repair or service is required, remove and send to appropriately rated avionics repair facility.

B. MF Controller (KMC 2220)

1) Ensure MFD CTRL (Q3) circuit breaker is closed.

2) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

3) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

4) If repair or service is required, send to appropriately rated avionics repair facility.

C. Flight Controller (KMC 9200)

1) Ensure FLT CTRL 1A (M1) & FLT CTRL 1B (M2) circuit breakers are closed.

2) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

3) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

4) If repair or service is required, send to appropriately rated avionics repair facility.

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Display Reversion Control Panel – General data

1. General

The Display Reversion Control Panel controls display brightness, power, and AGM reversion. Refer to Figure 1.



Display Reversion Control Panel
Figure 1

The display system allows the pilot to revert the Display Units (DU) and Advanced Graphics Module 1 and 2 (AGM) in the event of failure of an individual display unit or an AGM.

2. Description

A. Display Reversion Control Panel

The Display Reversion Control panel has potentiometers for the PILOTS PFD, UPPER MFD, LOWER MFD and CO-PILOTS PFD. The four potentiometers are used to adjust the individual Display Unit (DU) brightness and, at their most counterclockwise position, to switch the displays to OFF/REV.

The right side PFD has no Reversion (REV) function. Turning off the right side PFD simply turns off the display, and does not affect the contents of the other three display units. When the PILOTS PFD, UPPER MFD, or LOWER MFD potentiometers are moved to the OFF/REV position, the selected Display Unit (DU) goes blank and the information is moved to another display.

In some reversion cases (typically when two DUs have been turned off) the PFD will enter a composite mode. The PFD composite format shows the ADI/HSI, up to twelve CAS messages, the Systems Summary and the Radio windows, all on a single screen. Messages can be scrolled.

The two PFD controls also have a rotating switch surrounding their potentiometer that can be used to select the opposite side Advanced Graphics Module (AGM) in the event of a failure of the AGM normally assigned to drive that PFD. When the right hand rotating switch has been moved away from the NORM position and the Co-pilot PFD is being supplied with data from AGM 1, the engine instruments and radio window will appear on the outboard side of the Co-pilot's PFD.

The Display Reversion Control Panel is installed directly above the MF controller, on the aft face of the control column. Bezel and face lighting is controlled by the INSTRUMENT PANEL dim potentiometer.

3. Adjustment/Test

A. Function Test Display Reversion Control Panel

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Wait several minutes for the aircraft electrical and avionics systems to initialize.

3) Turn the PILOTS PFD dim control from the counterclockwise (MIN) position, to the fully clockwise position on the Reversion Control Panel. Verify that the Pilot's PFD display goes from dark to full brightness.

4) Turn the CO-PILOTS PFD dim control from the counterclockwise (MIN) position, to the fully clockwise position on the Reversion Control Panel. Verify that the Co-pilot's PFD display goes from dark to full brightness.

5) Turn the Multi-Function Displays (MFD) dim controls from the counterclockwise (MIN) position, to the fully clockwise position on the Reversion Control Panel. Verify that the Multi-Function Displays (MFD) display goes from dark to full brightness.

6) Set the PILOTS PFD, CO-PILOTS PFD, and Multi-Function Display dim controls from the full brightness position to a comfortable viewing dim level.

7) Set the PILOTS PFD revision control to AGM2. Verify the following:

   a) Pilot's PFD shows normal flight information.

   b) Co-pilot's PFD shows the pilot PFD format (radio window on the right side).

   c) Lower MFD shows a normal system multi-function display or situation awareness information with a Crew Alerting System (CAS) display in the upper left window.

   d) Upper MFD shows invalid (red cross).

8) Set the PILOTS PFD reversion control to NORM. Ensure the PFD and MFD displays return to normal.

9) Set the CO-PILOTS PFD revision control to AGM1. Verify the following:

   a) Co-pilot’s PFD shows normal flight information.

   b) Pilot’s PFD shows the co-pilot PFD format (radio window on the right side).

   c) Lower MFD shows invalid (red cross).

   d) Upper MFD shows a normal multi-function display with a Crew Alerting System (CAS) display in the upper left window.

10) Set the CO-PILOTS PFD reversion control to NORM. Ensure the PFD and MFD displays return to normal.

11) Set the PILOTS PFD dim control to OFF/REV. Verify that the Pilot's PFD is shown on the upper MFD.

12) Set the upper MFD dim control to OFF/REV. Verify that the Pilot's PFD is shown on the lower MFD.

13) Set the lower MFD dim control to OFF/REV. Verify that the composite PFD display is shown on the Co-pilot's PFD.

14) Set the CO-PILOTS PFD dim control to OFF. Verify that the composite PFD display is shown on the lower MFD.

15) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal and Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – PANEL / PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Display Reversion Control Panel

1) Removal of Display Reversion Control Panel

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) To remove the Reversion Control Panel, turn the quick release fasteners.

c) Pull the Reversion Control Panel away from the controller mount pedestal to gain access to the electrical connector.

d) Disconnect electrical connector.

e) Remove unit from aircraft.

2) Installation of Display Reversion Control Panel

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Reconnect the electrical connector to the Reversion Control Panel.

c) Slide the Reversion Control Panel into the controller mount.

d) Turn the quick release fasteners to secure the Reversion Control Panel to the controller mount.

e) Do a function test, refer to Para 3.A.

5. Troubleshooting

A. Display Reversion Control Panel

1) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

2) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

3) If repair or service is required, send to appropriately rated avionics repair facility.

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Independent Instruments – General Data

1. General

The independent instruments consists of the aircraft clock and an Electronic Standby Instrument System (ESIS).

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Clock (Mod 6/2049) – General Data

1. General

The Thommen Clock is used to keep accurate time and functions as a chronometer.

2. Description

An illuminated Thommen model Q13 quartz chronometer is located on the instrument panel, to the right of the upper multi-function display. A setting knob is provided beside the clock face. The clock provides a standard 3 function, single push-button operation of a 60 minute timer (a chronometer) using start/stop/reset-to-zero in a fixed, repeating sequence. The 60 minute timer has its own minute hand (with a triangular pointer on the end) and also uses the sweep second hand for the timer function. The clock is internally illuminated and equipped with antireflective coated glass. Power for illumination is provided from the same circuit that controls all other bezel lights and is controlled by the INSTRUMENT PANEL potentiometer.

28 Volt DC power is provided from the hot battery bus through a 2 amp CLOCK AND DOME LIGHT (S1) circuit breaker on the station 332.00 circuit breaker panel (refer to 24-00-00, Electrical Power – General Data). Electrical power consumption is negligible; approximately 4 milliamps. The clock has an integral backup battery to provide power in the event of a loss of power from the hot battery bus. The backup battery will support operation of the clock for approximately one month of cumulative use.

3. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/EIL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – PANEL/PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal of Clock

1) Ensure the aircraft electrical system is OFF.

2) Pull circuit breaker CLOCK AND DOME LIGHT (S1), on the station 332.00 circuit breaker panel and fit safety clip.

3) Gain access to the rear of the clock by removing the Upper MFD, refer to 31-11-00, Display Units – General Data.

4) Disconnect electrical connector.

Note Stow connectors of the removed components.

5) While supporting the clock from the rear of the panel, remove the (2) mounting screws.

B. Installation of Clock

1) Ensure the aircraft electrical system is OFF.

2) Ensure CLOCK AND DOME LIGHT (S1) circuit breaker is pulled and fitted with safety clip.

3) Set unit in position from behind the panel and install (2) mounting screws.

4) Reconnect electrical connector.

5) Install the Upper MFD, refer to 31-11-00, Display Units – General Data.

6) Verify that clock is operating, on battery power.

7) Remove the safety clip and close the CLOCK AND DOME LIGHT (S1) circuit breaker on the station 332.00 circuit breaker panel.

C. Removal of Battery

1) Remove the clock per Para 3.A.

2) Gently remove the battery case cover at rear of unit.

3) Remove the battery.

4) Visually inspect battery case for signs of corrosion or leakage from the battery.

D. Installation of Battery

1) Replace battery.

2) Install battery cover case, ensure case is secure.

3) Confirm clock is functioning properly.

4) Install clock as per Para 3.B.

4. Troubleshooting

A. Clock

1) Ensure CLOCK AND DOME LIGHT (S1) circuit breaker is closed.

2) Visually inspect the clock connector for damaged wiring or contacts. Ensure electrical connector on the clock is fully seated and locked.

3) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

4) If repair or service is required, send to appropriately rated avionics repair facility.

5. Servicing

A. Battery

1) Change the battery after any of the following conditions:

a) When the clock cannot maintain the correct time when aircraft power is removed.

b) There is any evidence of corrosion or leakage from the battery in the battery case.

Electronic Standby Instrument System (ESIS) (Mod 6/2049 and Mod 6/2170) – General data

1. General

The Electronic Standby Instrument System (ESIS) gives displays for attitude, heading, altitude and airspeed in case of primary display or systems failures. The ESIS is powered by its own dedicated PS–850B backup battery, and is entirely independent of the Apex system. Unless noted otherwise, the descriptions and procedures in this section are common to both the GH-3100 and the GH-3900.2 ESIS units.

2. Description

A. GH–3100 or GH–3900.2 (Mod 6/2170) Electronic Standby Instrument System

The ESIS is a 3–ATI size instrument with no moving parts. It utilizes flat panel, Active Matrix Liquid Crystal Display (AMLCD) technology. The ESIS contains a 3 axis inertial measurement sensor cluster that is used to compute attitude information. Airspeed and altitude information is displayed on either side of the attitude display in a digital vertical scale. Magnetic heading is displayed at the bottom. Airspeed and altitude information is computed using an integrated air data computer module that is supplied with pitot and static pressure from the right hand pitot tube and the upper set of static vents. In the event of obstruction of the upper static vents, the ESIS STATIC SELECTOR may be moved to the EMERGENCY position. When this is done, static pressure from the lower static vents is supplied to the ESIS. The pitot pressure input to the ESIS is not switchable. The design of the display has been customized to replicate as closely as possible the display of information on the Apex PFD.

The ESIS is controlled by the ESIS POWER switch on the fuel panel. The switch has an ARM, OFF, and TEST position. The TEST position is used to do a battery capacity test, refer to Para 2.D for details. When the ESIS POWER switch is in the ARM position, the aircraft supplies necessary power to charge the battery pack and operate the ESIS. In the event of an aircraft power failure, the PS–850B automatically takes over the duties of supplying the main power input and lighting input to the ESIS and ensures an uninterrupted flow of power. Whenever power is applied to the ESIS, a complete internal self test sequence of the ESIS unit is carried out. Any faults found will be displayed on the ESIS screen.

28 Volt DC power is provided from the ESIS bus through a 3 amp ESIS INST (P5) circuit breaker on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

B. MAG–3000 Magnetometer

The magnetometer uses a three-axis magnetic sensor to sense magnetic fields. The field data is transmitted via an RS–422 bus to the ESIS. The data is used to compute the magnetic heading of the aircraft. The magnetometer is located in the outboard end of the left wing.

28 Volt DC power is provided from the ESIS bus through a 1 amp ESIS MAG (Q5) circuit breaker on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

C. Configuration Module

The configuration module contains sufficient non-volatile memory to retain all the programming codes that describe the specific hardware and software configuration of each installation, such as display format configuration. When the ESIS is removed from the aircraft, the configuration module remains in the aircraft. This eliminates the need to reprogram a new ESIS if the instrument is replaced.

The GH–3100 (Pre Mod 6/2170) uses a DCM–3100 Detachable Configuration Module (DCM), which attaches to the ESIS as the P2 connector. The DCM–3100 is tethered to the aircraft harness, to ensure it remains in the aircraft when the ESIS is removed.

The GH–3900.2 (Post Mod 6/2170) uses a Data Configuration Module (DCM), which is tied into the wiring harness and is wired directly into the P1 connector.

D. PS–850B Emergency Power Supply

The PS–850B Emergency Power Supply is a self-contained power supply mounted in the rear avionics compartment. It is used to power the ESIS bus in the event of total aircraft electrical power failure. When fully charged, the backup battery has sufficient capacity to operate the ESIS for approximately 6 hours.

The emergency power supply has a test feature which tests battery capacity. Using either the push button on the front panel of the power supply, or the ESIS POWER switch, TEST position places a 7 ohm load (battery heater) across the output of the battery. If the voltage under load is acceptable (50% or greater than total capacity) the LED will light and the battery will provide a ground to the MAU, which triggers the ESIS Batt Test Pass CAS message.

28 Volt DC power is provided from the left bus through a 15 amp ESIS BATTERY (N5) circuit breaker on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

28 Volt DC power is provided to the ESIS bus through a 5 amp ESIS POWER circuit breaker located near the emergency power supply.

3. Adjustment/Test

A. Electronic Standby Instrument Test

CAUTION **DO NOT LEAVE THE ESIS POWER SWITCH IN THE ARM POSITION WHEN AIRCRAFT ELECTRICAL POWER IS NOT ON. BATTERY DAMAGE CAN OCCUR IF THE BATTERY DISCHARGES FULLY.**

1) ESIS Operational Test

Note No voltage shall be present on the 28 Volt DC input during test.

a) Ensure NO electrical power is applied to the aircraft.

b) Select ESIS POWER switch to the ARM position.

c) Verify the ESIS powers up, receiving electrical power ONLY from the emergency battery.

d) Verify the Built-in Test (BIT) and unit alignment complete (up to 3 minutes).

Note If any of the following are out of tolerance, refer to Para 5.A.

e) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

f) Allow avionics to fully initialize including ADAHRS alignment.

g) Compare the ESIS to the PFD's and verify the following:

1 Attitude will be ± 1.5°

2 Heading will be ± 6.0°

3 Airspeed will be at minimum setting (30 KIAS for GH–3100 and blank airspeed field for GH–3900.2)

4 Altitude will be ± 20 feet (ensure proper baro setting)

h) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

i) Verify the ESIS continues normal operation.

j) Select ESIS POWER switch to the OFF position.

B. PS–850B Battery Check

1) Periodic Battery Check

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical systems to initialize.

c) Select ESIS POWER switch to the momentary TEST position and hold.

d) After approximately 5 seconds, the ESIS Batt Test Pass CAS message should appear.

e) If no CAS message appears, do the Battery Capacity Test as per Para 3.B.2).

f) Select ESIS POWER switch to the OFF position.

g) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Battery Capacity Test

a) Ensure the aircraft electrical system is OFF, and ESIS POWER switch is in the OFF position.

b) Gain access to the Emergency Power Supply, which is accomplished through the aft baggage compartment into the rear avionics compartment.

c) Depress and hold the TEST switch on the front panel of the unit for 5 seconds.

1 If the LED on the front panel is illuminated for the entire duration of the tests, the battery capacity charge is greater than 50%.

2 If the LED on the front of the panel extinguishes before the switch is released, or does not illuminate at all, the battery capacity charge is lower than 50%.

d) Battery requires recharging as per Para 3.B.4).

Note 1 If the battery capacity charge is greater than 50% – battery requires up to 8 hours of recharging.

2 If the battery capacity charge is lower than 50% – battery requires up to 16 hours of recharging.

e) Do a discharge test, refer to Para 3.B.3).

3) Discharge Test

Discharge test is done to determine if the ESIS Emergency Battery requires servicing. Battery to be tested must be at full capacity.

Equipment required:

- Digital voltmeter Fluke Model 8050A or equivalent
- 7 Ohm Load Resistor 7Ω + 1% 150 watt (minimum) ceramic wire wound WireMasters Inc.
- WM910053 or equivalent
- Stopwatch

a) Place the digital voltmeter across pins J1–11 (+) and J1–7 (–) on the ESIS Emergency Battery.

b) Place the 7Ω load resistor across pins J1–11 (+) and J1–7 (–) on the ESIS Emergency Battery. Start Stopwatch.

c) Time the discharge period. Output voltage observed on the digital voltmeter should be greater than 20 VDC at 48 minutes.

d) If output is lower than 20 VDC, the battery requires servicing at an approved avionics facility. If output is greater than 20 VDC, continue to Para 3.B.3).e).

e) Remove the digital voltmeter and load resistor from the ESIS Emergency Battery pins and recharge battery (refer to Para 3.B.4)) within 2 hours of completing discharge test.

4) Recharging the Battery

WARNING **DO NOT CHARGE A BATTERY ON THE AIRCRAFT. THERE MAY BE A SERIOUS RISK OF INJURY TO PERSONNEL AND/OR DAMAGE TO THE AIRCRAFT OR AIRCRAFT SYSTEMS.**

Refer to the battery manufacturer's information for detail procedures, alternative methods of charging and procedures for storage.

Equipment required:

- DC Power Supply Hewlett Packard Model 6291 or equivalent
- Digital Voltmeter Fluke Model 8050A or equivalent
- Local Manufactured wire harness comprised of:

  • Mating Connector DPXB–13–33S–001 or equivalent
  • Isolation Diode 15 amp, 50 Volt PIV WireMasters Inc. WM1N3208 or equivalent
  • Wire 16 AWG

a) Remove ESIS battery from aircraft, refer to Para 4.D.1).

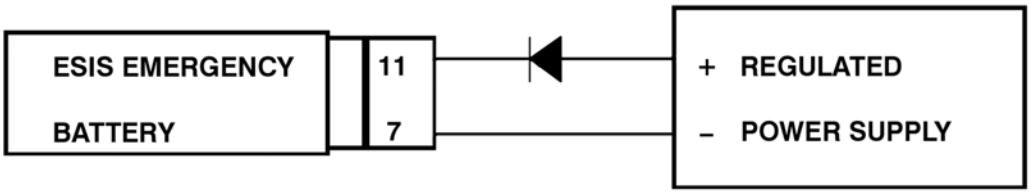
b) Set the voltage 29.0 ± 0.5 VDC on the regulated power supply. Ensure current is limited to 10 amp max, as shown below.

c) Connect power supply to battery using the local manufactured wiring harness.

d) Start regulated power supply, charge battery up to 16 hours.

Note
- Batteries are 80% capacity after 1 hour charging
- Batteries are at full capacity after 16 hours charging
- Charging for 23 hours is recommended for optimum cell balancing and maximum capacity

e) Stop regulated power supply and disconnect wiring harness from battery.



CAUTION **USE PIN J1–11 INSTEAD OF PIN J1–10 DURING CHARGING PROCEDURES TO AVOID ACTIVATING THE INTERNAL HEATING BLANKET.**

f) Reinstall ESIS battery in aircraft, refer to Para 4.D.2).

C. Electronic Standby Instrument System Functional Test

1) ESIS Function Test

a) Select ESIS POWER switch to the ARM position.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Allow the Built-In Test (BIT) to finish, and the unit alignment to complete (approximately 3 minutes).

Note If any of the following are out of tolerance, refer to Para 5.A.

d) Compare the ESIS to the PFD's and verify the following:

1 Attitude will be ± 1.5°

2 Heading will be ± 6.0°

3 Airspeed will be at minimum setting (30 KIAS for GH–3100 and blank airspeed field for GH–3900.2)

4 Altitude will be ± 20 feet (ensure proper baro setting)

e) Set the barometric correction to “STD” (29.92 in hg) by pressing the indicator adjustment knob.

f) Connect the air data test set the aircraft pitot and static ports (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4. Adjustment/Test).

g) Set the air data test set to achieve a 1000 – 3000 fpm climb. Ensure the altitude tape moves smoothly and displays the correct altitude.

h) Stop the test set at 5,000, 10,000, and 25,000 ft. Indicated altitude should match the following:

1 5,000 ft ± 25 ft

2 10,000 ft ± 30 ft

3 25,000 ft ± 50 ft

i) Set the air data test set to achieve an increasing airspeed. Ensure the airspeed tape moves smoothly and displays the correct airspeed. Indicated airspeed should match the following:

1 60 kts ± 5.0 kts

2 100 kts ± 2.0 kts

3 150 kts ± 2.0 kts

j) Adjust airspeed to greater than aircraft V_{MO}, a red bar will be displayed from V_{MO} to the high-end of the airspeed tape. Ensure the airspeed readout digits change to red when the airspeed is increased greater than V_{MO}.

k) Set the air data test set to achieve a 1000 – 3000 fpm descent. Ensure the altitude tape moves smoothly and displays the correct altitude.

l) Set the air data test set to achieve a decreasing airspeed. Ensure the airspeed tape moves smoothly and displays the correct airspeed.

m) Vent the pitot and static pressure to ambient conditions. Confirm the indicator displays field altitude and minimum airspeed setting (30 KIAS for GH–3100 and blank airspeed field for GH–3900.2).

n) Disconnect the air data test set from aircraft pitot and static ports (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data).

o) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

p) Set the ESIS POWER switch to OFF.

2) ESIS Display Cleaning

The ESIS bezel, softkey and display glass can be cleaned with a damp lint-free, static-free cloth. If required, clean with a soft cloth moistened with a mild solution of soap and water. Care should be taken to avoid scratching the surface of the display or getting water inside the softkey.

D. Calibration

Note Deleted

1) Attitude Calibration

a) ESIS GH–3100 (Pre Mod 6/2170)

Note The attitude calibration does not change the attitude display. These procedures allow compensation of attitude differences between the indicator and magnetometer for improved calibration.

Refer to Section 3.2.2.2.1 of the GH–3100 Electronic Standby Instrument System Installation and Operation Manual, part number TP–559, at the latest applicable revision.

b) ESIS GH–3900.2 Panel Angle (Post Mod 6/2170)

Note The nominal panel installation angle is set within the configuration file, and normally should not need adjustment. The following procedure should only be used if adjustment is deemed necessary.

1 Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2 To access the ESIS maintenance menu, press and hold the M key and BARO knob; select ESIS POWER switch to ARM.

3 Wait several minutes for the aircraft electrical and avionics systems to initialize.

4 Turn on the maintenance laptop computer and boot up to the desktop.

5 Connect the maintenance laptop computer to the ADAHRS using an RS–232 maintenance cable from an available computer COM port, to the ADAHRS 1B maintenance port on the aircraft maintenance panel located in the forward avionics bay.

6 Start the KSG 7200 Maintenance Program on the maintenance laptop computer.

7 Select the appropriate COM port and click ‘Start Maintenance’.

8 Click ‘Attitude’ to bring up a window with the digital pitch and roll values for ADAHRS channel B. Wait for pitch and roll values to stabilize.

9 Within the ESIS maintenance menu, ensure ANGLES is highlighted, press the BARO knob to enter the ANGLES menu.

10 When the ANGLES menu is first entered, the MOUNT ANGLE field displays the panel mounting angles stored in the configuration module.

11 Rotate the BARO knob to select pitch or roll AIRCRAFT ATTITUDE data fields.

12 Press the BARO knob to select the field, rotate the BARO knob to adjust the value in each field, then press BARO to enter the selected value.

13 Adjust the AIRCRAFT ATTITUDE pitch angle until the CALC AC ANGLE pitch value matches the ADAHRS pitch value displayed on the maintenance computer.

14 Adjust the AIRCRAFT ATTITUDE roll angle until the CALC AC ANGLE roll value matches the ADAHRS roll values displayed on the maintenance computer.

15 Press the M button to exit the ANGLES menu. Press BARO to select SAVE CHANGES before returning to the SETUP menu.

16 On the maintenance laptop, click 'Close'.

17 Click 'Exit Maintenance'.

18 Close the maintenance program on the laptop maintenance computer.

19 Remove the maintenance laptop computer and RS-232 maintenance cables from the aircraft.

20 Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

21 Set the ESIS POWER switch to OFF.

c) ESIS GH–3900.2 Magnetometer Angle (Post Mod 6/2170)

1 Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2 To access the ESIS maintenance menu, press and hold the M key and BARO knob; select ESIS POWER switch to ARM.

3 Rotate the BARO knob to select MAG INSTALL, press the BARO knob to enter the MAG INSTALL menu.

4 From the MAG INSTALL menu, use the BARO knob to select MOUNTING.

5 Rotate the BARO knob to highlight desired parameter.

6 Push the knob to select the item, then rotate the knob clockwise to increment, and counterclockwise to decrement the value. Enter the magnetometer installation settings (refer to Table 1 — Magnetometer Installation Settings).

Table 1 Magnetometer Installation Settings

Parameter	Setting
PITCH	1.9°
ROLL	2.8°
YAW	0.0°

7 Press the M button to exit the MAG MOUNTING screen and return to the MAG INSTALL menu.

8 From the MAG INSTALL menu, use the adjustment knob to select WING FLEX COMPENSATION.

9 Rotate the BARO knob to highlight desired parameter.

10 Push the knob to select the item, then rotate the knob clockwise to increment, and counter clockwise to decrement the value. Enter the wing flex compensation settings (refer to Table 2 — Wing Flex Compensation Settings.

Table 2 Wing Flex Compensation Settings

Parameter	Setting
NO COMPENSATION	20 kts
FULL COMPENSATION	40 kts
PITCH COMPENSATION	0.0°
ROLL COMPENSATION	1.3°

11 Press the M button to exit the WING FLEX COMPENSATION screen and return to the MAG INSTALL menu. Press the M button (EXIT) to enter the MAG INSTALL SAVE menu.

12 Press the SAVE button (SAVE) to save changes or press the M button (EXIT) to exit without saving before returning to the SETUP MENU.

13 Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

14 Set the ESIS POWER switch to OFF.

2) Heading Calibration

a) ESIS GH–3100 (Pre Mod 6/2170)

Refer to Section 3.2.2.2.2 of the GH–3100 Electronic Standby Instrument System Installation and Operation Manual, part number TP–559, at the latest applicable revision.

b) ESIS GH–3900.2 (Post Mod 6/2170)

It is recommended that the entire procedure be reviewed prior to beginning MAG swing.

Note
- All installation angles (ANGLES menu and MAG MOUNTING menu) must be entered and saved, and the indicator must be re-booted prior to starting the MAG swing procedure.
- The MAG swing procedure must be done in an open area that allows the aircraft engines to be running, and can be safely swung.
- The MAG swing procedure must be done in a similar ambient magnetic field on the ground, as exists when the aircraft is in-flight (e.g. aircraft engine(s) should be running and all avionics equipment should be powered).
- Avoid being in the proximity of objects that may alter the magnetic field around the Magnetometer (e.g. rebar in concrete, metal building, etc).

1 Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2 Start both engines (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

3 Position the aircraft in an area (preferably a compass rose) free of distortions (no rebar), in the earth's magnetic field where it can swing 360°.

4 To access maintenance menu, press and hold the M key and adjustment knob; select ESIS POWER switch to ARM.

5 Rotate the adjustment knob to select MAG SWING; press the adjustment knob to enter the MAG SWING menu. A PRESS KNOB TO START message is shown at the top of the screen if valid magnetometer data is being received by the indicator.

6 With the aircraft in position (on any heading) and stationary, press the adjustment knob to begin alignment.

7 A WAIT message is shown at the top of the screen and an ALIGNING message is shown below it. The bar to the right of the ALIGNING message represents the alignment progress.

8 When the WAIT message is replaced with 360° message, begin swinging the aircraft. The ALIGNING message is replaced by SWINGING. The bar to the right of the SWINGING message represents the progress of the swing procedure.

9 Swing the aircraft in a complete 360° clockwise turn, until VERIFYING message is shown. During the swing procedure, monitor the Turn Rate bar. The pointer (white triangle) must not enter the red area of the bar.

Note
- If the pace of the swing is too fast (greater than 9° per second), a FAILED message is shown at the top of the screen resulting in a brief OVERRATE message, followed by the removal of the SWINGING message and progress bar.
- If the pace of the swing is too slow (takes longer than 300 seconds), a FAILED message is shown at the top of the screen resulting in the removal of the SWINGING message and progress bar.
- If a problem with magnetometer data is detected during the swing procedure, a FAILED message is shown at the top of the screen.
- If FAILED message is witnessed, restart the procedure from Para 3.D.2).b).5.

10 When the MAG swing procedure is complete, the progress bar is removed and a VERIFYING message is shown at the top of the screen.

11 If the FIELD VAR is less than or equal to 5mG, the message SWING OK is shown at the top of the screen. If the FIELD VAR is greater than 5mG, the message FAILED is shown at the top of the screen. If FAILED message is witnessed, restart the procedure fromPara 3.D.2).b).5

12 Press the M button to exit the MAG SWING menu. Select SAVE CHANGES when returning to the SETUP MENU.

13 Cycle power to the GH–3900.2 to apply swing data to unit.

14 Do a calibration check, refer to Para 3.D.3).

3) Calibration Check

a) Select ESIS POWER switch to the ARM position.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Start both engines (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

d) Allow ESIS to complete self-test/alignment process.

e) Position the aircraft in an area (preferably a compass rose) free of distortions (no rebar), in the earth's magnetic field where it can swing 360°.

Note To speed up the following steps, when aligned at the desired heading, instead of waiting 3 minutes for the heading to stabilize, initiate a manual alignment by selecting "Alignment >Initiate Alignment" from the ESIS menu.

f) Align the aircraft to a cardinal heading (e.g. North, East, South, or West). Allow the heading to stabilize by waiting a minimum of 3 minutes. Ensure heading is within ±4.0° of the known magnetic heading.

g) Align the aircraft to a heading 45° apart from first position. Allow the heading to stabilize by waiting a minimum of 3 minutes. Ensure heading is within ±4.0° of the known magnetic heading.

h) Align aircraft to remaining headings at 45° increments. Allow the heading to stabilize by waiting a minimum of 3 minutes. Ensure heading is within ±4.0° of the known magnetic heading.

E. Gear Configuration Change

1) ESIS GH–3100 (Pre Mod 6/2170)

The GH–3100 incorporates a Detachable Configuration Module (DCM) to change between various gear configurations.

To switch configurations, do the following:

a) Remove existing DCM, refer to Para 4.C.1).

b) Install new DCM, refer to Para 4.C.2).

c) Do a configuration verification, refer to Para 3.F.

2) ESIS GH–3900.2 (Post Mod 6/2170)

The GH–3900.2 incorporates a Data Configuration Module (DCM), tied into the ESIS wiring harness, to store gear configuration and a maintenance connector to load various gear configurations.

To switch configurations, do the following:

a) Apply external power to aircraft (refer to 12-00-00, Servicing – General Data).

b) To access MAINTENANCE menu, press and hold the M key and adjustment knob; select ESIS POWER switch to ARM.

c) Rotate the adjustment knob to select DATA LOAD; press the adjustment knob to enter the DATA LOAD menu.

d) Press the adjustment knob to Activate Data Load and place the unit in Date Load Mode.

e) Connect the maintenance computer to the ESIS connector on the maintenance panel using a serial cable, or RS–232/USB converter.

f) On the maintenance computer, start the GH–3900.2 Field Loader software (Part Number 576–6158–02).

g) Select the appropriate serial (COM) port, press the Connect button to establish communication between the GH–3900.2 and the maintenance computer.

h) Press the Browse button and locate desired configuration file (.bin file extension) that matches the aircraft gear configuration. Select the file and press the Update button.

i) During the transfer of data the indicator displays “Data Load: TRANSFER IN PROGRESS”.

j) If the transfer of data is successful, the indicator shows the message “Data Load: TRANSFER COMPLETE, READY”.

Note If an error message is displayed, refer to Section 3.3.1.1 of GH–3900.2 Installation Manual, part number 0040–34401–01, at the latest applicable revision.

k) Press the Disconnect button on the GH–3900.2 Field Loader software and remove RS–232 cable.

l) Select ESIS POWER switch to OFF. Remove external power from aircraft.

m) Do a configuration verification, refer to Para 3.F.

F. Configuration Verification

1) Select ESIS POWER switch to the ARM position.

2) When ESIS splash screen appears, note the aircraft model name, DHC-6-400 XXXXXx. (XXXXXX = LAND, SKIS, or FLOATS).

3) Confirm LAND, SKIS, or FLOATS matches the current gear configuration.

G. Altimeter System Test

1) Select ESIS POWER switch to the ARM position.

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Wait several minutes for the aircraft electrical system to initialize and the altitude flags to clear.

4) Leak test the Pitot-Static System. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test).

5) Ensure that the (ESIS) STATIC SOURCE valve is in the NORMAL position.

6) Connect the air data test set to the RH pitot head and upper static port on the aircraft. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test).

7) Complete the Altitude Calibration Table (refer toTable 3 — Altitude Calibration Table).

Table 3 Altitude Calibration Table

TEST SET ALTITUDE (FEET)	TOLERANCE (± FEET)	RECORD ERROR	REMARKS	INITIALS
		ESIS		
– 1,000	20			
0	20			
500	20			
1,000	20			
1,500	25			
2,000	25			
3,000	25			
4,000	25			
5,000	25			
6,000	25			
8,000	30			
10,000	30			
12,000	35			
14,000	40			
16,000	40			
18,000	45			

Table 3 Altitude Calibration Table (continued)

TEST SET ALTITUDE (FEET)	TOLERANCE (± FEET)	RECORD ERROR	REMARKS	INITIALS
		ESIS		
20,000	50			
22,000	50			
25,000	50			

4. Removal/Installation

WARNING INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:

CHASSIS GROUNDING – PANEL/PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

Note The Electronic Standby Instrument System has some additional shielding requirements; refer to the notes on the applicable wiring diagram.

A. Electronic Standby Instrument System (ESIS)

1) Removal of ESIS

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the ESIS INST (P5) circuit breaker and fit safety clip.

c) Loosen the (2) large clamp screws on the top right and bottom left corners of the indicator.

d) Carefully pull the ESIS out of the instrument panel to give access to the electrical connector and the pitot and static fittings.

e) Disconnect electrical connector (P287) and configuration module (P288). Ensure configuration module chain is attached to electrical harness.

f) Disconnect pitot and static fittings on rear of instrument.

g) Remove instrument from aircraft.

Note Stow connectors and fittings as required.

2) Installation of ESIS

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the ESIS INST (P5) circuit breaker is opened and fitted with a safety clip.

c) Hold the ESIS near its mount in the instrument panel.

d) Connect the pitot and static fittings to the ESIS.

e) Connect the electrical connector (P287) and configuration module (P288) to the ESIS.

f) Slide the ESIS fully into its mount.

g) While pushing on the ESIS towards the instrument panel, tighten the (2) large clamp screws.

h) Remove the safety clip and close the ESIS INST (P5) circuit breaker.

i) Do pitot and static leak tests. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data).

j) Do an ESIS Function Test, refer to Para 3.C.1).

B. Magnetometer

CAUTION **DO NOT EXPOSE THE MAGNETOMETER TO MAGNETIC FIELDS GREATER THAN 4 GAUSS DURING PACKAGING, STORAGE, OR TRANSPORT IN ORDER TO PREVENT MAGNETIZATION OF THE MAGNETOMETER.**

1) Removal of Magnetometer

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the ESIS MAG (Q5) circuit breaker and fit safety clip.

c) Gain access to the magnetometer by removing the (4) non-ferrous mounting screws from the access panel in the underside of the left wing. (Refer to 12-00-00, Servicing – General Data).

d) Disconnect electrical connector (P286).

Note Stow connectors as required.

e) Remove the (3) non-ferrous magnetometer mounting screws.

f) Lower Magnetometer from shelf while feeding the pigtail through the opening.

2) Installation of Magnetometer

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the ESIS INST (P5) circuit breaker is opened and fitted with a safety clip.

c) Feed connector pigtail through opening in mounting shelf and connect electrical connector.

d) Install the (3) non-ferrous magnetometer mounting screws.

e) Install access panel with (4) non-ferrous mounting screws.

f) Remove the safety clip and close ESIS MAG (Q5) circuit breaker.

g) Do an ESIS calibration check, refer to Para 3.D.3). If check fails, do an ESIS heading calibration, refer to Para 3.D.2).

C. Detachable Configuration Module (Pre Mod 6/2170)

1) Removal of Detachable Configuration Module

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the ESIS INST (P5) circuit breaker and fit safety clip.

c) Loosen the (2) large clamp screws on the top right and bottom left corners of the indicator.

d) Carefully pull the ESIS out of the instrument panel to give access to the configuration module (P288).

e) Disconnect configuration module (P288).

f) Detach configuration module chain from electrical harness.

2) Installation of Detachable Configuration Module

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the ESIS INST (P5) circuit breaker is opened and fitted with a safety clip.

c) Attach configuration module chain to electrical harness.

d) Connect configuration module (P288) to J2 of the ESIS.

e) Remove the safety clip and close ESIS INST (P5) circuit breaker.

f) Do an ESIS attitude calibration, refer to Para 3.D.1).

g) Do an ESIS heading calibration, refer to Para 3.D.2).

D. ESIS Emergency Battery (PS-850B)

1) Removal of ESIS Emergency Battery

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the ESIS BATTERY (N5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the rear avionics shelves through the rear baggage compartment.

d) Loosen retaining screw clamp from the front and slide the unit out of the ATR Rack.

e) Remove unit from aircraft.

Note Stow connectors as required.

f) Visually inspect ATR Rack. If damaged, replace as per Para 4.E.1).

2) Installation of ESIS Emergency Battery

Note If ESIS emergency battery has been in storage or, the battery capacity is unknown, do the Battery Capacity Test detailed in Para 3.B.3) prior to installation into aircraft.

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the ESIS BATTERY (N5) circuit breaker is opened and fitted with safety clip.

c) Slide the unit onto the ATR Rack. Tighten the retaining screw clamp at the front lip of the ESIS Emergency Battery unit.

d) Remove safety clip and close ESIS BATTERY (N5) circuit breaker on the avionics circuit breaker panel.

e) Do the Periodic Battery Check, refer to Para 3.B.1).

f) Do the ESIS Operational Test, refer to Para 3.A.1).

E. ESIS Emergency Battery ATR Rack

1) Removal of ESIS Emergency Battery ATR Rack

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the rear avionics shelves through the rear baggage compartment.

c) Remove ESIS Emergency Battery from ATR Rack, refer to Para 4.D.1).

d) Remove the (4) screws that secure the ATR Rack to the bottom of the avionics rack.

e) Remove the (4) mounting screws from the electrical connector. Remove connector from the ATR Rack.

f) Remove ATR Rack from aircraft.

2) Installation of ESIS Emergency Battery ATR Rack

a) Ensure the aircraft electrical system if OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Install (4) mounting screws to secure the electrical connector to the ATR Rack.

c) Align the ATR Rack to the mounting holes on the rear avionics rack.

d) Install the four screws that secure the ATR Rack to the avionics rack.

5. Troubleshooting

A. Electronic Standby Instrument System

1) Ensure ESIS BATTERY (N5), ESIS INST (P5), and ESIS MAG (Q5) circuit breakers are closed.

2) Visually inspect ESIS and magnetometer connectors for damaged wiring or contacts. Ensure electrical connector on ESIS and magnetometer are fully seated and locked.

3) Do a pitot/static leak test. Refer to 34-11-00, Pitot Static System (Mod 6/2033) – General Data.

4) Ensure STANDBY INSTRUMENT STATIC SOURCE lever is in the NORMAL position.

5) Ensure pitot and static fittings are secure.

6) As applicable to the ESIS type installed, refer to Section 4.3 of the GH–3100 Electronic Standby Instrument System Installation and Operation Manual, part number TP–559, or Section 4.3 of the GH–3900.2 Electronic Standby Instrument System Installation and Operation Manual, part number 0040–34401–01, at the latest applicable revision.

7) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Diagrams Manual PSM 1-64-2W.

8) If repair or service is required, send to appropriately rated avionics repair facility.

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Recorders – General Data

1. General

The Recording System consists of two types of crash protected digital recording devices; the Cockpit Voice Recorder (CVR), and the Flight Data Recorder (FDR). These devices capture the cockpit communications and the flight data information for retrieval in the event of a crash.

For aircraft incorporating customer standard option Mod S.O.O. 6292, the Cockpit Voice Recorder (CVR) is removed from the aircraft and associated systems.

For aircraft incorporating customer standard option Mod S.O.O. 6299, the Flight Data Recorder (FDR) is removed from aircraft and associated systems.

An optional recorder is offered for Engine Condition Trend Monitoring (ECTM)/Flight Operations Quality Assurance (FOQA) Data Capture (S.O.O. 6288) system which consists of an IONode recording unit and a multi element GPS/WiFi antenna. The IONode continuously records data from the aircraft Flight Data Recorder, and its own internal sensors. Data is subsequently downloaded to a PC for analysis.

For aircraft incorporating Mod 6/2303, provisions have been made for 512 Word FDR outputs for Apex ARINC 717 output data stream only.

For aircraft with customer Standard Option Order S.O.O. 6388 installed, the Cockpit Voice Recorder (CVR) and the Flight Data Recorder (FDR) are not installed.

For aircraft with S.O.O. 6388 incorporated, the Cockpit Voice Recorder (CVR) and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6384.

For aircraft with S.O.O. 6388 incorporated, the 512 Word Flight Data Recorder (FDR) and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6385.

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Flight Data Recorder (FDR) – General Data

1. General

The Flight Data Recorder (FDR) is used to record aircraft system data for the purpose of investigating aircraft accidents and incidents. The FDR uses solid state flash memory modules to record the data for up to 25 hours. The unit is also equipped with an Underwater Locator Beacon (ULB).

For aircraft with S.O.O. 6388 incorporated, the Flight Data Recorder (FDR) is not installed.

The Flight Data Recorder (FDR) is installed as part of the IFR Operations Bundle Customer Order Option S.O.O. 6209.

Pre Mod 6/2332 aircraft uses a Honeywell AR–256 FDR. Post Mod 6/2332 aircraft uses an L–3 FA2100 FDR.

For aircraft with S.O.O. 6209 incorporated, the Flight Data Recorder (FDR) and associated systems may be removed from the aircraft, by incorporating customer Standard Order Option S.O.O. 6299.

For aircraft with S.O.O. 6388 incorporated, the Flight Data Recorder (FDR) and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6385.

2. Description

The Honeywell AR–256 Flight Data Recorder (FDR) (Pre Mod 6/2332) is a 256 word per second FDR with 25 hour recording capacity. The AR–256 uses flash memory modules to record aircraft data, and has no moving parts. It is equipped with an underwater location beacon.

The L–3 FA2100 Flight Data Recorder (FDR) (Post Mod 6/2332) is a 512 word per second FDR with 25 hour recording capacity. The FA2100 uses flash memory modules to record aircraft data, and has no moving parts. It is equipped with an underwater location beacon.

The Flight Data Recorder is contained in an environmentally protective enclosure. The recorder is located in the rear avionics compartment, at approximately station 429, where the survival rate is high. For Pre Mod 6/2332, the recorder has a flange which allows for the unit to be mounted directly onto the avionics shelf. For Post Mod 6/2332, the recorder uses a mounting tray which is mounted to the avionics shelf.

28 Volt DC power is provided from the left bus through a 5 amp FDR circuit breaker on the avionics circuit breaker panel refer to 24-00-00, Electrical Power – General Data.

The following aircraft systems interface with the FDR system:

A. Data Acquisition Unit (DAU)

The DAU monitors & conditions the signals from the aircraft system sensors and send the information to the MAU via ARINC 429 refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data.

B. Accelerometer

A tri-axial accelerometer is installed to provide the FDR with normal acceleration data via the MAU. The accelerometer is located in the forward avionics compartment on the port side of the middle avionics shelf.

C. Modular Avionics Unit (MAU)

The MAU accepts FDR data from various aircraft systems and compiles it onto the ARINC 717 data bus.

D. Impact Switch

The impact switch is an inertia operated device which removes the power to the recorder in the event of a crash, thus keeping the recorded data intact. A deceleration of 3g or greater will trigger the switch. The

impact switch is located in the rear avionics compartment, outboard of the FDR, and is the aft of the two switches. For Pre Mod 6/2256, a red lamp on the switch will illuminate when the switch has been tripped.

E. FDR Power Relay

Under normal operation, the FDR Fail wire will provide a ground to the MAU. If a fault condition is detected by the Built-In Test Equipment (BITE), the output will open and cause a caution level CAS message to be displayed. The FDR Fail output remains grounded when power is removed from the FDR, which would indicate normal operation. Therefore, a relay is connected to FDR power to open the FDR Fail wire when power is removed.

On aircraft with Mod 6/2303 incorporated, the stowed FDR Status wire is connected and the FDR Power Relay is removed. The FDR Status MAU input triggers the caution level CAS when power is removed from the FDR, eliminating the need for the FDR Power Relay.

F. Engine Condition Trend Monitoring (ECTM)/Flight Operations Quality Assurance (FOQA) Data Capture

The ECTM/FOQA Data Capture (S.O.O. 6288) receives the data from the FDR via ARINC 717 output port on the FDR (refer to 31-33-00, Engine Condition Trend Monitoring (ECTM)/Flight Operations Quality Assurance (FOQA) Data Capture (S.O.O. 6288)).

G. Underwater Location Beacon

The Underwater Location Beacon (ULB) is a cylindrical transmitter mounted to the front of the crash survivable memory unit. When the sensor comes in contact with water the ULB emits ultrasonic pulses to assist in the location of the submerged recorder.

3. Adjustment/Test

A. Flight Data Recorder

If a fault condition is detected by the Built-In Test Equipment (BITE), a caution level CAS message will be displayed. Pulling the FDR (M3) circuit breaker should trigger the FDR Fail CAS.

B. System Function and Correlation Check

Note The FDR correlation check must be accomplished with at least one activated GPS for valid time parameters.

Equipment and Materials:

The following equipment, or other units having equivalent capabilities, are required to do the function tests as outlined from Para 3.B.4) to Para 3.B.25). The equipment should be calibrated and so identified in accordance with local specifications and requirements:

- Digital Voltmeter (Fluke 8000A or equivalent)
- Air Data Test Equipment (ADTS 505 or equivalent)
- 0-25 VDC variable power supply
- 0-100 Hz frequency generator
- DC ground power unit
- HHDLU (Handheld Download Unit) – Honeywell PN 964–0446–001 (Pre Mod 6/2332)
- HHDLU Cable Assembly – Honeywell PN 704–2688–001 (Pre Mod 6/2332)
- HHDLU PCMCIA Memory Card – Honeywell PN 700–1679–006 (Pre Mod 6/2332)
- PI/2 (Portable Interface) – L–3 PN 17TES0062 (or equivalent) (Post Mod 6/2332)

Preparation:

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

   Note Ensure a GPS signal is being received during the following tests.

2) Complete the tests described in the following tables.

3) Upon completion of the testing, download the FDR data using the HHDLU (Pre Mod 6/2332) or PI/2 (Post Mod 6/2332) and forward the data to an accredited FDR facility for correlation.

   Note If the data cannot be downloaded following the completion of the test, disable recording by opening the FDR circuit breaker until such time the recorder can be downloaded.

4) Test Altitude Parameter (Table 1 — Altitude Parameter)

   a) Connect Air Data Test Equipment to static ports. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data).

   b) Set altitude per values in Table 1 — Altitude Parameter and record in space provided. Verify test set readout is within the specified tolerances and that the Apex display reads within the tolerances specified. If readouts are not within the specified tolerances, troubleshoot system per Para 5.

      Note Rate of change of pressure should not be more than 3000 feet per minute during test.

      When moving from one test point to the next, the pressure or vacuum should at no time exceed the specified test points.

      After one minute at each specified test pressure, error should not exceed tolerances given in Table 1 — Altitude Parameter.

Table 1 Altitude Parameter

(Aircraft SN/Registration): _____/_____					
Altitude Tester (feet)	**Pilots Altitude Readout (feet)**	**Co-Pilots Altitude Readout (feet)**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Tolerances**	**Initials / Stamp**
5,000			Date: Time:	± 25 feet	
10,000			Time:	± 35 feet	
15,000			Time:	± 45 feet	
20,000			Time:	± 50 feet	
25,000			Time:	± 50 feet	

   Note Tolerances per KSG–7200 Installation Manual.

5) Test Airspeed Parameter (Table 2 — Airspeed Parameter)

   a) Connect Air Data Test Equipment to pitot heads. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data).

   b) With the aircraft on ground power and Apex powered up, set airspeed per values in Table 2 — Airspeed Parameter and record in space provided.

c) Tolerance for correlation to be ± 5 KIAS of the readout value.

Table 2 Airspeed Parameter

(Aircraft SN/Registration): _____/_____					
Airspeed Tester (KIAS)	**Pilots Airspeed Readout (KIAS)**	**Co-Pilots Airspeed Readout (KIAS)**	**Date (DD /MM/YY and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
50			Date: Time:		
100			Time:		
150			Time:		
170			Time:		

6) Test Heading Parameter (Table 3 — Heading Parameter)

a) With the aircraft on ground power and Apex powered up, or alternatively with the engines running, orientate the aircraft so that the heading displayed on Apex corresponds to the values listed in Table 3 — Heading Parameter.

Note The heading source type is automatically recorded, so selection of heading source is not required.

b) Set heading per values in Table 3 — Heading Parameter and record on calibration record card.

Table 3 Heading Parameter

(Aircraft SN/Registration): _____/_____					
Target Heading (Degrees)	**Pilots HSI Readout (Degrees)**	**Co-Pilots HSI Heading Readout (Degrees)**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
0			Date: Time:	± 2 degrees	
90			Time:	± 2 degrees	
180			Time:	± 2 degrees	
270			Time:	± 2 degrees	

7) Test Pitch Attitude Parameter (Table 4 — Pitch Attitude Parameter)

a) Release ADAHRS from mounting tray. With the aircraft on ground power and Apex powered up, angle ADAHRS to produce the desired angular orientation on the ADI display and record in space provided in Table 4 — Pitch Attitude Parameter.

b) Tolerance for correlation to be ± 2 degrees of the readout value.

Table 4 Pitch Attitude Parameter

(Aircraft SN/Registration): _____/_____					
Desired Pitch Angle (Degrees)	**Pilots ADI Readout (Degrees)**	**Co-Pilots ADI Readout (Degrees)**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
–30			Date: Time:		
0			Time:		
30			Time:		

8) Test Manual Radio Transmitter Keying Parameter (Table 5 — Radio Transmitter Keying Parameter)

   a) With the aircraft on ground power and Apex powered up, set radio to appropriate test frequency for testing radio function.

   b) Key radio per Table 5 — Radio Transmitter Keying Parameter and record data in space provided.

Table 5 Radio Transmitter Keying Parameter

(Aircraft SN/Registration): _____/_____				
XMIT Radio Switch to Key	**Radio Frequency Used**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Pilot's		Date: Time:		
Co-pilot's		Time:		

9) Test Engine Torque Parameter (Table 6 — Torque Parameter)

   a) Connect 0-5 volt power supply to pins A (+) and B (–) of the left and right torque transducer connector leading to the DAUs.

   b) With the aircraft on ground power and Apex powered up, set output voltage (pressure) to those listed in Table 6 — Torque Parameter and record in space provided.

   c) Tolerance for correlation to be ± 1.0 PSI of the readout value.

Table 6 Torque Parameter

(Aircraft SN/Registration): _____/_____					
Voltage Applied to Pins A (+) and B (-) (VDC)	**Left Engine Torque Readout (PSI)**	**Right Engine Torque Readout (PSI)**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Intials / Stamp**
0.5 VDC = 0 PSI			Date: Time:		
2.75 VDC = 35 PSI			Time:		
4.68 VDC = 65 PSI			Time:		

10) Test Engine Propeller RPM Parameter (Table 7 — N_P Parameter)

a) Connect frequency generator to pins A (+) and B (–) of the left and right N_P transducer connector leading to the DAUs.

Note Frequency is to be sine wave and a minimum of 1 V_{P-P} in amplitude, but not exceeding 5 V_{P-P}.

b) With the aircraft on ground power and Apex powered up, set frequency per values in Table 7 — N_P Parameter and record in space provided.

c) Tolerance for correlation to be ± 2% of the readout value.

Table 7 N_P Parameter

(Aircraft SN/Registration): _____/_____					
Frequency Applied to Pins A (+) and B (-) (Hz)	**Left Engine N_P Readout (%)**	**Right Engine N_P Readout (%)**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
35 Hz = 50% N_P			Date: Time:		
70 Hz = 100% N_P			Time:		
77 Hz = 110% N_P			Time:		

11) Test Pitch/Roll/Yaw Control Parameters (Table 8 — Pitch/Roll/Yaw Parameter)

a) With the aircraft on ground power and Apex powered up, starting in the neutral position, set Pitch/Roll/Yaw Control (elevator) position per values in Table 8 — Pitch/Roll/Yaw Parameter and record in space provided. Maintain each position for a minimum of 10 seconds. Note the time for each position.

b) Record all control surface position values during correlation.

Table 8 Pitch/Roll/Yaw Parameter

(Aircraft SN/Registration): _____/_____					
Parameter	**Position 1**	**Position 2**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Pitch (Elevator)	Full Up	Full Down	Date: Time:		
Roll (Aileron) (Post Mod 6/2332)	Full Left	Full Right	Time:		
Yaw (Rudder) (Post Mod 6/2332)	Full Left	Full Right	Time:		

12) Test Pitch/Roll/Yaw Trim Parameters (Table 9 — Pitch/Roll/Yaw Trim Parameter)

a) With the aircraft on ground power and Apex powered up, starting in the neutral position, set trims per values in Table 9 — Pitch/Roll/Yaw Trim Parameter and record in space provided. Maintain each position for a minimum of 10 seconds. Note the time for each position.

b) Record all control surface position values during correlation.

Table 9 Pitch/Roll/Yaw Trim Parameter

(Aircraft SN/Registration): _____/_____					
Parameter	**Position 1**	**Position 2**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Pitch Trim (Elevator)	Full Up	Full Down	Date: Time:		
Roll Trim (Aileron)	Full Left	Full Right	Time:		
Yaw Trim (Rudder)	Full Left	Full Right	Time:		

13) Test Flap Parameter (Table 10 — Flap Position Parameter)

a) With ground power turned on and the hydraulic pump active; actuate flaps to the values listed in Table 10 — Flap Position Parameter and record results in the space provided. Maintain each position for a minimum of 10 seconds. Note the time for each position.

b) Record all control surface position values during correlation.

Table 10 Flap Position Parameter

(Aircraft SN/Registration): _____/_____					
Flap Lever Position	**Angle Measured on Inboard Flap (Degrees)**	**Apex Flap Position Indicator (Degrees)**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
0 Degrees			Date: Time:		
10 Degrees			Time:		
20 Degrees			Time:		
37 Degrees (refer to Note)			Time:		

Note 37 Degree position corresponds to an inboard flap angle of 40 degrees. All other angles correspond to the listed flap lever position angle.

14) Test Propeller Beta Parameter (Table 11 — Propeller Beta Parameter)

a) Lower the left and right engine nacelle cowlings. With the aircraft on ground power and Apex powered up, depress the left, then the right prop beta microswitches for 5 seconds each and record any remarks in the space provided.

Table 11 Propeller Beta Parameter

(Aircraft SN/Registration): _____/_____				
Beta Switch to Depress	**Press and hold on for:**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Left Beta	5 seconds	Date: Time:		
Right Beta	5 seconds	Time:		

15) Test OAT Parameter (Table 12 — OAT Parameter)

a) With the aircraft on ground power and Apex powered up, record the ambient OAT displayed on Apex. It is not necessary to remove the probe from the aircraft for this test and an insulated (Styrofoam) cup is all that is required. Place the probe in an ice bath until the OAT readout on Apex approaches 0°C and record the results. The probe and readout can be within 4°C of zero and that will be sufficient, as the element is difficult to submerge while mounted on the aircraft.

b) Place the probe in sufficiently hot water until the OAT displayed on Apex reads 40°C, record the OAT displayed on Apex.

c) Tolerance for correlation to be ± 2C of the readout value.

Table 12 OAT Parameter

Aircraft SN/Registration): _____/_____				
Target Temp.	**Apex OAT Reading (°C)**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Ambient		Date: Time:		
0°C		Time:		
40°C		Time:		

16) Test Selected Barometric Setting Parameter (Table 13 — Selected Barometric Setting Parameter)

a) With the aircraft on ground power and Apex powered up, starting at the standard setting (STD), set barometric setting per values in Table 13 — Selected Barometric Setting Parameter and record in space provided. Ensure Baro Sync option is enabled.

b) Tolerance for correlation to be ± 0.01 inHg or ± 1 HPa of the readout value.

Table 13 Selected Barometric Setting Parameter

(Aircraft SN/Registration): _____/_____					
Selected Baro Setting (in HG / mb)	**ADI Readout (in HG)**	**ADI Readout (mb)**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
STD (29.92 / 1013)			Date: Time:		
30.92 / 1047			Time:		
28.92 / 979			Time:		

17) Test Selected Altitude Parameter (Table 14 — Selected Altitude Parameter)

a) With the aircraft on ground power and Apex powered up, adjust selected altitude using the ALT knob on the Flight Controller per values in Table 14 — Selected Altitude Parameter and record in space provided. Selected altitude adjustment is made with the selection of the ALT knob. Refer to Altitude Select Readout and Bug section in the Primus Apex Pilots Guide (Page 6-36) for further altitude function explanation.

b) Tolerance for correlation to be ± 25 ft of the readout value.

Table 14 Selected Altitude Parameter

(Aircraft SN/Registration): _____/_____				
Selected Altitude (feet)	**Apex Readout (feet)**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
0		Date: Time:		
5,000		Time:		

Table 14 Selected Altitude Parameter (continued)

(Aircraft SN/Registration): _____/_____				
Selected Altitude (feet)	**Apex Readout (feet)**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
10,000		Time:		
15,000		Time:		
20,000		Time:		
25,000		Time:		

18) Test Selected Heading Parameter (Table 15 — Selected Heading Parameter)

a) With the aircraft on ground power and Apex powered up, adjust selected heading using the HDG knob on the Flight Controller per values in Table 15 — Selected Heading Parameter and record in space provided. Selected altitude adjustment is made with the selection of the ALT knob.

b) Tolerance for correlation to be ± 1 degree of the readout value.

Table 15 Selected Heading Parameter

(Aircraft SN/Registration): _____/_____				
Selected Heading (Degrees)	**Apex Readout (Degrees)**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
0		Date: Time:		
90		Time:		
180		Time:		
270		Time:		

19) Test Selected Flight Path Parameter (Table 16 — Selected Flight Path Parameter)

a) Activate a direct-to waypoint by pressing the Direct-To button on the MF controller, entering a waypoint, and pressing ENT button to confirm and activate. Record the desired track displayed on the PFD in Table 16 — Selected Flight Path Parameter, in space provided. Repeat with additional waypoints.

b) Tolerance for correlation to be ± 5 degrees of the readout value.

Table 16 Selected Flight Path Parameter

(Aircraft SN/Registration): _____/_____				
Selected Waypoint	**Desired Track (Degrees)**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Intials / Stamp**
Waypoint 1		Date: Time:		
Waypoint 2		Time:		

Table 16 Selected Flight Path Parameter (continued)

(Aircraft SN/Registration): _____/_____				
Selected Waypoint	**Desired Track (Degrees)**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Intials / Stamp**
Waypoint 3		Time:		
Waypoint 4		Time:		

20) Test Selected Decision Height Parameter (Table 17 — Selected Decision Height Parameter)

a) With the aircraft on ground power and Apex powered up, adjust selected decision height per values in Table 17 — Selected Decision Height Parameter and record in space provided.

b) Tolerance for correlation to be ± 25 ft of the readout value.

Table 17 Selected Decision Height Parameter

(Aircraft SN/Registration): _____/_____				
Selected Decision Height (feet)	**Apex Readout (feet)**	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials /Stamp**
200		Date: Time:		
800		Time:		
1,600		Time:		
2,000		Time:		

21) Test Accelerometer (Table 18 — Normal Acceleration Parameter)

a) With the aircraft on ground power and Apex powered up, remove the accelerometer and adjust accelerometer orientation to match the values in Table 18 — Normal Acceleration Parameter and record in space provided. Hold each orientation for 5 seconds to ensure sufficient data is obtained by the FDR.

Table 18 Normal Acceleration Parameter

(Aircraft SN/Registration): _____/_____			
Orientation of Base Plate from Avionics Shelf	**Date (DD/MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Down (Base plate facing downward, as installed)	Date: Time:		
Forward (rotate base plate forward 90 degrees)	Time:		
Up (rotate base plate 180 degrees)	Time:		

22) Test EFIS Display Format Parameters (Table 19 — EFIS Display Format Parameter)

a) With the aircraft on ground power and Apex powered up, trigger and/or record the following on Apex.

b) Set parameters per values in Table 19 — EFIS Display Format Parameter and record on calibration record card. Leave each of the selected inputs on for at least 4 full seconds to ensure the FDR has read all the inputs.

c) Tolerance for correlation to be ± 1 degree of the readout value.

Table 19 EFIS Display Format Parameter

(Aircraft SN/Registration): _____/_____					
Parameter	**Aircraft Actions Required**	**Apex Readout**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Display Unit 1	Using reversion panel, turn off Pilot's display	N/A	Date: Time:		
Display Unit 2	Using reversion panel, turn off upper MFD display	N/A	Time:		
Display Unit 3	Using reversion panel, turn off lower MFD display	N/A	Time:		
Display Unit 4	Using reversion panel, turn off Co-Pilot's display	N/A	Time:		
	Turn all displays on				
AGM 1 Reversion	Set Co-Pilot's display to read from AGM 1	N/A	Time:		
AGM 2 Reversion	Set Pilot's display to read from AGM 2	N/A	Date: Time:		
	Reset AGM selection				
PFD 1 Selected ADAHRS	On the Pilot's PFD controller, select ADHRS button to switch to alternate ADAHRS side.	N/A	Time: Time:		

Table 19 EFIS Display Format Parameter (continued)

(Aircraft SN/Registration): _____/_____					
Parameter	**Aircraft Actions Required**	**Apex Readout**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
PFD 2 Selected ADAHRS	On the Co-Pilot's PFD controller, select ADHRS button to switch to alternate ADAHRS side.	N/A	Time:		
PFD 1 Selected Nav Source	Select LOC1, LOC2, FMS		Time:		
PFD 2 Selected Nav Source	Select LOC1, LOC2, FMS		Time:		
PFD 1 HSI Compass Format	ARC, ROSE		Time:		
PFD 2 HSI Compass Format	ARC, ROSE		Time:		
AGM 1 Composite Mode Selection	Turn off displays 2, 3, and 4.		Time:		
AGM 2 Composite Mode Selection	Turn off displays 1, 2, and 3.		Time:		
Selected Course	Select a course of 180 degrees on the Pilot's PFD controller. Tolerance for correlation to be ± 1 degree of the readout value.		Time:		
SVS-Pilot (For aircraft equipped with S.O.O. 6222 only)	Turn Pilot's SVS OFF, wait 10 seconds and then turn Pilot's SVS ON.				
SVS-Co-Pilot (For aircraft equipped with S.O.O. 6222 only)	Turn Co-Pilot's SVS OFF, wait 10 seconds and then turn Co-Pilot's SVS ON.				

23) Test Multi-Function/Engine/Alerts Display Format Parameters
(Table 20 — Multi-Function/Engine/Alerts Display Format Parameters)

a) With the aircraft on ground power and Apex powered up, trigger and/or record the following on Apex.

b) Set parameters per values in Table 20 — Multi-Function/Engine/Alerts Display Format Parameters and record on calibration record card. Leave each of the selected inputs on for at least 4 full seconds to ensure the FDR has read all the inputs.

Table 20 Multi-Function/Engine/Alerts Display Format Parameters

(Aircraft SN/Registration): _____/_____					
Parameter	**Aircraft Actions Required**	**Apex Readout**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
MFD Formats Swapped	Using the MFD button on the MFD controller, swap upper and lower MFDs.		Date: Time:		
MFD MAP /PLAN selected	Select North Up/Track Up		Time:		
INAV Selected Overlay Terrain	Select the MFD Map Overlay Terrain		Time:		
INAV Selected Overlay TCAS	Select the MFD Map Overlay TCAS		Time:		
INAV Lateral Map Half Range	Record the range value listed on the MFD map		Time:		

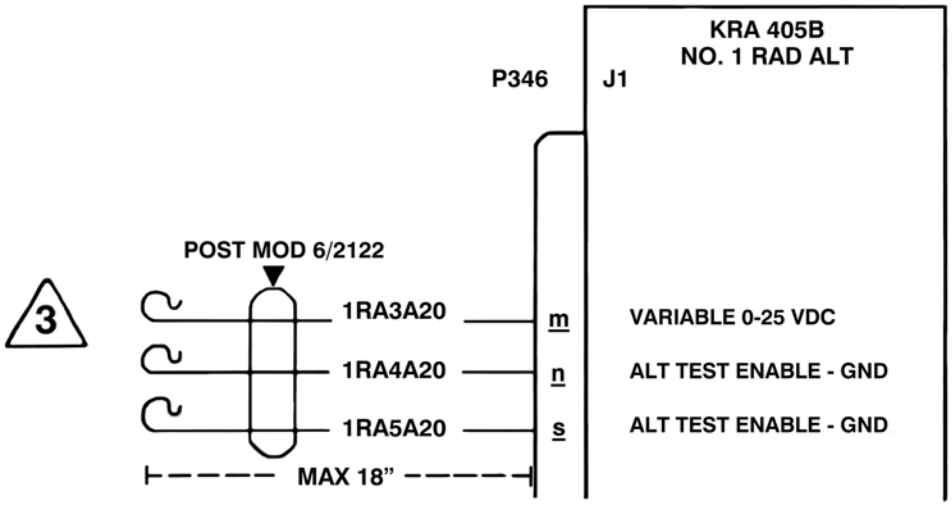
24) Radar Altimeter Display Format Parameters (Table 21 — Radar Altimeter Display Format Parameters)

a) With the aircraft on the ground power and Apex powered up, trigger and/or record the following on Apex.

b) Set the parameters per values in Table 21 — Radar Altimeter Display Format Parameters and record on calibration and record card. Leave each of the selected inputs on for at least 4 full seconds to ensure the FDR has read all the inputs.

Note Use connector P346 to ground pin n and pin s of J346 (J1 of KRA405B). Input the required DC test voltage with a 0-25 VDC variable power supply in pin m. Stub wires may be supplied, if not they can be installed (Max 18 inches in length). Cap and stow stub wires after test. Refer to Figure 1.

c) Tolerance for correlation to be ± 25 ft of the readout value.



3 **WIRES INSTALLED POST MOD 6/2122 ARE CAPPED AND STOWED DURING NORMAL OPERATION.**
FOR TESTING PURPOSES:
- 1RA3A20 CONNECTED TO EXTERNAL DC POWER SOURCE
- 1RA4A20, 1RA5A20 GROUNDED TO AIRFRAME

Wiring Diagram
Figure 1

Table 21 Radar Altimeter Display Format Parameters

(Aircraft SN/Registration):_____/_____					
Parameter	**Aircraft Actions Required**	**Apex Readout**	**Date (DD/MM /YY) and Time (UTC) Recorded**	**Remarks**	**Initials/Stamps**
RAD ALT #1	Test Input to Rad Alt per OEM. Input voltage required to record 0, 1000, and 2000 feet on Apex. Tolerance for correlation to be ± 25 ft of the readout value.	_____ft 0.00 VDC _____ft 10.00 VDC _____ft 20.00 VDC			
RAD ALT #2 (For aircraft equipped with S.O.O. 6218 only)	Test Input to Rad Alt per OEM. Input voltage required to record 0, 1000, and 2000 feet on Apex. Tolerance for correlation to be ± 25 ft of the readout value.	_____ft 0.00 VDC _____ft 10.00 VDC _____ft 20.00 VDC			

25) Test the following Miscellaneous Parameters (Table 22 — Miscellaneous Parameters)

a) With the aircraft on the ground power and APEX powered up, trigger and or record the following on Apex.

b) Set the parameter per values in Table 22 — Miscellaneous Parameters and record on calibration and record card. Leave each of the selected inputs on for at least 10 seconds to ensure FDR has read all the inputs.

c) Hydraulic Pressure

d) Brake Pressure

e) Left Bus Voltage and Current

f) Right Bus Voltage and Current

g) Battery Bus Voltage and Current

h) Fuel Quantity in Each Tank

i) Wing Tanks Installed (Y/N)

j) GPS position from Apex

k) Master Warning Reset (Pilot and Co-Pilot)

Table 22 Miscellaneous Parameters

(Aircraft SN/Registration): _____/_____					
Parameter	**Aircraft Actions Required**	**Apex Readout**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Hydraulic System Pressure	Tolerance for correlation to be ± 50 PSI of the readout value.	_____PSI	Date: Time:		
Brake Pressure	Engage parking brake. Tolerance for correlation to be ± 50 PSI of the readout value.	_____PSI	Time:		
Left Bus Voltage	Maintain steady ground power level. Tolerance for correlation to be ± 0.5 VDC of the readout value.	_____VDC	Time:		
Left Bus Current	Stabilized current draw, left engine start required. Switch left generator ON. Tolerance for correlation to be ± 2 Amps of the readout value.	_____Amps	Time:		
Right Bus Voltage	Maintain steady ground power level.	_____VDC	Time:		
Right Bus Current	Stabilized current draw, right engine start required. Switch right generator ON. Same voltage and current tolerances as listed above.	_____Amps	Date: Time:		

Table 22 Miscellaneous Parameters (continued)

(Aircraft SN/Registration): _____/_____					
Parameter	**Aircraft Actions Required**	**Apex Readout**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
Battery Bus Voltage	Switch to battery power and maintain a steady ground power level.	_____VDC	Time:		
Battery Bus Current	Switch to battery power and maintain a stabilized current draw. Return to ground power after recording current.	_____Amps	Time:		
Fuel Quantity Aft Tank	None	_____lbs	Time:		
Fuel Quantity Fwd Tank	Tolerance for correlation to be ± 5 lbs of the readout value.	_____lbs	Time:		
Fuel Quantity Left Wing Tank (if equipped)	None	_____lbs	Time:		
Fuel Quantity Right Wing Tank (if equipped)	None	_____lbs	Time:		
GPS position (Long/Lat)	Initialize GPS to current location.	Long: Lat:	Time:		
Pilot Master Warning Reset	Pull Hydraulic CB to trigger warning. Reset after test.	N/A	Time:		
Co-Pilot Master Warning Reset	Pull Hydraulic CB to trigger warning. Reset after test.	N/A	Time:		
AP Equipped AC: Manual Pitch Trim Up (For aircraft equipped with S.O.O. 6221 only)	Press and hold the Pitch Trim Up lever on the Pilot's control wheel.		Time:		

Table 22 Miscellaneous Parameters (continued)

(Aircraft SN/Registration): _____/_____					
Parameter	**Aircraft Actions Required**	**Apex Readout**	**Date (DD /MM/YY) and Time (UTC) Recorded**	**Remarks**	**Initials / Stamp**
AP Equipped AC: Manual Pitch Trim Down (For aircraft equipped with S.O.O. 6221 only)	Press and hold the Pitch Trim Down lever on the Pilot's control wheel.		Time:		
AP Equipped AC: Engage AP (For aircraft equipped with S.O.O. 6221 only)	Adjust air data test set to provide an airspeed greater than 60 knots, engage AP.		Time:		
AP Equipped AC: Auto Pitch Trim Up (For aircraft equipped with S.O.O. 6221 only)	With AP still engaged, apply steady Fwd force on control column until pitch trim moves Up.		Time:		
AP Equipped AC: Auto Pitch Trim Down (For aircraft equipped with S.O.O. 6221 only)	With AP still engaged, apply steady Aft force on control column until pitch trim moves Down.		Time:		
	Remove airspeed.				
ADS-B Out Equipped AC: ADS-B Out Off (For aircraft equipped with S.O.O. 6347 only)	In XPDR Tuning window, turn ADS-B Out Off.		Time:		

C. Underwater Locator Beacon (ULB) – Dukane DK100

1) ULB Battery Test

Note Use a high impedance voltmeter (input impedance of 10 M ohms) to do the following procedure.

a) Ensure the ULB is clean and dry (refer to Para 6.A.1)).

b) Place the negative lead of the voltmeter on the water switch pin, and the positive lead on the case of the ULB.

c) Read the battery voltage on the voltmeter.

d) Check the ULB label for the battery code. ULBs with codes B, C, or D, have a minimum acceptable voltage of 2.97 volts. If the battery voltage is below minimum acceptable, the battery requires replacement.

2) ULB Operational Test

A Dukane 42A12 Ultrasonic Test Set, or equivalent, is required to do the operational test.

a) Set the TUNING knob between 35 and 40 kHz.

b) Turn the GAIN knob clockwise and check for operation by making a noise (such as jingling keys or coins) near the microphone.

c) Activate the ULB by shorting the water switch pin to the case with a jumper.

d) Adjust the tuning knob, if required, to produce the best audible signal.

e) Proper operation is indicated by a pulsing, audible tone. The ULB should pulse at least 5 times in 10 seconds.

f) Remove the jumper and confirm the pulsing stops.

g) Turn the GAIN knob to OFF position.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT/REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LINE REPLACEABLE UNIT (LRU) MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDING – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR31-37**

Insert in 31-31-00 facing Page 21
FLIGHT DATA RECORDER (FDR)

REASON FOR ISSUE:

To introduce Underwater Locator Beacon DK120.

ACTION:

In Chapter 31, Section 31-31-00 ignore existing Para 3.C. and insert revised Para, as shown on Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

C. Underwater Locator Beacon (ULB) – Dukane DK100 and DK120

1) ULB Battery Test

Note Use a high impedance voltmeter (input impedance of 10 M ohms) to do the following procedure.

a) Ensure the ULB is clean and dry (refer to Para 6.A.1)).

b) Place the negative lead of the voltmeter on the water switch pin, and the positive lead on the case of the ULB.

c) Read the battery voltage on the voltmeter.

d) Check the ULB label for the battery code. ULBs with codes B, C, or D, have a minimum acceptable voltage of 2.97 volts. If the battery voltage is below minimum acceptable, the battery requires replacement.

2) ULB Operational Test

A Dukane 42A12 Ultrasonic Test Set, or equivalent, is required to do the operational test.

a) Set the TUNING knob between 35 and 40 kHz.

b) Turn the GAIN knob clockwise and check for operation by making a noise (such as jingling keys or coins) near the microphone.

c) Activate the ULB by shorting the water switch pin to the case with a jumper.

d) Adjust the tuning knob, if required, to produce the best audible signal.

e) Proper operation is indicated by a pulsing, audible tone. The ULB should pulse at least 5 times in 10 seconds.

f) Remove the jumper and confirm the pulsing stops.

g) Turn the GAIN knob to OFF position.

A. FDR

1) Removal of FDR (Pre Mod 6/2332)

a) Ensure the aircraft electrical system is OFF.

b) Pull the circuit breaker FDR (M3) (Left Bus) on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the FDR, through the aft baggage compartment into the rear avionics compartment.

d) Remove the (6) mounting screws and washers.

e) Remove unit.

Note Stow connectors of the removed components.

2) Installation of FDR (Pre Mod 6/2332)

a) Ensure the aircraft electrical system is OFF.

b) Ensure the FDR (M3) (Left Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clips.

c) Set unit in position and install (6) mounting screws and washers. Ensure unit is secure.

d) Reconnect electrical connector. Ensure connector is secure and sound.

e) Remove the safety clips and replace the FDR (M3) (Left Bus) circuit breaker on the avionic circuit breaker panel.

f) Close access panel.

3) Removal of FDR (Post Mod 6/2332)

a) Ensure the aircraft electrical system is OFF.

b) Pull the circuit breaker FDR (M3) (Left Bus) on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the FDR, through the aft baggage compartment into the rear avionics compartment.

d) Loosen the two (2) retaining screw clamps and slide unit out of the mounting tray.

Note The connector is secured to the mounting tray and does not require stowing.

e) Remove unit.

4) Installation of FDR (Post Mod 6/2332)

a) Ensure the aircraft electrical system is OFF.

b) Ensure the FDR (M3) (Left Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clips.

c) Ensure connector is securely fastened to mounting tray. Slide unit into mounting tray and ensure unit is fully seated onto connector. Tighten the two (2) retaining screw clamps at the front lip of the FDR unit.

d) Remove the safety clips and replace the FDR (M3) (Left Bus) circuit breaker on the avionic circuit breaker panel.

e) Close access panel.

B. Accelerometer

1) Removal of Accelerometer

a) Ensure the aircraft electrical system is OFF.

b) Pull circuit breaker FDR (M3) (Left Bus) on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the forward avionic shelves through the forward baggage compartment.

d) Remove the middle and bottom covers attached to bulkhead station 44.00 for full access.

e) Disconnect the electrical connector on the bottom of the accelerometer.

f) Remove the (3) mounting screws.

g) Remove unit.

2) Installation of Accelerometer

a) Ensure the aircraft electrical system is OFF.

b) Ensure the FDR (M3) (Left Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clips.

c) Locate accelerometer on the middle avionics shelf, align unit with mounting screw holes.

d) Install (3) mounting screws. Ensure unit is secure.

e) Connect the electrical connector. Ensure the connection is secured and sound.

f) Reinstall middle and bottom covers attached to bulkhead station 44.00 at the aft wall of the baggage compartment. Ensure covers are mounted correctly and secure.

g) Remove safety clip and replace circuit breaker FDR (M3) (Left Bus) on the avionics circuit breaker panel.

h) Close forward baggage compartment door.

C. Impact Switch

1) Removal of Impact Switch

a) Ensure the aircraft electrical system is OFF.

b) Pull circuit breaker FDR (M3) (Left Bus) on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the impact switch, which is accomplished through the aft baggage compartment into the rear avionics compartment.

d) Disconnect electrical connector.

e) Remove (4) mounting screws and washers.

f) Remove unit.

Note Stow connectors of the removed components.

2) Installation of Impact Switch

a) Ensure the aircraft electrical system is OFF.

b) Ensure the FDR (M3) (Left Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Set unit in position and install (4) mounting screws and washers. Ensure unit is secure.

d) Reconnect electrical connector. Ensure the connection is secure and sound.

e) Close access panel.

f) Remove safety clip and replace circuit breaker FDR (M3) (Left Bus) on the avionics circuit breaker panel.

5. Troubleshooting

A. FDR System

1) If both the FDR and CVR Fail messages are present, check that the FDR and CVR are installed in the correct locations.

2) Ensure FDR (M3) circuit breaker is closed.

3) Ensure the FDR impact switch (aft of CVR impact switch) has not tripped.

4) Ensure the MAU is installed and operational.

5) Visually inspect FDR connector for damaged wiring or contacts. Ensure electrical connector on FDR is fully seated and secured.

6) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

7) If repair or service is required, send to appropriately rated avionics repair facility.

6. Servicing

A. Underwater Location Beacon

1) The case and water switch should be cleaned when dust and dirt become apparent. The switch should be cleaned using a soft cloth and mild detergent, then dried with a clean cloth.

2) Change the battery after any of the following conditions:

a) After use in an emergency.

b) If the battery fails the periodic test.

c) On or before the battery replacement date entered on the case of the beacon.

d) There is any evidence of corrosion or leakage.

e) Operation checks of the ULB must be carried out upon installation in accordance with PSM 1–6–7.

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Cockpit Voice Recorder (CVR) – General Data

1. General

The Cockpit Voice Recorder (CVR) is used to record aircraft audio for the purpose of investigating aircraft accidents and incidents. The CVR can record the data for up to 120 minutes.

For aircraft with S.O.O. 6388 incorproated, the Cockpit Voice Recorder (CVR) is not installed.

The Cockpit Voice Recorder (CVR) is installed as part of the IFR Operations Bundle customer Standard Order Option 6209.

For aircraft with S.O.O. 6209 incorporated, the CVR and associated systems may be removed from the aircraft by incorporating customer Standard Order Option Mod S.O.O. 6292.

For aircraft with S.O.O. 6388 incorporated, the CVR and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6384.

2. Description

The Honeywell AR–120 Cockpit Voice Recorder is a four channel CVR with 120 minute recording capacity. The AR–120 uses flash memory modules to record aircraft audio, and has no moving parts. It is equipped with an underwater location beacon.

The Cockpit Voice Recorder is contained in an environmentally protective enclosure. The recorder is located in the rear avionics compartment, at approximately station 415, where the survival rate is high. The recorder has a flange which allows the unit to be mounted directly onto the avionics shelf. 28 Volt DC power is provided from the left bus through a 5 amp CVR circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The audio control panels accept various aircraft audio signals, and output all required audio (except the cockpit area microphone) to the CVR via ARINC 429 (refer to 23-53-00, Audio Control Panel – General Data).

The following aircraft systems interface with the CVR system:

A. Cockpit Area Microphone

The cockpit area microphone is mounted in the center of the instrument panel, above the glareshield. The cockpit area microphone is intended to capture the cockpit ambient environment on the CVR. The pre-amplification and conditioning are accomplished within the CVR.

B. Modular Avionics Unit (MAU)

Each channel of the Custom Input/Output Modules (CSIO) outputs alert audio to the audio control panels. This audio is included in the audio panel output to the CVR.

The Generic Input/Output Module (GIO) module transmits Greenwich Mean Time (GMT) to the CVR over A429 (label 150).

C. Impact Switch

The impact switch is an inertia operated device which removes the power to the recorder in the event of a crash, thus keeping the recorded data intact. A deceleration of 3 g or greater will trigger the switch. The impact switch is located in the rear avionics compartment, outboard of the FDR, and is the forward of the two switches. A red lamp on the switch will illuminate when the switch has been tripped.

D. CVR Monitor Jack

A CVR monitor jack is provided on a panel in the rear avionics compartment, bottom shelf, at approximately station 421. The purpose of the jack is to provide a means to test the various audio inputs to the CVR.

E. Underwater Location Beacon

The Underwater Location Beacon (ULB) is a cylindrical transmitter mounted to the front of the crash survivable memory unit. When the sensor comes in contact with water the ULB emits ultrasonic pulses to assist in the location of the submerged recorder.

3. Adjustment/Test

A. Cockpit Voice Recorder

1) If a fault condition is detected by the Built-In Test Equipment (BITE), a caution level CAS message will be displayed. Pulling the CVR (K3) circuit breaker should trigger the CVR Fail CAS.

2) Function Test

a) Gain access to the CVR audio jack on the rear maintenance panel, which is accomplished through the aft baggage compartment, into the rear avionics compartment.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) While monitoring the CVR audio jack, do the following and ensure they are heard clearly.

1 Pull both PILOT AUDIO (3E) and COPILOT AUDIO (4E) CB's and speak into the cockpit area mic.

2 Reset the PILOT AUDIO CB (3E) and:

1 Speak into the pilot's headset.

2 Speak into the co-pilot's headset

3 Speak into the observer jack.

4 Speak into the cockpit area mic.

5 Trigger various alert audio sources such as altitude alert, engine fire test, stall warning, etc.

6 Tune a VHF Com radio and do a radio check, ensure both the sidetone and receive audio are heard.

3 Pull PILOT AUDIO CB (3E) and reset the COPILOT AUDIO CB (4E) and:

1 Repeat steps in Para 3.A.2).c).2.

4 Reset the PILOT AUDIO CB (3E).

d) Pull the CVR (K3) circuit breaker and ensure the CVR Fail message is present.

e) Reset circuit breaker and confirm the CAS messages clear.

f) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

TEMPORARY REVISION

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Insert in 31-32-00 after Page 2

COCKPIT VOICE RECORDER (CVR)

REASON FOR ISSUE:

To introduce Underwater Locator Beacon DK120.

ACTION:

In Chapter 31, Section 31-32-00 ignore existing Para 3.B. and insert revised Para, as shown on Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

B. Underwater Locator Beacon (ULB) – Dukane DK100 and DK120

1) ULB Battery Test

Note Use a high impedance voltmeter (input impedance of 10 M ohms) to do the following procedure.

a) Ensure the ULB is clean and dry, refer to Para 6.A.1).

b) Place the negative lead of the voltmeter on the water switch pin, and the positive lead on the case of the ULB.

c) Read the battery voltage on the voltmeter.

d) Check the ULB label for the battery code. ULBs with codes B, C, or D, have a minimum acceptable voltage of 2.97 volts. If the battery voltage is below minimum acceptable, the battery requires replacement.

2) ULB Operational Test

A Dukane 42A12 Ultrasonic Test Set, or equivalent, is required to do the operational test.

a) Set the TUNING knob between 35 and 40 kHz.

b) Turn the GAIN knob clockwise and check for operation by making a noise (such as jingling keys or coins) near the microphone.

c) Activate the ULB by shorting the water switch pin to the case with a jumper.

d) Adjust the tuning knob, if required, to produce the best audible signal.

e) Proper operation is indicated by a pulsing, audible tone. The ULB should pulse at least 5 times in 10 seconds.

f) Remove the jumper and confirm the pulsing stops.

g) Turn the GAIN knob to OFF position.

B. Underwater Locator Beacon (ULB) – Dukane DK100

1) ULB Battery Test

Note Use a high impedance voltmeter (input impedance of 10 M ohms) to do the following procedure.

a) Ensure the ULB is clean and dry, refer to Para 6.A.1).

b) Place the negative lead of the voltmeter on the water switch pin, and the positive lead on the case of the ULB.

c) Read the battery voltage on the voltmeter.

d) Check the ULB label for the battery code. ULBs with codes B, C, or D, have a minimum acceptable voltage of 2.97 volts. If the battery voltage is below minimum acceptable, the battery requires replacement.

2) ULB Operational Test

A Dukane 42A12 Ultrasonic Test Set, or equivalent, is required to do the operational test.

a) Set the TUNING knob between 35 and 40 kHz.

b) Turn the GAIN knob clockwise and check for operation by making a noise (such as jingling keys or coins) near the microphone.

c) Activate the ULB by shorting the water switch pin to the case with a jumper.

d) Adjust the tuning knob, if required, to produce the best audible signal.

e) Proper operation is indicated by a pulsing, audible tone. The ULB should pulse at least 5 times in 10 seconds.

f) Remove the jumper and confirm the pulsing stops.

g) Turn the GAIN knob to OFF position.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND LIGHTNING. ENSURE INTEGRITY OF THE HIRF/LIGHTNING PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT/REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDING – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. CVR

1) Removal of CVR

a) Ensure the aircraft electrical system is OFF.

b) Pull circuit breaker CVR (K3) (Left Bus) on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the CVR through the rear baggage compartment to the aft avionics compartment.

d) Remove (6) mounting screws and washers.

e) Remove unit.

Note Stow connectors of the removed unit.

2) Installation of CVR

The CVR has the same form and fit as the FDR, and will accept the same electrical connector. Ensure the CVR is installed in the correct position.

a) Ensure the aircraft electrical system is OFF.

b) Ensure the CVR (K3) (Left Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clips.

c) Position unit onto rack and install (6) mounting screws and washers. Ensure unit is secure.

d) Reconnect electrical connection. Ensure the connection is secure and sound.

e) Close access panel.

f) Remove safety clip and close CVR (K3) circuit breaker on the avionics circuit breaker panel.

g) Do a function test, refer to Para 3.A.2).

B. Cockpit Area Microphone

1) Removal of Cockpit Area Microphone

a) Ensure the aircraft electrical system is OFF.

b) Open CVR (K3) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Loosen the (2) hex headset screws.

d) Remove the area microphone from the mount to gain access to the wire pigtail and connector.

e) Disconnect the electrical connector.

Note Stow the connectors as required.

2) Installation of Cockpit Area Microphone

a) Ensure the aircraft electrical system is OFF.

b) Ensure the CVR (K3) circuit breaker on the avionics circuit breaker panel is opened and fitted with safety clips.

c) Connect electrical connector. Ensure connection is secure and sound.

d) Position the area microphone on the mount.

e) Tighten the (2) hex headset screws.

f) Remove safety clip and close CVR (K3) circuit breaker on the avionics circuit breaker panel.

g) Do a function test, refer to Para 3.A.2). Only the cockpit area mic portion of Para 3.A.2).c) needs to be completed.

C. Impact Switch

1) Removal of Impact Switch

a) Ensure the aircraft electrical system is OFF.

b) Open CVR (K3) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the impact switch, which is accomplished through the aft baggage compartment, into the rear avionics compartment.

d) Disconnect electrical connector.

e) Remove (4) mounting screws and washers.

f) Remove unit.

Note Stow the connectors as required.

2) Installation of Impact Switch

a) Ensure the aircraft electrical system is OFF.

b) Ensure the CVR (K3) circuit breaker on the avionics circuit breaker panel is opened and fitted with safety clip.

c) Set unit in position and install (4) mounting screws and washers. Ensure unit is secure.

d) Reconnect electrical connector. Ensure the connection is secure and sound.

e) Close access panel.

f) Remove safety clip and close CVR (K3) circuit breaker on the avionics circuit breaker panel.

g) Do a function test, refer to Para 3.A.2), Para 3.A.2).c) may be omitted for the purposes of this test.

5. Troubleshooting

A. CVR System

1) If both the CVR and FDR Fail messages are present, check that the FDR and CVR are installed in the correct locations.

2) Ensure the CVR (K3) circuit breaker is closed.

3) Ensure the CVR impact switch (forward of FDR impact switch) has not tripped.

4) Ensure the audio panels are installed and operational.

5) Visually inspect CVR connector for damaged wiring or contacts. Ensure electrical connector on CVR is fully seated and locked.

6) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

7) If repair or service is required, send to appropriately rated avionics repair facility.

6. Servicing

A. Underwater Location Beacon

1) The case and water switch should be cleaned when dust and dirt become apparent. The switch should be cleaned using a soft cloth and mild detergent, then dried with a clean cloth.

2) Change the battery on any of the following conditions:

a) After use in an emergency.

b) If the battery fails the periodic test.

c) On or before the battery replacement date entered on the case of the beacon.

d) There is any evidence of corrosion or leakage.

3) Operation checks of the ULB must be carried out upon installation in accordance with PSM 1–6–7.

Engine Condition Trend Monitoring/Flight Operations Quality Assurance Data Capture (S.O.O. 6288) – General Data

1. General

The Engine Condition Trend Monitoring (ECTM)/Flight Operations Quality Assurance (FOQA) Data Capture (S.O.O. 6288) system consists of an IONode recording unit and a multi element GPS/WiFi antenna. The IONode continuously records data from the aircraft Flight Data Recorder, and its own internal sensors. Data is subsequently downloaded to a Personal Computer (PC) for analysis.

2. Description

The ECTM/FOQA Data Capture system is a data storing and forwarding system. The IONode unit provides multiple inputs (digital and analog) for gathering data from onboard sensory equipment. The IONode is designed to do local logging to internal, on-board storage or an optional USB flash drive. In this installation the IONode records FDR Data, GPS position data and data from its internal accelerators.

The system consists of two major components; the IONode unit and combined GPS/Iridium/Wi-Fi Antenna The IONode unit is located on the bottom surface of the upper, left hand avionics shelf in the forward cargo compartment. The combined GPS/Iridium/Wi-Fi antenna is located on the top surface of the fuselage at station 246.50 near the right hand side of the aircraft. The ECTM/FOQA circuit breaker (Q4), located on the avionics circuit breaker panel is powered through CB230.

The ECTM/FOQA Data Capture system is a standalone installation that interfaces with one other aircraft system. The IONode unit receives all of its information from the Flight Data Recorder, interfacing with the ARNIC 717 output port on the Flight Data Recorder.

3. Equipment and Materials

- Cleaner MIL–C–25769
- Isopropanol
- Multimeter, 1 megaohm range
- Personal Computer (PC) with a Wi-Fi adapter, internet connection and a web browser
- Latitude Technologies registered account

4. Cleaning

The antenna may be cleaned only with the approved cleaning products listed in Para 3. Refer to 12-30-01, Aircraft External Cleaning – General Data, for cleaning procedures. Limit cleaning to wiping with moistened cloths.

5. Adjustment/Test

A. Function Test of the ECTM/FOQA Data Capture System

1) General

a) Switch aircraft power ON.

b) Open the ECTM/FOQA (Q4) circuit breaker on the avionics circuit breaker panel and ensure that all the LEDs on the IONode are OFF.

c) Close the ECTM/FOQA (Q4) circuit breaker on the avionics circuit breaker panel and ensure that there is activity on the LED's on the IONode (LED L1 reports Wi-Fi status, LED L2 reports GPS status, LED L3 reports overall system status).

2) No Engine Test

Note This test requires availability of the Ground Support Computer. Communications between the aircraft and the person monitoring the Ground Support Computer are required. The computer software and the IONode must be configured prior to this test.

a) Switch the aircraft power ON.

b) Establish communication between the aircraft and the person at the Ground Support Computer. Link the IONode either by Wi-Fi or USB-B connection.

c) Check that the green L2 and L3 status lights on the IONode are ON and steady.

d) When using Wi-Fi to connect to the Ground Support Software, enable IONode Wi-Fi by holding the "ECTM/FOQA TRANSMIT SWITCH" ON for ten seconds.

e) Check that the green L1 status light on the IONode is ON steady indicating that a Wi-Fi connection is established.

f) Check that the Ground Support Computer is receiving data. Verify that the position, speed (0), and altitude are correct for the aircraft's current position.

g) Switch OFF aircraft electrical system power. (Refer to 24-00-00, Electrical Power – General Data).

6. Removal/Installation

A. Combined GPS/Iridium/Wi-Fi Antenna

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the ECTM/FOQA (Q4) circuit breaker on the avionics circuit breaker panel and fit safety clip. If S.O.O. 6298 is installed, open the SAT COM (N4) circuit breaker and fit safety clip.

c) Remove the four (4) screws securing the antenna to the airframe.

d) Carefully cut the sealant from around the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connections.

f) Disconnect the coaxial connectors.

Note Stow connectors as required.

g) Remove the antenna.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the ECTM/FOQA (Q4) circuit breaker is opened and fitted with safety clip. If S.O.O. 6298 is installed, ensure the SAT COM (N4) circuit breaker is opened and fitted with safety clip.

c) Remove non-conductive material (e.g. anodic films, primer, paint, etc) from the electrical bond contact area in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data. Then apply chemical conversion coating (C10) in accordance with 20-30-02, Chemical Conversion Coatings (Alodine) – Manual Application, within 8 hours of stripping high resistance films.

d) Solvent clean the electrical contact bond areas immediately before installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data).

e) Move the antenna close to the fuselage and connect the coaxial connectors.

f) Secure the antenna to the fuselage using the four (4) mounting screws.

g) Solvent clean the exposed areas of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data).

h) Touch up exposed areas of the airframe around the antenna, if required, with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

i) If the affected area requires paint touch up, brush coat the area with the top coat paint system matching the existing.

j) Fillet seal around the base of the antenna with MIL–PRF–81733 sealant in accordance with 20-50-01, Weather Sealing – General Data. The aircraft will be ready to fly 24 hours after application of the sealant.

k) Do a resistance measurement between the antenna and the airframe. Resistance must be less than 3 milliohms in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data.

l) Remove the safety clip and close the ECTM/FOQA (Q4) circuit breaker. If S.O.O. 6298 is installed, remove safety clip and close the SAT COM (N4) circuit breaker.

m) Do a function test of the ECTM/FOQA Data Capture system per Para 5.A.1).

B. IONode Unit

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open the ECTM/FOQA (Q4) circuit breaker on the avionics circuit breaker panel and fit safety clip. If S.O.O. 6298 is installed, open the SAT COM (N4) circuit breaker and fit safety clip.

c) Disconnect the connectors.

Note Stow connectors as required.

d) Remove the four (4) screws and washers securing the IONode in position on the bottom face of the upper left avionics shelf in the FWD cargo compartment.

e) Carefully remove the IONode unit.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the ECTM/FOQA (Q4) circuit breaker is opened and fitted with safety clip. If S.O.O. 6298 is installed, ensure the SAT COM (N4) circuit breaker is opened and fitted with safety clip.

c) Carefully position the IONode over the four (4) potted mounting holes on the bottom face of the upper left avionics shelf in the FWD cargo compartment.

d) Install the four (4) screws and washers to secure the IONode in position.

e) Connect the connectors.

f) Remove the safety clip and close the ECTM/FOQA (Q4) circuit breaker. If S.O.O. 6298 is installed, remove safety clip and close the SAT COM (N4) circuit breaker.

Note Calibrate in accordance with IONode Installation and Operational Manual #740–0193 Revision 1.16 or later.

g) Do a function test of the ECTM/FOQA system as per Para 5.A.1).

7. Troubleshooting

A. IONode Unit

1) If the IONode is deemed inoperative check the following:

a) Ensure that the ECTM/FOQA (Q4) circuit breaker is closed.

b) Check that the IONode is properly secured and grounded to the airframe.

c) Ensure that all connectors are properly mated and tightened.

d) Visually inspect the IONode for damaged wires or contacts and ensure no short circuits have formed.

e) Check that the Combined GPS/Iridium/Wi-Fi Antenna is properly secured and bonded to the airframe.

f) Visually inspect the IONode and antenna for damaged wires or contacts.

g) If troubleshooting at the wire level is required, refer to the applicable diagrams in the Wiring Diagrams Manual, PSM 1-64-2W.

h) Do a function test of the ECTM/FOQA system as per Para 5.A.1).

i) If repair or service is required, send the IONode to appropriately rated avionics repair facility.

2) The LED indicators on the IONode may assist in problem diagnosis:

LED L1	**Wi-Fi STATUS**
LED Off Solid	Wi-Fi subsystem disabled.
LED On Solid	Wi-Fi subsystem enabled and connected to Base Station Software (socket open).
LED Blinking once per second	Wi-Fi subsystem enabled, or searching for access point, or attempting to open socket.
LED Flickering	Wi-Fi subsystem enabled and transferring data.

LED L2	**GPS STATUS**
LED On Solid	GPS fix acquired.
LED Off Solid	No GPS source, or GPS using external source and cannot parse GPS string.
LED Blinking once per two seconds	GPS present and fix not acquired.

LED L3	**SYSTEM STATUS**
LED On Solid	System running normally.
LED Off Solid	System Off.

Table 1

Symptom	Diagnosis
IONODE L3 Light won't turn ON.	Verify correct power and ground wiring.
IONODE L2 LED Light indicates "No GPS Fix" blink pattern.	Make sure that the triple Iridium/GPS/Wi-Fi antenna is properly connected to the IONODE and that the antenna has a clear view of the sky.
IONODE L1 LED will not turn ON.	Make sure that the correct Configuration Settings file has been installed on the IONODE and that the associated Wi-Fi enable signals are present.
IONODE L1 LED blinking even though there is an access point nearby.	Make sure that the correct Configuration Settings file has been installed on the IONODE. Double check the SSID, Password and IP settings that were sent to Latitude Technologies Support to create the Configuration Settings. Make sure that the triple Iridium/GPS/Wi-Fi antenna is properly connected to the IONODE and that the antenna is located in an area with good exposure to the access point signal (this can be verified in the location of the Wi-Fi antenna using a smart phone or other Wi-Fi enabled mobile device).
IONODE L2 LED won't turn ON when SkyNode is installed in the same aircraft.	When S.O.O. 6288 and S.O.O. 6298 are installed in the same aircraft, the GPS position for the IONode is obtained from the S200 SkyNode. Verify GPS data wiring between the SkyNode and IONode.
IONODE log files do not show sensor readings.	If operating on the ground, make sure to place the IONode in Maintenance Mode (this can be done through the Base Station Software).
IONODE IMU yaw/pitch/roll readings are swapped or appear to be incorrect.	Make sure that the IONode orientation procedure has been completed, that the recorded values were passed to a Latitude Technical Support Representative, and that the returned Configuration Settings files has been installed in the IONode. If the values are still incorrect, an additional 2D calibration procedure is required. Contact Latitude Technologies Technical Support for calibration assistance. (Refer to 23-15-00 – Satellite Communication System (S.O.O. 6298) – General Data for full contact information).

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Quick Access Recorder (QAR) (S.O.O. 6356) – General Data

1. General

The Quick Access Recorder (QAR) Installation (S.O.O. 6356) consists of a removable QAR unit which is wired to the FDR DATAOUT channels. The QAR continuously records data from the aircraft Flight Data Recorder when the QAR switch is in the NORMAL position. The data is subsequently downloaded to a Personal Computer (PC) for analysis.

2. Description

The QAR is a data capture and storing system. The QAR unit provides multiple inputs (digital and analogue) for gathering data from onboard sensory equipment. The QAR comes with a 2 gigabyte (GB) removable CompactFlash Card, which will provide 1280 hours of recording time at 256 Words-Per-Second (WPS) data stream. The data from the QAR may be downloaded onto a computer while installed in the aircraft, using the built-in mini Universal Serial Bus (USB) port. In addition, the USB port may be used to supply power to the QAR from a host computer when aircraft power is not available.

The QAR is located above the rear baggage door on the left hand side of the aircraft at station 362.00. The QAR circuit breaker (T3), located on the station 332 circuit breaker panel, is powered through the left bus.

3. Equipment and Materials

Optional (required to download data directly from QAR via the USB port):

- Personal Computer (PC) with Microsoft Windows, standard USB port interface (ver. 1.1 or 2.0) or PCMCIA drive
- microQAR Flight Data Processor software utility program (L-3 Communications part number 883E2822–03 or above).
- USB cable with ‘standard’ connector and ‘mini-B’ connector.
- L-3AR Rose software.
- Database part number 851–E6415–00.

4. Adjustment/Test

A. Function Test of the QAR

1) Gain access to the QAR in the rear baggage compartment, above the baggage door.

2) Remove QAR cover per Para 5.A.4).

3) Set QAR switch to the OFF position.

4) Switch aircraft power ON.

5) Ensure that both LEDs on the QAR are not illuminated.

6) Set QAR switch to the NORMAL position. Ensure that both the LEDs on the QAR momentarily illuminate two times and then go out; this indicates normal operation.

7) Set QAR switch to the OFF position.

8) Open the QAR (T3) (Left Bus) circuit breaker on the stations 332 circuit breaker panel.

9) Set QAR switch to the NORMAL position. Ensure that both LEDs on the QAR are not illuminated.

10) Close the QAR (T3) (Left Bus) circuit breaker on the station 332 circuit breaker panel. Ensure both LEDs located on the label side of the QAR momentarily illuminate two times and then go out; this indicates normal operation.

11) Install QAR cover per Para 5.B.5).

5. Removal/Installation

A. Removal

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Open the QAR (T3) (Left Bus) circuit breaker on the station 332 avionics circuit breaker panel and fit with safety clip.

3) Gain access to the QAR in the rear baggage compartment, above the baggage door.

4) Remove the two (2) screws securing the QAR cover to the airframe.

5) Remove the QAR by disconnecting mating connector.

Note Removal of airframe mounted connector is not required.

B. Installation

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Ensure the QAR (T3) (Left Bus) circuit breaker on the station 332 avionics circuit breaker panel is open and fitted with safety clip.

3) Gain access to the QAR installation area in the rear baggage compartment, above the baggage door.

4) Install QAR to mating connector.

5) Install the two (2) screws and washers securing the QAR cover to the airframe.

6) Close the QAR (T3) (Left Bus) circuit breaker on the station 332 circuit breaker panel.

7) Do a function test of the QAR, per Para 4.A.

6. Troubleshooting

A. QAR Unit

1) If the QAR is deemed inoperative check the following:

a) Ensure that the QAR (T3) (Left Bus) circuit breaker is closed.

b) Ensure that the QAR switch is in the NORMAL position.

c) Check the CompactFlash card is not full.

d) Check that the QAR is properly secure and grounded to the airframe.

e) Ensure that all the connectors are mated and tight.

f) Visually inspect the QAR for damaged wires and contacts, and ensure no short circuits have formed.

g) If troubleshooting at a wire level is required, refer to the applicable diagrams in the Wiring Diagrams Manual PSM 1-64-2W.

h) If repair or service is required, return the QAR to L-3 Communications.

2) The LED indicators on the QAR may assist in problem diagnosis:

FAULT LED	**QAR STATUS**
LED On – Amber	QAR has stopped functioning and should be removed from aircraft and returned to L-3 Communications.
MEMORY FULL LED	
LED On – Amber	Memory card is full and recording will stop. CompactFlash card may be removed and replaced or the data from the card may be downloaded and cleared at the aircraft thorough the QAR micro USB interface.
MEMORY FULL AND FAULT LED	
LEDs momentarily flash two times on start up	QAR operating normally.

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Central Computers (Mod 6/2049) – General Data

1. General

The Central Computer is built on the Honeywell Primus APEX platform. The Central Computer systems architecture is built on modular hardware located in the aircraft and operational software. The Central Computer systems hardware is comprised of the following components:

- Modular Avionics Unit, MAU
- Data Acquisition Units, DAU 1 and DAU 2
- KDU 1080 Display Units – DU1, DU2, DU3, DU4
- KMC 2210 PFD Controllers
- KMC 2220 MFD Controller
- Display Reversionary Panel
- KMA 29 Audio Panels
- KMC 9200 Flight Controller

Data is communicated and received between the system along Avionic Standard Communication Bus, version D, (ASCB-D) are used to transfer data types between the MAU and the DAU and other peripherals. Maintenance data and software updates are transferred via the local area network, LAN.

2. Description

A. Modular Avionics Unit

The Modular Avionics Unit, MAU, with the modules installed acts like the central processing unit for Central Computer System. The MAU and modules, collect, compile and formats the data for the required system display. (Refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

B. Data Acquisition Unit

The Data Acquisition Units, DAU 1 and DAU 2 receives the analogue input signals from the aircraft sensors and transmitters, converts the analogue signals to a digital and send the digital signals to the Modular Avionics Unit, MAU to the Flight Data Recorder. (Refer to 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data).

C. Display Units

The data is displayed on one of the four KDU 1080 display units, DU1, DU2, DU3 and DU4, installed in the cockpit; two designated as the Primary Flight Displays, PFD, DU1, DU4 and two Multi-Function Displays, MFD, DU2, DU3. Under normal operations each Display Unit has a default format. (Refer to 31-11-00, Display Units – General Data).

D. Primary Flight Display Controllers

There are two KMC 2210 PFD Controllers, one for the Pilot's PFD one for the Co-pilot's PFD and is installed on the instrument panel to the right of the Pilot's PFD and to the left of the Co-pilot's PFD respectively. It allows the operator to navigate within the windows on the PFD screen and set the format for the system information to be displayed. (Refer to 31-11-00, Display Units – General Data).

E. Multi-Function Display Controller

The KMC 2220 multi-function controller is mounted on the aft face of the central control yoke in the cockpit. The multi-function controller provides the operator key access to menus, functions and formats for the information within each defined windows displayed on the MFDs. (Refer to 31-11-00, Display Units – General Data).

F. Display Reversionary Panel

The display system allows the pilot to revert the Display Units (DU) and Advanced Graphics Module 1 and 2 (AGM) in the event of failure of an individual display unit or AGM. The Display Reversion control panel is installed directly above the MFD controller, on the aft face of the control column.

G. Audio Panel

The KMA 29 audio control panels provide the pilot and co-pilot with monitoring of all the communication and navigation receivers, and microphones transmission over the selected radio transmitter. (Refer to 23-50-00, Audio Integration – General Data).

H. Flight Controller

The KMC 9200 Flight Controller is used to enter selected minima, set heading or track, and enter selected altitude.

I. ASCB–D Data Bus

Data is communicated and received between the system along Avionic Standard Communication Bus, version D, (ASCB–D) are used to transfer data types between the MAU and the DAU and other peripherals.

J. Local Area Network, LAN

The LAN is a standard general purpose data bus which is used for transferring maintenance data between components located on the LAN. The LAN also is used to loaded software updates via the aircraft maintenance panel located below the MAU in the Forward Avionics Compartment. (Refer to 45-00-00, Central Maintenance Computer – General data).

Modular Avionics Unit (MAU) (Mod 6/2049) – General Data

1. General

The Modular Avionics Unit (MAU), with the modules installed acts like the central processing unit for Central Computer. The MAU and modules, collect, compile and formats the data for the required system display. The Modular Avionics Unit (MAU) consists of a cabinet which houses a variety of line replaceable modules (LRUs).

2. Description

The MAU is installed on the upper of the three avionics racks between the nose baggage compartment and the flight compartment at approximately station 50. It consists of a cabinet/chassis containing a backplane circuit card assembly, two cooling fans and 12 user module slots that host a variety of line replaceable modules.

28 Volt DC power is provided from the left bus through 5 amp MAU 1A (E1) and MAU 2A (F1) left bus circuit breakers, and from the right bus through 5 amp MAU 1B (E2) and MAU 2B (F2) right bus circuit breakers on avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The MAU cabinet is divided into two channels (A and B), each channel is electrically isolated from the other with its own power supply module, Network Interface Controller (NIC) module and data communications backplane. The dual channel architecture of the MAU allows system functions to be distributed between channels. All modules are field replaceable and software updates can be loaded in the field. The modules communicate to the Avionics Standard Communications Bus (ASCB) and the Local Area Network (LAN) bus via the NIC modules.

The Avionics Standard Communications Bus (ASCB) consists of two independent busses, the No. 1 ASCB and No. 2 ASCB corresponding to pilot and co-pilot side primary data. Each NIC in the system reads and writes to the onside primary bus, and reads from the cross-side primary bus. One end of each ASCB is terminated within the wiring harness, the other end is terminated at a D series connector on the maintenance panel. When the maintenance panel connector is not being used, a terminating backshell must be installed. The ASCB bus uses special twisted pair wiring and is terminated at each point using an ASCB stub, refer to wiring manual PSM 1-64-2W for more details.

The LAN bus uses a 10base2 Ethernet communication scheme, and is used primarily for transferring data, such as MAU updates or changing MAU settings. It uses a triax cable which allows the outer shield to be terminated in multiple places, while the inner shield is only terminated at one end. Both ends of the LAN bus are available as BNC connectors on the maintenance panel. A 50 ohm BNC terminator must be connected to the BNCs when not in use. The LRUs connected to the LAN bus are connected in series; the bus enters the LRU on one pin and exits to the next LRU on another pin. Therefore, if the system is to be energized with any of the LRUs removed, the input and output pins must be jumpered.

The modules used in the Modular Avionics Unit include:

A. Two – Power Supply Modules (PS)

Each channel of the MAU has a dedicated, independent power supply. Either power supply can operate both of the MAU cabinet cooling fans. Each channel receives power from the left and right bus. The PS modules contain no processing or backplane communication capability.

B. Two – Network Interface Controller (NIC) Modules

The Network Interface Controller (NIC) modules provide a gateway for the MAU modules to access the ASCB and LAN bus. Two NIC modules are installed, one for each channel of the MAU.

Two Aircraft Personality Modules (APMs) are located in the MAU NIC backshells. They are configuration modules which contain Apex configuration data, such as system identifier, aircraft type, aircraft serial number, installed configuration options and system settings. The APM allows individual cards within the MAU to be removed and replaced quickly, without the need to reconfigure the card for a specific aircraft.

1) The NICs interface to the following aircraft systems:

   a) ID straps

   b) ASCB 1 & 2

   c) APMs

   d) LAN buses

C. Two – Advanced Graphics Module (AGM)

The Advanced Graphics Module (AGM) is a single channel module, one per MAU channel is installed. The AGM does general purpose processing as well as display processing and graphics generation. The Configuration Management Function (CMS), Flight Management System (FMS), electronic display of Jeppesen charts function and maintenance functions (CMC, ACMS) are also hosted on the AGM module. AGM 1 (MAU channel A) drives the pilot's PFD and upper MFD and AGM 2 (MAU channel B) drives the co-pilot's PFD and lower MFD.

A repeater capability allows the pilot's PFD to be displayed on the co-pilot's PFD (and vice-versa) in the event of a single AGM failure. The PFD controllers, MF controller and reversion panel are interfaced with the AGMs. AGM integrity is monitored by the Monitor Warning Function (MWF), which verifies that the data selected by the AGM for display on the DUs has high integrity.

1) The AGMs receive inputs from the following aircraft systems:

   a) Ethernet

   b) PFD Controllers

   c) MF Controller

   d) Reversion Panel

   e) EGPWS

   f) Weather Radar

   g) XM Weather (if installed)

   h) Stormscope (if installed)

2) The AGMs output to the following aircraft systems:

   a) Display Units

   b) Stormscope (if installed)

D. Custom Input/Output (CSIO) Modules

The Custom I/O (CSIO) Module is a dual channel module. Each channel is connected to a different MAU backplane (channel A and B). The CSIO module translates aircraft model-specific I/O data onto and off of the ASCB. The CSIO also translates selected fuselage signals from analog to digital, and then sends this digital data to the Data Acquisition Unit (DAU).

1) The CSIO module received inputs from the following aircraft systems:

   a) DMEs

   b) DAUs

c) Audio Panels

d) RAD ALT

e) EGPWS

f) FDR Accelerometer

g) XPDRs

h) GPSs

i) TCAS

j) Aircraft Voltages

2) The CSIO module outputs to the following aircraft systems:

a) DAUs

b) Audio Panels

c) FDR (if installed)

E. Two – Actuator Input/Output and Processor (AIOP) Modules

The Actuator I/O Processor (AIOP) Module is a single channel module, one per MAU channel is installed. The AIOP module is principally associated with the Automatic Flight Control System (AFCS) and flight controller operation. Currently this functionality is limited to selected minimums, heading or track, and selected altitude.

1) The AIOP modules receive inputs from the following aircraft systems:

a) Flight Controller

b) ADAHRS

2) The AIOP modules output to the following aircraft systems:

a) Flight Controller

F. Generic Input/Output (GIO) Module

The Generic Input/Output (GIO) Module is a dual channel module. Each module channel is connected to a different MAU backplane (channel A and B). The GIO module translates aircraft I/O data onto and off of the ASCB via the MAU's backplane. The GIO module also translates selected fuselage discretes for ARINC 429, which connects to the Data Acquisition Unit (DAU).

1) The GIO module receives inputs from the following aircraft systems:

a) ADAHRS

b) FDR (if installed)

c) CVR (if installed)

d) EGPWS

e) DAUs

f) Master Warning/Caution

g) XPDRs

h) GPSs

i) TCAS

j) Display Units

k) Aural Alert Power (mute)

l) Caution Dim Box (lamp test)

m) Various Aircraft Discrete Signals

2) The GIO module outputs to the following aircraft systems:

a) ADAHRS

b) CVR (if installed)

c) EGPWS

d) MMDRs

e) Master Warning/Caution

f) MF Controller

g) XPDRs

h) GPSs

i) TCAS

j) ELT

k) Weather Radar

3. Adjustment/Test

A. Modular Avionics Unit (MAU) Test

The MAU has extensive built in test and fault detection and exclusion capability. Failures of components or loss of system capability will be announced to the crew using the aircraft crew alerting system. Information may be presented as caution, advisory, or status messages, depending on the importance of the message or the severity of the condition.

1) Ensure MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel are closed.

2) Ensure none of the following CAS messages are present following system initialization:

a) MAU Fail

b) ASCB Fail

c) AIOP Fail

d) CSIO Fail

e) MAU Overheat

f) MAU FAN Fail

g) GIO Fail

h) AGM Fail

i) CMS Fail

j) FMS Fail

3) Ensure the following windows are displayed properly:

a) Altitude Director Indicator (ADI)

b) Airspeed

c) Altitude

d) Vertical Speed

e) Horizontal Situation (HSI) Displays and Annunciators

f) Radio Management

g) Situational Awareness and Flight Planning

B. Modular Avionics Unit (MAU) Software Load or Update

For the procedure for loading the APEX MAU Software refer to 45-00-02, Apex Modular Avionics Unit Software – Load Procedures.

C. Options File or Setting File Load or Update

For the procedure for loading the APEX Options or Setting File refer to 45-00-01, Apex Options and Settings File – Load Procedures.

D. FMS Performance Database File Load or Update

For the procedure for loading the APEX FMS Aircraft Performance Database File refer to 45-00-03, Apex FMS Aircraft Performance Database Files – Load Procedures.

E. Ferrite Bead Verification

1) Remove NIC 1 or 2 backshell. Refer to Para 4.C.

2) Remove fasteners from backshell cover and remove cover.

3) Verify presence of ferrite beads on LAN wires. The LAN wires are the coaxial cables terminated to the center post of the backshell.

4) If ferrite beads are not present, install according to PSM 1-64-2W, Chapter 20 for details.

5) Install cover onto NIC backshell and tighten fasteners.

6) Install NIC 1 or 2 backshell. Refer to Para 4.D.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT/REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDING – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

CAUTION **THE MAU AND MODULES ARE ELECTROSTATICALLY DISCHARGE SENSITIVE (ESDS). ENSURE PROPER ESDS PROCEDURES ARE FOLLOWED. FAILURE TO DO SO COULD DAMAGE UNIT.**

CAUTION **THE MAU AND COMPONENTS ARE ESDS SENSITIVE AND ANY WORK DONE ON THE MAU AND COMPONENTS AWAY FROM AIRCRAFT MUST BE DONE ON AN ESDS-PROTECTED WORK AREA. FAILURE TO DO SO MAY DAMAGE UNIT.**

CAUTION **ENSURE AN ESDS WRIST STRAP IS WORN AND IS CONNECTED TO THE GROUND LOCATED ON THE MAU BEFORE DOING THE REMOVAL OR INSTALLATION OF THE MAU.**

CAUTION **CARE MUST BE TAKEN NOT TO DAMAGE THE AIR OUTLETS SCREENS ON THE REAR OF THE MAU WHEN YOU REMOVE OR INSTALL UNIT.**

CAUTION **DAMAGE TO MAU AND COMPONENTS OR DEGRADATION OF PERFORMANCE CAN OCCUR IF THE CONNECTOR PINS OF THE MAU OR COMPONENTS ARE TOUCHED.**

A. Removal of the MAU

1) Ensure the aircraft electrical system is OFF.

2) Pull circuit breaker MAU 1A (E1), MAU 2A (F1) (Left Bus), MAU 1B (E2), MAU 2B (F2) (Right Bus), on grid 1 and 2 of the avionics circuit breaker panel and fit safety clips.

3) Gain access to the forward avionic shelves through the forward baggage compartment.

4) Remove top cover attached to bulkhead station 37.21.

5) Remove the MAU backshells, refer to Para 4.C.

6) Remove the (2) cable clamps and reposition cable harness to allow greater access.

7) Loosen the (4) retaining screw clamps which secure MAU to mounting tray.

8) Carefully move MAU forward until unit is clear of mounting wedge located at the rear of the mounting tray.

Note The MAU, with all the components installed, weighs approximately 40 lbs, personnel must take care in lifting the unit.

9) Lift MAU off avionics shelf and clear of aircraft.

10) Visually inspect the metal mesh covering the air intakes for dust or debris. Clean if required.

CAUTION DO NOT BLOW DUST OR DEBRIS INTO MAU COVER.

B. Installation of MAU

1) Visually inspect the metal mesh covering the air intakes for dust or debris. Clean if required.

CAUTION DO NOT BLOW DUST OR DEBRIS INTO MAU COVER.

2) Ensure the aircraft electrical system is OFF.

3) Ensure the MAU 1A (E1), MAU 2A (F1) (Left Bus), MAU 1B (E2), MAU 2B (F2) (Right Bus) on grid 1 and grid 2 of the avionics circuit breaker is pulled and fitted with safety clips.

4) Gain access to the forward avionic shelves through the forward baggage compartment.

5) Remove top cover attached to bulkhead station 37.21.

6) Verify that the static bonding points to the aircraft frame are clean.

7) If you are installing a different MAU into aircraft, make sure all 12 modules are installed correctly and are secure within the MAU cabinet, refer to Para 4.E and Para 4.F.

8) Inspect the condition of the MAU support bracket. If the tray is damaged, remove and replace as per Para 4.I and Para 4.J.

Note The MAU, with all the components installed, weighs approximately 40 lbs, personnel must take care in lifting the unit.

9) Lift MAU into the forward avionics compartment and onto the mounting tray.

10) Carefully move the MAU aft in the tray until rear of unit sits securely in the mounting wedge at the rear of the mounting tray.

11) Tighten the (4) retaining screw clamps, torque to 42 ± 2 in–lbs, ensure MAU is secure.

12) Reposition MAU cable harness and install (2) cable clamps.

13) Install MAU backshells, refer to Para 4.D.

14) Ensure all connections are secure.

15) Ensure proper bonding has been achieved between the MAU and the airframe.

16) Install top cover attached to bulkhead station 37.21. Ensure cover is secure.

WARNING ACCESS COVER IS REQUIRED TO MAINTAIN PROTECTION FEATURES THAT ARE ESSENTIAL TO MINIMIZING ADVERSE EFFECTS FROM HIGH INTENSITY RADIATED FIELDS (HIRF).

Note If new MAU is being installed or has one or more modules installed, MAU software will have to be loaded. (Refer to Chapter 45 for loading procedures).

17) Remove safety clips and close MAU 1A (E1), MAU 1B (E2), MAU 2A (F1) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

18) Do an operational test on the MAU, refer to Para 3.A.

C. Removal of MAU Backshell

CAUTION **ENSURE AN ESDS WRIST STRAP IS WORN AND IS CONNECTED TO THE GROUND LOCATED ON THE MAU BEFORE DOING THE REMOVAL OR INSTALLATION OF THE MAU BACKSHELLS.**

CAUTION **DAMAGE TO MAU AND COMPONENTS OR DEGRADATION OF PERFORMANCE CAN OCCUR IF THE CONNECTOR PINS OF THE MAU OR COMPONENTS ARE TOUCHED.**

These removal and installation procedures are applicable for all the backshells installed in the MAU Cabinet.

1) Ensure the aircraft electrical system is OFF.

2) Open MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel and fit safety clips.

3) Gain access to the forward avionics shelf through the forward baggage compartment.

4) Remove top cover attached to bulkhead station 37.21.

5) Loosen the top bolt from the backshell until it has disengaged from MAU.

6) Carefully tilt the backshell away from the MAU until the alignment pin is clear of the alignment hole, approximately mid way on the module.

7) Disengage the bottom backshell hook from the base of the MAU and move backshell away from unit.

8) Install protective covers on backshell.

D. Installation of MAU Backshell

CAUTION **DAMAGE TO THE MODULE CONTACTS MAY RESULT IF THE FOLLOWING PROCEDURES ARE NOT CARRIED OUT WITH PROPER CARE AND ATTENTION.**

1) Ensure the aircraft electrical system is OFF.

2) Ensure MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel are open and fitted with safety clips.

3) Remove protective cover from backshell.

4) Place the backshell to engage hook onto the base of the MAU.

5) Carefully align the pin into the alignment hole in the module.

6) Install the top bolt into MAU and torque to 32 in–lbs. Ensure the backshell is secure.

7) Install top cover attached to bulk head station 37.21. Ensure cover is secure.

8) Remove safety clips and close MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

9) Do an operational test on the MAU, refer to Para 3.A.

E. Removal of Modules

CAUTION **ENSURE AN ESDS WRIST STRAP IS WORN AND IS CONNECTED TO THE GROUND LOCATED ON THE MAU BEFORE DOING THE REMOVAL OR INSTALLATION OF THE MAU MODULES.**

CAUTION **DAMAGE TO MAU AND COMPONENTS OR DEGRADATION OF PERFORMANCE CAN OCCUR IF THE CONNECTOR PINS OF THE MAU OR COMPONENTS ARE TOUCHED.**

These removal and installation procedures are for all the modules installed in the MAU Cabinet. Please note the individual modules may have two or four D connectors securing module into the MAU Cabinet.

1) Remove backshell from module as per Para 4.C.

2) Install protective covers on the module connectors.

CAUTION **DAMAGE TO THE MODULE OR MAU MAY RESULT IF THE FOLLOWING PROCEDURES ARE NOT CARRIED OUT WITH PROPER CARE AND ATTENTION.**

3) Loosen the jackscrews gradually and evenly until they are all free from the MAU threads.

4) While gripping the jackscrews, carefully pull the module clear of the MAU cabin slot.

5) Place module into ESDS approved protective sleeve.

F. Installation of Modules

1) Ensure the aircraft electrical system is OFF.

2) Ensure MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel are open and fitted with safety clips.

3) Remove module from ESDS approved protective sleeve.

4) Carefully align module into applicable slot.

5) Gently push module into slot, ensure module is all the way in.

6) Tighten the jackscrews gradually and evenly to 42 in–lbs. Ensure the module is secure.

7) Remove protective covers from module connectors.

8) Install backshell, refer to Para 4.D.

9) Remove safety clips and close MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

Note If a new module has been installed, MAU software will have to be loaded. (Refer to Chapter 45 for loading procedures).

10) Do an operational test on the MAU, refer to Para 3.A.

G. Removal of MAU Cooling Fans

CAUTION **ENSURE AN ESDS WRIST STRAP IS WORN AND IS CONNECTED TO THE GROUND LOCATED ON THE MAU BEFORE DOING THE REMOVAL OR INSTALLATION OF THE COOLING FANS FROM THE MAU.**

CAUTION **CARE MUST BE TAKEN NOT TO DAMAGE THE AIR OUTLET SCREENS ON THE REAR OF THE MAU WHEN YOU REMOVE OR INSTALL A COOLING FAN.**

CAUTION **DAMAGE TO MAU AND COMPONENTS OR DEGRADATION OF PERFORMANCE CAN OCCUR IF THE CONNECTOR PINS OF THE MAU OR COMPONENTS ARE TOUCHED.**

These removal and installation procedures are applicable for the cooling fans located in the rear of the MAU.

1) Remove MAU as per Para 4.A.

2) Unlatch the (3) clamps which hold the rear fan hinge plate, located at the rear of the unit.

3) Pull the fan out of the slot, towards the top of the hinge. This disconnects the electrical connector on the fan, so some resistance may be felt.

4) Put fan in an ESDS approved protective sleeve.

5) Visually inspect the metal mesh covering the air intakes for dust or debris. Clean as required.

CAUTION **DO NOT BLOW DUST OR DEBRIS INTO THE INTERIOR OF MAU.**

H. Installation of MAU Cooling Fans

1) Visually inspect the metal mesh covering the air intakes for dust or debris. Clean if required.

CAUTION **DO NOT BLOW DUST OR DEBRIS INTO THE INTERIOR OF MAU.**

2) Remove fan from ESDS approved protective sleeve.

3) Slide fan into position on the fan hinge plate. Align the guide pin for the electrical connector and push the fan down until it is fully seated.

4) Close the fan hinge plate and secure with clamps.

5) Install MAU, as per Para 4.B.

6) Apply power to aircraft.

7) Ensure there is no “Mau Fan Failure Message” displayed through the CAS.

I. Removal of MAU Support Bracket

1) Remove the MAU as per Para 4.A.

2) Remove the (16) mounting screws which secure the bracket to the avionics shelf.

3) Remove bracket.

J. Installation of MAU Support Bracket

1) Visually inspect the mating surfaces of the tray and the avionics shelf for any corrosion or contaminants.

2) Ensure the mounting bracket is oriented correctly, mounting edge to the aft.

3) Install the (16) mounting screws which secure the bracket to the avionics shelf.

4) Ensure proper bonding has been achieved between the tray and the avionics shelf.

5) Install MAU as per Para 4.B.

5. Troubleshooting

A. Modular Avionics Unit (MAU) Software

1) Ensure MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers are closed.

If problem persists reboot MAU as follows:

a) Open MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

b) Wait 60 seconds.

c) Close MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

If problem persists:

d) Ensure the correct software is loaded by checking the systems window of the MFD.

e) Follow load procedures described in Chapter 45 if software is suspected.

B. MAU Connectors

1) Ensure both ASCB1 and ASCB2 dummy plugs are installed and seated correctly on the maintenance panel below the top FWD avionics shelf.

2) Ensure both LAN Bus Terminators are installed correctly on the maintenance panel.

If problem persists:

3) Open MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

4) Ensure each connector on the MAU is fully seated and tight.

5) If found not seated correctly, remove the electrical connector that is suspected and visually check for damaged or pushed back contacts or damaged wiring. Refer to Para 4.C.

CAUTION **DO NOT TOUCH MAU MODULE CONTACTS. PROPER ESDS PROCEDURES MUST BE FOLLOWED. AN ESDS WRIST STRAP MUST BE WORN AND GROUNDED AT ALL TIMES, WHEN HANDLING THE MAU EQUIPMENT.**

6) Ensure the MAU module connector contacts are not bent or damaged.

7) Install the connectors which had been removed. Refer to Para 4.D.

8) Close MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

9) Check if the problem has been resolved.

C. MAU System Wiring

1) Open MAU 1A (E1), MAU 2A (F1), MAU 1B (E2) and MAU 2B (F2) circuit breakers on the avionics circuit breaker panel.

2) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in PSM 1-64-2W.

D. MAU Hardware

1) If the problem can be isolated to one of the MAU modules the module should be replaced. Refer to Para 4.E and Para 4.F.

2) If further repair or service to the unit is required, send to appropriately rated avionics repair facility.

6. Servicing

A. Air Intake Screen

The metal mesh at the air inlet at the back of the MAU cabinet should be clean and free of lint, dust and other foreign objects which may block cooling air flow.

The metal mesh must be inspected for cleanliness after any of the following events:

1) MAU removal.

2) Servicing of the MAU cooling fans.

Data Acquisition Unit (DAU) (Mod 6/2049) – General Data

1. General

The Data Acquisition Units (DAUs), DAU 1 and DAU 2 receive analog signals from the aircraft sensors and transmitters, and converts the analog signals to digital signals.

2. Description

A. Data Acquisition Unit (DAU)

The aircraft is equipped with two dual-channel Data Acquisition Units (DAUs) that collect and process various discrete, resistive, variable voltage and other signals from the engines and airframe and then encode this data into digital format for transmission via ARINC 429 to the MAU. The DAUs also do any logic required to trigger CAS messages. The DAUs are mounted above the cabin headliner, at approximately station 250. The DAUs are flange mount units and attach directly to the shelf.

The Apex ARINC 429 outputs to the MAU are a combination of graphics commands to drive systems indications, CAS message triggers, monitors that trigger aural warnings and PFD alerts, and phase of flight inhibit logic bits that are used by the MAU Monitor and Warning Function (MWF) to inhibit predefined CAS messages during certain phases of flight. The ARINC 429 data to the MAU also includes data required by the MAU to do logic related to avionics systems. The FDR ARINC 429 outputs to the MAU contain all applicable FDR data which the MAU converts to ARINC 717 and combines with other APEX FDR data before it is sent to the FDR (if installed). The FDR ARINC 429 outputs are also brought to the cross side DAU and provide cross side communication.

28 Volt DC power is provided from the left bus through 5 amp DAU 1A (E5) and DAU 2A (F5) circuit breakers, and from the right bus through 5 amp DAU 1B (E6), and DAU 2B (F6) circuit breakers on avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

1) The DAUs interface with the following aircraft systems and sensors:

   a) Torque Transmitters

   b) T_5 Sensors

   c) Propeller RPM Transmitters

   d) Gas Generator RPM Transmitters

   e) Fuel Flow Transmitters

   f) Oil Pressure Transmitters

   g) Oil Temperature Transmitters

   h) Generator Shunts

   i) Battery Shunt

   j) Brake Pressure Transmitter

   k) Hydraulic Pressure Transmitter

   l) Stall Warning Vane

   m) Cockpit Temperature Sensor

   n) Cabin Temperature Sensor

o) Elevator Trim Tab Potentiometer

p) Elevator Angle Position Potentiometer

q) Aileron Trim Tab Position Potentiometer

r) Aileron Position Potentiometer

s) Flap Position Potentiometer

t) Rudder Trim Tab Potentiometer

u) Rudder Angle Potentiometer

v) Aircraft Voltages via MAU ARINC 429

w) Aircraft Discretes via MAU ARINC 429

2) Some parameters were included as provisions. These parameters are jumpered, either directly or with drop down resistors, to other working parameters to prevent a DAU Maintenance CAS message. The provisions include:

a) Pilot Oxygen Transmitter (provision only)

b) Passenger Oxygen Transmitter (provision only)

c) Flap Lever Potentiometer

d) Engine Control Lever Potentiometers

e) Avionics Battery Voltage via the MAU ARINC 429

B. DAU Configuration Plugs

Two pin D Series DAU configuration plugs are used to define various aircraft parameters. They are located on either side of the DAU maintenance panel, above the cabin headliner, just forward of the DAUs at approximately station 244.00.

Parameters controlled by the configuration plugs include:

1) Flaps 20 permitted (provision only)

2) Engine Type

3) De-ice Installed

4) Pitot Heat Warning

5) Oxygen Installed (provision only)

6) Fuel in Kgs (provision only)

7) Gear Configuration

3. Adjustment/Test

A. Data Acquisition Unit (DAU)

If a fault condition is detected by the Built-In Test Equipment (BITE), a caution level CAS message will be displayed.

1) Ensure no DAU Fail or DAU Maintenance CAS messages are present following system initialization.

2) Ensure the following system windows are displayed properly:

a) Engine Window

b) CAS Window

c) Flight Controls/Hydraulic Window

d) Fuel Window

e) Electrical/Misc. Window

f) Environmental Window

4. Removal and Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT/REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MOUNTING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDING – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Data Acquisition Unit (DAU)

1) General

These procedures are for the removal and installation of DAU 1 and DAU 2.

2) Removal of Data Acquisition Unit (DAU)

a) Ensure the aircraft electrical system is OFF.

b) Open the DAU 1A (E5), DAU 2A (F5) (Left Bus), DAU 1B (E6) and DAU 2B (F6) (Right Bus) circuit breakers and fit safety clips.

c) Gain access to the ceiling compartment aft of the bulkhead 240.

d) Disconnect (4) electrical connectors.

e) While supporting the DAU, remove (4) mounting screws.

f) Remove unit from shelf.

Note Stow connectors of the removed component.

3) Installation of Data Acquisition Unit (DAU)

a) Ensure the aircraft electrical system is OFF.

b) Ensure DAU 1A (E5), DAU 2A (F5) (Left Bus), DAU 1B (E6) and DAU 2B (F6) (Right Bus) circuit breakers on the avionics circuit breaker panel are open and fitted with safety clips.

c) Lift DAU into position and install (4) mounting screws.

d) Reconnect (4) electrical connectors. Ensure connectors are tight.

Note It is possible to swap connectors P2 and P3. It is also possible to swap P4 and P5. Ensure connectors are installed in the correct positions to prevent erroneous operation or damage.

e) Install interior as required.

f) Remove safety clips and close the DAU 1A (E5), DAU 2A (F5) (Left Bus), DAU 1B (E6) and DAU 2B (F6) (Right Bus) circuit breakers.

B. DAU Configuration Plugs Change

Note The DAU configuration plug must match current gear configuration. VSP18–XXXX. The VSP18–XXXX part number is unique to each aircraft's configuration options. Contact Viking Technical Support for configuration option.

1) Ensure the aircraft electrical system is OFF.

2) Open the DAU 1A (E5), DAU 2A (F5) (left bus), DAU 1B (E6), DAU 2B (F6) (Right Bus) circuit breakers.

3) Gain access to the ceiling compartments aft of bulkhead 240.

4) Change the DAU configuration plugs to match the gear configuration located of FWD DAU mounting plate left and right hand side.

5) Close the DAU 1A (E5), DAU 2A (F5) (left bus), DAU 1B (E6), DAU 2B (F6) (Right Bus) circuit breakers.

5. Troubleshooting

A. Data Acquisition Unit (DAU)

1) Ensure DAU 1A (E5), DAU 2A (F5) (Left Bus), DAU 1B (E6) and DAU 2B (F6) (Right Bus) circuit breakers are closed.

2) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

3) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

4) If repair or service is required, send to appropriately rated avionics repair facility.

Monitor Warning System – General Data

1. General

The Monitor Warning System (MWS) is responsible for all APEX warning and caution functions.

2. Description

A. Monitor Warning System

The Monitor Warning System (MWS) controls the CAS messages, aural warnings, and annunciation. The MWS also records information in the Fault History Data Base (FHDB) via the Central Maintenance Computer (CMC).

The Apex system includes two Monitor and Warning Function (MWF) systems, one in each AIOP module, that independently calculate the logic for Crew Alerting System (CAS) messages. This allows one channel to control the MWS functions, such as aural alerts and CAS messages, while the other channel monitors the controlling channel for redundancy. Comparison monitors alert the pilot when there is difference between the results generated from MWF system 1 and MWF system 2.

MWF system 1 displays DAU channel A CAS messages and MWF system 2 displays DAU channel B CAS messages. Both channels of a DAU should always output the same message but in the event of a malfunction the two DAU channels could disagree and trigger the miscompare monitor.

In normal operation there should be no discrepancy between the calculations of the two MWF systems. In the event of a fault in a DAU channel that results in different outputs between DAU channels, the CAS miscompare monitor will activate. The pilot may then toggle the CAS message window between MWF system 1 and MWF system 2 by pressing the bezel button beside the small amber ‘MW’ square to display and compare output from each of the two MWF systems. By identifying the difference between the CAS messages generated by the two MWF system, and then gathering further information about the affected aircraft component from other sources (e.g. engine indications, mechanical trim indications, visual observation of flap or control surface positions, etc.), it should be possible to determine which MWF system is most credible. The selected channel is annunciated by the larger number in the 1/2 indication directly below the MW square.

Under certain rare circumstances (for example, an advisory maintenance message concerning one of the two DAUs) both MWF systems will be correct. The difference between the MWF systems will trigger a MW annunciation because the DAU maintenance message sent to each individual MWF will contain fractionally different text (DAU channel A self-identification sent to MWF 1 will differ from DAU channel B self-identification sent to MWF 2).

B. Crew Alerting System (CAS)

The Crew Alerting System (CAS) provides visual warning to the crew regarding malfunctions or abnormal conditions which occur during operation.

The CAS messages are grouped by severity into four priority levels: warning messages, which are displayed in red text, caution messages, which are displayed in amber text, advisory messages, which are displayed in cyan text, and status messages, which are displayed in white text.

CAS messages are posted in the lower left 1/6 window of the system window on the multi-function display.

Warning messages are accompanied by illumination of the red MASTER WARNING lights and sounding of a triple chime. Caution messages are accompanied by illumination of the amber MASTER CAUTION lights and sounding of a single chime. Advisory and status messages are not annunciated.

Refer to the Section 7 of the Pilot Operating Handbook, PSM 1-64-POH for additional details on the CAS messages.

Some parameters are annunciated on the ADI instead of with a CAS message. These include STALL, FIRE, NO TAKEOFF, and SPEED warnings.

C. Aural Warning System

Various aural warnings are generated by the Apex avionics system. Each aural alert is prioritized and grouped with other audio alerts, called priority groups, as is shown in Section 7 of the Pilot Operating Handbook, PSM 1-64-POH. Group 1 is the highest priority group and Group 4 is the lowest priority group. When two alerts of different groups are enabled at the same time, only the aural alerts of the highest priority group will be sounded. When the alerts of the highest priority group stop, the aural alert of the next highest priority group will be sounded, if still active.

Aural warnings listed as 'mutable' in the AFM are silenced by pressing the Master Caution (MC) or Master Warning (MW) switch, as appropriate to the notification. When the logic that drives a mutable aural is enabled and the appropriate MC/MW switch is depressed, the mutable aural is silenced until the enabling logic transitions off. When multiple mutable aural warnings are active, all will be muted at once. Therefore, each aural cannot be muted individually. All aural warnings will be heard at least once before muting becomes effective.

Aural warnings are independently processed by both channels of the custom I/O board in the MAU, then compared to ensure validity. In the event of a passive failure of one channel, an Aural Warning Fault advisory (cyan) CAS message advising of the failure will be displayed, but full aural warning capability will continue to be available from the single remaining channel. In the event of an active failure of one channel that results in unwanted sound coming from one of the two channels, the pilot can silence the defective channel by selecting the AURAL 1 MUTE or AURAL 2 MUTE switch, as appropriate, to the MUTE position. If both switches are placed in the MUTE position, no aural warnings will be heard and an Aural Warning Fail caution (amber) CAS message will be displayed.

TAWS and TCAS aural warnings are separate from the Apex aural warnings and are connected to the pilot's and co-pilot's audio panels through an audio mixer, refer to 23-50-00, Audio Integration – General Data for details.

D. Master Warning and Caution Lights

Two 1 inch by 1.2 inch LED switchlights, internally labelled MASTER CAUTION and MASTER WARNING, are provided directly above the pilot's and co-pilot's Primary Flight Displays.

These lights will illuminate whenever a caution level (amber) CAS message or a warning level (red) CAS message appears. Pressing the light provides a ground to the MAU which acknowledges the alert. This will extinguish the light and remove the highlight from all new CAS messages. The caution lights are powered by the Caution Lights Dimming Control box, and triggered by a ground from the MAU. Refer to 33-10-11, Caution Lights Dimming Control Box – General Data for more information.

3. Adjustment/Test

A. Monitor Warning System (MWS)

The MWS has built in test and fault detection and exclusion capability. Failures of components or loss of system capability will be announced to the crew using the aircraft crew alerting system. Information may be presented as caution or advisory messages, depending on the importance of the message or the severity of the condition.

B. Master Warning and Caution Lights Function Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Open HYDRAULIC OIL PUMP (C6) circuit breaker to trigger the Master Warning annunciation and a highlighted Hydraulic Power Fail warning CAS.

3) Press the pilot's master warning pushbutton and confirm the warning annunciation clears and CAS is no longer highlighted.

4) Open FUEL CROSSFEED (D7) circuit breaker to trigger the Master Caution annunciation and a highlighted Fuel Crossfeed Fail caution CAS.

5) Press the pilot MASTER CAUTION pushbutton and confirm the Caution annunciation clears and CAS is no longer highlighted.

6) Repeat Para 3.B.2)to Para 3.B.5) for co-pilot's side light.

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal and Installation

A. Removal of Master Warning and Caution Lights

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Open CAUTION LTS NORMAL (N3) and CAUTION LTS BYPASS (N2) circuit breakers on the avionics circuit breaker panel and fit safety clips.

3) Gain access to the rear of the lights by removing the pilot's or co-pilot's PFD, as required. Refer to 31-11-00, Display Units – General Data.

4) Disconnect electrical connector.

5) Using the extraction slots, pull the pushbutton cap out of the switch body until it rotates down.

6) Loosen the (2) internal screws until the locking pawls release and rotate into the switch body.

7) While holding the mounting sleeve behind the panel, pull the switch body out of the panel from the front.

8) If aircraft power is required with the unit removed, ensure the connector is stowed as required.

B. Installation of Master Warning and Caution Lights

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Ensure the CAUTION LTS NORMAL (N3) and CAUTION LTS BYPASS (N2) circuit breakers on the avionics circuit breaker panel are open and fitted with safety clips.

3) Using the extraction slots, pull the pushbutton cap out of the switch body until it rotates down.

4) Ensure the locking pawls are fully retracted by loosening the (2) internal screws.

5) Insert the switch body though the panel from the front.

6) Slide the mounting sleeve over the switch body from the rear of the panel.

7) Tighten (2) internal screws until the locking pawls pull the mounting sleeve tight against the panel.

8) Insert the pushbutton cap into switch body.

9) Connect the electrical connector.

10) Install pilot's or co-pilot's PFD, as required. Refer to 31-13-00, Apex Controllers – General data.

11) Remove safety clips and close CAUTION LTS NORMAL (N3) and CAUTION LTS BYPASS (N2) circuit breakers on the avionics circuit breaker panel.

12) Do a function test of the lights, refer to Para 3.B.

5. Troubleshooting

A. Monitor Warning System (MWS)

The monitor warning system is contained within the MAU AIOP module, any errors will be annunciated with a CAS message. Refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data for information on the AIOP module.

If a system CAS message is present, refer to the applicable systems Chapter from this Maintenance Manual for troubleshooting information.

B. Master Warning and Caution Lights

1) Ensure CAUTION LTS NORMAL (N3) and CAUTION LTS BYPASS (N2) circuit breakers are closed.

2) Remove electrical connectors and visually inspect the connectors for damaged wiring or contacts.

3) Ensure connectors are fully seated and tight.

4) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

5) Swap pilot and co-pilot lights.

Remote Image Bus (RIB) (S.O.O. 6212) – General data

1. General

The Remote Image Bus (RIB) module interfaces with external video inputs and the Modular Avionics Unit (MAU). The RIB module converts input analog video signals to a format required by Apex and the system displays. The RIB module is installed in the nose avionics bay.

2. Description

The RIB Video system consists of RIB switch/interface module, which is installed in the nose avionics bay (underside of left hand middle shelf). The interface module is capable of receiving one VGA signal and one composite video signal, which are converted to two RIB outputs. These outputs are connected to AGM 1 and AGM 2. The video signal output from the RIB module can be switched between VGA and composite using the input select pin on connector P477 of the RIB module. Ground selects the RGB input and open selects the composite input to be displayed.

The video input is displayed on the right 2/3rd of the Systems MFD; this display may be cycled between fuel/environment and video (default configuration) or charts, video and fuel/environment (upper MFD). No video cameras or video sources are included in S.O.O. 6212.

3. Adjustment/Test

A. Function Test RIB Module

1) Ensure that VIDEO (F7) (Left Bus) circuit breaker on the avionics circuit breaker panel is closed.

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Wait several minutes for the aircraft electrical and avionics systems to initialize.

4) Set the system MFD to VIDEO by pressing the CHART/VIDEO button on the MF controller. If no video input source is connect to the RIB module, verify that a blank screen is displayed. If a video source is connected to the RIB module, verify the video feed is displayed.

5) Open circuit breaker VIDEO (F7) (Left Bus). Verify 'Unavailable' is displayed in the video window.

6) Close circuit breaker VIDEO (F7) (Left Bus). Verify a blank screen is displayed if there are not video inputs to the RIB module or the video feed is displayed in the video window if a video source is connected to the RIB module.

4. Removal/Installation

A. Removal of RIB Module

1) Ensure the aircraft electrical system is OFF.

2) Open VIDEO (F7) (Left Bus) circuit breaker on the avionics circuit breaker panel and fit safety clip.

3) Gain access to the forward avionics shelves through the forward baggage compartment.

4) Remove the middle HIRF cover.

5) Disconnect connectors connected to P1 (if installed), P2 and P3 of the RIB interface module.

6) While supporting the RIB interface module, remove the four mounting screws and remove unit from shelf.

Note Stow connectors as required.

B. Installation of RIB Module

1) Ensure the aircraft electrical system is OFF.

2) On the avionics circuit breaker panel ensure VIDEO (F7) (Left Bus) circuit breaker is open.

3) Set the RIB module in position and install the four mounting screws.

4) Reconnect electrical connectors.

5) Install the middle HIRF covers. Ensure the covers are mounted securely.

6) Remove the safety clip and close VIDEO (F7) (Left Bus) circuit breaker on the avionics circuit breaker panel.

7) Do a function test as per Para 3.A.

5. Troubleshooting

A. RIB Module

1) Ensure associated circuit breaker is closed.

2) Ensure connectors on the RIB module are fully seated and secured.

3) Visually inspect the connectors at the RIB module and MAU for damaged wiring or contacts.

4) If troubleshooting at the wire level is required, refer to the applicable wiring diagram in PSM 1-64-2W.

5) Verify input power voltage is 28 VDC.

6) If an input source is connected to the RIB module, verify the signal type is an RGB signal VGA (640x480) or composite signal (PAL or NTSC).

CHAPTER 32

LANDING GEAR

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CHAPTER 32 – LANDING GEAR

LIST OF EFFECTIVE PAGES

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Landing Gear – General Data

1. General

The aircraft is normally fitted with a non-retractable landing gear which consists of the left and right main gear assemblies, a steerable nose gear assembly, and a tail bumper. Each of the single main wheels incorporates a hydraulically-operated, disc type wheel brake unit. Refer to 32-10-00, Main Gear – General Data, 32-20-00, Nose Gear – General Data, and 32-50-00, Nose Wheel Steering System – General Data.

When main and nose wheel-skis are fitted (Mod S.O.O. 6001 and Mod 6/2182) the skis can be extended to permit operation on unprepared snow or ice surfaces, or retracted to permit normal wheel operation. Although the wheel-ski installation is removable, certain associated components must remain permanently installed in the aircraft. These include, the hydraulic piping in the fuselage and on the main landing gear legs, which provide hydraulic fluid from the aircraft hydraulic system to operate the skis and the electrical cables to provide ski position indication. At the time of installation, the gear configuration changes and flight limitation placard must be replaced by a flight limitation placard having markings and limitations appropriate to a wheel-skiplane. Refer to 32-11-00, Main Wheel-Skis – General Data, 32-21-00, Nose Wheel-Ski (Mod S.O.O. 6001) and 32-30-00, Extension and Retraction (Wheel-Skis) Mod S.O.O. 6001.

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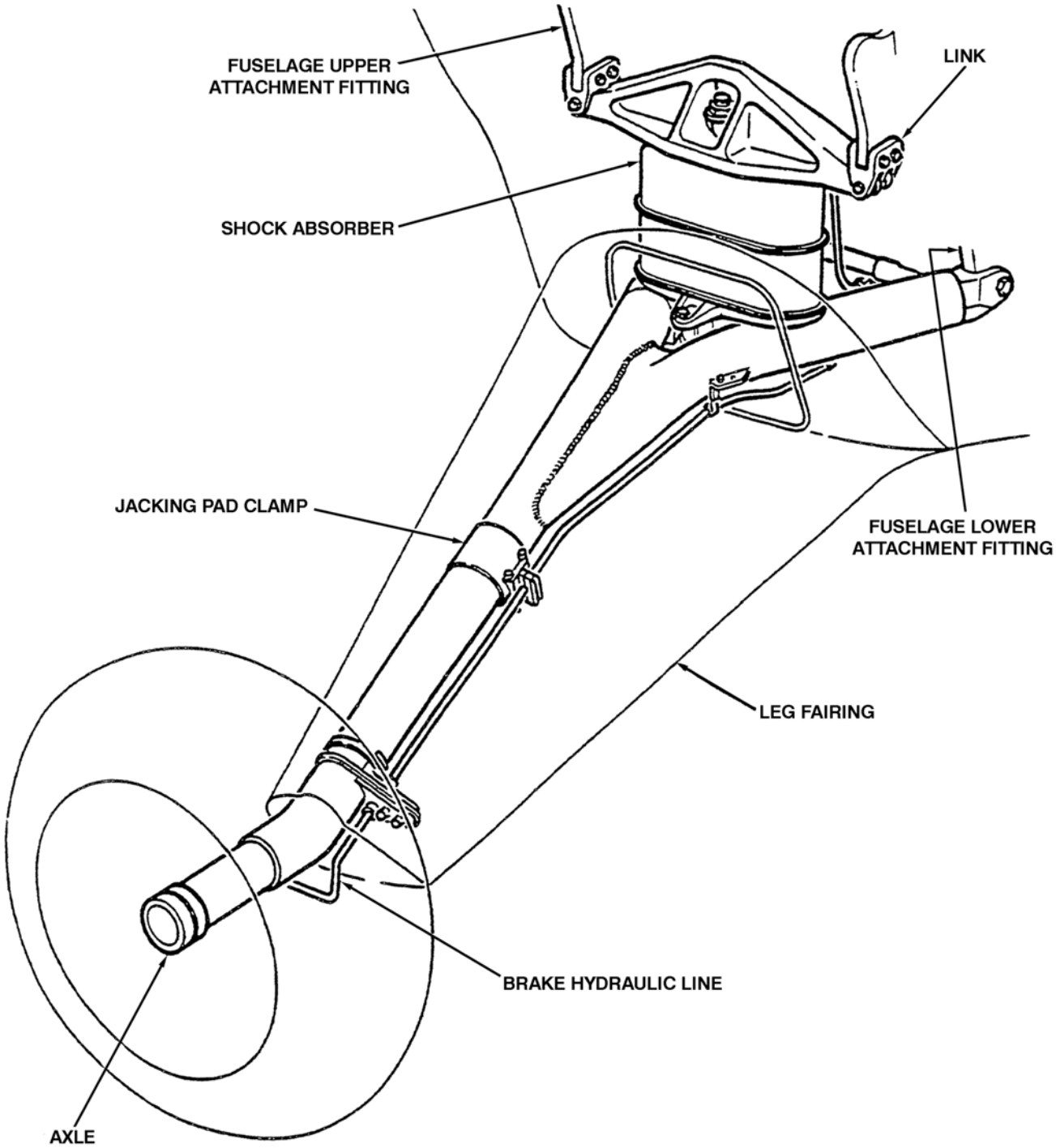
Main Gear – General Data

1. General

Each main gear consists of a leg assembly, shock absorber, and wheel and brake assembly. The leg assembly is bolted to two lower attachment fittings on the fuselage, and these provide the pivoted point for leg movement. Restriction and damping of leg movement is controlled by a shock absorber of the urethane block type. Each shock absorber is bolted to two fuselage upper attachment fittings and to metal pads on the leg assembly. Installed on the axle is a wheel and a hydraulically-operated brake assembly. A two piece fairing encloses the leg. Refer to Figure 1 for details.

A. Shock Absorber

The shock absorber consists essentially of urethane blocks, and absorbs the normal operating compression and rebound loads. The compression unit consists of an upper and lower platen between which are sandwiched two compression urethane blocks interspaced with a separator plate. Below the lower platen is the rebound unit comprising a rebound urethane block and end plate. The units are preloaded by a preload bolt and fork fitting. On landing, the leg is forced up against the compression blocks and upper platen which absorb the impact force, then as the leg rebounds, the rebound force is absorbed by the rebound block.



Main Gear
Figure 1

2. Removal/Installation

Refer to Figure 2 and Figure 3 for details.

A. Remove Shock Absorber

1) Remove main gear/fuselage fairings.

2) Jack aircraft until wheel is almost clear of ground (Refer to 07-00-00, Lifting and Shoring – General Data).

3) Remove cotter pin from preload bolt and remove preload nut.

4) Remove three bolts from underside of end plate and withdraw preload bolt.

5) Raise aircraft higher on jacks, and lift out compression blocks and separator plate.

B. Install Shock Absorber

1) Jack aircraft sufficiently, and position compression blocks and separator plate on lower platen.

2) Smear preload bolt, fork fitting and lower end of preload nut with grease, Specification MIL–G–21164. Assemble preload bolt and preload nut to main gear. Tighten three bolts on underside of end plate.

Note Bonding of urethane rebound block to metal end plates by operators is prohibited. A rebound block must be renewed, as necessary, with a complete assembly. Part No. C6U1137–3.

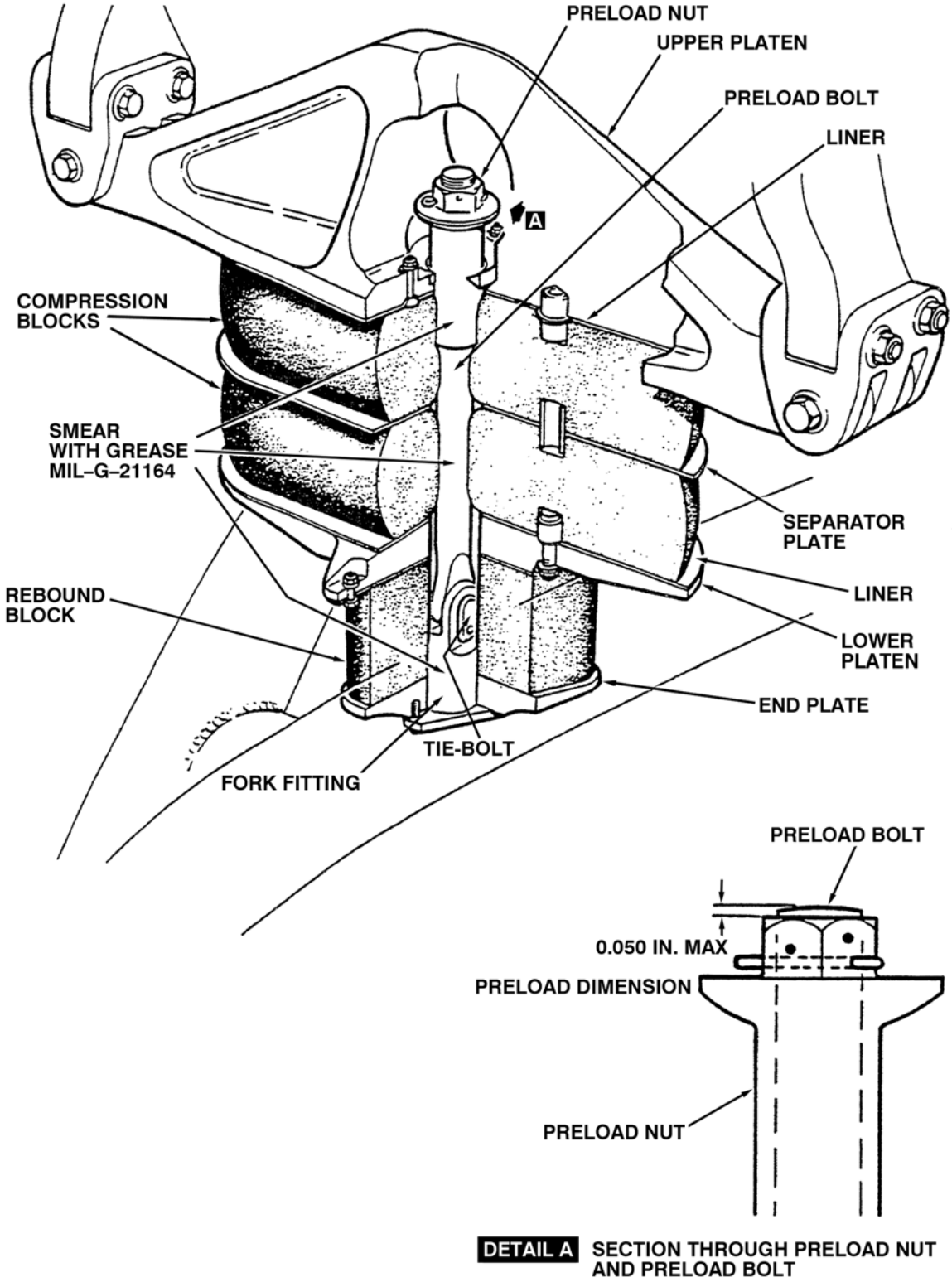
CAUTION - ENSURE THAT PRELOAD BOLT DOES NOT PROTRUDE MORE THAN 0.050 INCH THROUGH PRELOAD NUT.
- INSTALL ONE PACKING PAD ONLY. IF GAP IS NOT REMOVED BY USING PAD, COMPRESSION BLOCKS MUST BE REPLACED.

3) With wheel clear of ground, tighten preload nut until end of preload bolt (C6SC1025–5) is level with top of nut. Lock by installing cotter pin through hole in preload nut and slot in preload bolt.

Note After a period of operation permanent set of shock absorbers can occur. If, after tightening preload nut as in Para 2.B.3), a gap is found between the compression blocks and separator plates, the shock absorber must be dismantled and the compression blocks checked for cracks or pieces missing; replace if found defective. If shock absorbers have permanent set but are otherwise serviceable, a packing pad Part No. C6UM1219–27 (see Figure 3) must be installed between the bottom surface of the lower compression block, and top surface of lower platen. Simultaneously, the pan headed screws (MS27039-5–27), and lower platen spigots Part No. C6UM1142–27 must be replaced with longer spigots Part No. C6UM1142–29 (see Figure 3) and longer pan headed screws (MS27039-5–31), using existing nuts. After reinstalling shock absorber, adjust preload nut as in Para 2.B.3).

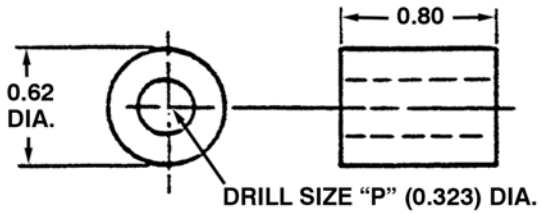
4) Lower aircraft to ground. Install main gear/fuselage fairings.

Note For aircraft incorporating Mod 6/2340, the fairings have rubber washers installed, torque the screws to compress the rubber washers (which are underneath the metal washers) to approximately one-half of their original thickness. For fairings without rubber washers (Pre Mod 6/2340) torque the #10 screws to 10 – 16 in–lbs.



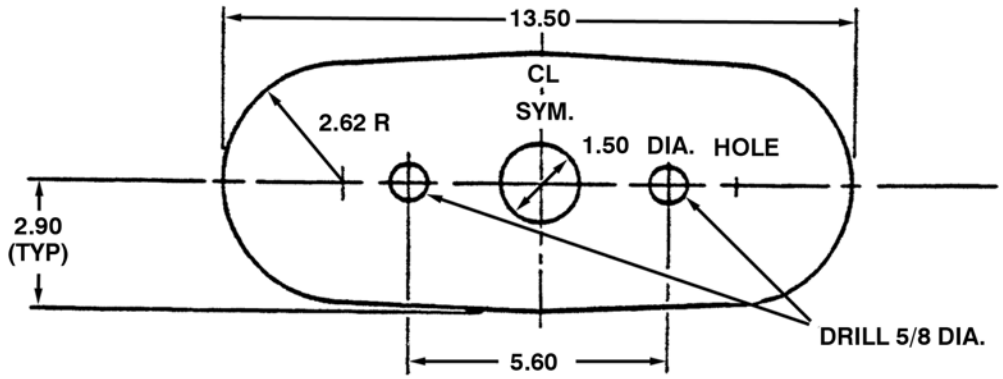
Shock Absorber Installation
Figure 2

SPIGOT (C6UM1142–29)



MATERIAL: ALUMINUM ALLOY BAR 2024–T4
SPEC: QQA–200/3
FINISH: ANODIZED (MIL–A–8625)

PACKING PAD (C6UM1219–27)



MATERIAL: 1/4 INCH THICK PHENOLIC SHEET
SPEC: MIL–P–15035 F.B.M.

Spigot and Packing Pad
Figure 3

C. Remove Upper Platen and Link Assemblies

Refer to Figure 4 for details.

1) Remove main gear/fuselage fairings.

2) Jack aircraft until wheel is almost clear of ground (refer to 07-00-00, Lifting and Shoring – General Data) and remove shock absorber (refer to Para 2.A).

3) Support aircraft on suitable cradle and/or trestle then remove jack.

4) Remove nuts and washers from upper platen mounting bolts.

5) Remove mounting bolts, washers and sleeves, then remove platen.

6) Remove nuts and washers from forward link assembly mounting bolts.

7) Remove mounting bolts, washers and sleeves, then remove forward link assembly.

8) Remove nuts and washers from aft link assembly mounting bolts.

9) Remove mounting bolts, washers and sleeves, then remove aft link assembly and shim (if shim installed).

Note Ensure shim is retained.

D. Install Upper Platen and Link Assemblies

Refer to Figure 4 for details.

1) Lightly smear platen and link assembly mounting bolts and sleeves with anticorrosion compound, Specification MIL–C–16173.

2) Install forward link assembly on landing gear forward attachment frame using sleeves, bolts, washers and nuts. Tighten nuts.

3) Temporarily install aft link assembly on landing gear aft attachment frame using sleeves.

Note Do not install shim or mounting bolts at this time.

4) Temporarily install upper platen on link assemblies using sleeves.

5) Using feeler gauge, measure and note gap between rear face of landing gear aft attachment frame and rear inner face of aft link.

6) Remove upper platen and aft link assembly.

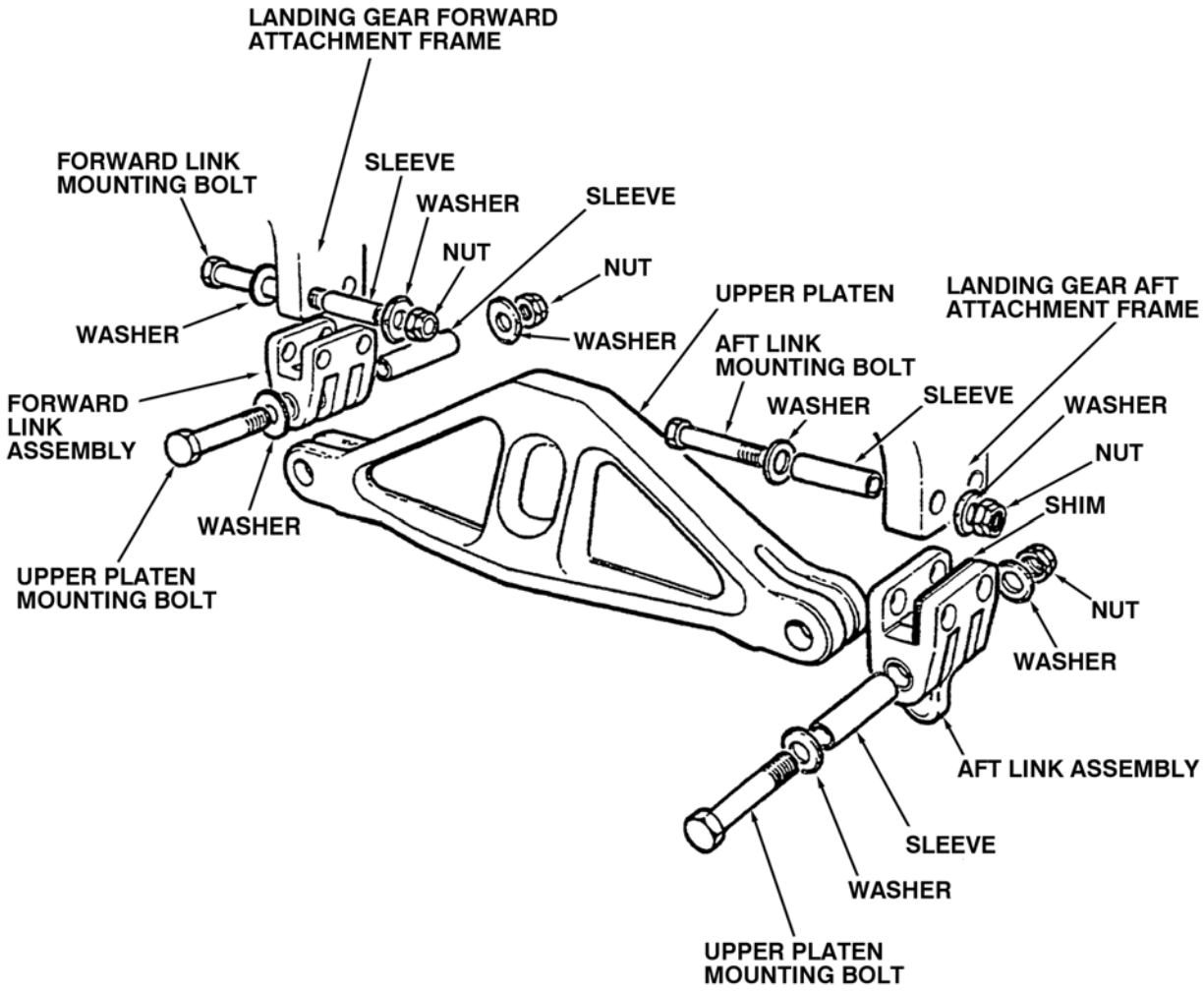
7) Prepare shim sufficient to reduce gap noted in Para 2.D.5) to 0.005 inch maximum.

8) Install aft link assembly and shim on aft landing gear attachment frame using sleeves, bolts, washers and nuts. Tighten nuts.

9) Install upper platen using sleeves, bolts, washers and nuts. Tighten nuts.

10) Jack aircraft (refer to 07-00-00, Lifting and Shoring – General Data) and remove cradle and/or trestle.

11) Install shock absorber (refer to Para 2.B).



Upper Platen Link Assemblies – Removal/Installation
Figure 4

3. Inspection/Check

A. Shock Absorber Compression Block Bubble Criteria

1) Limitations for bubbles within a shock absorber compression block under preload or free state condition are as follows:

a) There shall not be more than ten visible bubbles.

b) No single bubble shall be greater than 0.10 inch diameter.

c) No bubble shall be located less than two and one half times its diameter from any vertical surface of the block.

d) No bubble shall be separated from any adjacent bubble by less than one inch except as described under cluster limitation exceptions (refer to Para 3.A.2)).

2) Bubble cluster limitation exceptions are as follows:

a) Any cluster with a diameter less than 0.10 inch (which may be considered as a single defect) provided there are not more than three such clusters in the block.

b) Any cluster formed by not more than three bubbles, none of which exceed 0.10 inch diameter, provided that the separation between the bubbles is greater than four times the diameter of the largest bubble, and that there are not more than three such clusters in the block.

Main Gear Leg – General Data

1. General

Refer to Figure 1 for details.

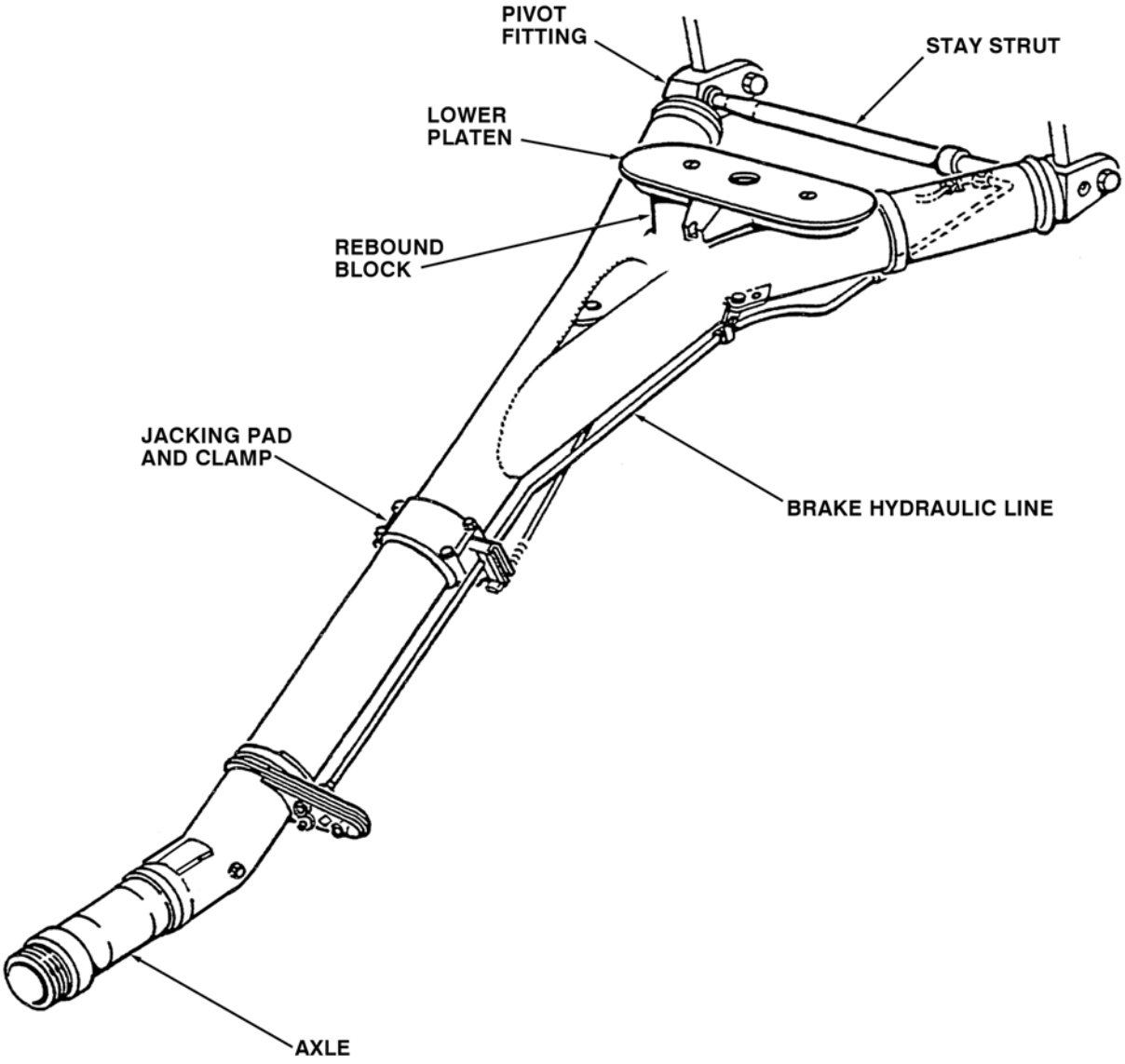
Each main gear leg consists of a Y-shaped tubular strut leg machining (C6UM1110) (Hy-Tuf Material), a stay strut and an axle. The Y-shaped strut is comprised of a front strut to which is welded a short rear strut. At the upper ends of both front and rear struts are pivot fittings which are welded to the struts. The two pivot fittings are interconnected by a stay strut. An axle is bolted to the lower end of the leg. Midway between the axle and the pivot fittings, a clamp type jacking pad is bonded to the strut. Four pads, each containing a threaded hole, are welded to the leg for attachment of the shock absorber lower platen. Each main gear leg is enclosed by front and rear section fairings.

A. Stay Strut

A stay strut is provided for each main gear leg and consists of a tube with two flanged end fittings. Each strut is positioned between the main gear leg pivot fittings and is shimmed equally at both ends to maintain the proper dimension between the pivot fittings. Screws and special washers secure the strut in position.

B. Main Gear Leg Fairings

Each main landing gear leg fairing is of fiberglass construction and is in two sections, front and rear, which are joined together by screws. The fairings are centered at the lower end of the leg by attachment to a split seal. The upper ends are attached to a frame which is bolted to two lugs on the leg. Contained in the frame is a roller which is free to turn in two nylon bearing cups. A blind which is attached to the fuselage structure rides over the roller and is held in tension by an elastic cord which is anchored to the front fairing. The blind provides protection for the shock absorber from dirt, water and debris entering the leg fairings during take-off and landing when the leg is in motion.



Main Gear Leg
Figure 1

2. Removal/Installation

Refer to Figure 2 for details.

A. Remove Main Gear Leg

1) Discharge brake system accumulator pressure by pumping brake pedals.

2) Remove main gear/fuselage fairing. Place container to receive spillage, and disconnect brake system flexible hose at elbow on stay strut. Cap open connection and line.

3) Jack aircraft until wheel is almost clear of ground (refer to 07-00-00, Lifting and Shoring – General Data), and remove shock absorber (refer to 32-10-00, Main Gear – General Data).

4) Remove pivot bolts at front and rear struts.

5) Disconnect brake system hydraulic line at lower end of leg. Cap open line. Remove wheel brake unit (refer to 32-40-31, Wheel Brakes Unit (Mod 6/2002) – General Data).

6) Remove leg fairings by removing attaching screws and unhooking blind elastic cord from inside of front fairing.

B. Install Main Gear Leg

1) Install wheel brake unit (refer to 32-40-31, Wheel Brakes Unit (Mod 6/2002) – General Data). Connect brake system hydraulic line at lower end of leg.

2) Position leg, insert pivot bolts and spacers at front and rear strut, using alignment pin SD12576–1 and acorn SD12576–3, and install washers and nuts. Tighten bolts to 200 to 400 pound–inches (dry thread) and install cotter pins.

3) Install shock absorber (refer to 32-10-00, Main Gear – General Data).

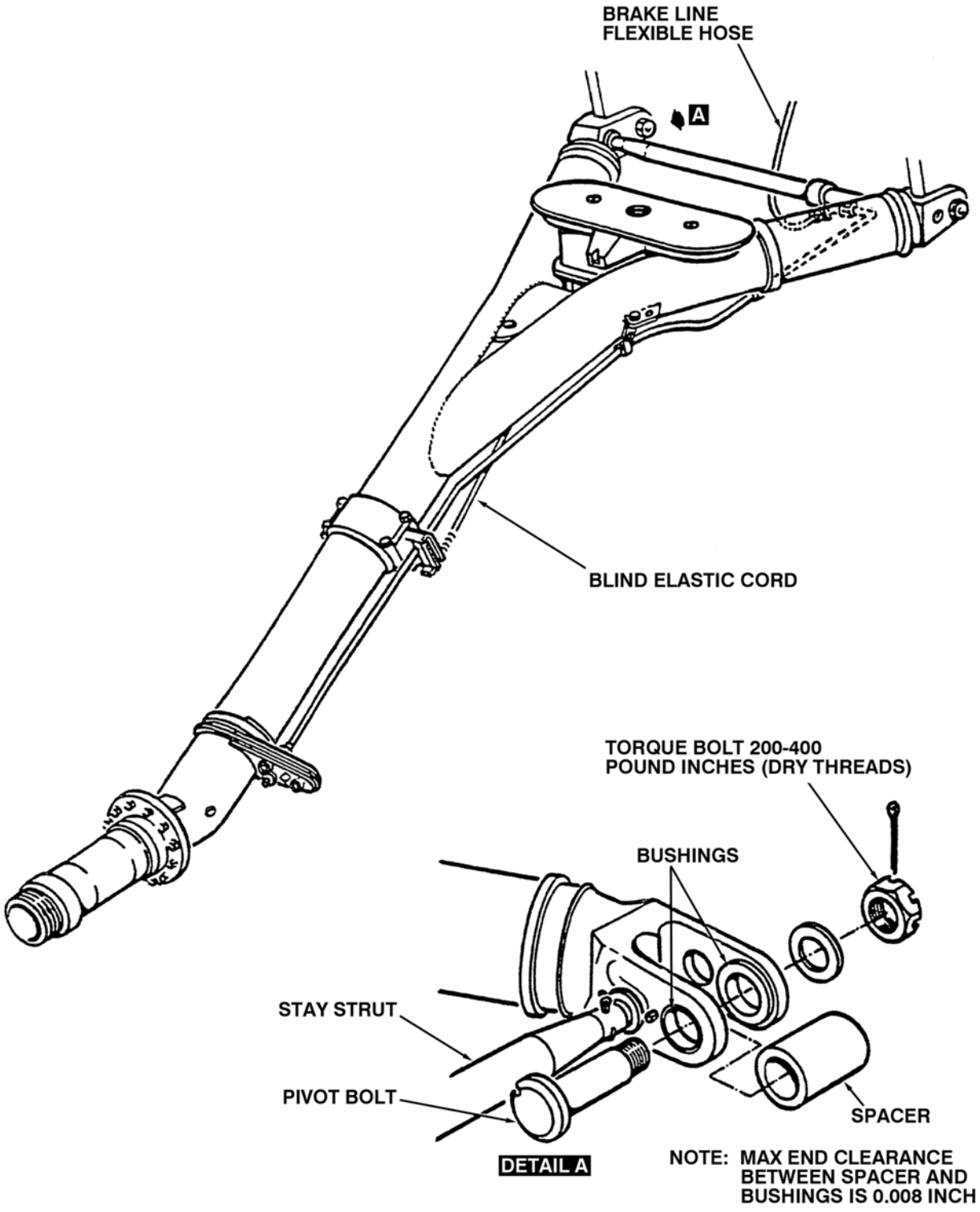
4) Connect brake system flexible hose at elbow on stay strut.

5) Bleed brake system (refer to 32-40-00, Wheels and Brakes (Mod 6/2002) – General Data, Para 2.).

6) Position leg fairings, secure attaching screws, and hook blind elastic cord to inside of fairing.

Note When a replacement main gear leg is installed, and fairing support frame is transferred from the unserviceable leg, the frame must be realigned with the two lugs on the leg by relocating the six washers on each bolt as required. The aircraft must be jacked up during this operation to ensure that leg will be in flight position (i.e., wheel clear of ground).

7) Install main gear fuselage fairing.



Main Gear Leg Installation
Figure 2

C. Gear Configuration Change

1) ESIS GH–3100 (Pre Mod 6/2170)

a) Change the DAU configuration plugs to match the gear configuration as per 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data, Para 4.B.2).

b) Change the ESIS configuration plugs as per 31-20-21, Electronic Standby Instrument System (ESIS) (Mod S.O.O. 6209, Mod 6/2049 and Mod 6/2170) – General Data, Para 3.E.1).

Note If this is a first install for the ESIS configuration module, a compass swing must be done to ensure the ESIS is calibrated correctly.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note If file identification is required for the MAU option files - aircraft Serial number specific, and ADAHRS certification files - gear configuration type specific, contact Viking Air Ltd. Technical Support.

d) Load the MAU option file to match gear configuration. (Refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

e) Load ADAHRS certification file to match gear configuration. (Refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data).

f) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

g) Do a configuration check, refer to Para 3.A.

2) ESIS GH–3900.2 (Post Mod 6/2170)

a) Change the DAU configuration plugs to match the gear configuration as per 31-44-00, Data Acquisition Unit (DAU) (Mod 6/2049) – General Data, Para 4.B.2).

b) Load GH–3900.2 configuration file to match gear configuration. (Refer to 31-20-21, Electronic Standby Instrument System (ESIS) (Mod S.O.O. 6209, Mod 6/2049 and Mod 6/2170) – General Data, Para 3.E.2)).

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Note If file identification is required for the MAU option files - aircraft Serial number specific, and ADAHRS certification files - gear configuration type specific, contact Viking Air Ltd. Technical Support.

d) Load the MAU option file to match gear configuration. (Refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

e) Load ADAHRS certification file to match gear configuration. (Refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data).

f) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

g) Do a configuration check, refer to Para 3.A.

3. Adjustment/Test

A. Configuration Check

1) Ensure the aircraft electrical system is OFF.

2) Open the DAU 1A (E5), DAU 2A (F5) (left bus), DAU 1B (E6), DAU 2B (F6) (right bus) circuit breakers.

3) Gain access to the ceiling compartments aft of bulkhead 240.

4) Confirm that the configuration plugs match the current gear configuration, VSP18–XXXX.

Note The VSP18–XXXX part number is unique to each aircraft's configuration options. Contact Viking Technical Support for configuration option confirmation.

5) Close the DAU 1A (E5), DAU 2A (F5) (left bus), DAU 1B (E6), DAU 2B (F6) (right bus) circuit breakers.

6) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

7) Confirm CAS Advisory message with current gear configuration in table below:

CAS Advisory Message	Gear Type
Gear Wheels	Standard Wheel
Gear Amphibious	Float 1 Amphibious
Gear Wipline Floats	Float 2 Wipline Floats
Gear IFG	Intermediate Flotation Gear
Gear Wheel-Ski	Wheel-Ski

8) If there is CONFIG_MISCOMPARE Warning CAS messages for the gear type. Refer to Para 6.

9) Confirm ESIS display matches the current gear configuration as per 31-20-21, Electronic Standby Instrument System (ESIS) (Mod S.O.O. 6209, Mod 6/2049 and Mod 6/2170) – General Data), Para 3.F.

10) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Inspection/Check

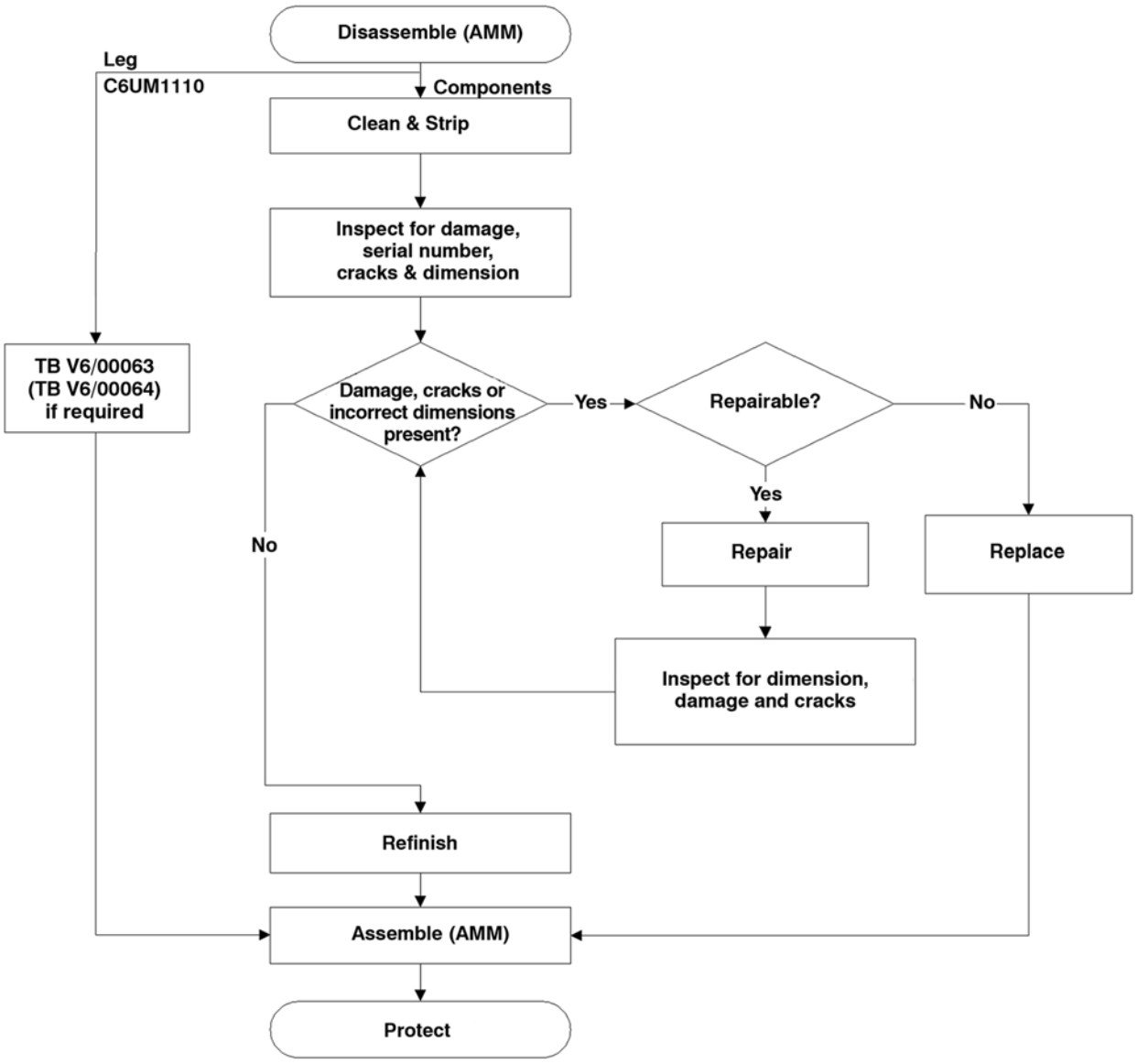
A. General

The general sequence for inspection and repair of the DHC-6 Main Landing Gear (refer to Figure 1) consists of the following:

1 The inspection and repair of the Main Landing gear components, in accordance with Aircraft Maintenance Manuals.

2 The inspection and repair of the DHC-6 Main Landing Gear Leg Machining (C6UM1110) (Hy-Tuf Material) and the pivot fittings, in accordance with TB V6/00063 for inspection, and (if required) TB V6/00064 for repair.

3 Protection of the MLG for short term storage (up to 6 months) in accordance with Para 5.B should proceed immediately after overhaul. Either the leg machining (C6UM1110) alone or the assembled MLG leg can be protected. If time will elapse between overhaul and assembly, protect the leg machining (C6UM1110) as per Para 5.B in the interim.


Disassemble (AMM)
Leg
C6UM1110
Components
Clean & Strip
Inspect for damage, serial number, cracks & dimension
TB V6/00063 (TB V6/00064) if required
Damage, cracks or incorrect dimensions present?
Yes
Repairable?
No
Yes
No
Repair
Replace
Inspect for dimension, damage and cracks
Refinish
Assemble (AMM)
Protect

1) Equipment and Materials

Facilities shall be equipped, in addition to standard shop tools, with the equipment listed below. Where applicable, equipment shall carry a current calibration and/or certification certificate and date.

Materials required for the removal, cleaning, inspection, finishing, assembly, and packing of the main landing gear leg assembly are provided below.

Note Use of qualified materials, where applicable, is mandatory.

a) Equipment and Tooling

- Compressed air supply, clean and dry
- Hot and cold water supplies
- Vapour degreasing equipment conforming to ASTM D3698
- Micrometer calibrated to ±0.0001 inch or better
- Telescoping gauge set
- Magnetic particle inspection equipment per ASTM E1444
- Fluorescent penetrant inspection equipment per ASTM E1417
- Heat lamp
- Paint spray/application equipment
- Hydraulic pressurization test equipment
- Coating thickness gauge
- Torque wrench
- Lint-free cloths (MIL–DTL–2467/C)
- Lint-free cotton gloves
- Solvent resistant gloves
- Stir sticks
- Tack rags
- Steel wire wheel
- Steel wool

b) Materials

- Abrasive paper, aluminum oxide, 180 – 220 grit (Commercial)
- Adhesive, RTV730 (Dow Corning)
- Adhesive, EC1357 (3M)
- Celastic tape, 0.050 inch thick recommended (Commercial)
- Conversion coating, Alodine 1200S or 1201 (MIL–DTL–81706)
- Corrosion preventive compound, Grade 1 (MIL–PRF–16173 Grade 1)
- Corrosion preventive compound, Grade 2 (MIL–PRF–16173 Grade 2)
- Degreasing solvent (MIL–PRF–680, Type I)
- Developer, fluorescent penetrant inspection (QPL–AMS–2644)
- Diestone DLS (Socomore)
- Diestone HFP (Socomore)
- DS108 (Dysol)
- Enamel, polyurethane (MIL–PRF–85285, Type I, Class H)
- Epoxy primer (MIL–PRF–23377)
- Fluorescent dye (QPL–AMS–2644)
- Grease (MIL–PRF–23827)
- Hydraulic fluid (MIL–H–5606)
- Kraft paper, oil-resistant and water repellent (Commercial)
- Loctite ‘H’ Sealant (ASTM D5363, AN0161)
- Loctite ‘N’ 7649 Primer (ASTM D5363, AN0321)
- Lubricant (MIL–L–7870)
- Magnetic wet particles, fluorescent (AMS 3045 or AMS 3044)
- Masking tape (Commercial)
- Methyl Ethyl Ketone (MEK) (ASTM D740)

- Non-acid paint stripper (Commercial grade)
- Sodium hydroxide (Commercial grade)
- Stoddard solvent (MIL–PRF–680)
- Suspension for magnetic particles (DOD–F–87935 or AMS 2641, Type I)
- Toluene (A–A–59107)
- Trichloroethane (MIL–T–81533)
- Trichloroethylene (ASTM D4080)
- EnSolv–5408 (stabilized n-Propyl Bromide based solvent), Enviro Tech
- EnTron–Aero 150 (stabilized n-Propyl Bromide based solvent)
- Leksol (stabilized n-Propyl Bromide based solvent), Amity International
- Varsol (ExxonMobil Chemical)
- VCI paper (MIL–PRF–3420 or meet the requirements of NACE TM0208)

c) Standards

- ASTM D3698, Standard Practice for Solvent Vapour Degreasing Operations
- ASTM E1417, Standard Practice for Liquid Penetrant Testing
- ASTM E1444, Standard Practice for Magnetic Particle Testing
- ISO/IEC 17025, General Requirements for the Competence of Testing and Calibration Laboratories

2) Standard Cleaning Methods

a) Solvent Cleaning External Surfaces

1 Use solvent applicable to material and cleaning purpose as listed in Table 1 — Solvent Use by Purpose and Material or Finish.

2 Apply solvent to a clean, dry, lint-free cloth and wipe the surface.

3 Dry with a clean cloth before solvent evaporates using a clean part of the cloth for each pass.

b) Solvent Cleaning Internal Surfaces

1 Plug one end of the part to be washed, as applicable.

2 Fill with solvent applicable to material and cleaning purpose as listed in Table 1 — Solvent Use by Purpose and Material or Finish.

3 Plug other part opening(s).

4 Hold the part horizontally and revolve part to ensure solvent reaches all internal surfaces.

5 Drain.

6 Repeat Para 4.A.2).b).2 through Para 4.A.2).b).5 until all dirt, and/or coatings and/or residue are removed.

Table 1 Solvent Use by Purpose and Material or Finish

Purpose	Material or Finish	Applicable Solvent(s)
Removal of oil, dirt, dust, etc.	Primed surfaces	DS108, Diestone DLS, Diestone HFP
	Painted parts other than axle, leg machining	DS108, Degreasing solvent (MIL–PRF–680, Type I)
	Painted axle, leg machining	DS108, MEK
	Plated parts	DS108, MEK
	Bare metal	DS108, Diestone DLS, Diestone HFP, MEK
	Bushings	DS108
Removal of uncured corrosion preventive compound or uncured adhesive	All metal parts	DS108, Diestone DLS, Diestone HFP, MEK
Solvent wash removal of cured corrosion preventive compound	Steel parts	MEK, trichloroethylene, trichloroethane, or equivalent
Preparation for installation	Fasteners	DS108 or MEK
Preparation for hydraulic test	Brake tube	Stoddard solvent or Varsol
Cleaning after hydraulic test	Brake tube	Stoddard solvent
Preparation for painting	Fully cured primed surfaces	DS108, Diestone DLS, Diestone HFP
	Partially cured primed surfaces	DS108, Diestone DLS, Diestone HFP, MEK
Preparation for application Celastic tape	Primed or painted	DS108
Preparation for re-anodizing	Aluminum	DS108, Diestone DLS, Diestone HFP, MEK
Vapour degreasing	All metal parts	Trichloroethylene or equivalent

c) Vapour Degreasing

Note Vapour degreasing is the preferred method for cleaning internal surfaces.

1 Mask bushings and bearings.

2 Vapour degrease per ASTM D3698 using trichloroethylene, methyl chloroform, or equivalent.

3 Rack or arrange parts vertically.

4 Vertical rate of entry not to exceed 11 ft/min.

5 After degreasing, raise part(s) and drain until no solvent remains on the part(s).

d) Mechanical Cleaning

1 Surfaces to be mechanically cleaned must be free of oil, grease, and other shop contaminants.

2 Use a steel wire wheel, steel wool, or abrasive paper (180 – 220 grit) to remove scale, oxide, or rust.

Note Cleaning tools shall be clean, free of contamination and/or particles of other metal types.

3) Standard Temporary Protection Methods

a) Temporary Protection of Exposed Steel Surfaces

1 If subsequent operations or inspections are not to occur immediately, temporarily protect exposed steel surfaces with oil conforming to MIL–L–7870, or with MIL–PRF–16173, or equivalent. Ensure parts have a continuous film of oil on all internal and external surfaces, as applicable. Alternatively, wrap part(s) in VCI paper (MIL–PRF–3420 or meeting the requirements of NACE TM0208).

2 Do not coat with oil, parts that have been cadmium or chrome plated, primed, or painted.

b) Temporary Protection of All Parts

1 If subsequent operations or inspections are not to occur immediately, wrap part(s) coated with MIL–L–7870 or MIL–PRF–16173 in kraft paper and secure with masking tape.

4) Standard Inspection Methods

a) Dimension Measurements

1 Measure dimensions using calibrated calipers with a tolerance of ±0.0001 inch.

b) Magnetic Particle Inspection

1 Carry out Magnetic Particle Inspection per ASTM E1444 using Fluorescent wet method with DC current.

2 Prior to inspection, ensure all paints and coatings (metallic and otherwise) have been removed and that surfaces are free of oil, dust, and shop contaminants.

3 Use best practices. Refer to Table 2 — Parameters for Magnetic Particle Inspection.

Note Values are provided for reference only. Parameters may vary depending on equipment and operator experience.

Table 2 Parameters for Magnetic Particle Inspection

Parameter	Value
Magnetizing current	DC
Field Strength	30 - 60 Gauss (2.4 - 4.8 kA/m)
Magnetization method	2 shots 0.5s - 1.0s 2 directions 90° apart
Particle	Wet fluorescent per AMS 3045 or AMS 3044
Suspension vehicle	DOD–F–87935 or AMS 2641, Type I
Particle concentration	0.1 - 0.4% by volume
Light source	Black light (UV-A), ≥1200 μW/cm^2 @ 15 inches

Table 2 Parameters for Magnetic Particle Inspection (continued)

Parameter	Value
Background illumination	≤2 ft-candles (20 lux)
Maximum residual magnetic field intensity	3 Gauss (0.24 kA/m)

4 Inspect exterior and interior surfaces. For interior surfaces, use a boroscope with the appropriate light source.

5 Inspect the whole of the part, paying particular attention to stress concentrations (e.g. holes, abrupt changes in shape), and weld areas.

6 De-magnetize.

7 Remove all magnetic particle solution by solvent cleaning, in accordance with Para 4.A.2).a) and Para 4.A.2).b).

8 Temporarily protect, in accordance with Para 4.A.3).

c) Fluorescent Penetrant Inspection

1 Carry out Fluorescent Penetrant Inspection per ASTM E1417 for all non-magnetic parts.

2 Prior to inspection, ensure all paints and coatings have been removed and that surfaces are free of oil, dust, and shop contaminants.

3 Use Type I (fluorescent dye), Method A (water washable), Sensitivity Level 3 (high) penetrant system conforming to QPL–AMS–2644.

4 Use a suitable developer, conforming to QPL–AMS–2644.

5 Use best practices.

6 Remove all inspection materials by solvent cleaning, in accordance with Para 4.A.2).a) and Para 4.A.2).b).

7 Temporarily protect in accordance with Para 4.A.3).

B. Inspection of Main Landing Gear Components

1) General

a) This section describes the minimum requirements for inspection of the main landing gear assembly components.

b) Fits and clearances for close parts must meet the requirements of Para 4.B.8) and PSM 1-6-3, Table 1, and Figure 1.

c) All parts retained for reassembly must be inspected for cracks. Standard parts (bolts, washers, pins, nuts, etc.), should be discarded and replaced with new parts.

2) Remove Main Landing Gear Leg from aircraft (refer to Para 2.A).

3) Remove Main Landing Gear Components C6U1103–3/–4 (refer to Figure 3 and Figure 4 for details) as follows:

a) Remove Brake Tube Assembly

1 Remove nut (1), tube clamp (2), and spacer (3) from clamp (38) at top of leg assembly.

2 Remove nut (4), washer (5), screw (6), clamp (7), nut (8), washer (9), angle (10), and bolt (11) from machined leg pad.

3 Remove bolt (12), washer (13), channel (14), and fairlead (15) from bracket at jacking pad.

4 Disconnect the tube from the union at rib, and from elbow at stay assembly. Remove the tube assembly (16).

Note If the leg assembly is fitted for skis, remove tube assembly and electrical wiring for ski position indication.

5 Remove nut (17), washer (18) and elbow (19) from stay assembly at top of leg assembly.

6 Remove nut (20) and union (21), two plugs (22), and two packings (23) from rib (25) at bottom of leg assembly.

b) Remove Lower Leg Rib

1 Release bond seal (26) from leg and rib (25), using toluene.

2 Loosen and remove clamp (24) and remove rib (25).

3 Remove Celastic tape (27) from rib (25).

Note Celastic tape will not be present for leg assemblies fitted for skis. This tape covers the hole provided in the rib for ski position electrical wiring.

c) Remove Axle

1 Remove nut (28), washer (29), and bolt (30).

2 Remove axle (31).

d) Remove Jacking Pad

1 Remove four nuts (32), four washers (33), two bolts (34), two bolts (34A), and remove bracket assembly (35) from clamp (36) and jacking pad (37).

2 Using MEK or DS108, loosen bond of clamp and jacking pad to leg, and remove clamp (36) and jacking pad (37).

e) Remove Upper Clamp

1 Using MEK or DS108, loosen and remove clamp (38) above lower platen.

2 Remove rubber sheet (39).

f) Remove and Disassemble Platen and Rebound Assemblies

1 Remove two nuts (40), two washers (41), two screws (42), two spigots (43), scuff plate (44) and liner (45) from platen assembly (48).

2 Remove four twelve-point bolts (46) and four washers (47).

3 Remove platen assembly (48), including rebound rubber assembly (55).

4 Remove four screw locking inserts (49) from leg machining pads.

5 Remove nut (50), washer (51), bolt (52), two bolts (53), and two washers (54).

6 Remove rebound rubber assembly (55) from platen assembly (48).

7 Remove two screw locking inserts (56) from platen assembly.

Note Do not remove four bushings (57) from platen assembly. Bushings need only be removed if repair is required (refer to Para 4.B.8)).

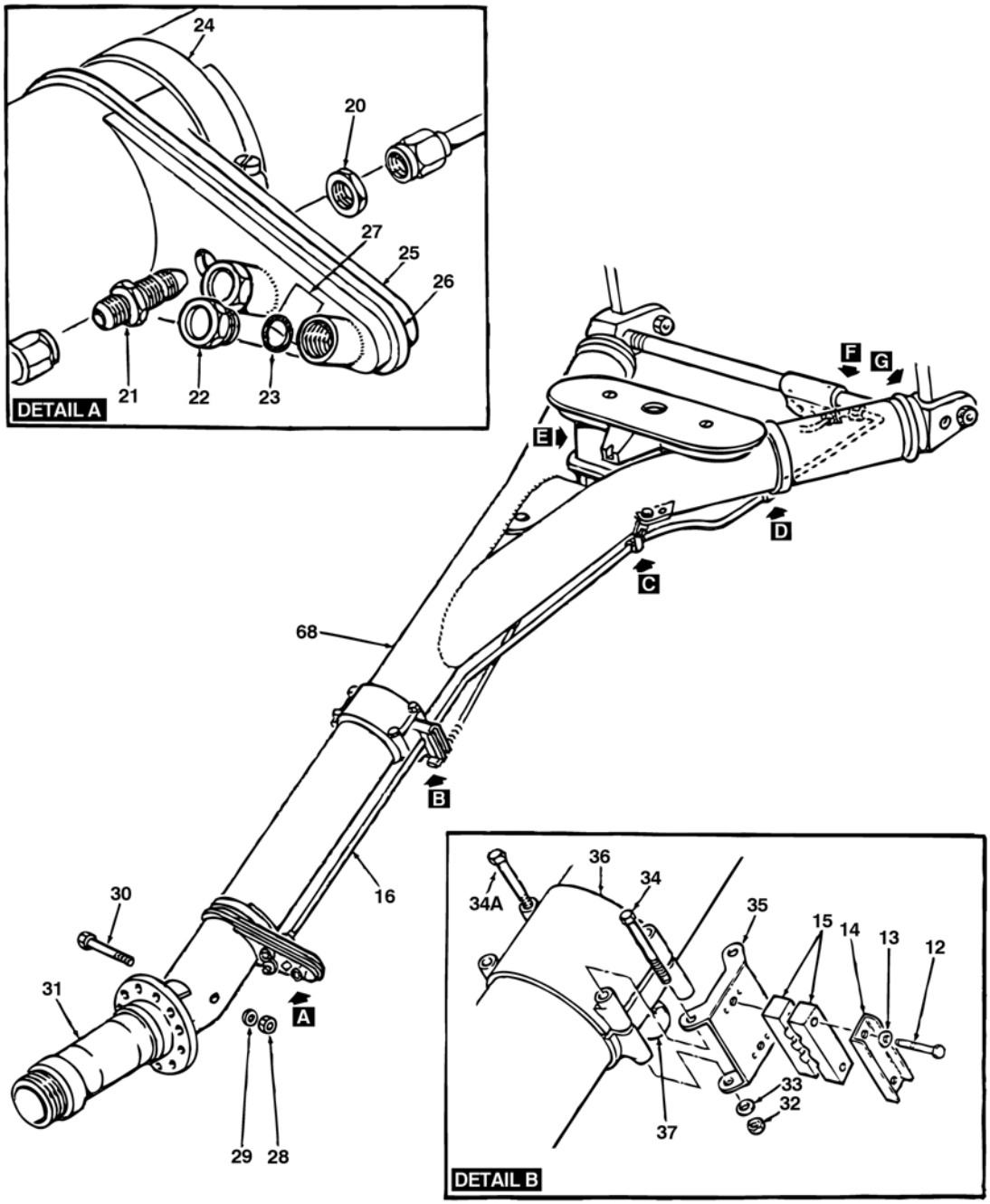
g) Remove Stay Assembly and Pivot Fitting Bushings

1 Remove spring pin (58) at rear stay pivot fitting.

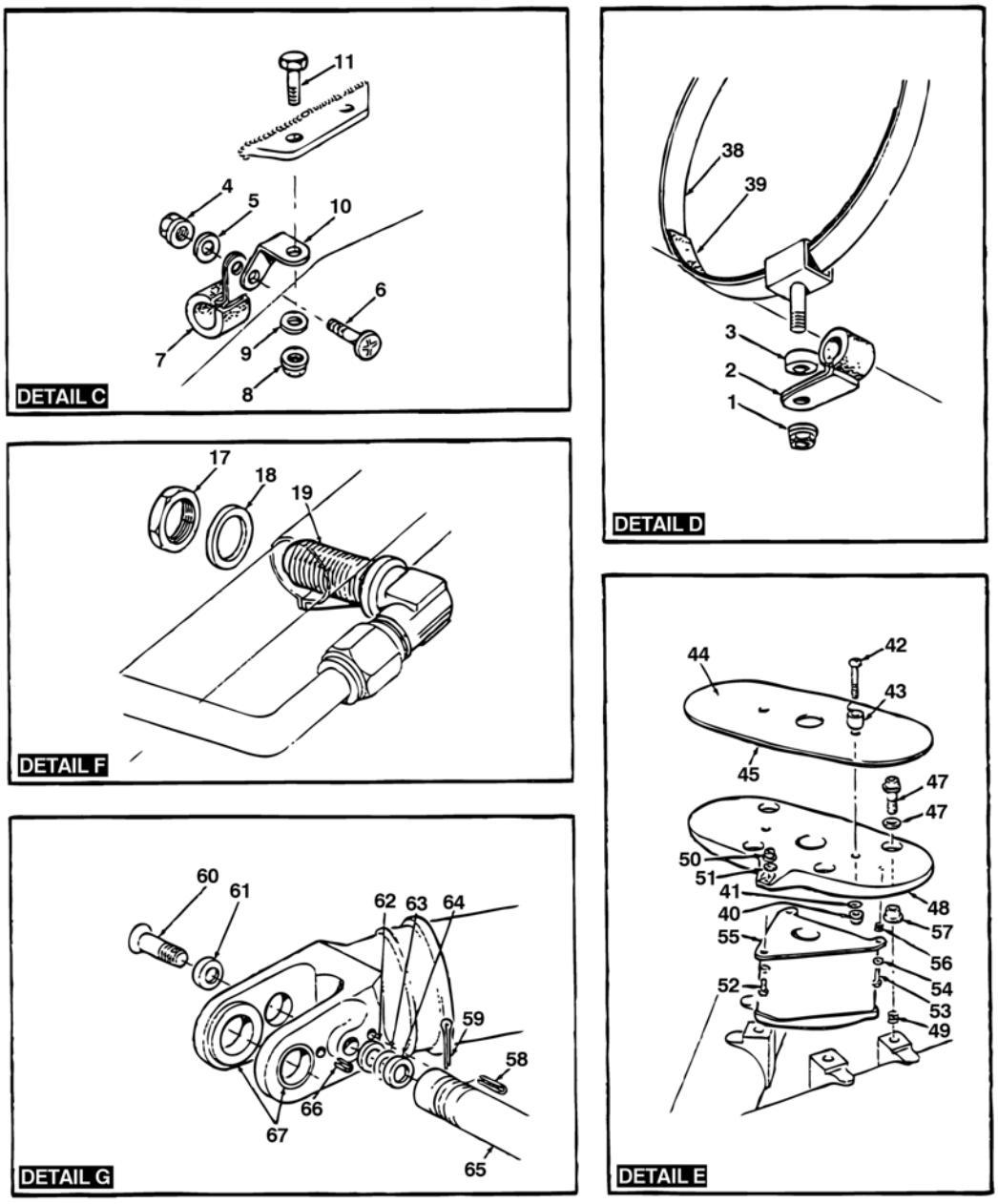
2 At front and rear pivot fittings, remove cotter pin (59), bolt (60), washer (61), washer (62) (if fitted), shim (63), washer (64), and remove stay assembly (65).

3 At front and rear pivot fittings, remove spring pin (66).

4 Remove four bushings (67) from pivot fittings.



Main Landing Gear C6U1103–3/–4 with Labeled Sub-Assemblies and Exploded Views
Figure 3



Additional Exploded Views for Main Landing Gear C6U1103–3/–4
Figure 4

4) Main landing gear leg machining (C6UM1110) (Hy-Tuf Material) and the pivot fittings are to be inspected at an appropriately rated inspection facility, in accordance with TB V6/00063, and, if required, repaired at an appropriately rated repair facility, in accordance with TB V6/00064.

5) Clean and Strip Main Landing Gear Components

   a) Cleaning on Receipt

      1 Remove dirt and surface contaminants by solvent cleaning parts, in accordance with Para 4.A.2).a) and Para 4.A.2).b), as applicable.

   b) Remove Corrosion Preventative Compound from Internal Surfaces

      1 Remove internal protective treatment by vapour degreasing, in accordance with Para 4.A.2).c), or by solvent washing, in accordance with Para 4.A.2).b).

      2 Visual inspect to ensure all traces of coating are removed. If not, repeat Para 4.B.5).b).1.

      3 Temporarily protect interior surface, in accordance with Para 4.A.3).

   c) Remove Paint from External Surfaces

      1 Clean exterior surfaces by vapour degreasing, in accordance with Para 4.A.2).c) (preferred) or by solvent cleaning, in accordance with Para 4.A.1).

      2 Remove external paint using a non-acid type paint stripper (Turco 5469, Cee–Bee R–256, Trebor 4500–PR55, Stripit 772, Oakite 157, Ardox 20, or equivalent).

         Note Paint must be removed from all parts to be inspected by Magnetic Particle Inspection or Fluorescent Penetrant Inspection.

      3 Immediately after stripping, solvent clean the external surface in accordance with Para 4.A.2).b).

      4 If subsequent operations are not to occur immediately, temporarily protect, in accordance with Para 4.A.3).

6) Inspect Main Landing Gear Components – General

   a) Visually inspect all part surfaces (outside and inside) for corrosion, plating defects, evidence of fatigue, scratches, cracks, and salt entrapment.

7) Inspect Main Landing Gear Components – Crack Detection

   a) Parts Compromised of Magnetic Metal Material

      1 Ensure that all paints and protective coatings have been removed and that parts are free of oil, dust, dirt and any other shop contaminants, in accordance with Para 4.B.5).a).

      2 Plated parts having electroplated finishes of 0.0008 inch or less, may be inspected without removal of the plating.

      3 Magnetic Particle Inspect, in accordance with Para 4.A.4).b).

      4 Parts having cracks in the parent material shall be replaced.

      5 Parts showing cracks in the plating can be repaired (refer to PSM 1-6-3, Section 32-10-00, Para 3.C.), or replaced.

b) Parts Compromised of Non-Magnetic Metal Material

1 Ensure that all paints and protective coatings are removed and that parts are free of oil, dust, dirt and any other shop contaminants in accordance with Para 4.B.5).a).

2 Fluorescent Penetrant Inspect, in accordance with Para 4.A.4).c).

3 Parts having cracks shall be replaced.

8) Inspect Main Landing Gear Components – Specific Components

a) Dimensional Inspections

1 Do all dimensional measurements in accordance with Para 4.A.4).a).

2 Many of the main landing gear component parts are close tolerance parts and are subject to wear.

3 Dimensions for fits and clearances are provided in PSM 1-6-3, 32-10-00, Table 1, and Figure 1. The table lists the “Original Manufacturing Limits” and “In-Service Wear Limits” for the corresponding part.

b) Main Landing Gear Axle (C6UM1134)

1 Visually inspect the plating for wear, scratches, and other damage or defects.

2 Visually inspect holes and diameters for wear, eccentricity, and other defects.

3 Visually inspect threads for wear and other defects.

4 Measure and record outer diameter of the axle bearing bands (refer to PSM 1-6-3, Section 32-10-00, Figure 1, Item A).

5 Measure and record axle bolt hole inner diameter (both holes), and axle bolt external diameter (refer to PSM 1-6-3, Section 32-10-00, Figure 1, Item B). Calculate and record clearance.

6 Axle not meeting outer diameter or plating requirements can be repaired (refer to PSM 1-6-3, Section 32-10-00, Para 3.C.).

7 Repair bearing surfaces in accordance with PSM 1-6-3RS, Chapter 32, RD6–32–016.

c) Ski Axle and Plug (C6US1015 and C6US1016)

1 Visually inspect the plating for wear, scratches, and other damage or defects.

2 Visually inspect holes and diameters for wear, eccentricity, and other defects.

3 Visually inspect threads for wear and other defects.

4 Measure and record outer diameter of the axle bearing bands (refer to PSM 1-6-3, Section 32-10-00, Figure 1, Item A).

5 Measure and record axle bolt hole inner diameter (both holes), and axle bolt external diameter (refer to PSM 1-6-3, Section 32-10-00, Figure 1, Item B). Calculate and record clearance.

6 Measure and record ski axle spline minor, major, and pitch diameters (refer to PSM 1-6-3, Section 32-10-00, Figure 1, Item E).

7 Ski axle and plug not meeting outer diameter or plating requirements can be repaired (refer to PSM 1-6-3, Section 32-10-00, Para 3.C.).

d) Lower Platen Assembly

1 Measure and record inner diameter of all four bushings (refer to PSM 1-6-3, Section 32-10-00, Figure 1, Item D).

2 Check that locking inserts are secure.

3 Visually inspect threaded holes for thread stripping, wear, or other defects.

4 Lower platen bushings and locking inserts not meeting dimensional tolerance must be replaced (refer to PSM 1-6-3RS, Chapter 32, RD6–32–016 for bushing repair limits or PSM 1-6-3, Section 32-10-00, Para 3.E.).

5 Repair lower platen holes and threads in accordance with PSM 1-6-3, Section 32-10-00, Para 3.E.

Note If the part cannot meet the requirements of PSM 1-6-3RS, Chapter 32, RD6–32–016 for bushing repair limits or PSM 1-6-3, Section 32-10-00, Table 1 upon repair, replace.

e) Brake Tube Assembly Inspection and Test

1 Visually inspect for holes or evidence of wear.

2 Flush line with Stoddard solvent or Varsol.

3 Ensure tubes inside and outside are dry.

4 Using hydraulic fluid conforming to MIL–H–5606, pressure test at 1.5 times the system operating pressure.

5 Ensure brake tube does not burst, leak, or otherwise deform under the test pressure. Replace brake tubes not meeting acceptance criteria.

6 Drip drain brake tubes and cap the ends.

7 Remove excess hydraulic fluid from the outside by solvent cleaning in accordance with Para 4.A.2).a).

9) Finish Axle Internal Surface as follows:

a) Prepare Internal Surfaces

1 Solvent clean internal surfaces, in accordance with Para 4.A.2).b) or vapour degrease in accordance with Para 4.A.2).c).

2 Visually inspect the internal surfaces to ensure all surfaces are free of corrosion products or other forms of contamination.

b) Apply Corrosion Preventive Compound to Main Landing Gear Axle (C6UM1134)

1 Plug one end of the axle using a suitable adhesive tape.

2 Completely fill the axle with undiluted corrosion preventive compound, conforming to MIL–PRF–16173, Grade 1.

3 Plug the open end of the axle using a suitable adhesive tape.

4 With the axle in the horizontal position, rotate the sealed axle 360° about the longitudinal axis several times to ensure all of the internal surface is covered. Let stand 5 minutes.

5 Drain and let stand for 16 hours.

6 Repeat protective treatment, Para 4.B.9).b).2 to Para 4.B.9).b).4.

7 Drain and let stand for a minimum of 8 hours.

8 Inspect all internal surfaces to ensure complete coverage of coating, if necessary, repeat Para 4.B.9).b).1 through Para 4.B.9).b).7 to obtain a satisfactory coating.

9 Remove excess compound from exterior surfaces by solvent cleaning, in accordance with Para 4.A.2).a).

c) Apply Corrosion Preventive Compound to Main Landing Gear Ski Axle (C6US1015)

1 Mask the splines and plug one end of the axle using a suitable adhesive tape.

2 Fill the axle with undiluted corrosion preventive compound, conforming to MIL–PRF–16173, Grade 1, taking care to not get any compound on the splines. Fill slowly to prevent air bubble formation. Let stand 5 minutes.

3 Puncture the plugged end and allow compound to slowly drain.

4 Position the part vertically and let stand for 16 hours.

5 Repeat protective treatment, Para 4.B.9).c).2 to Para 4.B.9).c).3.

6 Position the part vertically and let stand for a minimum of 8 hours.

7 Inspect all internal surfaces to ensure complete coverage of coating. If necessary, repeat Para 4.B.9).c).1 through Para 4.B.9).c).6 to obtain a satisfactory coating.

8 Brush a thin, uniform coating of undiluted corrosion preventive compound, conforming to MIL–PRF–16173, Grade 2, onto the ski axle splines, overlapping slightly with the Grade 1 coating. Inspect to ensure complete coverage.

9 Remove excess compound from exterior surfaces by solvent cleaning, in accordance with Para 4.A.2).a).

10) On parts other than leg machining or axle, restore damaged external finish as necessary using fluid resistant epoxy primer to MIL–PRF–23377 and grey polyurethane enamel to MIL–PRF–85285, Type I, Class H. As an alternative, zinc chromate primer may be used in place of epoxy primer. The use of etch primer is not recommended.

11) Install Main Landing Gear Components C6U1103–3/–4 (refer to Figure 3 and Figure 4 for details) as follows:

Note Ensure all parts are clean and free of defects prior to assembly. Only newly inspected and repaired, or replaced, main landing gear components are to be installed on newly inspected and, if required, repaired leg machining (C6UM1110), or new leg machining.

a) Install Stay Assembly

1 Ensure bushings (67) (or oversize bushings) are secure in pivot fittings.

2 Apply corrosion protective compound (Grade 2) to holes for spring pins, and install spring pins (66) at front and rear pivot fittings, and spring pin (58) at rear pivot fitting.

3 Coat a new bolt (60) with corrosion preventive compound (Grade 2).

4 Position stay assembly (65), and install washer (64), shim (63), washer (62) (if fitted), washer (61) and bolt (60) at front and rear pivot fittings. Torque bolt (60), at each end 100 to 150 pound-inches.

5 Drill the new bolt (60) for cotter pin installation. Install a new cotter pin (59).

b) Install Platen Assembly

1 Install two screw locking inserts (56) to platen (48).

2 Position rebound rubber assembly (55) on platen (48) and secure with two washers (54), two bolts (53), bolt (52), washer (51), and nut (50).

3 Coat four screw-locking inserts (49) with corrosion protective compound (Grade 2) and install in pads on leg.

4 At pads on leg, clean threads in each pad locking insert, and threads of twelve-point bolts (46) by brushing thoroughly with DS108. Wipe dry with lint free cloth.

5 Coat all threads with Loctite 'N' 7649 Primer (ASTM D5363, Grade N), and allow to air dry for five minutes.

6 When activator is dry, apply an even coat of Loctite Grade 'H' (ASTM D5363, AN0161) sealant to all threads.

7 Position platen assembly complete with rebound assembly on pads, and install four washers (47) and four twelve-point bolts (46). Torque bolts 480 to 500 pound-inches.

8 Cure assembly at 120° - 150°F for 30 minutes under a heat lamp.

9 Position scuff plate (44) with liner (45), against face of platen assembly and install two spigots (43), two screws (42), two washers (41), and two nuts (40).

c) Install Axle

1 Apply corrosion preventive compound (Grade 2) to bolt holes in axle and leg.

2 Position axle (31) and secure to leg with bolt (30), washer (29), and a nut (28).

d) Adhesive Bond Jacking Pad to Leg Machining

WARNING **USE SOLVENTS AND ADHESIVES IN A WELL VENTILATED AREA AND KEEP AWAY FROM OPEN FLAME. AVOID TOUCHING PREPARED BOND SURFACE AREAS WITH BARE HANDS. WEAR CLEAN, SOLVENT RESISTANT GLOVES WHEN CLEANING AND HANDLING SURFACES.**

CAUTION **DO NOT USE PROTECTIVE HAND CREAM AS IT MAY CONTAMINATE CLEAN OR ADHESIVE COATED SURFACES.**

1 Clean jacking pad bond area of leg machining, jacking pad (37), and clamp (36), using a cloth dampened with DS108 or MEK, and wipe dry with a lint-free cloth before solvent evaporates.

2 Thoroughly stir adhesive EC1357 and apply a thin, even coat to bond surfaces.

3 Allow surfaces to air dry until adhesive is tacky, but does not transfer to gloved fingers when touched lightly.

4 Measure 15.00 inches from center of hole in lug toward axle.

5 Position jacking pad and clamp so that the lower face of jacking pad and clamp is coincident with the 15.00 inch mark (refer to Figure 5). Press parts together.

6 Position bracket (35) and secure jacking pad and clamp with two bolts (34), two bolts (34A), four washers (33) and four nuts (32).

7 Remove surplus adhesive using toluene or MEK.

e) Adhesive Bond Upper Clamp to Leg Machining

1 Clean bond surface areas on leg machining (above the shoulder line, short tube side), rubber sheet (39), and clamp (40), using a cloth dampened with DS108 or MEK. Wipe dry with a clean, dry lint-free cloth before solvent evaporates.

2 Measure 0.75 inch from the shoulder line on the leg machining (short tube side) toward the pivot fitting as per Figure 5.

3 Apply a thin, even coat of adhesive RTV730 to the rubber sheet and clamp.

CAUTION RTV730 ADHESIVE IS COMPLETELY SOLVENT RESISTANT. HANDLE WITH CARE AND AVOID SPILLAGE.

4 Install rubber sheet and clamp so that the bottom face is coincident with the measured 0.75 inch mark.

f) Mount Lower Rib and Adhesive Bond Seal to Rod and Leg Machining

1 Cut Celastic tape to a dimension of 1.0 x 1.0 inch.

Note Use of Celastic tape is not applicable to leg assemblies fitted for skis.

2 Clean rib (25) area to be covered with tape, using a cloth dampened with MEK and dry with a clean cloth before solvent evaporates.

3 Thoroughly soak contact side of tape with MEK and hand smooth tape on rib. Use a non-porous rubber block or pad to shape the tape to rib surface and ensure intimate contact throughout.

4 Measure 27.33 inches from center of hole in lug toward the axle (refer to Figure 5).

5 Position lower face of rib (25) coincident with the 27.33 inches measured point. Secure rib (25) to leg machining with clamp (24).

6 Temporarily install seal (26) and trim to suit leg and rib.

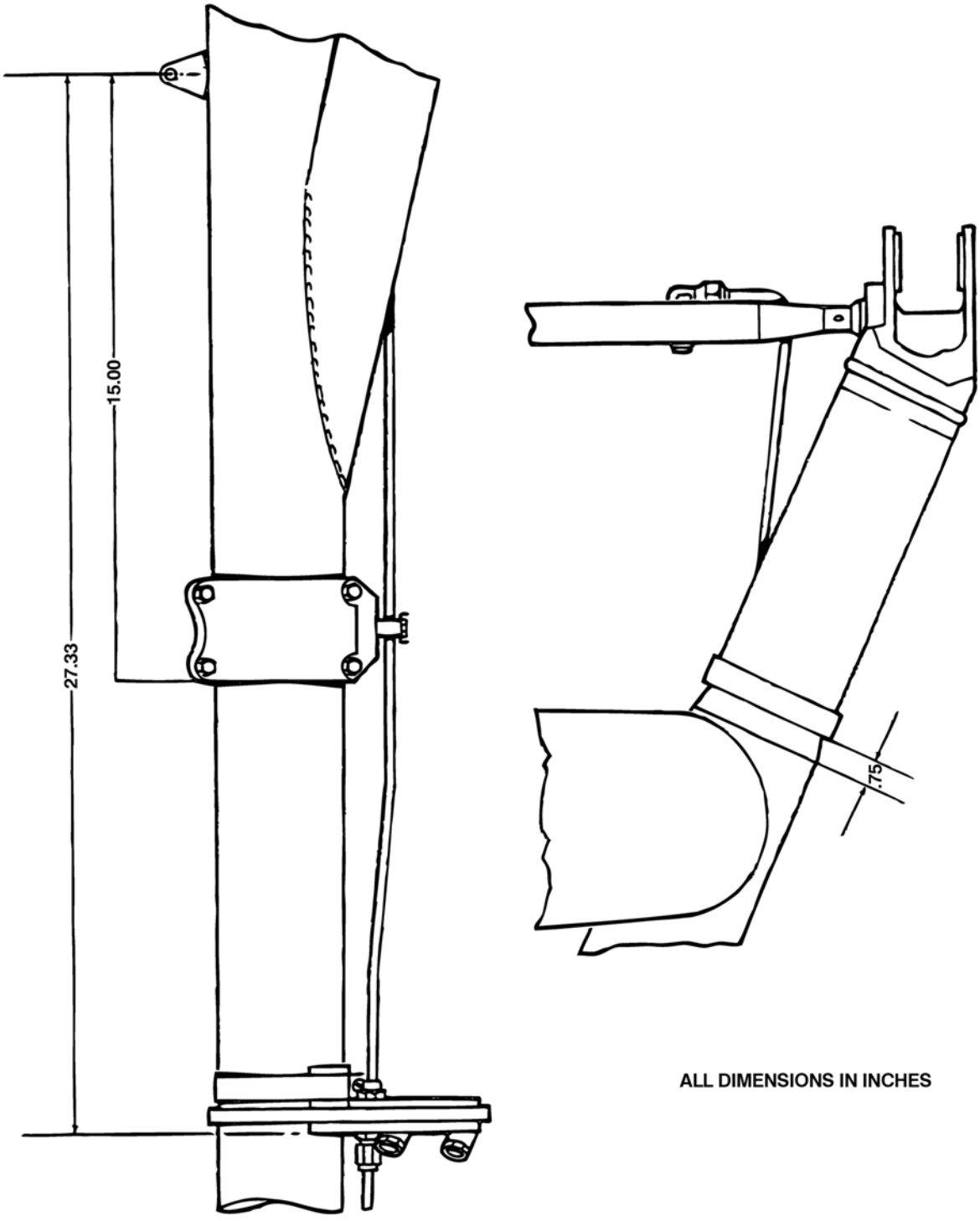
7 Clean bonding surfaces of rib, leg machining, and seal, using a cloth dampened with DS108 or MEK and wipe dry with a clean cloth before solvent evaporates.

8 Apply a thin, even coat of adhesive RTV730 to the rib and leg machining, position seal and press firmly against leg and rib.

CAUTION RTV730 ADHESIVE IS COMPLETELY SOLVENT RESISTANT. HANDLE WITH CARE AND AVOID SPILLAGE.

9 Cure adhesive bonds for 24 hours at room temperature.

WARNING ALLOW ALL BONDS TO SET BEFORE SUBJECTING ASSEMBLY TO FURTHER WORKING OR INSTALLATION.



Positioning of the Clamps for Main Landing Gear C6U1103–3/–4
Figure 5

g) Complete Rib Installation

1 Install two packings (23), two plugs (22), union (21) and nut (20) to rib.

h) Install Tube Assembly

1 Position elbow (19) on stay assembly at top of leg and secure with washer (18) and nut (17).

2 Position tube assembly (16) parallel to the leg machining.

3 Connect tube assembly to elbow (19) on stay assembly and to union (21) on rib.

Note If leg assembly is fitted for skis, install two tube assemblies and electrical wiring for ski position indication.

4 Position fairlead (15) and channel (14), and secure tube assembly to bracket (35) on jacking pad with washer (13) and bolt (12).

5 At machined lug, position angle (10) and secure with bolt (11), washer (9) and nut (8). Secure tube assembly to angle with clamp (7), screw (6), washer (5), and nut (4).

6 Secure tube assembly to clamp (38) at top of leg assembly with spacer (3), clamp (2), and nut (1).

12) Locate and record the 4 digit serial number and 2 letter suffix on the aft face of the aft pivot fitting for the inspected and repaired leg machining (C6UM1110).

13) Install main landing gear leg (refer to Para 2.B).

5. Protection for Storage

A. Preparation

1) Cap the brake tube assembly at both ends.

2) Lightly lubricate bushings with grease, conforming to MIL–PRF–23827, and wrap front and rear pivot fittings and stay assembly joints in water vapour resistant barrier material.

3) Wrap platen assembly and rebound rubber assembly in water vapour resistant barrier material.

4) Lightly lubricate axle with grease, conforming to MIL–PRF–23827, and wrap in water vapour resistant barrier material.

B. Protection for Temporary Storage (New or Newly Overhauled MLG Only)

The following steps protect and prepare the main landing gear leg machining (C6UM1110) or leg assembly for short-term storage (less than 6 months).

Note For temporary storage of up to 6 months, provision for protection is applicable only to new or newly overhauled MLG.

1) Prepare on a flat soft table.

2) Visually inspect to ensure all surfaces are appropriately plated, painted, and/or protected.

CAUTION ENSURE THAT THE SELECTED HEAT SEALING MACHINE CAN PROVIDE THE CORRECT TEMPERATURE AND PRESSURE FOR THE SELECTED FILM. FAILURE TO PROPERLY SEAL THE PACKAGE WILL COMPROMISE THE CORROSION PREVENTIVE PROPERTIES OF THE PACKAGING.

CAUTION **DO NOT PLACE MATERIALS THAT MAY CARRY OR ABSORB MOISTURE, E.G. PAPER PRODUCTS OR FOAMS, INSIDE THE PACKAGING. MOIST MATERIALS IN CONTACT WITH THE MLG MAY INITIATE CORROSION.**

3) Unroll a suitable length of multi-metal Vapor Corrosion Inhibiting (VCI) film to MIL–PRF–22019 Type I or NACE–TM0208 Grade 4 (e.g. CorTec VpCI–126), with a minimum thickness of 0.004 in (0.1 mm). Sleeving or bags made of this material can also be used.

4) Carefully place the main landing gear leg on the film or in the sleeving or bag. Use a hand-held heat sealing machine to seal the open edges, leaving a small gap on one edge. Gently press out excess air. Seal the remaining opening using the heat sealing machine. Refer to manufacturer instructions for the recommended sealing conditions (temperature, time, and pressure) for the selected film/sleeve/bag.

5) Label the exterior with the serial number, part number, overhaul date and packaging date.

6) Store the MLG indoors, away from direct sunlight. Avoid contact with moisture. Do not stack. Enclosing the MLG in a protective crate after bagging is recommended to prevent contact with other objects.

6. Troubleshooting

The DAU configuration plugs are unique to an aircraft specific gear configuration, VSP18–XXXXXX. (XXXXXX = LAND, SKIS or FLOATS). The DAU compares the configuration plug to the gear configuration data held in the MAU options file.

If the data does not match the DAU generates the following CAS Warning:

CAS Warning	Possible Cause	Check	Remedial Action
CONFIG-MISCOMPARE	Mismatch between the DAU configuration plugs and the selected MAU configurations files.	Check that the correct configuration plugs are used on DAU 1 and DAU 2. Check correct option file is selected on MAU for gear configuration.	Confirm DAU configuration plugs and MAU option file settings to match the current gear configuration.

If the configuration error continues, contact Viking Air Ltd. Technical Support.

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Main Wheel-Skis (Mod S.O.O. 6001 and Mod 6/2182) – General data

1. General

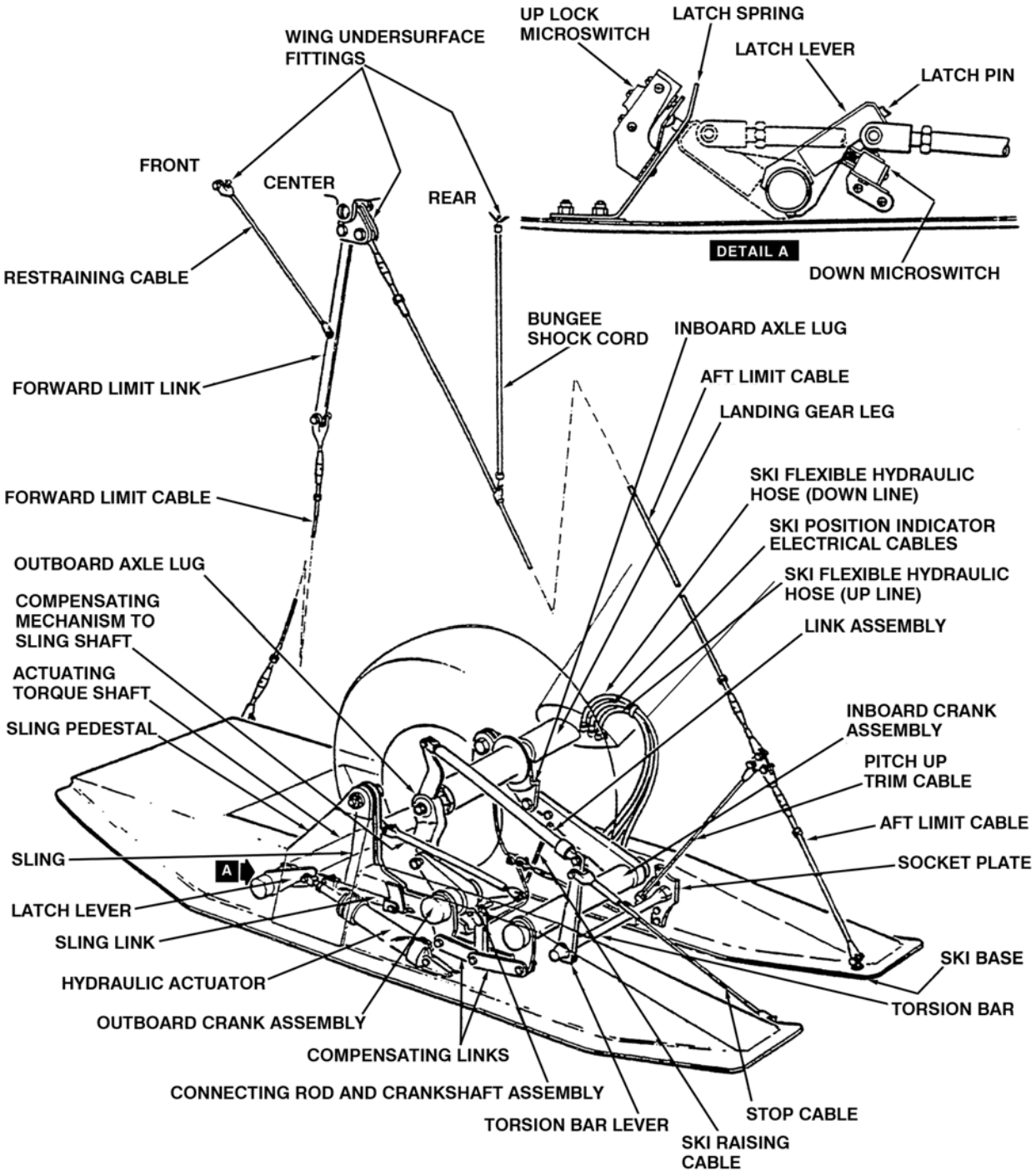
The left and right-hand Main Wheel-Skis form part of the Wheel-Ski installation (refer to 32-00-00 Landing Gear – General Data). Each ski is connected to its appropriate main leg axle ski attachments by its inboard and outboard crank assembly levers, and by cables, link and bungee cord to three fittings on the lower surface of the wing, directly above each ski inboard beam. The skis, controlled by a lever in the flight compartment, are hydraulically actuated from the aircraft's hydraulic system to the "DOWN/SNOW" or "UP/LAND" positions (refer to 32-30-00, Extension and Retraction (Wheel-Ski) (Mod S.O.O. 6001) – General Data). The skis are rigged level in flight and are self-trimming in pitch for either the "DOWN/SNOW" or "UP/LAND" configurations. Both skis embody a device for controlling the lateral attitude of the skis, regardless of the amount of main gear leg deflection, throughout its range of travel from in-flight to maximum landing load conditions. This feature applies only when the skis are in the "DOWN/SNOW" position. A selection to the "UP/LAND" position cancels this feature and introduces a device for re-trimming the lateral attitude for optimum ground clearance during taxiing. An indication of ski position is provided, on the same panel incorporating the Wheel-Ski selector lever.

The Main Wheel-ski base, (Mod 6/2199) consists of a UHMW layer retained by a series of steel retainer strips, screws and fittings. The UHMW base is trimmed out around fittings where tension cables are attached. Fasteners run through the retainer strips bolting the base to the ski. The base material provides a low friction surface on the bottom of the ski and helps protect the skis from damage experienced during landings. The retainer strips are wear components and should be inspected in accordance with PSM 1–6–7 and replaced as required.

2. Description

Refer to Figure 1 for details.

Each main Wheel-Ski consists of two beams structurally joined at the forward end and connected at the rear by the inboard crank assembly tube. A double acting hydraulic actuator is housed in the outboard beam of the ski, and is connected by its cylinder end to the beam structure. The rod end of the actuator connects to a latch lever to operate the ski sling, through actuating torque shaft levers and sling links. A latch pin on the latch lever penetrates the latch to retain the up lock when hydraulic power is lost. Hydraulic lines to the actuator are routed from connections on the inner side of the ski inboard beam, through the beam and the ski front structure. Two pedestals, one on each ski beam, form the attachment and pivot points for the ski sling. The inboard and outboard crank assemblies, which connect to the wheel axle inboard and outboard lugs, are interconnected within the outboard beam by two sets of links. The center of the two sets of links is operated by a connecting rod and crankshaft assembly, which in turn is operated by a shaft connected to the ski sling. This mechanism allows freedom of vertical and pitch motion, but controls the lateral angle of the ski, which is changed from level in the ski down configuration to an anhedral attitude when retracted. A torsion bar, mounted transversely between the beams at the rear of the ski, is secured in a socket plate attached to the inboard ski beam, and pivots in a pivot bearing attached to the outboard ski beam. A lever, secured to the outboard end of the torsion bar is connected to the upper arm of the outboard wheel axle lug by a link assembly, and to an eye bolt at the trailing edge of the outboard ski beam by a pre-tensioning cable. A cable interconnecting the ski sling actuating mechanism and the inboard crank assembly lever, provides the means of raising the ski for a "wheel - landing". A pitch down limit link and cable is connected between an eye bolt at the front of the ski and a center fitting on the undersurface of the wing. Two cables, connected together, are connected between an eye bolt at the rear of the ski, and the center fitting on the undersurface of the wing, to limit the pitch up angle of the ski in flight. A cable interconnecting the aft limit cable is routed through a pulley and fairlead to the ski sling. Two limit microswitches are installed within the outboard ski beam, both operated by the latch lever in its extremes of travel, to provide ski position indication.



Main Wheel-Ski
Figure 1

3. Operation

When skis “DOWN/SNOW” is selected, the actuator retracts, withdrawing the latch-pin from the latch spring, unlocking the latch lever and, through the connecting linkage, moves the sling down under the wheel, in doing so the sling impinges on the tire and forces the wheel to roll over the top of it so moving the ski downward into the extended position. Simultaneously, as the sling moves rearward, the ski raising cable, connected between the inboard crank assembly lever and the sling actuating linkage, is slackened progressively with sling movement. On selecting “UP/LAND”, the actuator extends moving the sling up and forward, allowing the wheel to roll from it, at the same time engaging the latch, simultaneously the actuating linkage operates the ski raising cable, which lifts the ski. Pitch down angle of the ski is restricted by the limit link and cable. The restoring force required to counteract external pitch down loads is provided by the pre-loaded torsion bar, geometrically linked to the landing gear leg, to yield equal forces in both the “UP/LAND” and “DOWN/SNOW” configurations, regardless of landing gear leg deflection. The pitch up angle of the ski in flight is fixed by the cables connecting the aft end of the ski to the center undersurface wing fitting. The restoring force, required to counteract external pitch up loads while taxiing on wheels, or skis, is provided by the spring loaded link connected between the torsion bar lever and the upper arm of the outboard wheel axle lug. To compensate for the change in levels between the “DOWN/SNOW” and “UP/LAND” positions of the ski, the trim cable interconnecting the two aft limit cables and the ski sling takes up slack in the aft limit cables during the transition between positions. The inboard and outboard crank assembly levers, which are of a different length, compensate for landing gear and tire deflection.

4. Servicing

A. Main Wheel-Ski Lubrication

Note The lubrication points, method, and frequency of application are given in the Main Wheel-Ski Lubrication Diagram. (Refer to 12-20-10, Lubrication – General Data, Figure 2).

5. Removal/Installation

Refer to Figure 2 for details.

A. Remove Main Wheel-Skis

Note The following removal procedures are based on the assumption that the aircraft is in the “UP/LAND” configuration.

1) Remove fuselage to landing gear leg fairings (refer to 32-10-11, Main Gear Leg – General Data), and using main inboard jacking points and nose jacking beam (SD12538), jack aircraft until all wheels are at least 8 inches clear of ground (refer to 07-00-00, Lifting and Shoring – General Data).

2) Using a board to protect plastic bottom surface of ski, jack ski under inboard beam, below axle, sufficiently to take weight of ski.

3) Select POWER SOURCE and MASTER switches to OFF (Pre Mod 6/2314); or EXTERNAL and BATTERY switches to OFF (Post Mod 6/2314).

4) Disengage HYDRAULIC OIL PUMP circuit breaker and discharge hydraulic accumulator pressure by operating brake pedals.

5) Disconnect ski hydraulic lines and electrical connector from main gear leg. Blank open hydraulic lines.

6) Slacken and disconnect ski raising cable from inboard crank assembly lever. Disconnect ski pitch up trim cable from inboard side of sling.

7) Lower and remove jack from under ski.

8) Disconnect link assembly from upper arm of the axle outboard lug.

9) Disconnect restraining cable, forward limit link, and aft limit cable and bungee cord, from the three fittings on the undersurface of the wing.

10) Support (or chock) ski and disconnect inboard and outboard crank assembly levers from their respective axle lugs, noting position of washers. Lower ski to ground.

11) Slide ski forward past wheel.

12) Remove outboard lug from axle.

13) Remove main wheel assy as per 32-40-51, Main Wheels (Mod 6/2002) – General Data, Para 2.A.

14) Loosen torque plate bolts, rotate torque plate 60° FWD (2 bolt positions) to standard gear configuration on axle.

15) Remove inboard lug and replace lower portion ski hardware with original gear hardware (103–22100 bolts, Qty 12).

16) Install wheel assy as per 32-40-51, Main Wheels (Mod 6/2002) – General Data, Para 2.B.

17) Repeat Para 5.A.2) through Para 5.A.16) for opposite ski and, if necessary, remove nose Wheel-Ski (refer to 32-21-00, Nose Wheel-Ski (Mod S.O.O. 6001) – General Data).

18) Lower aircraft to ground and remove jacks.

19) Switch on electrical power by selecting POWER SOURCE switch to EXTERNAL or BATTERY and the MASTER switch to ON (Pre Mod 6/2314). Ensure that BUS TIE switch is set to NORMAL.

    Switch on electrical power by selecting EXTERNAL or BATTERY switch to ON (Post Mod 6/2314). Ensure that BUS TIE switch is set to NORMAL.

20) Engage HYDRAULIC OIL PUMP circuit breaker.

    Note Observe that motor pump charges accumulators, and that motor cuts out when pressure reaches 1575 ± 50 psi.

21) Using a suitable container to receive oil spillage, bleed both main Wheel-Ski hydraulic lines, by slackening blanking caps on ski down lines with ski selector lever selected to DOWN/SNOW. When fluid is free of air, tighten blanking cap while fluid is still flowing.

22) Repeat Para 5.A.21) on ski up lines with an UP/LAND selection made.

23) Ensure on completion that all landing gear leg hose blanks are tight and secure.

24) Remove ski selector valve from flight compartment.

25) Install cover plate with mounting screws (4) on instrument panel.

26) Switch OFF electrical supply. Lock out GEAR IND circuit breaker on avionics circuit breaker panel.

27) Install leg fairings and store Wheel-Skis complete with attachment fittings and hardware for future use.

28) Replenish hydraulic system reservoir as necessary (refer to 12-10-15, Hydraulic System – General Data).

29) Change “wheel-ski” airspeed programming instructions with standard gear airspeed program instructions (refer to 32-10-11, Main Gear Leg– General Data, Para 2.C.).

B. Remove Ski Base (Mod 6/2199)

Refer to Figure 2 for details.

1) Remove the Main Wheel-Ski from aircraft as per Para 5.A.

2) Remove the screws securing the skids and skegs to the bottom of ski.

3) Remove skids and skegs.

4) Remove the screws (3) securing the retainer clip to the bottom of ski.

Note It is not necessary to remove doubler, eye bolt or related hardware for the Main Wheel-Ski forward limit cable attachment point.

5) Remove the screws securing the ski base retainers.

6) Remove retainers from ski base.

7) Remove remaining screws securing the ski base to ski.

8) Remove ski base from ski.

C. Preparation of Replacement Ski Base Components

1) Clamp together the old ski base to the replacement ski base.

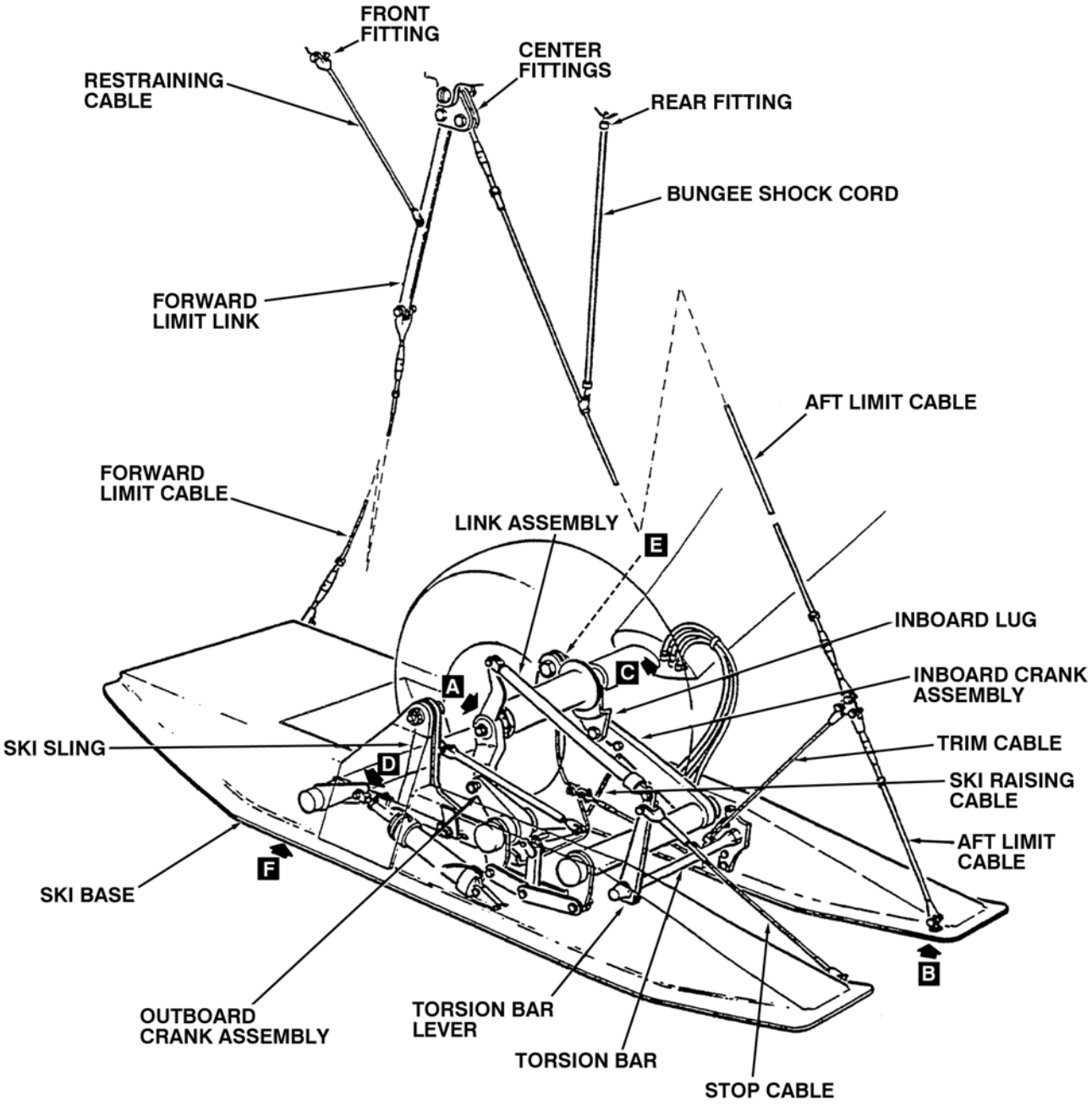
2) Copy profile and hole locations.

3) Transfer hole pattern from ski to replacement ski base. Match hole size and countersunk as required.

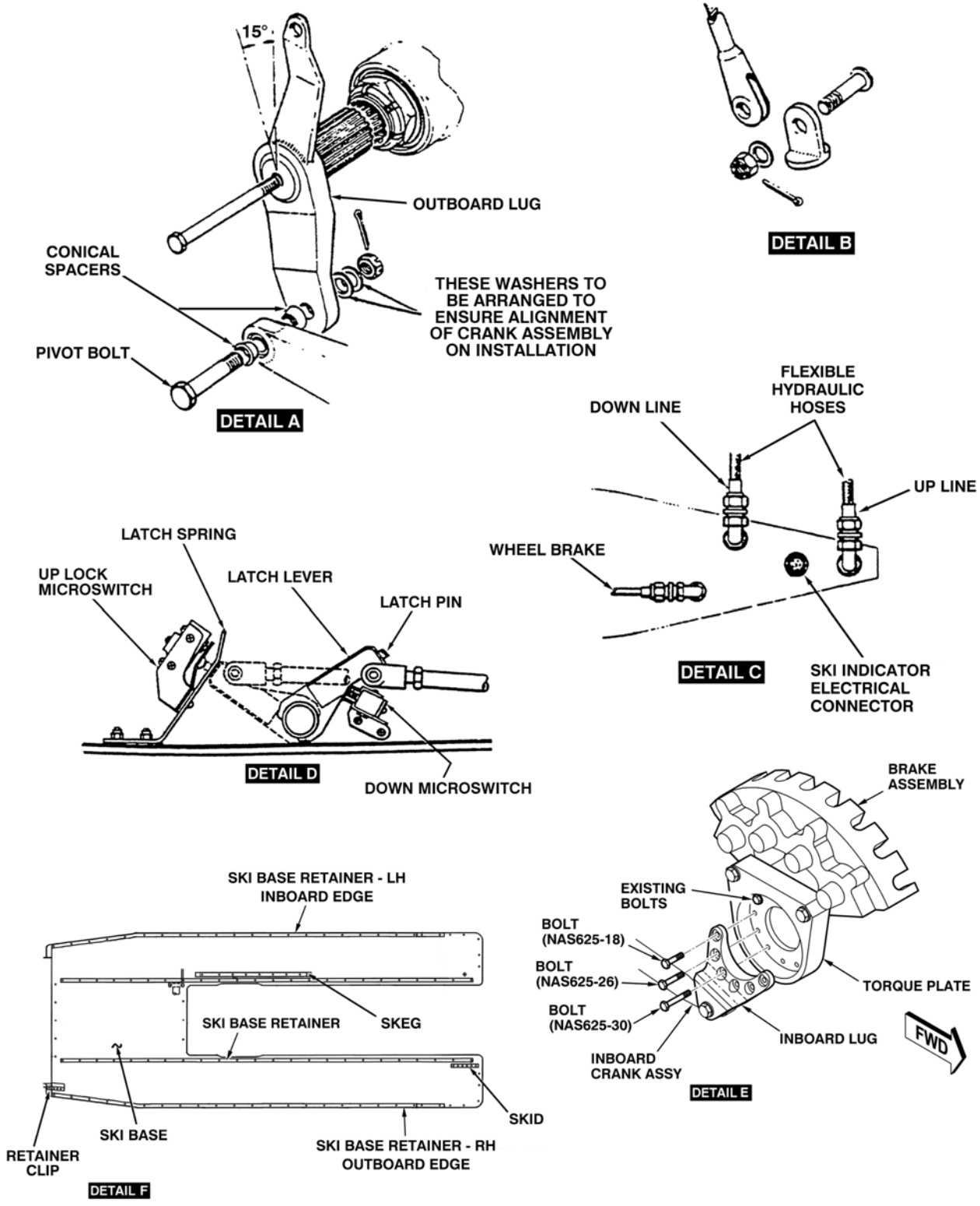
4) Carefully trim replacement base to match old base profile.

5) Using old ski base retainers, bend new retainers to match profile.

6) Clamp retainers securely together and transfer hole patterns. Match hole size and countersunk as required.



Main Wheel-Ski Installation
Figure 2 (Sheet 1 of 2)



Main Wheel-Ski Installation
Figure 2 (Sheet 2 of 2)

D. Install Main Wheel-Skis

Note For the purpose of the following procedures it is assumed that the aircraft has the necessary structural provisions for a Wheel-Ski installation.

1) Select POWER SOURCE and MASTER switches to OFF (Pre Mod 6/2314); or EXTERNAL and BATTERY switches to OFF (Post Mod 6/2314).

2) Disengage HYDRAULIC OIL PUMP circuit breaker and discharge hydraulic accumulator pressure by operating brake pedals.

3) Remove fuselage to landing gear leg fairing (refer to 32-10-11, Main Gear Leg – General Data) and using main inboard jacking points and nose jacking beam (SD12538), jack aircraft until all wheels are at least 8 inches clear of ground (refer to 07-00-00, Lifting and Shoring – General Data). Level aircraft (refer to 08-00-00, Leveling and Weighing – General Data).

4) Remove main wheel assy as per 32-40-51, Main Wheels (Mod 6/2002) – General Data, Para 2.A.

5) Loosen torque plate bolts, rotate torque plate 60° AFT (2 bolt positions) to clear the ski assembly.

6) Install the inboard lug with the supplied hardware.

Note The wheel-ski installation requires the original torque plate hardware for the upper portion of the torque plate and new hardware for the lower portion, refer to Figure 2, Detail E. (Store the original hardware not used).

7) Install wheel assembly as per 32-40-51, Main Wheels (Mod 6/2002) – General Data, Para 2.B., steps 1 to 4.

8) Reinstall brake assembly as per 32-40-31, Wheel Brakes Unit (Mod 6/2002) – General Data, Para 2.B.

9) Install outboard lug. When installing lug, ensure that centerline of lower arm is 15° to the rear of vertical, before engaging splines. Tighten retaining bolt.

10) Inspect entire ski assembly for corrosion and damage. Ensure ski mechanism is in the “UP/LAND” position (sling up). Connect ski raising cable and leave slack. Check ski moving parts for freedom of movement.

11) Lubricate ski (refer to 12-20-10, Lubrication – General Data, Figure 2).

12) Position ski under main wheel, lift and chock ski until inboard and outboard crank assembly levers align with inboard and outboard axle lug arms (lower arm on outboard lug).

13) Install conical spacers and insert pivot bolts to secure ski inboard and outboard crank assembly levers to axle lugs. Observe the position of the conical spacers, and if necessary, add or interchange washers as required to maintain natural alignment and freedom of movement of the crank assembly levers after installation. Install and tighten pivot bolt nuts and secure with cotter pins.

14) Using a board to protect plastic surface of ski bottom, jack ski under inboard beam, below axle to clear chocks and until the rubber pad on the ski beam contacts landing gear leg. Compression of the rubber pad by 0.125 inch maximum is permissible. Remove chocks.

15) Adjust tension of ski raising cable so that ski will remain up when jack is removed. Remove jack supporting ski.

16) Pitching ski up as required, install link assembly between outboard axle lug upper arm and upper hole in torsion bar lever.

17) Pitching ski down against torsion bar load, install stop cable between eye bolt on outboard ski beam and lower hole in torsion bar lever. Adjust cable as required to bring ski approximately level.

18) Install forward limit link to front hole in center fitting on wing undersurface. Connect restraining cable to wing undersurface forward fitting and center of forward limit link. Connect forward limit cable to forward limit link, but do not at this stage connect to ski.

19) Install aft limit cables to rear hole in center fitting on wing undersurface. Do not at this stage connect to ski.

20) Install hydraulic flexible hoses between landing gear leg and ski inboard beam connections. When tightening, position hoses to prevent any possible contact with wheel.

21) Repeat Para 5.D.9) through Para 5.A.20) to install opposite ski.

22) Switch on electrical power by selecting POWER SOURCE switch to EXTERNAL or BATTERY and the MASTER switch to ON (Pre Mod 6/2314). Ensure that BUS TIE switch is set to NORMAL.

    Switch on electrical power by selecting EXTERNAL or BATTERY switch to ON (Post Mod 6/2314). Ensure that BUS TIE switch is set to NORMAL.

23) Engage HYDRAULIC OIL PUMP circuit breaker.

    Note Observe that motor pump charges accumulators and that motor cuts out when pressure reaches 1575 ± 50 psi.

24) Select ski selector to DOWN/SNOW and, using a suitable container, bleed both main ski down hydraulic lines by slackening both down lines, at bottom of the landing gear legs, in turn. When fluid is clear of air, retighten each connection while fluid is flowing. Repeat this procedure on both main ski up lines after making an UP/LAND selection on ski selector.

    Note If nose Wheel-Ski is being installed at same time as main Wheel-Skis, bleed nose Wheel-Ski hydraulic lines when carrying out Para 5.D.24) (refer to 32-21-00, Nose Wheel-Ski (Mod S.O.O. 6001) – General Data).

25) With ski selector still selected at UP/LAND, and hydraulic pressure maintained, adjust the stop cable to bring ski level with aircraft datum.

26) Select ski selector to DOWN/SNOW and check level of ski. If a discrepancy of more than 0.5° from level exists, readjust stop cable to average the two.

27) Connect aft limit cables to eye bolt at rear of ski inboard beam. Adjust turnbuckles to take up slack only.

28) Adjust forward limit cable to allow 7 inches of slack and connect to eye bolt at front of ski.

29) Select ski selector to UP/LAND, connect sling trim cable to link joining aft limit cables. Adjust sling cable to take up slack in aft limit cables. Ensure rigged angle of ski is not affected by this adjustment.

30) Measure 53 inches down aft limit cable from center of cable attachment fork end, at wing undersurface center fitting. Secure clamp to cable, connect bungee cord to clamp and rear fitting at wing undersurface.

31) Lock all cable turnbuckles and attachments.

32) Repeat Para 5.D.25) through Para 5.D.31) for opposite main Wheel-Ski.

33) Connect both ski position indicator electrical cables between connectors on inboard ski beams and landing gear legs. Unlock and engage GEAR IND avionics circuit breaker.

34) Function test main and nose Wheel-Skis (refer to 32-30-00, Extension and Retraction (Wheel-Ski) (Mod S.O.O. 6001) – General Data).

35) Test Wheel-Ski indication system (refer to 32-61-00, Wheel-Ski Indication – General Data).

36) Lower aircraft to ground and remove jacks. Install fuselage to leg fairings.

37) Check hydraulic system fluid level (refer to 12-10-15, Hydraulic System – General Data).

38) Change standard gear airspeed program instructions with “wheel-ski” airspeed programming instructions (refer to 32-10-11, Main Gear Leg – General Data, Para 2.B.).

E. Install Ski Base (Mod 6/2199)

1) Ensure bottom of ski is clean and undamaged.

2) Install ski base to ski and secure using new hardware in countersunk holes on base.

Note Over torquing of screws will damage holes in UHMW ski base.

3) Install retainer clip with new hardware.

4) Install the inner ski base retainers. Start at forward end of ski, use new hardware.

5) Install the outer edge ski base retainers. Start at forward end of ski, use new hardware.

6) Install the skids and skegs with associated screws.

6. Adjustment/Test

Note The access panels form part of the stressed structure of the ski and must be installed as soon as adjustments are completed.

A. Adjust Main Ski Attitude Compensating Mechanism

Note Adjustment of compensating mechanism may be required after component replacement and can be done with ski installed or removed from aircraft.

1) If skis are installed jack up aircraft (refer to 07-00-00, Lifting and Shoring – General Data) to take load off skis.

2) Remove outboard aft access panel.

3) Select ski down position.

4) Adjust shaft connecting compensating mechanism to sling (refer to Figure 1) until compensating links are in line with each other. Lock fork end of shaft.

5) Install access panel and, if applicable, lower aircraft.

B. Adjust Sling Actuating Mechanism

1) Jack up aircraft (refer to 07-00-00, Lifting and Shoring – General Data).

2) Remove forward outboard access panels.

3) Select ski UP/LAND position and observe behavior of ski and latch mechanism.

4) If latch pin does not engage locking hole of latch spring, disconnect and adjust fork end of hydraulic actuator until latch pin engages locking hole (refer to Figure 2).

5) If latch pin engages locking hole of latch spring, release hydraulic pressure and check that clearance between tire and sling is between 0.10 inch and 0.60 inch.

Note Tire clearance is adjusted during manufacture and assembly, and should not be affected appreciably in normal service or by the replacement of related components. Ensure that inadequate tire clearance has not been caused by damage to the latch mechanism or ski bottom in area of latch assembly before attempting tire clearance adjustment.

6) Adjust tire clearance by removing or adding shims under base of latch spring (removing shims increases tire clearance).

Note Adjusting fork end of hydraulic actuator will not alter tire clearance and will only lead to erratic or faulty latch function. If, however, shims have been added or removed, adjustment of hydraulic actuator is also required as in Para 6.B.7).

7) Adjust fork end of hydraulic actuator to give 0.015 inch to 0.060 inch overtravel after latch engagement.

8) Check adjustment of Ski Up and Ski Down limit switches. (Refer to 32-61-00, Wheel-Ski Indication – General Data).

9) Reinstall access panels.

10) Lower aircraft.

7. Inspection/Check

A. Check Main Wheel-Ski Rigging

1) Remove fuselage to leg fairings (refer to 32-10-11, Main Gear Leg – General Data), jack aircraft well clear of ground using inboard jacking points and nose jacking beam (SD12538) (refer to 07-00-00, Lifting and Shoring – General Data). Level aircraft (refer to 08-00-00 Leveling and Weighing – General Data).

2) Check tire inflation pressure (refer to 12-10-20, Landing Gear – General Data).

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4) Engage GEAR IND and HYDRAULIC OIL PUMP circuit breakers.

Note Observe that motor pump charges accumulators and that motor cuts out when pressure reaches 1575 ± 50 psi.

5) Operate skis UP/LAND and DOWN/SNOW several times, and check ski mechanical linkage for freedom of movement.

6) Select ski selector lever to UP/LAND and check that main skis are level with aircraft datum and that rubber pad on ski contacts landing gear leg.

Note If ski does not contact leg, adjust ski raising cable (refer to Para 5.D).

7) Select ski selector lever to DOWN/SNOW and check main skis are level with aircraft datum.

Note If a discrepancy of more than 1/2° from level exists as checked in Para 7.A.6) and Para 7.A.7), readjust the stop cable to average the two.

8) Check aft limit cables are adjusted so that only cable slack is taken up.

9) Check forward limit cable has 7 inches of slack to accommodate additional ski down travel.

10) Check that sling trim cables are adjusted to take up slack in aft limit cables. Ensure rigged angle of ski is not affected by this adjustment.

11) Check all cable turnbuckles are in safety and that they are lock pinned or wire locked.

12) Lower aircraft to ground and remove jacking equipment. Install fuselage to leg fairings.

13) Check hydraulic system fluid level (refer to 12-10-15, Hydraulic System – General Data).

14) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

Nose Gear – General Data

1. General

The nose gear consists of a non retractable, pneudraulic shock strut mounted on the forward face of the nose compartment bulkhead, and a single wheel supported on an axle within the fork of the shock strut. The shock strut is equipped with a hydraulically operated steering mechanism to steer the nose wheel. When the torque links are disconnected (as for towing) the nose wheel can be rotated 360°. A detachable cover is provided in the nose compartment to prevent loose objects obstructing the steering actuator, which is mounted on the shock strut.

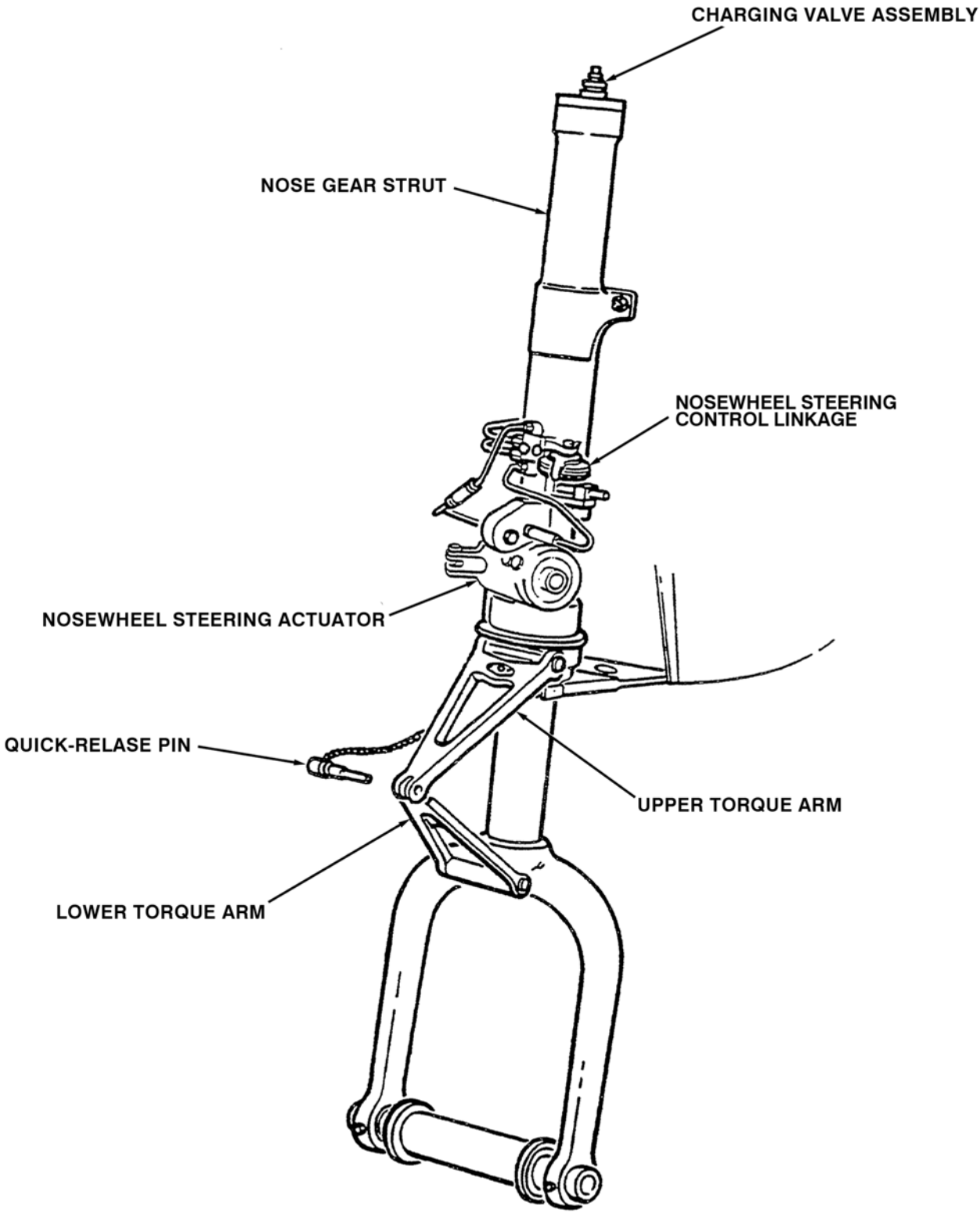
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Nose Gear Shock Strut – General Data

1. General

Refer to Figure 1 for details.

The nose gear shock strut consists of an outer cylinder, piston assembly, nose wheel fork and a steering collar. The cylinder forms the main structure of the strut and provides the mounting for the nose wheel steering actuator (refer to 32-50-00, Nose Wheel Steering System – General Data). The nose wheel fork is secured to the piston and telescopes within the cylinder which is charged with oil and air, damping being achieved by restricting the oil and air flow through orifices in the piston. A valve, through which the strut may be recharged, is located on top of the strut.



Nose Gear Shock Strut
Figure 1

2. Servicing

A. Charge Nose Gear Shock Strut

The nose gear strut must be serviced in accordance with the instructions on the nose gear identification plate.

Note Dry compressed nitrogen is preferred to compressed air for charging the shock strut.

3. Removal/Installation

Refer to Figure 2 for details.

A. Remove Nose Gear Strut

CAUTION ENSURE ALL AIR PRESSURE IS RELEASED BEFORE REMOVING NOSE WHEEL FORK FROM NOSE GEAR SHOCK STRUT.

1) Open nose baggage compartment. Remove screens and covers as required.

2) Jack aircraft until nose wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data).

3) Discharge hydraulic system accumulator pressure by operating wing flaps.

4) Disconnect hydraulic lines at swivel fittings. Cap open ports and lines.

5) Disconnect nose wheel steering cables at turnbuckles.

6) Support strut and remove two lower attachment bolts and cap fitting. Remove two upper attachment bolts.

B. Install Nose Gear Strut

1) Position nose gear strut on forward face of nose compartment bulkhead and install two upper attachment bolts. Install cap fitting and two lower attachment bolts.

2) Torque strut attachment bolts to 250 to 300 pound–inches, and ensure attachment bolts have 0.02 inch minimum clearance between end of bolt and side of hole. Ensure cap fitting has 0.03 inch minimum gap on each side. Lockwire bolts.

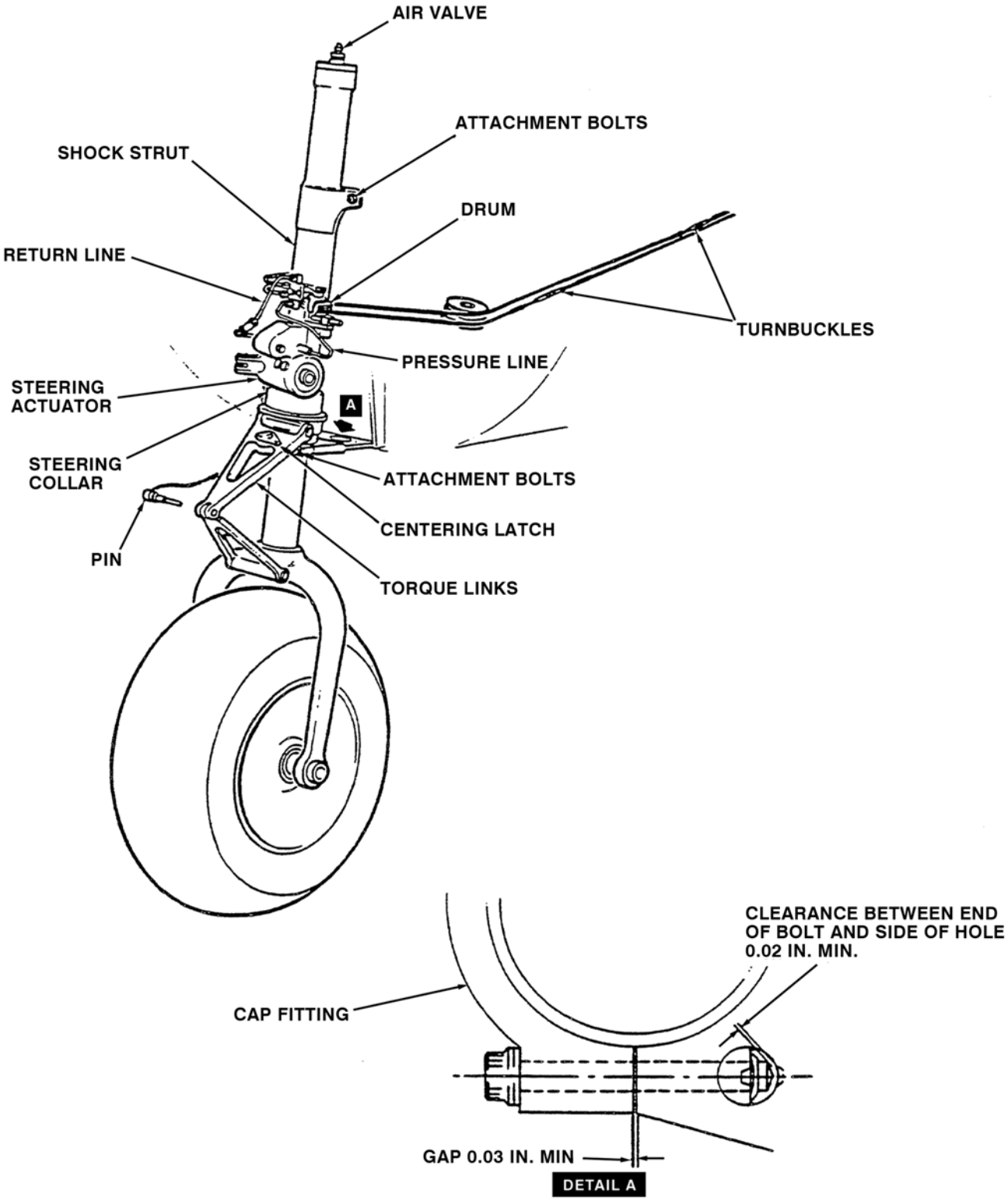
3) Connect nose wheel steering cables at turnbuckles.

4) Connect hydraulic lines at swivel fittings.

5) Bleed nose wheel steering system. (Refer to 32-50-00, Nose Wheel Steering System – General Data).

6) Rig and function test nose wheel steering system. (Refer to 32-50-00, Nose Wheel Steering System – General Data).

7) Install screens and covers as necessary. Close nose baggage compartment.



Nose Gear Shock Strut Installation
Figure 2

4. Adjustment/Test

A. Pressure Test

CAUTION POUR HYDRAULIC FLUID. DO NOT FILL WITH HYDRAULIC FLUID UNDER PRESSURE.

1) Jack nose wheel clear of ground. Remove valve cap and release air slowly.

2) Remove air charging valve and slowly compress strut fully to expel hydraulic fluid.

Note If no hydraulic fluid is expelled, the strut is almost empty. Add approximately 10 fluid ounces (300 cc) of hydraulic fluid, Specification MIL–H–5606 and repeat Para 4.A.2).

3) Extend strut fully with valve open to permit air to enter strut. Ensure ground clearance with strut fully extended. Compress strut to expel additional oil.

4) Repeat Para 4.A.3) five times until no additional oil is expelled.

5) Add 25 fluid ounces (725 cc) of hydraulic fluid, Specification MIL–H–5606, while slowly extending strut.

6) Install pressure gauge, and with strut fully extended, inflate with dry compressed nitrogen or dry compressed air to 95 psi. There must be no pressure loss after 48 hours.

7) Ensure there is no sign of fluid leakage at upper or lower ends of cylinder, or at joint of piston and fork.

8) Remove pressure gauge and install valve cap. Lower aircraft to ground.

5. Inspection/Check

A. Wear Check-Nose Gear Torque Links

1) Remove quick-release pin and separate upper and lower torque links.

2) Move upper torque link so that one lug is tight against face of steering collar; using feeler gauges, check gap between other lug and face on steering collar. Gap must not exceed 0.008 inch.

3) Repeat Para 5.A.2) and check for wear on lower torque link.

4) Check for wear between lug of lower torque link and two lugs of upper torque link. Gap must not exceed 0.008 inch.

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Steering Collar (with Mods S.O.O. 6001 and 6048) (Pre Mod 6/2331) – Maintenance Practices

1. Adjustment/Test

Refer to Figure 1 for details.

Note The following adjustment procedures are only applicable to nose legs incorporating shimmy damper provisions for use with wheel-skis or intermediate flotation gear.

A. Adjust Steering Collar Torque Loading

1) With the nose wheel jacked clear of ground (refer to 07-00-00, Lifting and Shoring – General Data), disconnect steering actuator from steering collar and remove pin from torque links. Slacken locking clip and remove key.

2) If new friction disc has been installed, wear in and seat disc as follows:

a) Torque preload nut until torque required to turn steering collar is 800 to 1000 pound–inches (70 to 90 pounds to move steering collar with spring balance attached to end of torque link).

b) Operate steering collar under load for a minimum of ten cycles.

Note Steering collar may be operated manually or, if hydraulic power is available, by connecting and operating steering actuator. Disconnect actuator, if used, after operation.

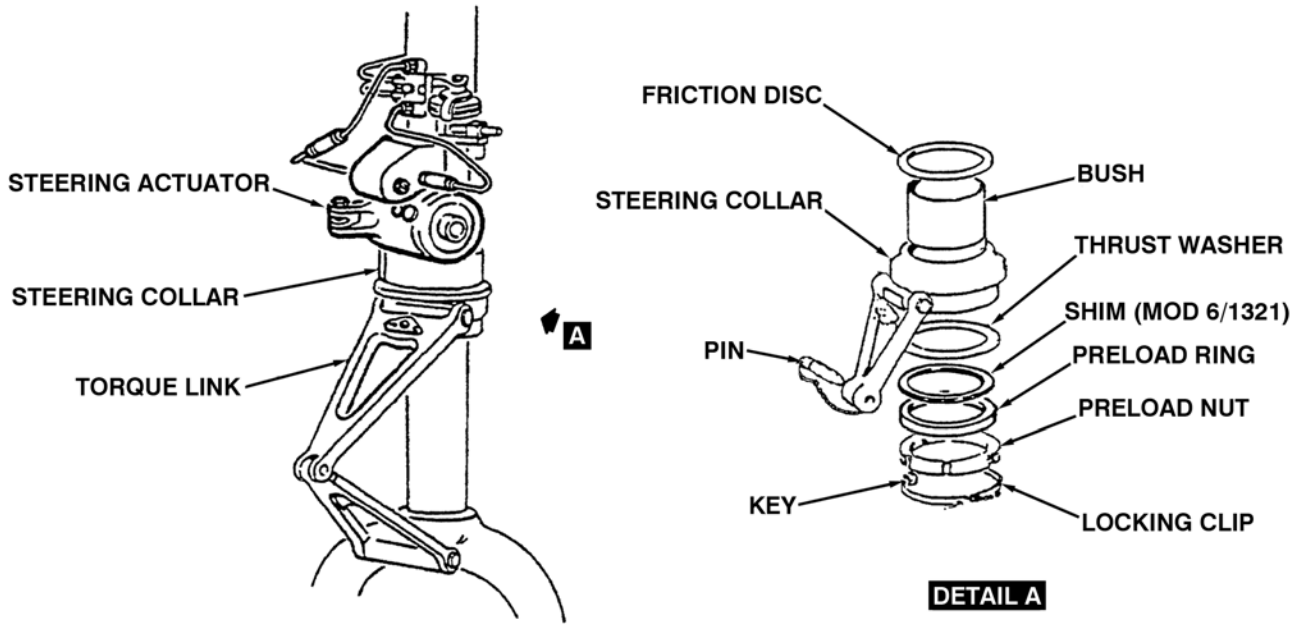
c) Back off torque on preload nut until steering collar rotates freely.

d) Torque preload nut to 30 pounds–feet, then back off to align with slots, ensuring steering collar is snug but free to rotate (approximately 8 pounds to move steering collar with a spring balance attached to end of torque link).

3) When skis or intermediate flotation gear are installed, torque preload nut until shimmy damper torque is 800 to 1000 pound–inches (70 to 90 pounds to move steering collar with a spring balance attached to end of torque link).

4) If skis or intermediate flotation gear are not fitted, or have been removed, back off torque on preload nut until steering collar rotates freely. Torque preload nut to 30 pounds–feet, then back off to align slots, ensuring steering collar is snug but free to rotate (approximately 8 pounds to move steering collar with a spring balance attached to end of torque link).

5) Install key and secure with locking clip. Reconnect steering actuator to steering collar and install pin in torque links. Lower aircraft to ground.



NOTE: REFER TO PSM 1-6-6A FOR ASSEMBLY CONFIGURATION

Steering Collar Torque Adjustment
Figure 1

Steering Collar (with Mods S.O.O. 6001 and 6048) (Post Mod 6/2331) – Maintenance Practices

1. Adjustment/Test

Refer to Figure 1 for details.

Note The following adjustment procedures are only applicable to nose legs incorporating shimmy damper provisions for use with wheel-skis or intermediate flotation gear. If wheel-skis or intermediate flotation gear are not fitted. Refer to 32-20-34, Steering Collar – Maintenance Practices.

A. Adjust Steering Collar Torque Loading

1) With the nose wheel jacked clear of ground (refer to 07-00-00, Lifting and Shoring – General Data), disconnect steering actuator from steering collar and remove pin from torque links. Slacken locking clip and remove key.

2) If new friction disc has been installed, wear in and seat disc as follows:

a) Torque ring nut until torque required to turn steering collar is 800 to 1000 pound–inches (70 to 90 pounds to move steering collar with spring balance attached to end of torque link).

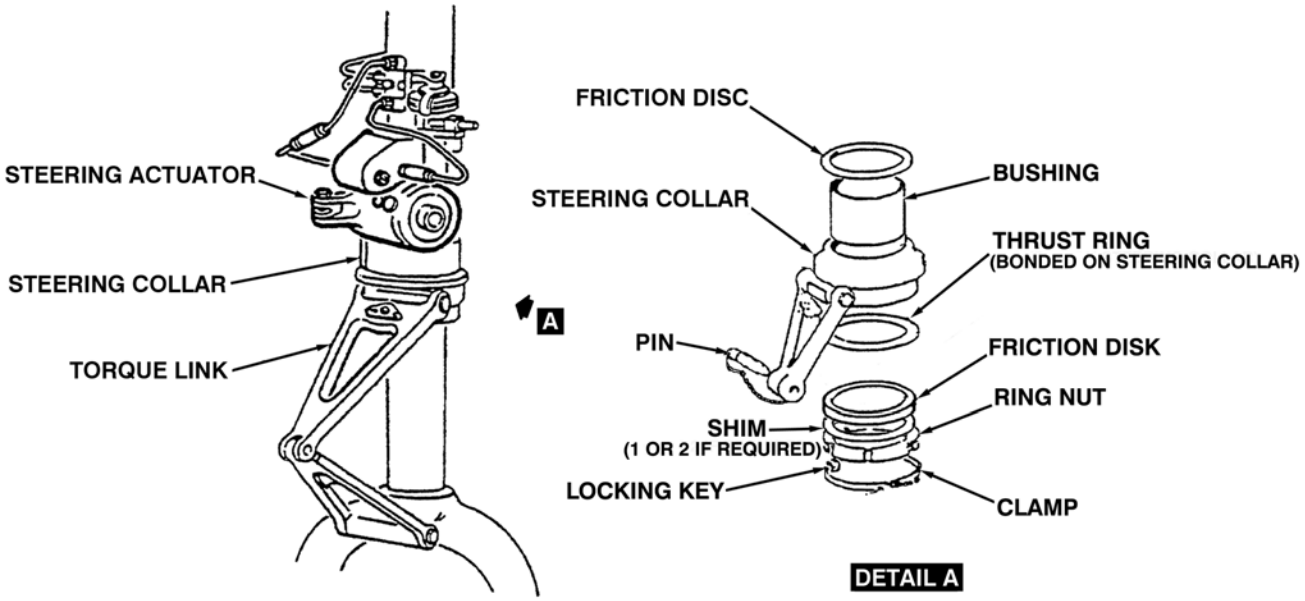
b) Operate steering collar under load for a minimum of ten cycles.

Note Steering collar may be operated manually or, if hydraulic power is available, by connecting and operating steering actuator. Disconnect actuator, if used, after operation.

c) Back off torque on ring nut until steering collar rotates freely. Proceed to Para 1.A.3).

3) Torque ring nut to 800 to 1000 pound–inches, to align with slots, ensuring steering collar is snug but free to rotate. If required, add one or two shims between the friction disk and the ring nut to achieve torque preload (70 to 90 pounds to move steering collar with a spring balance attached to end of torque link).

4) Install locking key and secure with clamp. Reconnect steering actuator to steering collar and install pin in torque links. Lower aircraft to ground.



NOTE: REFER TO PSM 1-6-6A FOR ASSEMBLY CONFIGURATION

Steering Collar Torque Adjustment
Figure 1

Nose Wheel Fork – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

Equipment Required:

- Locknut Tool Adapter T–187 (Heroux Machine Parts Ltd.) (for use with 3/4–inch square drive)

A. Remove Nose Wheel Fork

1) Jack aircraft until nose wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data).

2) Release air pressure from shock strut.

WARNING ENSURE ALL AIR PRESSURE IS RELEASED BEFORE REMOVING NOSE WHEEL FORK FROM NOSE GEAR SHOCK STRUT.

3) Remove nose wheel. (Refer to 32-40-61, Nose Wheel – General Data).

4) Remove nut, washer and bolt securing lower torque arm to fork and retain with torque arm for installation on replacement fork. On forks used with intermediate flotation tires retain the two additional Teflon washers.

5) If applicable, remove taxi light from fork.

6) Remove nut, washer and lockbolt.

7) Unscrew locknut, using special tool T–187, and remove locknut with washer and shim. Remove diaphragm and remove fork from piston tube.

B. Install Nose Wheel Fork

1) Position piston tube in fork, aligning slot with holes in bushings. Ensure that cartridge assembly is installed within piston tube and insert diaphragm. Lubricate surface of fork to be contacted by locknut with machine oil and install shim, washer and locknut to piston tube.

Note To maintain alignment between piston tube and bushings during following locknut tightening procedure, it is recommended that a headed pin, 0.25 inch diameter and approximately 0.625 inch in length be inserted through bushings in fork and slot in piston tube.

2) Using special tool T–187, torque locknut to 400 pound–feet. If the elongated hole in the locknut is visible through the bushing, increase torque until holes in locknut and bushing align.

Note If hole in locknut is not visible through bushing after initial torque loading, reduce thickness of shim as necessary and repeat tightening procedure.

3) Remove alignment pin (refer to Para 1.B.1)) if used.

4) Coat bores of bushings, lockbolt and washer with Chromate Epoxy Primer to MIL–PRF–23377 and install in fork while primer is still wet. Lock nut with cotter pin.

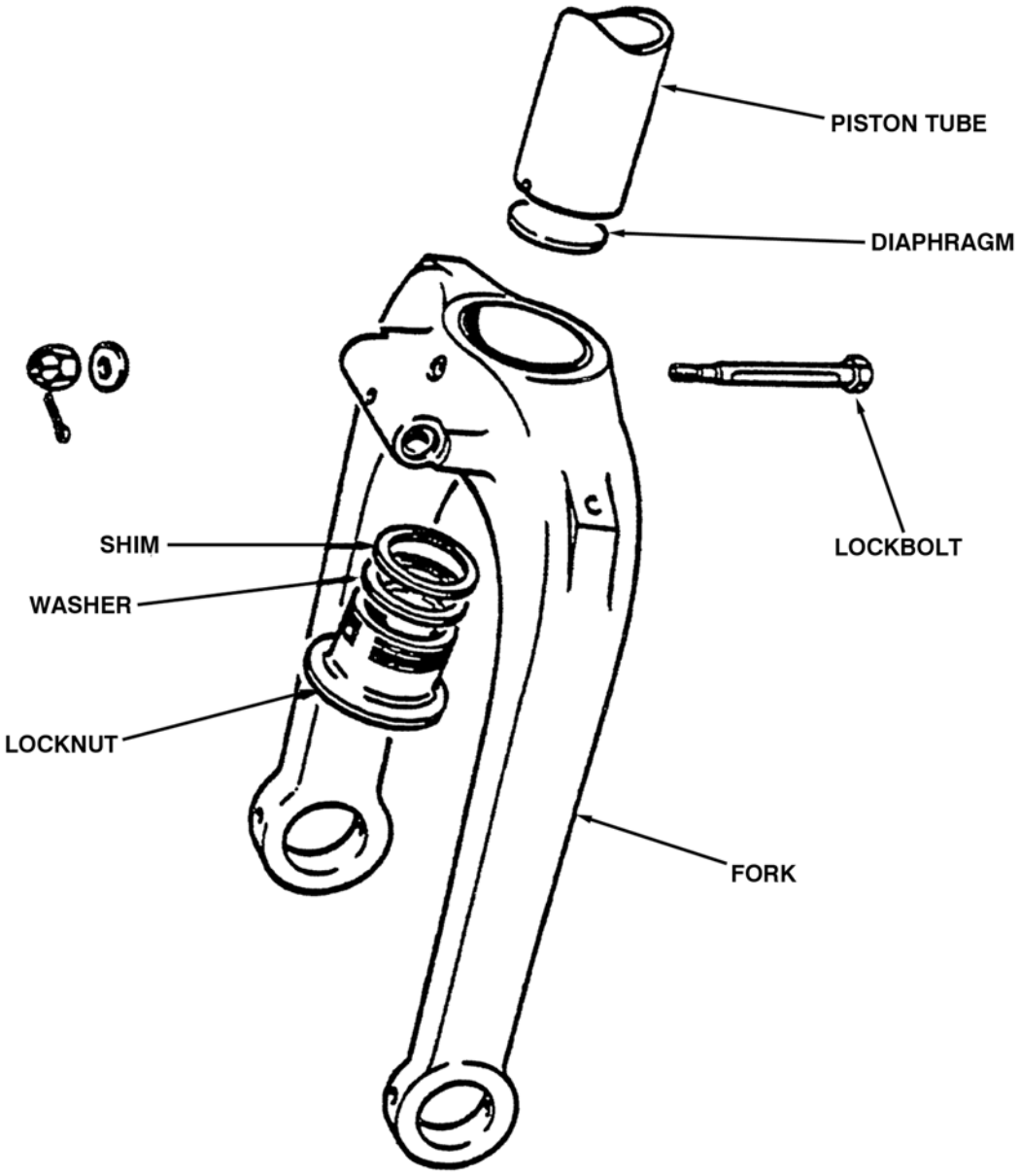
5) If applicable, install taxi light on fork.

6) Attach lower torque arm to fork with bolt, washer and nut. On fork used with intermediate flotation tires, install two Teflon washers, one at each lower torque arm lug, between torque arm and fork. Finger tighten nut, tighten to next slot and install cotter pin. Check torque arm for freedom of movement.

7) Service shock strut. (Refer to 12-10-20, Landing Gear – General Data).

8) Install nose wheel. (Refer to 32-40-61, Nose Wheel – General Data).

9) Lower aircraft to ground.



Nose Wheel Fork – Removal and Installation
Figure 1

Nose Wheel Fork (Mod 6/1618 and Mod S.O.O. 6048) – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

Equipment Required:

- Locknut Tool Adapter T–187 (Heroux Machine Parts Ltd) (for use with 3/4–inch square drive)

A. Remove Nose Wheel Fork

1) Jack aircraft until nose wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data).

2) Release air pressure from shock strut.

WARNING ENSURE ALL AIR PRESSURE IS RELEASED BEFORE REMOVING NOSE WHEEL FORK FROM NOSE GEAR SHOCK STRUT.

3) Remove nose wheel. Refer to 32-40-61, Nose Wheel – General Data or 32-40-62, Nose Wheel (Mod S.O.O. 6048 and Mod 6/2183) – Maintenance Practices).

4) Remove nut, washer and bolt securing lower torque arm to fork and retain with torque arm for installation on replacement fork. On forks used with intermediate flotation tires, retain the two additional Teflon washers.

5) If applicable, remove taxi light from fork.

6) Remove nuts, washers, profile washers and lockbolts securing locknut.

7) Unscrew locknut, using special tool T–187, and remove locknut with washer and shim. Remove diaphragm and remove fork from piston tube.

B. Install Nose Wheel Fork

1) Position piston tube in fork, aligning slot with holes in bushings. Ensure that cartridge assembly is installed within piston tube and insert diaphragm. Lubricate surface of fork to be contacted by locknut with machine oil and install shim, washer and locknut to piston tube.

Note To maintain alignment between piston tube and bushings during following locknut tightening procedure, it is recommended that a headed pin, 0.25 inch diameter and approximately 0.625 inch in length be inserted through a bushing in fork and slot in piston tube.

2) Using special tool T–187, torque locknut to 400 pounds–feet. Align two holes (180° apart) in locknut with holes in bushings in fork.

3) Remove alignment pin (refer to Para 1.B.1), if used.

4) Coat bores of bushings, lockbolts and profile washers with Chromate Epoxy Primer to MIL–PRF–23377 and install in fork while primer is still wet. Install washer and self-locking nut on each lockbolt.

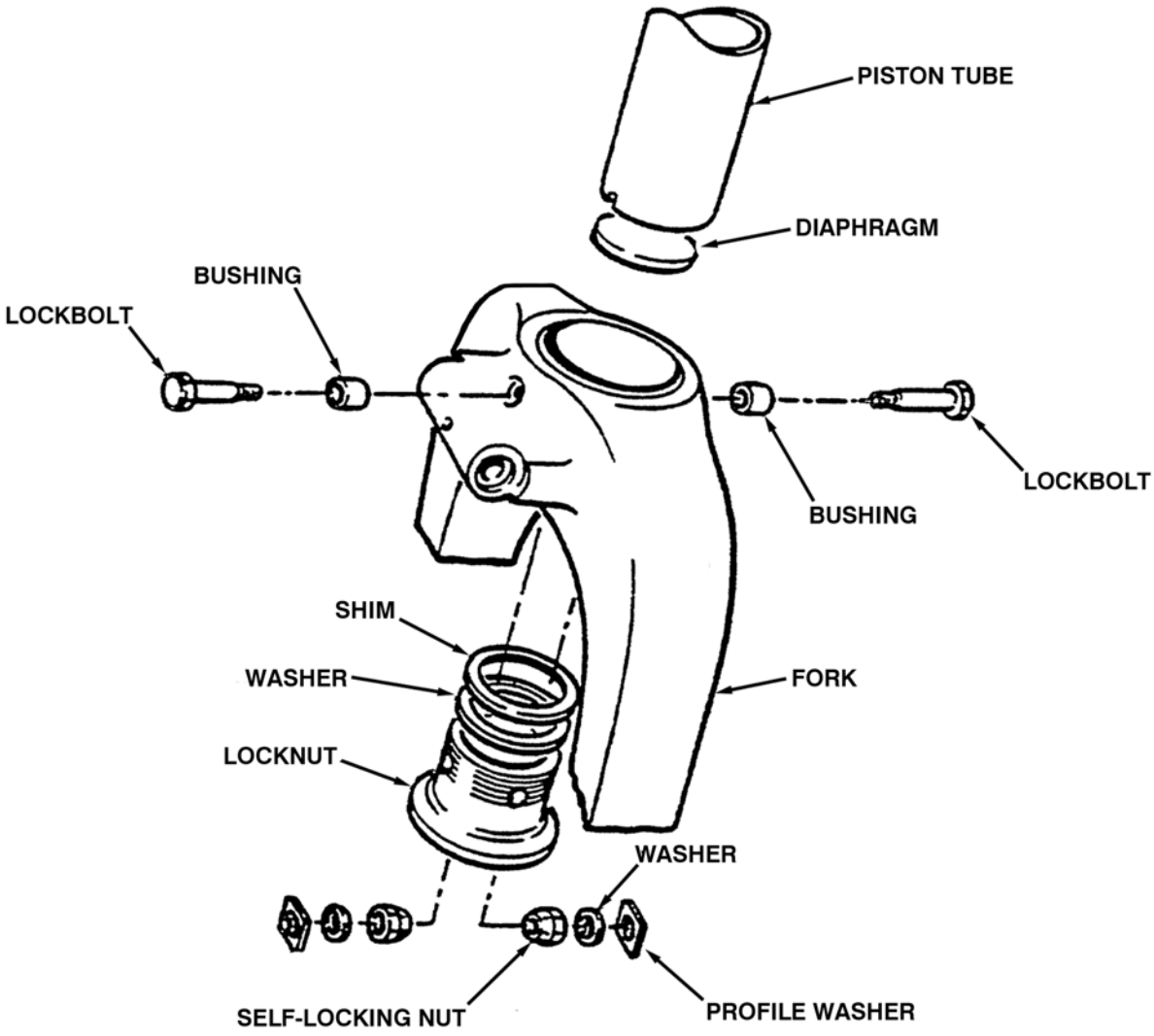
5) If applicable, install taxi light on fork.

6) Attach lower torque arm to fork with bolt, washer and nut. On fork used with intermediate flotation tires (Mod S.O.O. 6048), install two Teflon washers, one at each lower torque arm lug, between torque arm and fork. Finger tighten nut, tighten to next slot and install cotter pin. Check torque arm for freedom of movement.

7) Service shock strut. (Refer to 07-00-00, Lifting and Shoring – General Data).

8) Install nose wheel. Refer to 32-40-61, Nose Wheel – General Data or 32-40-62, Nose Wheel (Mod S.O.O. 6048 andMod 6/2183) – Maintenance Practices.

9) Lower aircraft to ground.



Nose Wheel Fork - Removal and Installation (Mod S.O.O. 6048)
Figure 1

Steering Collar (Pre Mod 6/2331) – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Steering Collar

WARNING BEFORE DOING ANY MAINTENANCE TO, IN, OR AROUND THE MAIN LANDING GEAR AND/OR THE NOSE LANDING GEAR, REFER TO ALL SAFETY PRECAUTIONS.

1) Jack aircraft until nose wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data, Para 9.).

2) Remove cover on underside of fuselage at nose leg and retain hardware.

3) Place container under nose landing gear to receive spillage.

WARNING ENSURE ALL NITROGEN PRESSURE IS RELEASED BEFORE REMOVING NOSE WHEEL FORK FROM NOSE GEAR SHOCK STRUT. FAILURE TO DO SO COULD CAUSE SERIOUS INJURY.

4) Remove nose wheel. (Refer to 32-40-61, Nose Wheel – General Data, Para 2.A).

5) Remove nose wheel fork. (Refer to 32-20-32, Nose Wheel Fork – Maintenance Practices, Para 1.A).

6) Remove screws securing cover.

7) Remove spring, cam, cotter pin, nut, washer and bolt securing upper torque arm to steering collar. Retain hardware.

8) Remove nuts, bolts and washers on cap fitting. Retain hardware.

9) Remove nut, washer, contour block, screw and clamp. Retain hardware.

10) Remove ring nut, locking key, wear washer, thrust ring and bushing from steering collar. Retain hardware.

11) Remove cotter pin, nut, and washer securing steering lever to steering collar.

12) Remove steering collar.

B. Install Steering Collar

1) Install bushing in the steering collar and install in the shock strut cylinder.

2) Align the stud on the steering collar with actuator steering lever.

3) Install the thrust ring, wear washer and the ring nut on the shock strut cylinder and tighten ring nut finger tight.

4) Install washer and nut on the steering collar stud and the steering actuator lever. Torque nut to 240 pound–inches and advance nut to the nearest hole and install new cotter pin.

5) Install the upper torque arm to the steering collar with torque link bolt, spring, cam, washer and nut. Tighten nut finger tight and advance to nearest hole. Install a new cotter pin.

6) Install cover with screws.

7) Torque the ring nut to 360 pound–inches and back off to the nearest matching slots. Install the locking key.

8) The steering collar must be snug but free to rotate. The steering collar should take approximately 8 pounds to move with a spring balance at the end of the extended upper torque arm.

9) Install clamp with screw, contour block, washer and nut.

10) Install cap fitting with bolts, washers and nuts. For gap settings and torquing instructions refer to 32-20-11, Nose Gear Shock Strut – General Data, Para 3.

11) Install nose wheel fork. (Refer to 32-20-32, Nose Wheel Fork – Maintenance Practices, Para 1.B).

12) Install nose wheel. (Refer to 32-40-61, Nose Wheel – General Data, Para 2.B).

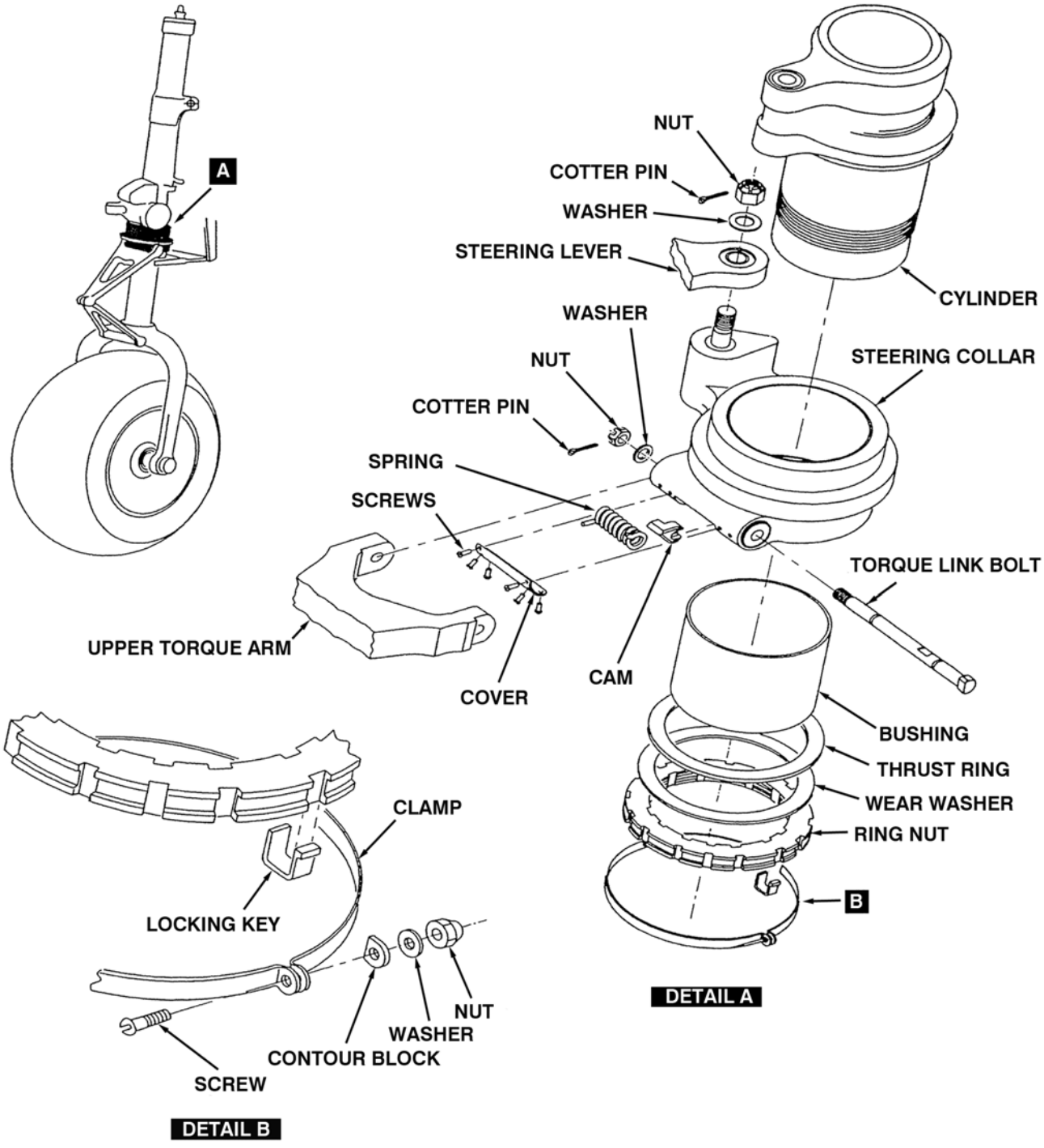
13) Charge the nose gear shock strut. (Refer to 32-20-11, Nose Gear Shock Strut – General Data, Para 2).

14) Install cover on underside of fuselage at nose leg.

15) Lower aircraft to ground.

16) Install screens and covers as necessary. Close nose baggage compartment.

17) Function test nose wheel steering. (Refer to 32-50-00, Nose Wheel Steering System – General Data, Para 3.B).



Steering Collar – Removal/Installation
Figure 1

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Steering Collar (Post Mod 6/2331) – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Steering Collar

WARNING BEFORE DOING ANY MAINTENANCE TO, IN, OR AROUND THE MAIN LANDING GEAR AND/OR THE NOSE LANDING GEAR, REFER TO ALL SAFETY PRECAUTIONS.

1) Jack aircraft until nose wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data, Para 9.).

2) Remove cover on underside of fuselage at nose leg and retain hardware.

3) Place container under nose landing gear to receive spillage.

WARNING ENSURE ALL NITROGEN PRESSURE IS RELEASED BEFORE REMOVING NOSE WHEEL FORK FROM NOSE GEAR SHOCK STRUT. FAILURE TO DO SO COULD CAUSE SERIOUS INJURY.

4) Remove nose wheel. (Refer to 32-40-61, Nose Wheel – General Data, Para 2.A).

5) Remove nose wheel fork. (Refer to 32-20-32, Nose Wheel Fork – Maintenance Practices, Para 1.A).

6) Remove screws securing cover.

7) Remove spring, cam, cotter pin, nut, washer and bolt securing upper torque arm to steering collar. Retain hardware.

8) Remove nuts, bolts and washers on cap fitting. Retain hardware.

9) Remove nut, washer, contour block, screw and clamp. Retain hardware.

10) Remove clamp, locking key, ring nut, shims (if installed), and friction disk from shock strut cylinder. Retain hardware.

11) Remove cotter pin, nut, and washer securing steering lever to steering collar.

12) Remove steering collar and bushing.

13) Remove bushing from steering collar.

B. Install Steering Collar

1) Install bushing in the steering collar and install on the shock strut cylinder.

2) Align the stud on the steering collar with actuator steering lever.

3) Install the friction disk and the ring nut on the shock strut cylinder and tighten ring nut finger tight.

4) Install washer and nut on the steering collar stud and the steering actuator lever. Torque nut to 240 pound–inches and advance nut to the nearest hole and install new cotter pin.

5) Install the upper torque arm to the steering collar with torque link bolt, spring, cam, washer and nut. Tighten nut finger tight and advance to nearest hole. Install a new cotter pin.

6) Install cover with screws.

7) Torque the ring nut to 360 pound–inches and back off to the nearest matching slots.

8) The steering collar must be snug but free to rotate. The steering collar should require a minimum of 8 pounds to move with a spring balance at the end of the extended upper torque arm.

   Note If required, add one or two shims between the friction disk and the ring nut to achieve torque preload and repeat Para 1.B.7) and Para 1.B.8).

9) Install locking key and secure the clamp with screw, contour block, washer and nut.

10) Install cap fitting with bolts, washers and nuts. For gap settings and torquing instructions refer to 32-20-11, Nose Gear Shock Strut – General Data, Para 3.

11) Install nose wheel fork. (Refer to 32-20-32, Nose Wheel Fork – Maintenance Practices, Para 1.B).

12) Install nose wheel. (Refer to 32-40-61, Nose Wheel – General Data, Para 2.B).

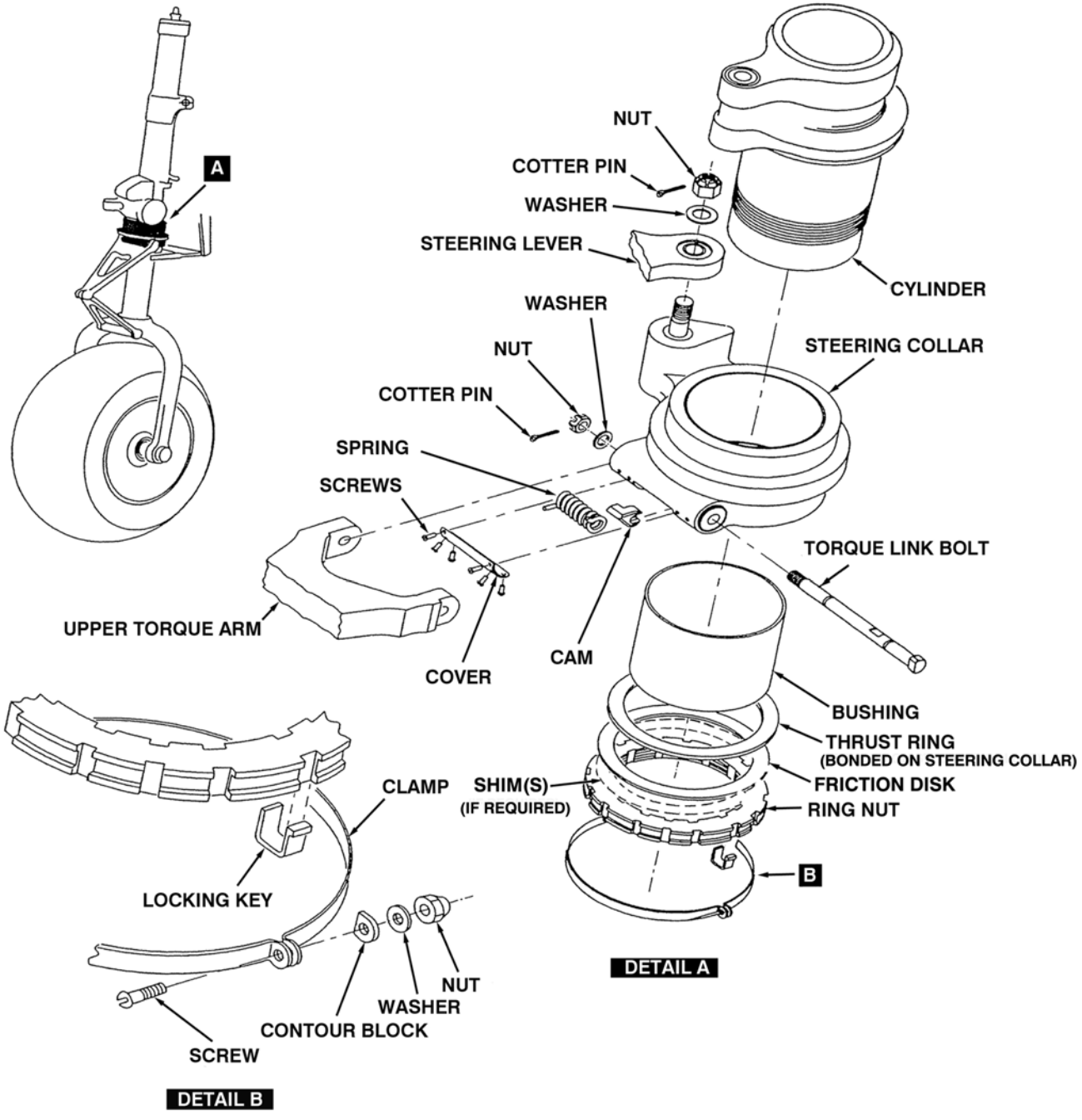
13) Charge the nose gear shock strut. (Refer to 32-20-11, Nose Gear Shock Strut – General Data, Para 2).

14) Install cover on underside of fuselage at nose leg.

15) Lower aircraft to ground.

16) Install screens and covers as necessary. Close nose baggage compartment.

17) Function test nose wheel steering. (Refer to 32-50-00, Nose Wheel Steering System – General Data, Para 3.B).



Steering Collar – Removal/Installation
Figure 1

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Nose Wheel-Ski (Mod S.O.O. 6001) – General data

1. General

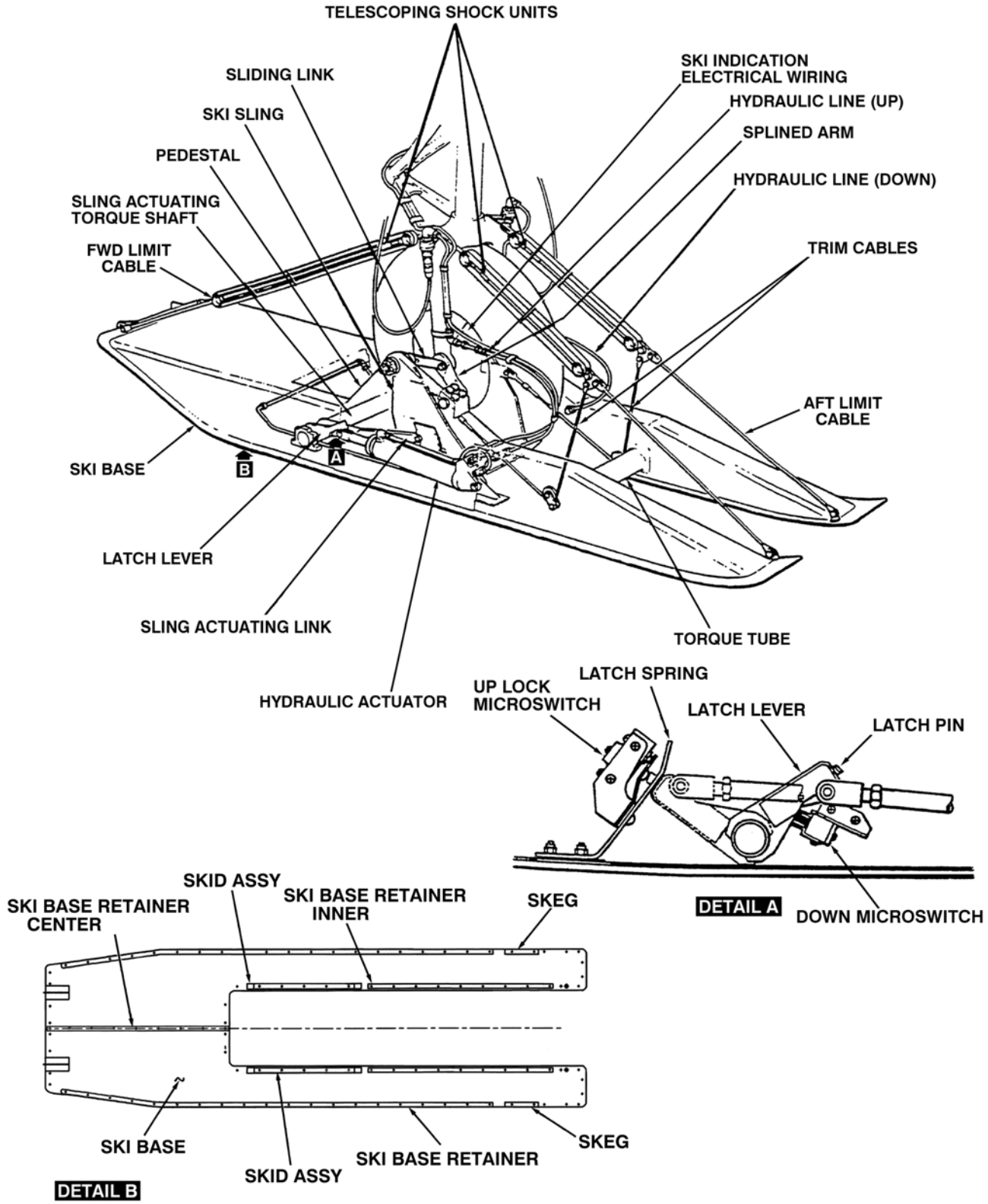
The Nose Wheel-Ski forms part of the Wheel-Ski installation (refer to 32-00-00, Landing Gear – General Data). The ski is attached to the nose wheel axle by two splined arms, and to the nose wheel fork by two pairs of telescoping shock units. The ski, which is controlled by the same lever as the Main Wheel-Skis, is hydraulically actuated from the aircraft's hydraulic system to the “DOWN/SNOW” or “UP/LAND” position (refer to 32-30-00, Extension and Retraction (Wheel-Ski) (Mod S.O.O. 6001) – General data). The ski is rigged level in flight and is self trimming in pitch for either a ski or wheel landing. An indicator, which shows the ski position, is provided on the ski selector panel in the flight compartment. When installing the nose ski the torque loading on the shimmy damper at the steering collar must be increased, and when the skis are removed the torque must be reduced (refer to 32-20-34, Steering Collar – Maintenance Practices or 32-20-21, Steering Collar (with Mods S.O.O. 6001 and 6048) – Maintenance Practices).

The Nose Wheel-ski base, (Mod 6/2199) consists of a UHMW layer retained by a series of steel retainer strips, screws and fittings. The UHMW base is trimmed out around fittings where tension cables are attached. Fasteners run through the retainer strips bolting the base to the ski. The base material provides a low friction surface on the bottom of the ski and helps protect the skis from damage experienced during landings. The retainer strips are wear components and should be inspected in accordance with PSM 1–6–7 and replaced as required.

2. Description

Refer to Figure 1 for details

The Nose Wheel-Ski consists of two beams structurally joined at the front, and connected at the rear by a torque tube secured to the left and right beams. A double-acting hydraulic actuator is housed in the left ski beam, and is connected by its cylinder to the beam structure. The rod end of the actuator connects to a latch lever attached to the end of a sling actuating torque shaft having two arms within the ski tunnel. The two arms are connected to either side of a sling by actuating links. A pedestal on each beam provides the pivot points for the ski sling and forms the mounting for the pivot blocks which attach the ski to the nose gear axle splined arms. Two hydraulic connections are provided, one on the inner side of each beam. The right-hand beam hydraulic pipe is routed through the right beam and the front section of the ski into the left beam, to the “down” side of the hydraulic actuator. The left-hand hydraulic pipe is routed within the left beam to the “up” side of the actuator. Flexible hydraulic hoses are installed between the connections on each side of the ski to pipe connections on the respective sides of the nose wheel shock strut fork. A ski position electrical cable connector is installed on the left ski beam and is wired internally to the two limit microswitches operated by the latch lever. Two trim cables, one connected to each side of the ski sling, are routed over pulleys to their respective rear left and right telescoping shock units. The four telescoping shock units, two connected to eye bolts at the front of the ski and two at the rear, connect to attachment brackets on the nose wheel shock strut fork. Each shock strut unit consists of a telescoping unit provided with positive stops to limit the maximum pitch attitudes of the ski, on which are mounted endless shock cord (bungee) rings, to restrain the ski in the neutral position and to provide a means of retracting the ski when the sling is actuated.



Nose Wheel-Ski
Figure 1

3. Operation

When skis "DOWN/SNOW" is selected, the actuator retracts unlocking the latch and, through the connecting linkage, moves the sling down under the wheel, in doing so the sling impinges on the tire and forces the wheel to roll over the top of it. This action causes the ski to move down into its extended position. Simultaneously as the sling moves rearward, the two trim cables slacken off, thus relieving the tension on the rear trim units. On selecting "UP/LAND", the actuator extends and moves the sling up and forward, allowing the wheel to roll from it. At the same time the trim cables, attached to the sling, tension the rear shock units. When the latch lever reaches the limit of its travel, a plunger on the end of the latch lever engages in a stop plate to lock the ski up. The two pairs of telescoping shock units each serve two purposes. The forward pair, by means of the shock cords, exert a restoring force to oppose external pitch down forces arising from aerodynamic maneuvering or handling loads, and by means of the internal stops, restrict the pitch down angle of the ski. The rear pair, by means of the shock cords, predetermine (after adjustment) the level trim attitude of the ski in fight, and through the trim cables, allow sufficient pitch up travel to accommodate surface irregularities encountered on snow or ice when selected to "DOWN/SNOW". The pitch up angle of the ski is also restricted by the rear shock unit internal stops.

4. Servicing

A. Nose Wheel-Ski Lubrication

Note The lubrication points, method and frequency of application are given in the Nose Wheel-Ski Lubrication diagram, refer to 12-20-10, Lubrication – General Data, Figure 2.

5. Removal/Installation

Refer to Figure 2 for details.

A. Remove Nose Wheel-Ski

Note The following removal procedures are based on the assumption that the aircraft is in the "UP/LAND" configuration.

1) Remove fuselage to landing gear leg fairings (refer to 12-00-00, Servicing – General Data) and, using main inboard jacking points and nose jacking beam (SD12538), jack aircraft until all wheels are at least 8 inches clear of ground (refer to 07-00-00, Lifting and Shoring – General Data).

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.) and ensure HYDRAULIC OIL PUMP circuit breaker is engaged.

Note Observe that motor pump charges accumulators, and that motor cuts out when pressure reaches 1575 ± 50 psi.

3) With the skis selected to the "DOWN/SNOW" position, disconnect sling trim cables from rear telescoping shock unit junction links.

4) Install chocks to forward and rear telescoping shock units.

Note The chocks referred to are supplied with the shock units as an aid to installation. For reference purposes, the length of the chocks are 17 9/16 inches for the rear, and 23 inches for the forward shock units.

5) With the skis selected to the "UP/LAND" position, block up ski level with aircraft datum, and remove rear telescoping shock units.

6) Remove forward telescoping shock units.

WARNING **SERIOUS INJURY COULD OCCUR SHOULD THIS HIGHLY-STRUNG SHOCK UNIT BE ACCIDENTLY DROPPED AND FRACTURED IN AN UNTENSIONED OR UNCHOCKED STATE.**

7) Select POWER SOURCE and MASTER switches OFF (Pre Mod 6/2314); or EXTERNAL and BATTERY switches OFF (Post Mod 6/2314).

8) Disengage HYDRAULIC OIL PUMP circuit breaker, and discharge hydraulic pressure from accumulators by operating brake pedals.

9) Using a suitable container to collect oil spillage, disconnect and remove hydraulic lines and electrical connections between ski and aircraft. Install blanking caps to open hydraulic lines.

10) Remove pivot bearing blocks attaching splined arms to ski.

11) Remove tire valve access cap and position wheel to provide clearance to remove left center bolt from pedestal pivot. Remove center bolt.

    Note The nose wheel is normally installed with the tire valve on the left, if wheel has been installed with valve on right, for Para 5.A.11) read “right center bolt”, and in Para 5.A.12) read “ski to left” and “left center bolt”.

12) Remove right center bolt nut, move ski to right, and remove right center bolt from pedestal pivot.

13) Remove blocks supporting ski. Remove ski.

14) Remove tie rod and splined arms from axle.

15) Install towing spools into axle and insert axle retaining bolts. Tighten axle retaining nuts and bolts.

16) At steering collar, back off torque on preload nut (refer to 32-20-34, Steering Collar – Maintenance Practices or 32-20-21, Steering Collar (with Mods S.O.O. 6001 and 6048) – Maintenance Practices.

17) If necessary, remove main wheel-skis before carrying out next step (refer to 32-11-00, Main Wheel-Skis (Mod S.O.O. 6001 and Mod 6/2182) – General data).

18) Lower aircraft to ground and remove jacks.

19) Switch on electrical power by selecting POWER SOURCE switch to EXTERNAL or BATTERY and MASTER switch to ON (Pre Mod 6/2314); or by selecting EXTERNAL or BATTERY switch to ON (Post Mod 6/2314). Check that BUS TIE switch is set to NORMAL.

20) Engage HYDRAULIC OIL PUMP circuit breaker.

    Note Observe that motor pump charges accumulators, and that motor cuts out when pressure reaches 1575 ± 50 psi.

21) Using a suitable container to receive oil spillage, bleed the right nose wheel-ski hydraulic line at fuselage, by slackening blanking cap with ski selector lever selected to DOWN/SNOW. When fluid is free of air, tighten blanking cap with fluid still flowing.

22) Repeat Para 5.A.21) on left nose wheel-ski hydraulic line with an UP/LAND selection made.

    Note If main wheel-skis are being removed at the same time as the nose wheel-ski, bleed main wheel-ski system in conjunction with Para 5.A.21) and Para 5.A.22). (Refer to 32-11-00, Main Wheel-Skis (Mod S.O.O. 6001 and Mod 6/2182) – General data).

23) Ensure after bleeding, that blanking caps on hydraulic lines are tight and secure.

24) Replenish hydraulic system as necessary (refer to 12-10-15, Hydraulic System – General Data).

25) Store wheel-skis complete with attachment fittings for future use.

26) Ensure Apex is reprogrammed with standard gear airspeed programming instructions to suit the gear configuration in accordance with 32-10-11, Main Gear Leg – General Data, Para 3.

27) Install taxi light and connect wiring.

B. Remove Nose Wheel-Ski Base (Mod 6/2199)

Refer to Figure 1 for details.

1) Remove the Nose Wheel-Ski from aircraft as per Para 5.A.

2) Remove the screws securing the skids and skegs from bottom of ski.

3) Remove skids and skegs.

4) Remove the screws securing the retainers to ski base.

5) Remove retainers.

Note It is not necessary to remove the Tow Lug Assembly from the forward end of skis.

6) Remove remaining screws securing ski base to ski.

7) Remove ski base.

C. Preparation of Replacement Ski Base Components

1) Clamp together the old ski base to the replacement ski base.

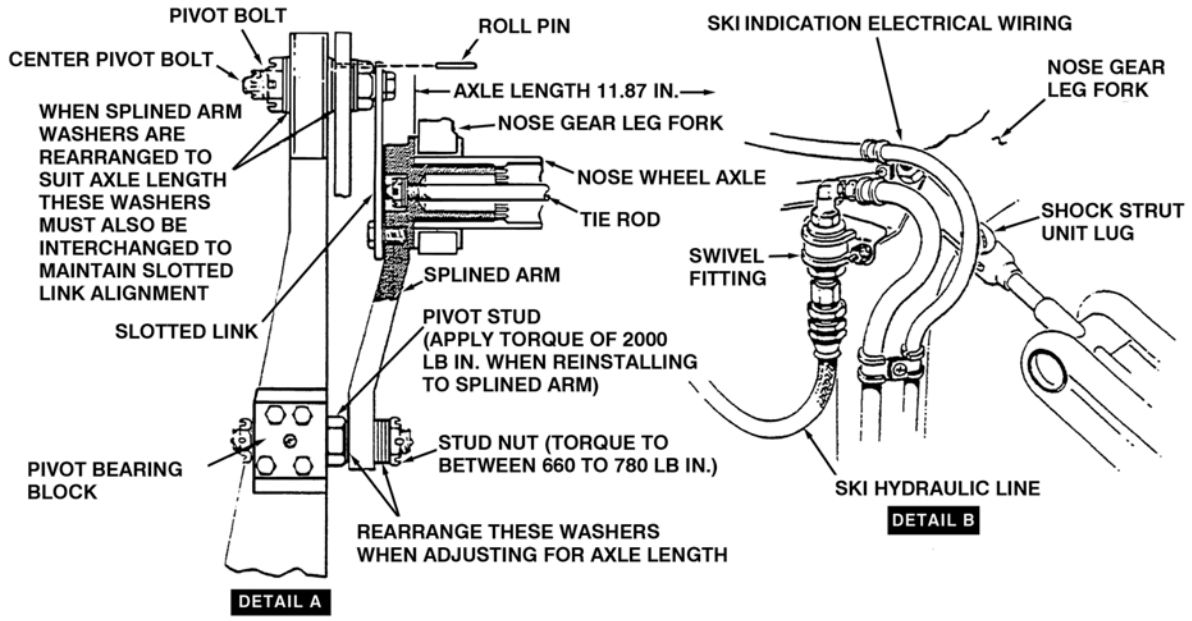
2) Copy profile and hole locations.

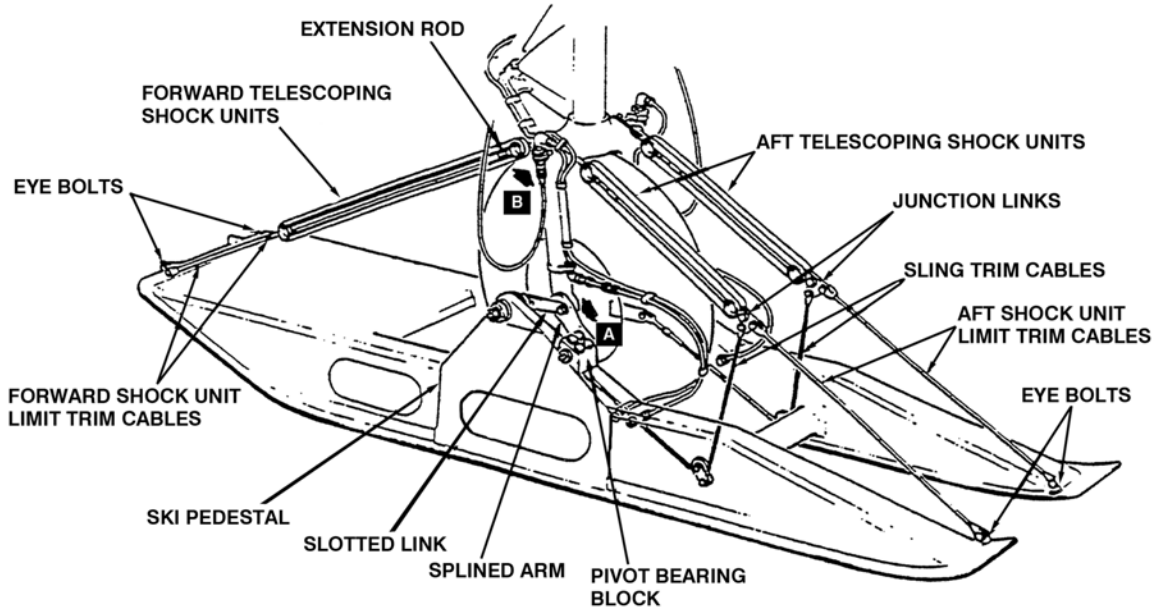
3) Transfer hole pattern from ski to replacement ski base. Match hole size and countersunk as required.

4) Carefully trim replacement base to match old base profile.

5) Using old ski base retainers, bend new retainers to match profile.

6) Clamp retainers securely together and transfer hole patterns. Match hole size and countersink as required.





Nose Wheel-Ski Installation
Figure 2

D. Install Nose Wheel-Ski

Note For the purpose of the following procedures, it is assumed that the aircraft has the necessary structural and nose leg shimmy damper provisions for a wheel-ski installation.

1) Remove fuselage to landing gear leg fairings (refer to 12-00-00, Servicing – General Data) and, using main inboard jacking points and nose jacking beam (SD12538), jack aircraft well clear of ground (refer to 07-00-00, Lifting and Shoring – General Data). Level aircraft (refer to 08-00-00, Leveling and Weighing – General Data).

2) Remove retaining bolts and tow bar spools from nose wheel axle.

3) Check overall length of nose wheel axle. This should agree with the measurement (11.87 inches) between faces of splined arm flanges when splined arms are installed on ski. If axle length is incorrect, obtain correct dimensions by rearranging the washers at pivot stud on either side of splined arm. The washers at the pivot bolts must also be relocated to align and allow freedom of movement of the slotted links. (Refer to Figure 2, Detail A). The pivot stud must be unscrewed from splined arm to relocate washer so when reassembling stud to arm apply a torque of 2000 pound–inches to stud against arm. Torque stud nut to a value of 660 to 780 pound–inches and lock with cotter pin.

4) Inspect entire ski assembly for corrosion and damage and check mechanism for freedom of movement. Lubricate ski (refer to 12-20-10, Lubrication – General Data, Figure 2).

5) Remove both center bolts from the ski pedestal pivots, and four bolts attaching each pivot bearing block to its ski pedestal. Lift off splined arm assemblies complete with slotted links. Separate the two splined arms by removing tie rod.

6) Assemble splined arms (complete with slotted links) to splines inside nose wheel axle, ensuring that arms on each side of wheel are aligned with each other. Check location of tire inflation valve and, because there is insufficient clearance later, insert left or right center pedestal pivot bolt, with washer under head of bolt, through slot in appropriate slotted link.

7) Install tie rod through center of splined arms and axle, tighten tie rod nuts, but ensure tie rod nuts are not torqued to restrict freedom of rotation of splined arms and axle assembly. Secure nuts with cotter pins.

8) With nose wheel-ski in the “UP/LAND” condition (sling up), position ski under wheel. Shift ski sideways as required to facilitate installation of center pivot bolt (refer to Para 5.D.6)), and position wheel tire valve access hole to allow the left center pedestal pivot bolt to be installed.

9) Check before bolting the pivot bearing blocks to ski, that bearing block holes align with holes in ski pedestals. If holes do not align, recheck axle length and distance between faces of splined arms and rectify as in Para 5.D.3).

10) Secure pivot bearing blocks to ski pedestals, with four bolts through each block.

11) Secure left and right pedestal center bolts with nuts and cotter pins. Ensure pivot bolts are not overtightened, slotted links must be allowed to slide freely. Ensure pivot bolts and slotted links are well lubricated.

12) Move ski up and down by hand and check for any indications of binding linkage.

13) Select POWER SOURCE and MASTER switches OFF (Pre Mod 6/2314); or EXTERNAL and BATTERY switches OFF (Post Mod 6/2314).

14) Disengage HYDRAULIC OIL PUMP circuit breaker and release hydraulic accumulator pressure by operating brake pedals.

15) Install left and right hydraulic pipes, and ski indication electrical wiring to nose gear leg fork. Install telescopic shock strut lugs on fork shoulder (refer to Figure 2, Detail B).

16) Connect left and right hydraulic pipes to ski and fuselage connections.

17) Connect ski indication electrical wiring to ski and nose gear leg connections.

18) Bleed nose wheel-ski hydraulic system as follows:

    a) Switch on electrical power by selecting POWER SOURCE switch to EXTERNAL or BATTERY and MASTER switch to ON (Pre Mod 6/2314); or by selecting EXTERNAL or BATTERY switch to ON (Post Mod 6/2314). Ensure that BUS TIE switch is set to NORMAL.

    b) Engage HYDRAULIC OIL PUMP circuit breaker.

        Note Observe that motor pump charges accumulators and that motor cuts out when pressure reaches 1575 ± 50 psi.

    c) Select ski selector to DOWN/SNOW, and using a suitable container, bleed the right nose wheel-ski hydraulic line by slackening flexible hose between ski and fork. When fluid is free of air retighten hose while fluid is still flowing.

    d) Repeat Para 5.D.18).c) on left hose making an UP/LAND selection.

        Note If main wheel-skis are being installed at the same time as the nose wheel-ski, bleed main wheel-skis in conjunction with Para 5.D.18) (refer to 32-11-00, Main Wheel-Skis (Mod S.O.O. 6001) – General Data).

19) Hydraulically actuate skis to the “DOWN/SNOW” position (sling down).

20) Position and firmly block up nose wheel-ski until it is parallel to the aircraft datum and sling is in firm contact with tire.

21) On left and right sides of nose wheel-ski measure distance between centers of eye bolt hole at aft end of ski and hole in rear lug on nose gear leg fork shoulder.

22) With chocks fitted, preassemble left and right aft telescoping shock units and cables with junction links.

    WARNING SERIOUS INJURY COULD OCCUR SHOULD THIS HIGHLY-STRUNG SHOCK UNIT BE ACCIDENTLY DROPPED AND FRACTURED IN AN UNTENSIONED OR UNCHOCKED STATE.

23) Adjust fork at upper end of each aft telescoping shock unit, until overall length of each assembly agrees with measurements taken in Para 5.D.21), taking into account that assembly will shorten approximately 1/16 inch, when the wood chocks are removed.

24) Install telescoping shock unit assemblies between nose gear leg fork lugs and eye bolts at rear of each ski beam. Do not remove chocks.

25) Actuate ski to the “UP/LAND” position and firmly block up ski level with aircraft datum +2° –0°. Center pivot bolts must contact end of slot in slotted links.

26) On left and right sides of nose wheel-ski, measure distance between centers of eye bolt at front of ski and forward lug attachment on nose gear leg fork shoulder.

27) With chocks fitted preassemble left and right forward telescoping shock units and forward limit cables.

28) Adjust fork at upper end of each forward shock unit assembly until overall length agrees with dimension obtained in Para 5.D.26).

29) Install telescoping shock unit assemblies between forward nose gear leg fork lugs and eye bolts at front of ski.

30) Remove wood chocks from all four telescoping shock unit assemblies.

31) Actuate ski to the “DOWN/SNOW” position and mark the forward shock unit extension rods. Tilt ski nose down to ensure the minimum overtravel of 1/2”. If required, readjust length of forward telescoping shock unit assemblies.

    Note It is necessary to replace chocks and return skis to the “UP/LAND” position, in order to carry out adjustment in Para 5.D.31).

32) Recheck rigging, making adjustments as necessary to the aft telescoping shock units, to maintain skis level with aircraft datum +2° –0°, ensuring that they remain under equal tension and that neither is pulled from its minimum limit length.

    Note These units are normally non-performing, but provide the necessary additional travel (2 inches) to accommodate severely undulating motions sometimes encountered in the “ski - landing” and “ski - taxiing” cases.

33) Loosely connect the sling trim cables between the sling attachments and the telescoping shock strut assembly junction links, through pulleys.

34) Actuate ski to the “wheel - landing” position and, with hydraulic pressure maintained, adjust both sling trim cable turnbuckles to position ski level with the aircraft datum +2° –0° ensuring equal tension of both cables.

    Note Turnbuckle adjustment can be facilitated by pitching nose of ski down.

35) Functionally test main and nose wheel-skis (refer to 32-30-00, Extension and Retraction (Wheel-Ski) (Mod S.O.O. 6001) – General data).

    Note In the “ski - landing” position the ski will assume an attitude approximately 2° nose up.

36) Lockwire the forks at the upper ends of the shock units so as to prevent chafing or fouling the bungee rings.

37) Test wheel-ski indication system (refer to 32-61-00, Wheel-Ski Indication (Mod S.O.O. 6001) – General data).

38) At steering collar, increase torque loading at preload nut (refer to 32-20-34, Steering Collar – Maintenance Practices or 32-20-21, Steering Collar (with Mods S.O.O. 6001 and 6048 – Maintenance Practices).

39) With skis selected to the desired position lower aircraft to ground, remove jacking and leveling equipment. Install fuselage to leg fairings.

40) Check hydraulic system fluid level (refer to 12-10-15, Hydraulic System – General Data).

41) Ensure Apex is reprogrammed with “Wheel-Ski” airspeed programming instructions to suit the gear configuration in accordance with 32-10-11, Main Gear Leg – General Data, Para 3.

E. Install Nose Wheel-Ski Base (Mod 6/2199)

1) Ensure bottom of ski is clean and undamaged.

2) Install base to ski and secure with new hardware in countersunk holes on base.

    Note Over torquing of screws will damage holes in UHMW ski base.

3) Install center ski base retainer. Start at forward end of ski and use new hardware.

4) Install the inner ski base retainers. Start at forward end of ski and use new hardware.

5) Install the outer edge ski base retainers. Start at forward end of ski and use new hardware.

6) Install skids and skegs with associated hardware.

6. Adjustment/Test

Note The access panels form part of the stressed structure of the ski and must be reinstalled as soon as adjustments are completed.

A. Adjust Sling Actuating Mechanism

1) Jack up aircraft (refer to 07-00-00, Lifting and Shoring – General Data).

2) Remove forward left-hand panel.

3) Select ski “UP/LAND” position and observe behavior of ski and latch mechanism.

4) If latch pin does not engage locking hole of latch spring, disconnect and adjust fork end of hydraulic actuator until latch pin engages locking hole.

5) If latch pin engages locking hole of latch spring, release hydraulic pressure and check that clearance between tire and sling is between 0.10 inch and 0.60 inch.

Note Tire clearance is adjusted during manufacture and assembly, and should not be affected appreciably in normal service or by the replacement of related components. Ensure that inadequate tire clearance has not been caused by damage to the latch mechanism or ski bottom in area of latch assembly before attempting tire clearance adjustment.

6) Adjust tire clearance, if necessary, by removing or adding shims under base of latch spring (removing shims increases tire clearance).

Note Adjusting fork end of hydraulic actuator will not alter tire clearance and will only lead to erratic or faulty latch function. If, however, shims have been added or removed, adjustment of hydraulic actuator is also required as in Para 6.A.7).

7) Adjust fork end of hydraulic actuator to give 0.015 inch to 0.060 inch overtravel after latch engagement.

8) Check adjustment of Ski Up and Ski Down limit switches. (Refer to 32-61-00, Wheel-Ski Indication – General Data).

9) Reinstall access panel.

10) Lower aircraft.

7. Inspection/Check

A. Check Nose Wheel-Ski Rigging

1) Remove fuselage to leg fairings (refer to 12-00-00, Servicing – General Data), jack aircraft well clear of ground using inboard jacking points and nose jacking beam SD12538 (refer to 07-00-00, Lifting and Shoring – General Data). Level aircraft (refer to 08-00-00, Leveling and Weighing – General Data).

2) Switch on electrical power by selecting POWER SOURCE switch to EXTERNAL or BATTERY, and MASTER switch to ON (Pre Mod 6/2314); or by selecting EXTERNAL or BATTERY switch to ON (Post Mod 6/2314). Ensure that BUS TIE switch is set to NORMAL.

3) Ensure GEAR IND and HYDRAULIC OIL PUMP circuit breakers are engaged.

Note Observe that motor pump charges accumulators and that motor cuts out when pressure reaches 1575 ± 50 psi.

4) Select ski selector lever to “UP/LAND”. Check ski is level with aircraft datum +2° –0°, and that center bolts at sling pivot are at end of slot in slotted link.

5) Actuate ski to the “DOWN/SNOW” position and mark the forward shock unit extension rods. Tilt ski nose down to ensure the minimum overtravel of 1/2”. If required, readjust length of forward telescoping shock unit assemblies.

   Note If an adjustment is required to achieve the 1/2 inch overtravel, adjust in accordance with ski installation procedures (refer to Para 5.D).

6) Check that neither of the aft telescoping shock units are pulled from their minimum length.

7) Select ski selector lever to UP/LAND and check that sling trim cables position ski level with aircraft datum +2° –0°.

8) At steering collar check, by applying spring balance to end of torque link, with steering actuator disconnected, that torque required to turn steering collar is between 800 and 1000 pound–inches or 70 to 90 pounds at end of torque link. Reconnect steering actuator.

9) Lower aircraft to ground, remove jacking equipment. Install fuselage-to-leg fairings.

10) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

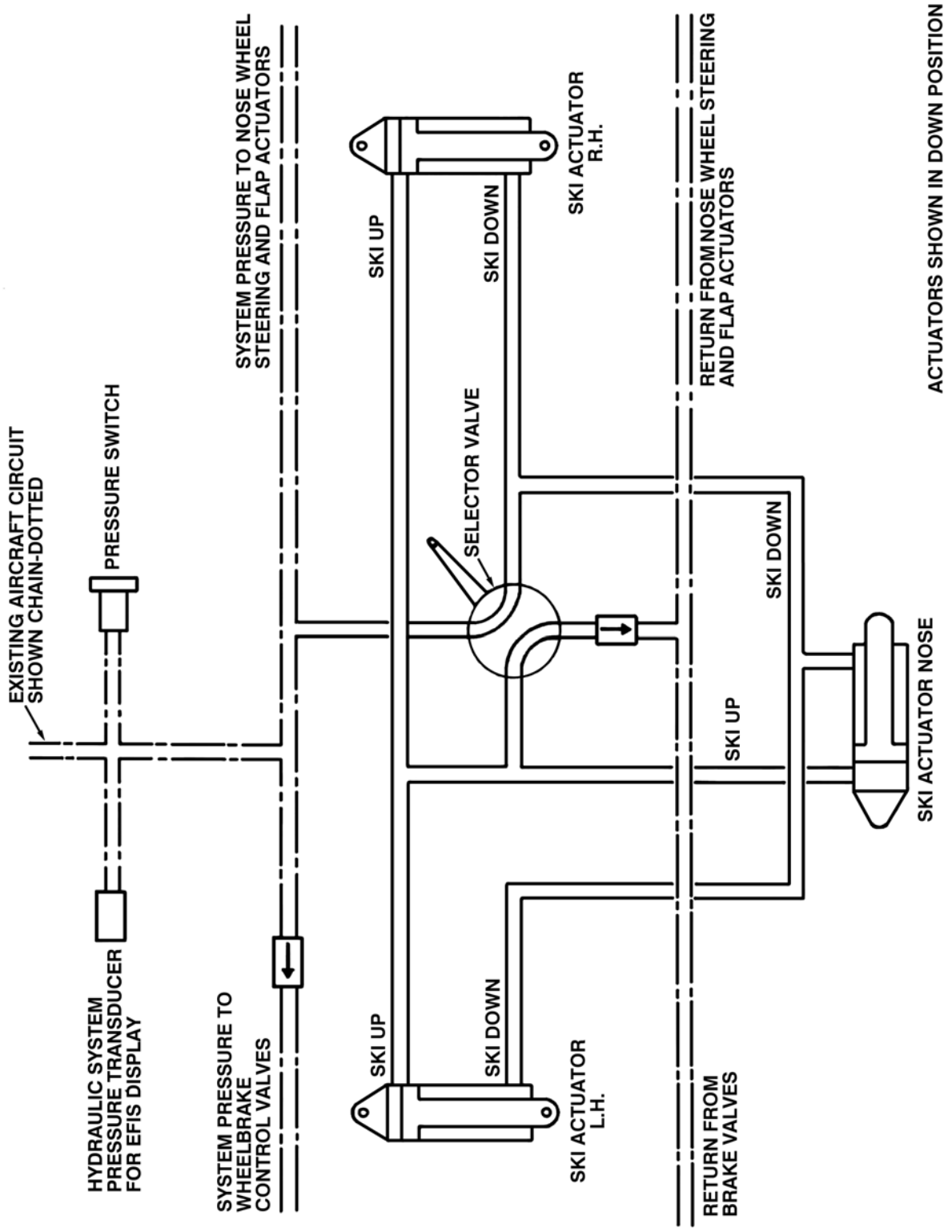
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Extension and Retraction (Wheel-Skis) (Mod S.O.O. 6001) – General data

1. General

Refer to Figure 1 for details.

Extension and retraction of the wheel-skis is accomplished by directing hydraulic fluid, under pressure from the aircraft's hydraulic system (refer to 29-00-00, Hydraulic Power – General Data), to either the “up” or “down” side of the wheel-ski hydraulic actuators. The direction of system fluid flow, and the return of the displaced fluid, is determined by the position of a four-way selector valve. The selector valve, mounted behind the ski position selector valve on the instrument panel. The ski position selector valve has two positions marked UP/LAND and DOWN/SNOW. The panel also includes indicator lights to provide ski position indication.



Wheel-Ski Hydraulic System – Schematic
Figure 1

2. Adjustment/Test

A. Rig Wheel-Ski Selector Control

Refer to Figure 2 for details.

1) Install Wheel-Ski position selector control in instrument panel cutout on the right hand side of the flight compartment.

2) Connect hydraulic hoses to ski position valve using straight/inline fittings.

3) If installed, remove caps from hydraulic bulkhead fittings for pressure lines labeled 'UP', 'DN', 'P' and 'R'.

4) From ski position valve, connect hydraulic hoses to bulkhead fittings labeled 'UP', 'DN', 'P' and 'R' using the 90° fittings.

5) Function test Wheel-Ski system, refer to Para 2.B.

B. Function Test Wheel-Ski Installation

1) Jack aircraft clear of ground. Use the main inboard jacking points and a narrow base jack (or jacking beam SD12538) to jack the nose (refer to 07-00-00, Lifting and Shoring – General Data).

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Ensure GEAR IND circuit breaker is engaged.

4) If the electrical hydraulic pump is to be used, engage the HYDRAULIC OIL PUMP circuit breaker.

Note Observe that motor pump charges accumulators and that motor cuts out when pressure reaches 1575 ± 50 psi.

5) If the electrical hydraulic pump is not to be used, operate the hydraulic handpump.

6) Select ski position selector lever “UP/LAND” and “DOWN/SNOW” several times and check SKI POSITION indicators display three lights when at UP, and three when at DOWN, for the appropriate selection.

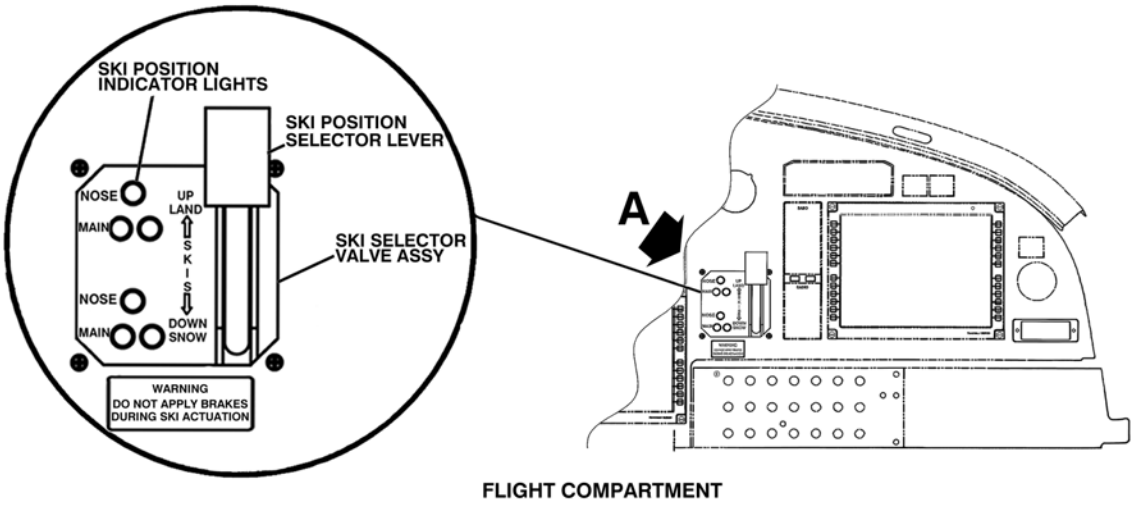
Note During function test, a hydraulic low pressure caution or warning CAS message might be generated, if so do steps Para 2.B.7) and Para 2.B.8).

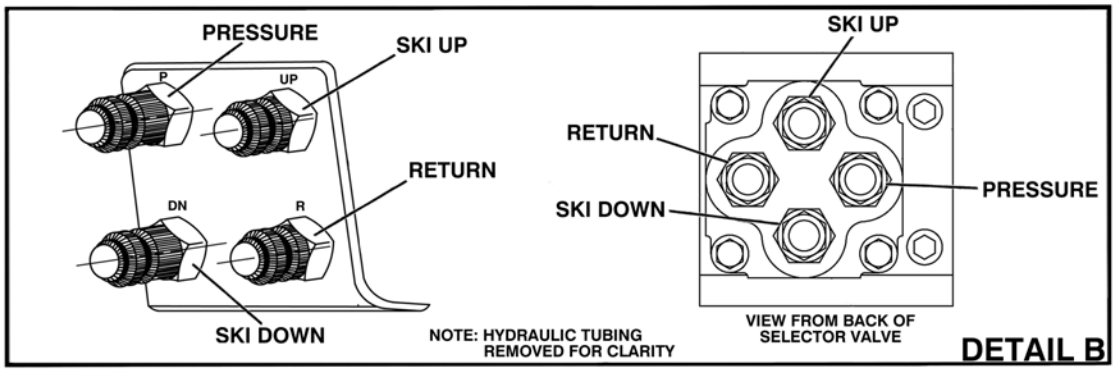
7) Confirm when skis are full “UP” or full “DOWN” the CAS message clears.

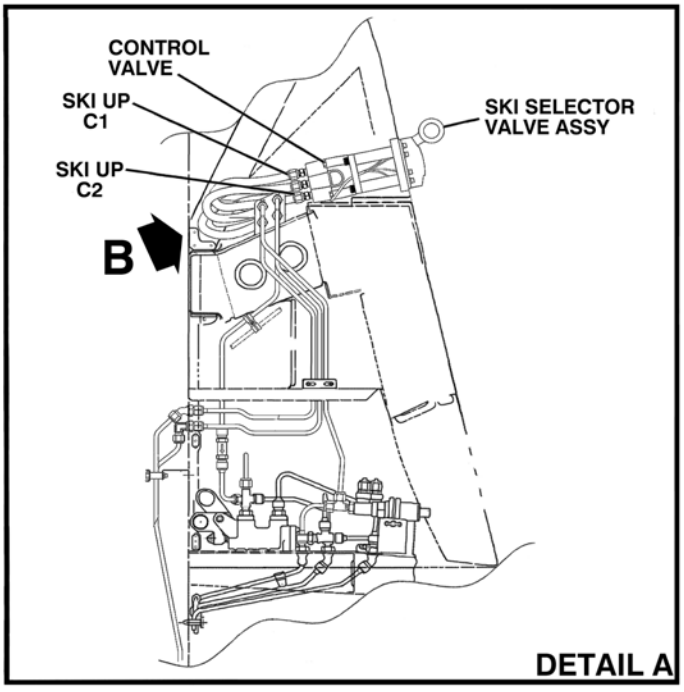
8) Check hydraulic system for leaks. If none, system is acceptable.

9) Lower aircraft to ground in the “UP/LAND” configuration, and remove jacking equipment.

10) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).







Ski Selector - Rigging
Figure 2

Wheels and Brakes (Mod 6/2002) – General Data

1. General

A. Wheels

The main landing gear wheels are carried on the axle of each main gear leg, and the nose wheel is carried on the axle mounted in the fork of the nose gear. The wheels are of the split hub type to facilitate removal and installation of tubeless tires.

B. Wheel Brakes System

The wheel brakes system consists of hydraulically-operated wheel brake units controlled from the pilot's and co-pilot's rudder pedals. The fluid pressure is metered to the wheel brake units in proportion to brake pedal depression. When the hydraulic system electric motor-driven pump or handpump is operated, fluid is supplied from the main hydraulic system to the control valves and to charge a wheel brakes accumulator (refer to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data). Return fluid from the control valves is fed back to the hydraulic system common return line. A check valve is installed upstream of the supply line to the accumulator. The accumulator supplies brake pressure when the pumps are not in operation.

C. Brake Hydraulic Pressure Indicator

The brake pressure transducers send a signal to display the amount of hydraulic pressure on the cockpit multi-function display screen.

2. Servicing

A. Bleed Hydraulic Brake System

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Check nose wheel steering lever corresponds with position of nose wheel, and wing flaps selector with position of wing flaps. Ensure parking brake is OFF.

2) Set POWER SOURCE switch to EXTERNAL, and MASTER switch to ON (Pre Mod 6/2314) or set EXTERNAL switch to ON (Post Mod 6/2314). Check electric motor-driven hydraulic pump charges damping accumulator and wheel brakes accumulator, and motor cuts out when reading on pressure indicators is 1575 ± 50 psi.

3) Depress both left and right brake pedals and engage the parking brake.

4) Place container to receive spillage, and slacken bleeder plug at top rear end of left brake unit. When fluid is clear of air, retighten bleeder plug.

5) Repeat Para 2.A.4) for right wheel brake unit.

6) Release parking brake.

7) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

8) Check Hydraulic reservoir fluid level.

3. Adjustment/Test

A. Function Test Wheel Brakes

1) Ensure parking brake is OFF.

2) Remove rear bleeder plug at top rear end of left and right-hand brake unit, and install 0 to 2000 psi pressure gauge at each bleeder port.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Discharge brake system accumulator hydraulic pressure by pumping brake pedals. Check nose wheel steering lever corresponds with position of nose wheel, and wing flaps selector with position of wing flaps.

4) Set POWER SOURCE switch to EXTERNAL, and MASTER switch to ON (Pre Mod 6/2314) or set EXTERNAL switch to ON (Post Mod 6/2314). Check electric motor-driven hydraulic pump charges damping accumulator and wheel brakes accumulator, and motor cuts out when reading on pressure indicators is 1575 ± 50 psi.

5) Depress pilot's left brake pedal and check pressure gage reading at left brake unit settles at 1000 ± 50 psi. Release brake pedal. Repeat test using co-pilot's left brake pedal.

6) Depress pilot's right pedal and check pressure gage reading at right brake unit settles at 1000 ± 50 psi. Release brake pedal. Repeat test using co-pilot's right brake pedal.

   Note Adjust pressure if necessary at turnbuckles on push rods.

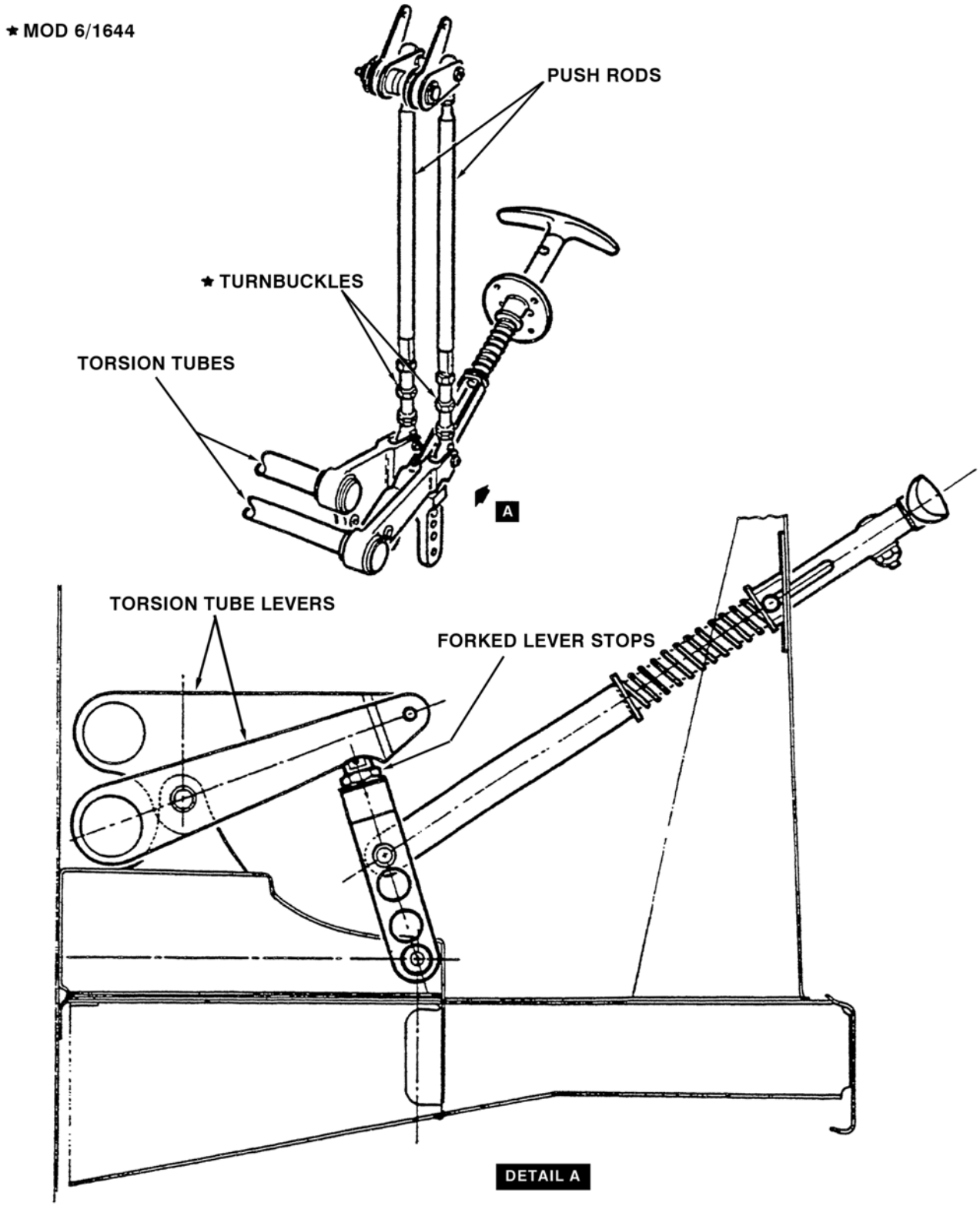
7) With system accumulator and wheel brakes accumulator fully charged, disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

8) Apply both brakes fully, pull up on parking brake handle and release brake pedals. Adjust fork lever stops to obtain a pressure reading of 850 ± 50 psi (refer to Figure 1).

9) Allow system to stand for 30 minutes. Check that system pressure is maintained. Release parking brake.

10) Remove pressure gauge from left and right-hand brake unit, and install bleeder plugs.

11) Set parking brake, if required.



Parking Brake
Figure 1

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Brake Control Valve – General Data

1. General

The brake control valves are connected through a system of push rods, levers, and torsion rods to the brake pedals on each rudder pedal. A turnbuckle is installed in each push rod to increase pressure adjustment for the brake control valve. The valves are located on the flight compartment floor, and are in effect variable pressure reducers which meter fluid from the hydraulic system to the brake pressure lines in proportion to brake pedal movement. Each valve consists of a cast aluminum alloy body containing a lever-operated floating piston and a spring-loaded poppet valve. The valves are connected into the system through three ports, inlet pressure brake pressure, and return (refer to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data).

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Brake Control Valve

1) Discharge brake system accumulator hydraulic pressure by pumping brake pedals.

2) Remove access panels in flight compartment.

3) At brake control valve, disconnect brake system indicator hydraulic line (right-hand valve only); disconnect inlet pressure, brake pressure and return hydraulic lines. Cap open connections and lines.

4) Remove cotter pin, nut, washer and bolt attaching brake control linkage levers to mounting bracket. Withdraw spring torsion rod (short) from brake control valve lever.

Note When removing right-hand brake control valve remove bolt at right-hand linkage levers and mounting bracket. For left-hand valve, remove bolt at left-hand linkage levers and mounting bracket.

5) Remove four bolts securing brake control valve to flight compartment floor, and move valve outboard to disengage valve lever from spring torsion rod (long).

B. Install Brake Control Valve

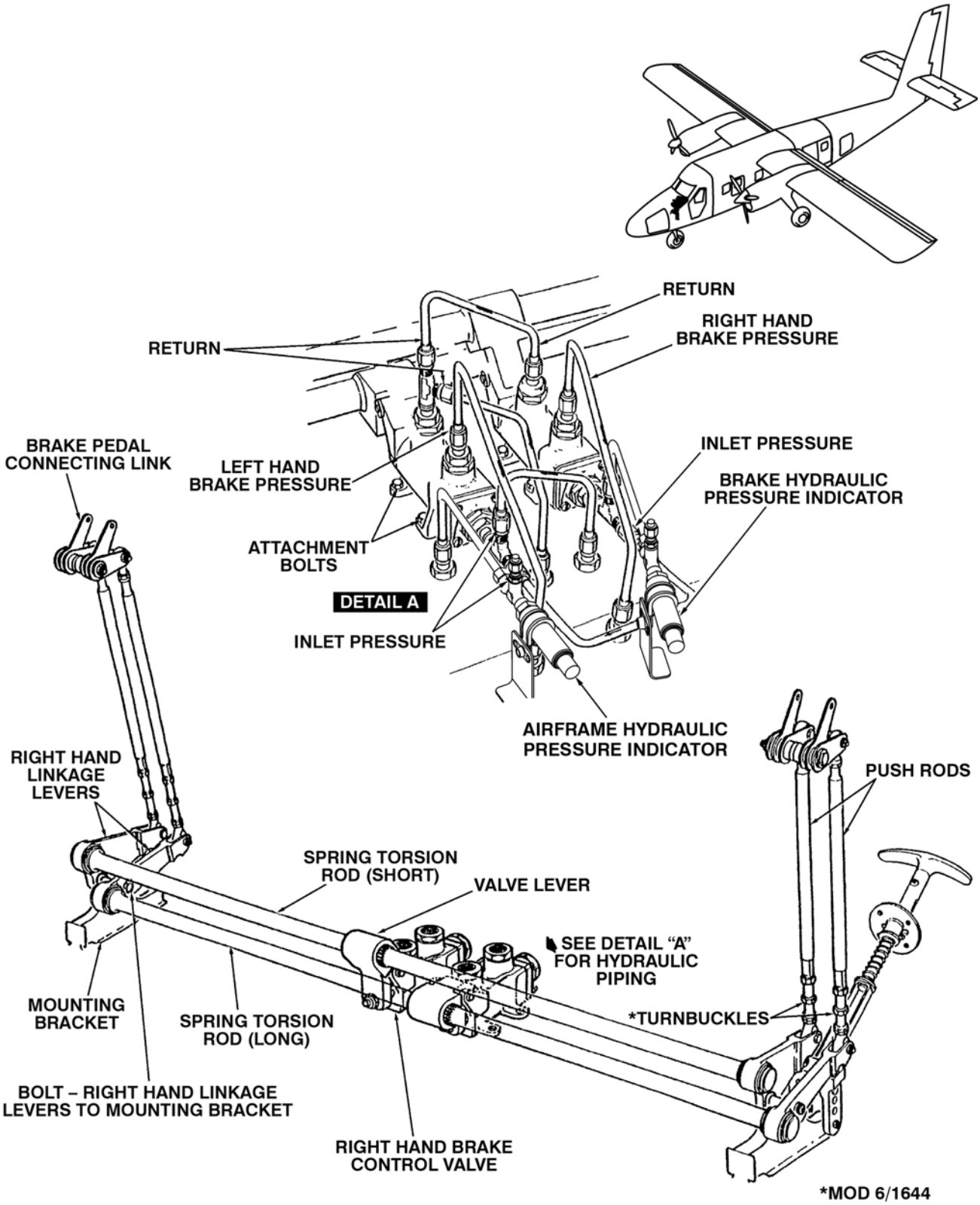
1) Engage valve lever on spring torsion rod (long), and position brake control valve on flight compartment floor. Install four bolts securing valve to floor.

2) Engage spring torsion rod (short) with valve lever. Position control linkage levers at mounting bracket and install bolt, washer, nut and cotter pin.

3) At brake control valve, connect brake system indicator hydraulic line (right-hand valve only); connect inlet pressure, brake pressure and return hydraulic lines.

4) Bleed brake system and function test wheel brakes. (Refer to 32-40-00, Wheels and Brakes (Mod 6/2002) – General Data).

5) Install access panels in flight compartment.

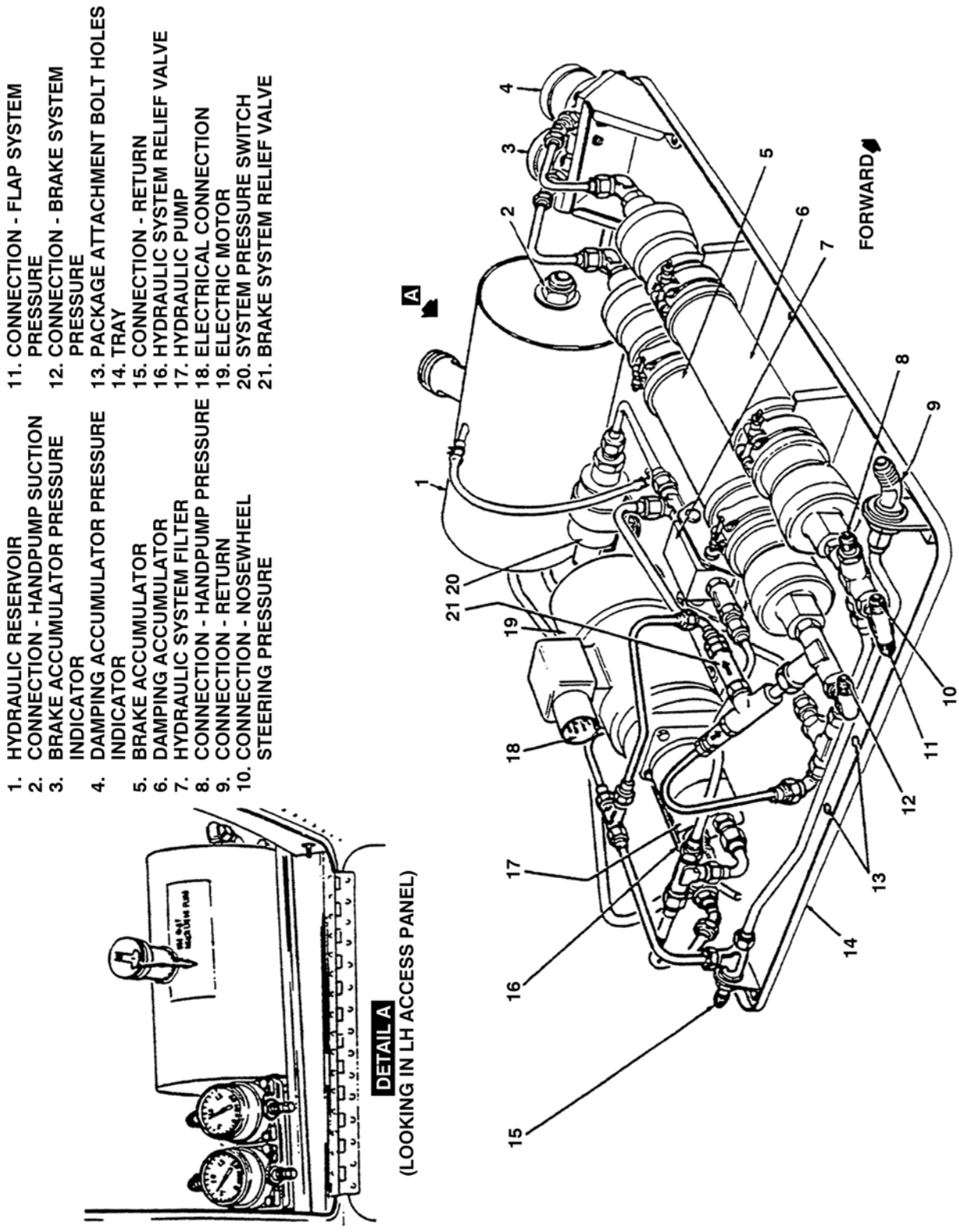


Brake Control Valve Installation
Figure 1

Wheel Brakes Accumulator (Mod 6/2039) – General Data

1. General

The wheel brakes accumulator mounted on the hydraulic power package beneath the flight compartment floor, provides the wheel brakes with supplemental pressure in an emergency or during limited periods of peak demand, and provides pressure when the hydraulic system is shutdown while parked. The accumulator is connected into the hydraulic system main pressure supply line to the wheel brakes control valves, and a combined air pressure gauge and air charging valve is mounted on a bracket on the left side of the package. (Refer to Figure 1).



Hydraulic Power Package
Figure 1

2. Removal/Installation

A. Remove Wheel Brakes Accumulator

1) Discharge brake system accumulator hydraulic pressure by pumping brake pedals.

2) Open left and right-hand access doors on underside of nose section.

3) Discharge all air pressure from brake system accumulator by loosening air valve one-quarter turn and waiting until all air has escaped. Check indicator on accumulator reads zero.

WARNING FAILURE TO PROPERLY DISCHARGE ALL AIR PRESSURE FROM ACCUMULATOR MAY RESULT IN SERIOUS INJURY.

4) Disconnect air line at head of accumulator. Cap and plug open connection and line.

5) Place container to receive spillage. Remove nuts and release two clamps, and unscrew accumulator with union from tee (refer to Figure 1).

B. Install Wheel Brakes Accumulator

1) If new accumulator is to be installed, transfer fittings from unserviceable accumulator and use new O-rings.

2) Install new O-ring at tee, and screw accumulator union into tee (refer to Figure 1).

3) Attach accumulator to tray with two clamps and nuts.

CAUTION OVERTIGHTENING OF ACCUMULATOR MOUNTING CLAMPS COULD RESULT IN DAMAGE TO ACCUMULATOR.

4) Overcome run down torque of clamp nuts then add 15 – 20 pound–inches torque maximum to seat nut.

5) Connect air line at head of accumulator.

6) Charge accumulator with dry nitrogen to 750 ± 50 psi.

7) Check reservoir fluid level. Function test hydraulic power package (refer to 29-10-00, Hydraulic Power Package (Mod 6/2039) – General Data).

8) Bleed brake system (refer to 32-40-00, Wheels and Brakes (Mod 6/2002) – General Data).

9) Close left and right-hand access doors on underside of nose section.

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Wheel Brakes Unit (Mod 6/2002) – General Data

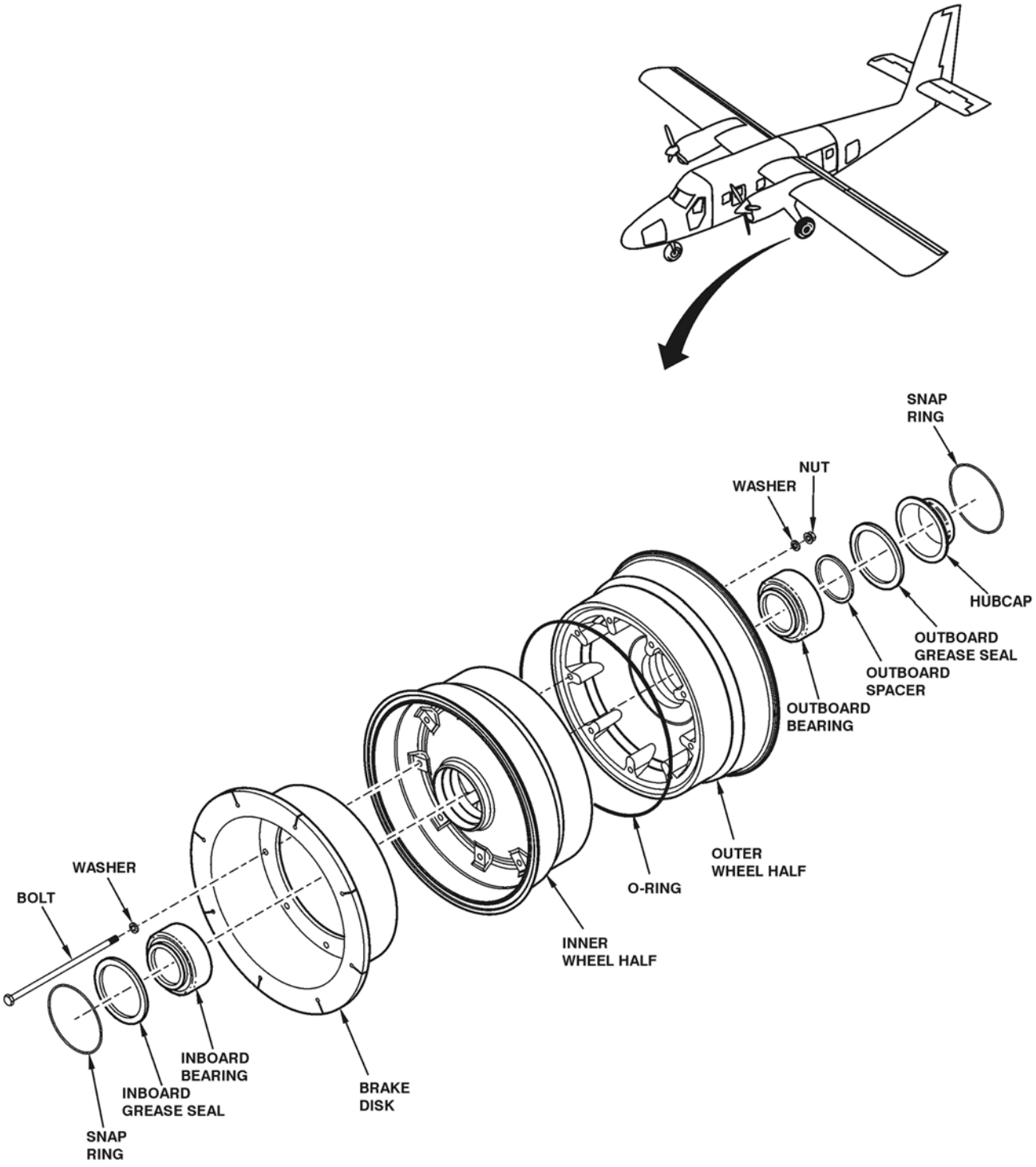
1. General

A wheel and brake unit is bolted to the torque plate on each mainwheel axle. The brake assembly is a piston actuated, hydraulically operated, external disc unit designed for use with a petroleum base hydraulic fluid. The right and left brake assemblies can be interchanged by switching the bleeder fitting with an inlet fitting.

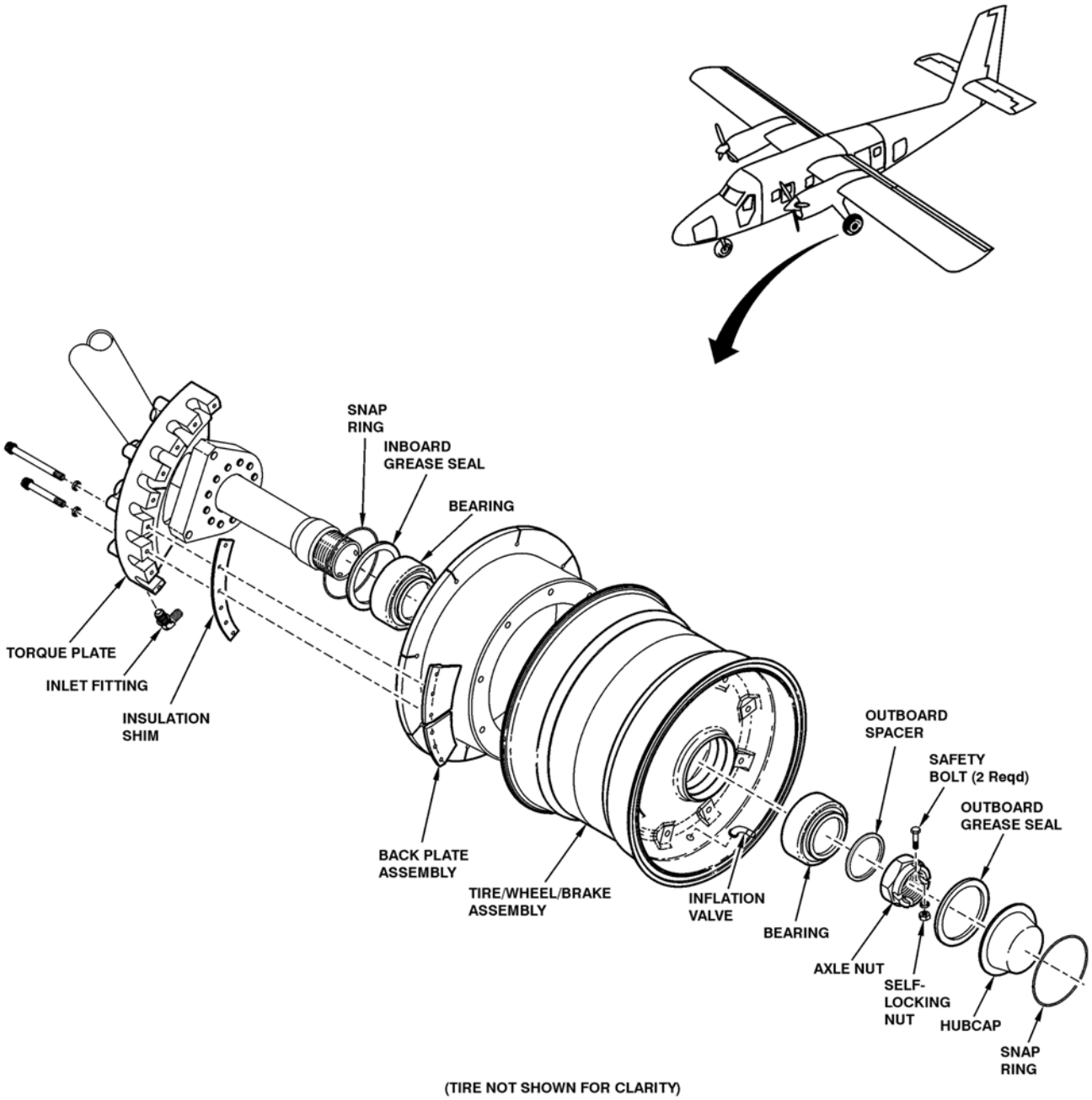
Each brake assembly is composed of a pressure plate assembly, four pistons, four backing plate assemblies, a torque plate, a cast aluminum alloy cylinder housing, four piston insulators to minimize the conduction of heat into the hydraulic fluid, four O-rings, four piston boots to seal out contaminates, four friction springs to maintain ideal piston positioning, two sets of anchor bolt hardware, a hydraulic fitting, one set of bleeder fitting hardware, and twelve high strength bolts, nuts, and washers to fasten the four back plate assemblies to the cylinder housing.

Braking action occurs when hydraulic pressure is applied to the brake via the pilot's or co-pilot's brake control valves. As hydraulic pressure reaches the caliper, the pistons move outward in the caliper forcing the pressure plate against the surface of the brake disc. This causes the caliper to align itself so that the pressure plate linings and backplate linings provide uniform contact against both rubbed surfaces of the brake disc.

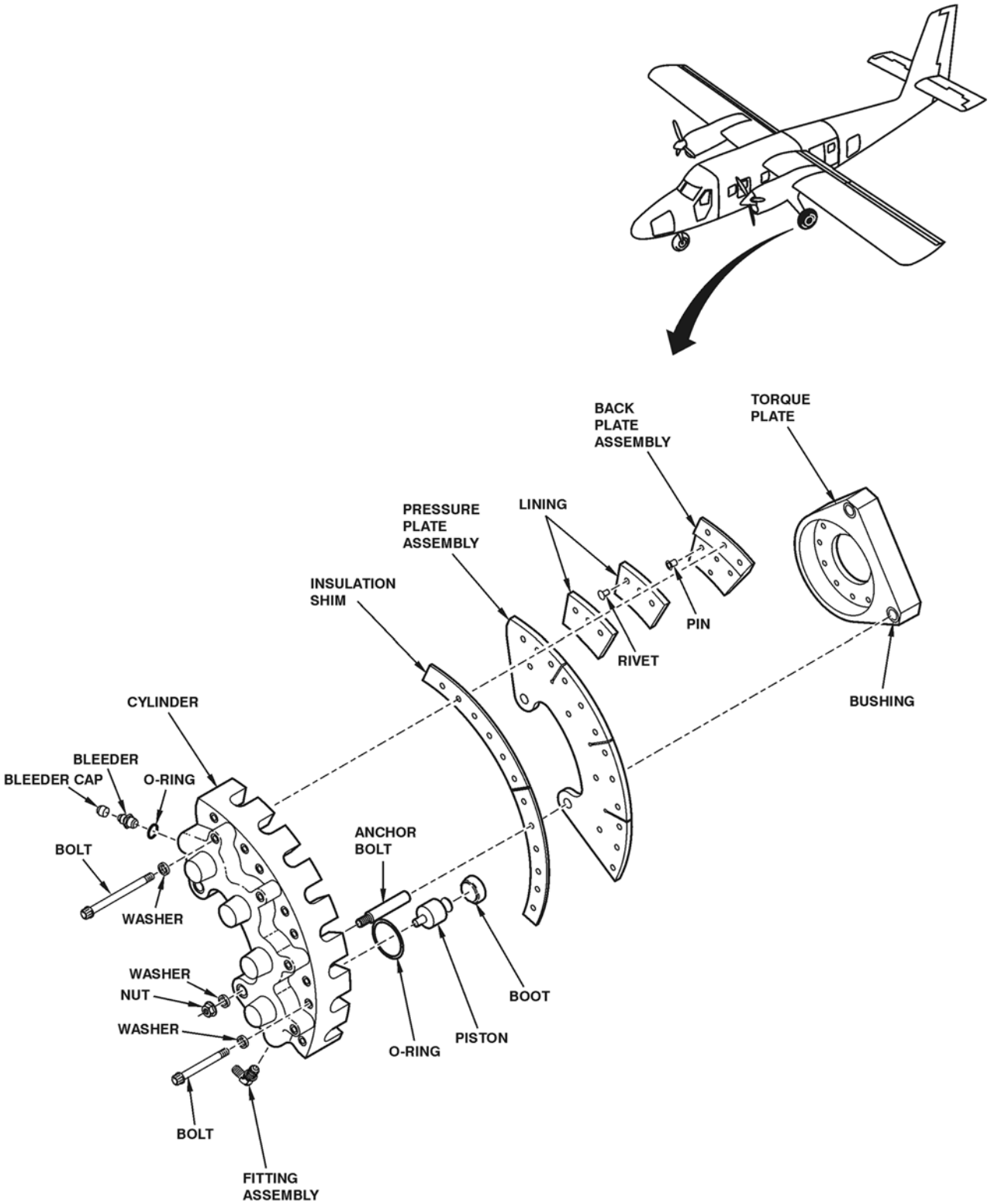
Refer to manufacturers information regarding maintenance, servicing repair of the brake assembly.



Wheel and Brake Assembly
Figure 1 (Sheet 1 of 3)



Wheel and Brake Assembly
Figure 1 (Sheet 2 of 3)



Wheel and Brake Assembly
Figure 1 (Sheet 3 of 3)

2. Removal/Installation

A. Brake Unit Removal

1) Discharge brake accumulator by pumping brake pedals.

2) Remove and cap brake hydraulic line.

3) Remove the twelve brake cylinder tie bolts and washers, and four back plate assemblies. (Refer to Figure 1 Sheet 3 of 3).

4) Slide the cylinder out of the torque plate. (The torque plate will remain mounted to the axle).

5) Place the assembly on a clean flat surface.

6) Remove the pressure plate assembly.

B. Brake Unit Installation

1) Install the pressure plate assembly (the lining side must NOT face the pistons) by aligning the bolt holes with the anchor bolts and slide the pressure plate onto the cylinder. The pressure plate MUST float freely on the anchor bolts.

2) Install the brake housing into torque plate by aligning the anchor bolts with the torque plate holes and sliding the brake assembly onto the torque plate. It MUST slide freely.

3) Place a countersunk washer on each of the twelve brake tie bolts, and insert these bolts into the respective brake housing holes.

Note The countersunk side of the washer must face the bolt head or bolt failure may result.

4) Install two new insulator shims onto brake housing.

5) Install the four backplate assemblies onto the brake housing and lubtork (MIL–T–5544A) the brake tie bolts to 80 – 85 in–lbs.

Note Be sure that the brake disc is sandwiched between the backplate and the pressure plate.

6) Reconnect the hydraulic lines.

7) Bleed brake with procedure from 32-40-00, Wheels and Brakes (Mod 6/2002) – General Data.

8) Check the pedal for proper feel and travel.

Note Wheels should rotate freely. There should be no evidence of binding or excessive brake drag.

3. Inspection/Check

Do inspections in accordance to OVH 030–17000 Section 3 Detail Check.

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Parking Brake – General Data

1. General

A push-pull type parking brake handle is located on a pedestal between the pilot's rudder pedals. The handle has restricted movement on the parking brake rod assembly and is spring-loaded to the off position. The rod assembly is connected to a forked lever fitted with two adjustable stops. These stops ride in machined portions of the brake control valves torsion tube levers, and when the brake pedals are fully depressed and the parking brake applied, the forked lever holds the torsion tube levers and brake control valves in brakes on position. To release the brakes, pressure on the brake pedals will allow the spring loading of the parking handle to push the forked lever clear of the torque levers. Refer to Figure 1 for details.

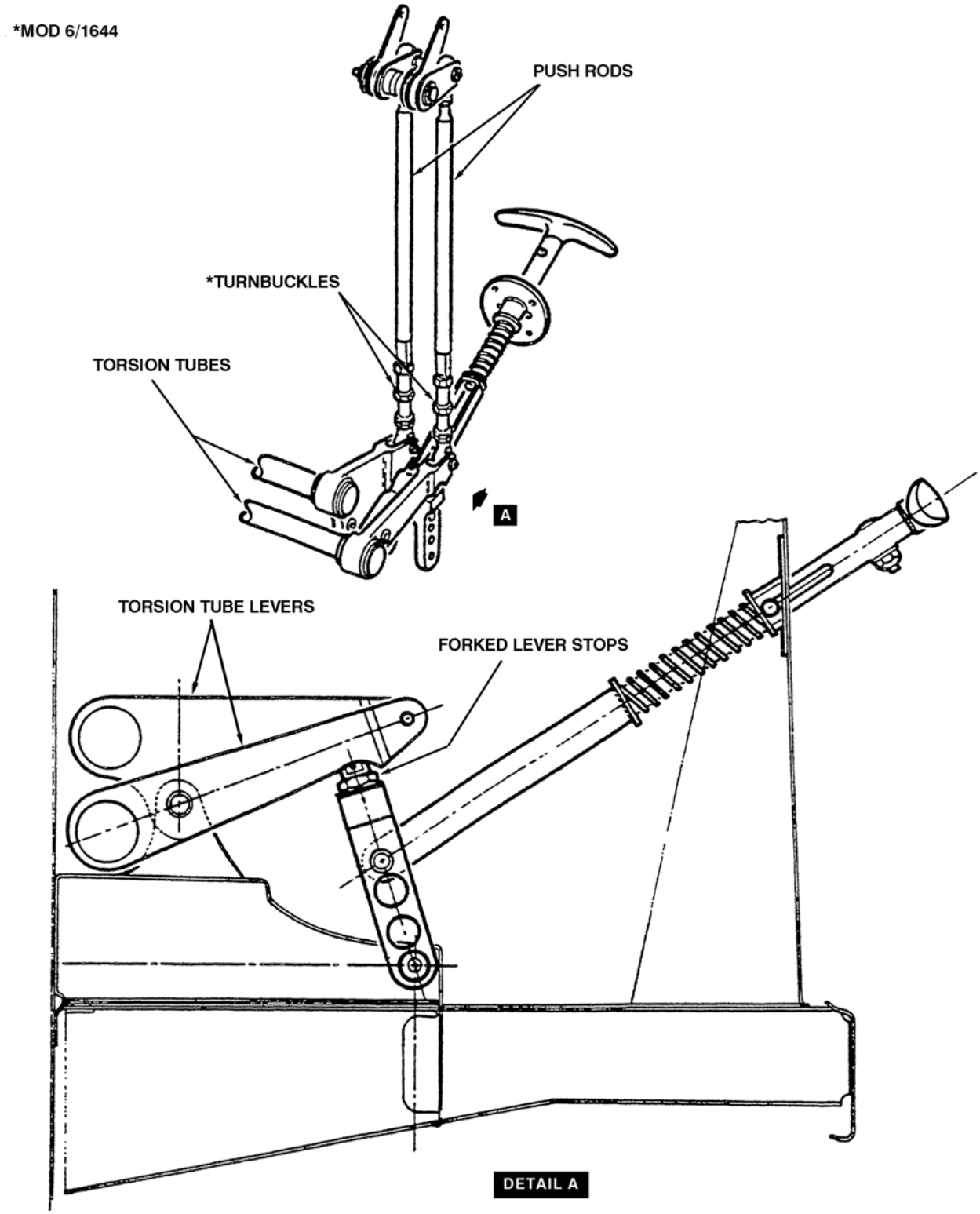
2. Adjustment/Test

A. Parking Brake Adjustment

1) Slacken jam nuts on forked lever stops (refer to Figure 1).

2) Have operator depress brake pedals and pullout parking brake handle. While maintaining position, adjust stops to contact torsion tube levers.

3) Tighten jam nuts. Depress pedals and observe that parking brake handle releases. Re-adjust stops, if necessary, to obtain this condition.



Parking Brake
Figure 1

Main Wheels (Mod 6/2002) – General Data

1. General

The wheel is cast magnesium and conforms to all tire and rim association standards for a 11.00-12 divided type wheel, suitable for use with all 11.00-12, 10 ply tires. It is a tubeless design only. The wheel incorporates inboard and outboard halves which are fastened together with bolts, washers, and nuts. The brake disc is attached to the wheel by bolts. The wheel rotates on two tapered roller bearings, which seat in bearing cups in the wheel half hubs. Rubber lip seals provide protection and lubricant retention for the bearing.

2. Removal/Installation

A. Remove Main Wheel

1) Remove the brake assembly and place it on a clean flat surface, as per 32-40-31, Wheel Brakes Unit (Mod 6/2002) – General Data.

2) Jack aircraft until wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data).

3) Deflate tire.

4) Remove snap ring and hubcap.

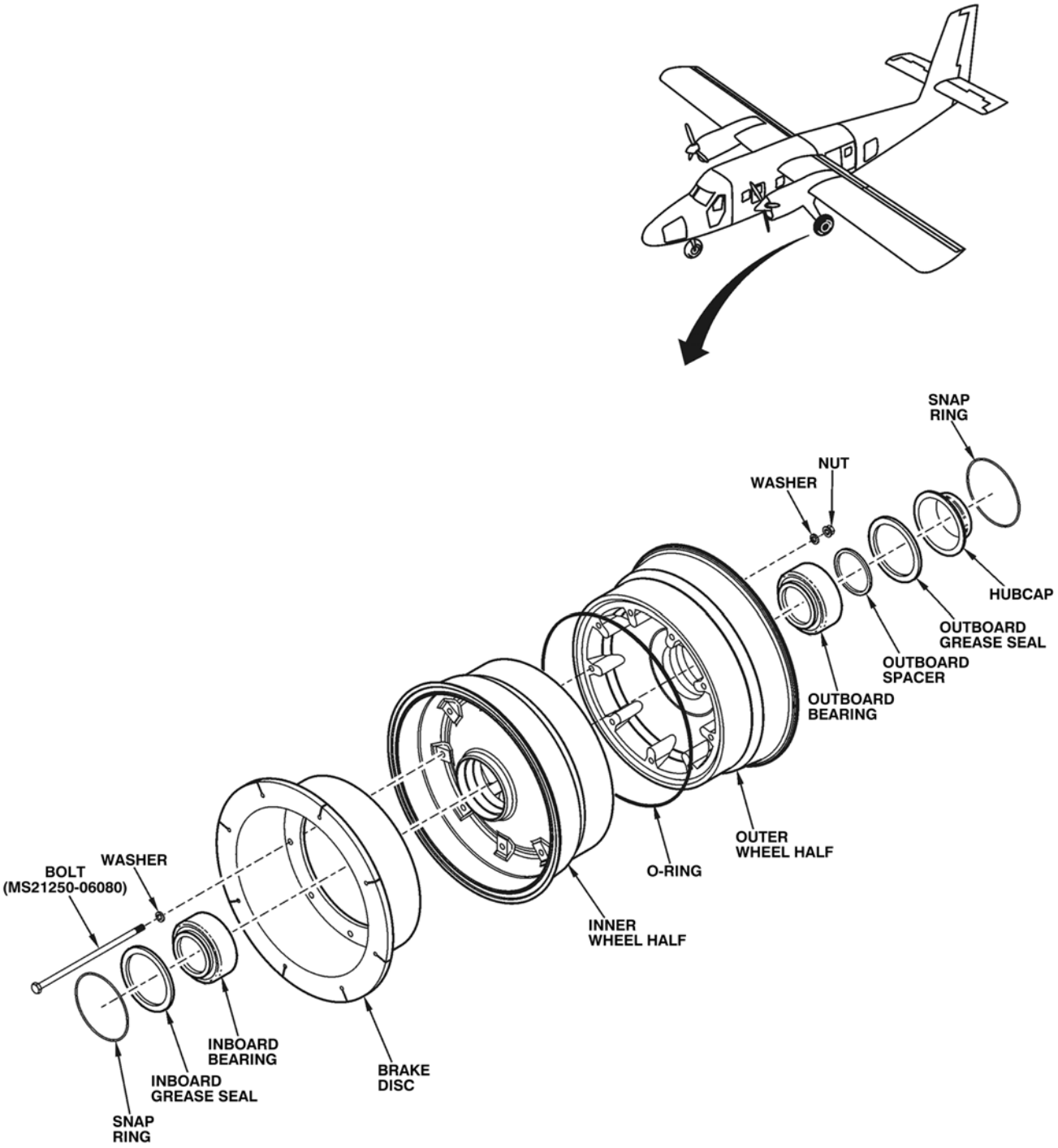
5) Remove two safety bolts and self locking nuts from axle nut. Remove axle nut using wrench SD5523.

6) Remove outboard grease seal, spacer and bearing.

Note Place wheel seals, spacer and bearings on a clean, flat surface to prevent any contamination.

7) Carefully slide wheel/tire assembly off of axle.

8) Remove inboard bearing, seal and retaining clip.



Main Wheel Assembly
Figure 1

B. Install Main Wheel

1) Apply a thin coat of grease (MIL–PRF–81322) or MOBIL SHC 100 (refer to 12-20-10, Lubrication – General Data, Table 1) to axle and axle threads.

2) Install lubricated inboard wheel bearing in wheel assembly with grease seal and retaining clip.

3) Carefully slide wheel/tire assembly on to axle.

4) Install lubricated outboard bearing into wheel assembly with spacer and grease seal.

a) Install axle nut using wrench SD5523 and torque to 150 in–lbs to 200 in–lbs while seating the bearing.

b) Back off axle nut.

c) Retighten to 30 in–lbs to 40 in–lbs while rotating.

d) Rotate axle nut to nearest axle slot and install two safety bolts and self locking nuts.

Note Wheel must rotate freely, without perceptible play.

5) Install hubcap and snap ring.

6) Inflate tire.

7) Refer to 32-40-31, Wheel Brakes Unit (Mod 6/2002) – General Data, to reinstall brake assembly.

Note If installation of main wheel on aircraft is part of gear configuration change then do step as per Para 2.B.8).

8) Do procedure for gear configuration change as per 32-10-11, Main Gear Leg – General Data, Para 2.C.

C. Remove Main Wheel Tire (Tubeless)

1) Remove wheel from aircraft. (Refer to Para 2.A).

2) Release all pressure from tire by slackening air valve core then finally remove core completely.

3) Using tire demounters, wooden wedges, or an approved tire machine, separate edges of tire from wheel hub beading.

4) Remove bolts, self-locking nuts, washers and countersunk washers, holding halves of hub together. (Refer to Figure 1).

5) Carefully separate hub halves, and remove O-ring and tire. Check O-ring for serviceability and discard if damaged.

D. Install Main Wheel Tire (Tubeless)

1) Check hub for scratches, scores and cleanliness according to OVH Manual 040–21101. Clean tire area of hub with a dry cleaning solvent. Dust area with french chalk (talc).

2) Assemble wheel on a clean flat surface according to OVH Manual 040–21101.

3) Position hub half containing air valve on bench. Lightly lubricate serviceable O-ring with silicone grease (DOW Molykote 55M) and install on hub. Position serviceable tire on hub with red spot on tire aligned with air valve.

CAUTION ENSURE BOLTS ARE MS21250–06080. DO NOT USE BOLTS, VALVE STEM BUSHING AND BRAKE DISC SPACER FROM S.O.O. 6048 AND MOD 6/2183.

4) Lubricate threads of bolts and bearing surfaces of nuts, bolt heads and washers with anti-seize compound (MIL–T–5544A).

WARNING **DO NOT INFLATE TIRE UNTIL ALL BOLTS HAVE BEEN PROPERLY TIGHTENED. IT IS RECOMMENDED THAT ALL MOUNTED TIRES BE INFLATED INSIDE A METAL CAGE TO AVOID POSSIBLE INJURY TO PERSONNEL.**

CAUTION **DO NOT USE IMPACT OR POWER WRENCHES TO TIGHTEN OR TORQUE WHEEL BOLTS OR NUTS.**

5) Install core in air valve and inflate tire to correct pressure. (Refer to 12-10-20, Landing Gear – General Data).

6) Apply a 1” x 3” creep mark using a suitable contrasting coloured paint between tire and rim.

7) Install wheel on aircraft. (Refer to Para 2.B).

3. Inspection/Check

A. Inspect Main Wheel Hub

1) Remove main wheel and main wheel tire. (Refer to Para 2.A and Para 2.C).

2) Clean hub halves with dry cleaning solvent (P–D–680 or equivalent) and dry with filtered and dried compressed air. Refer to OVH Manual 040–21101.

3) Inspect hub halves for cracks (using the fluorescent penetrant method or an approved dye penetrant method), chipping of rims, damage and corrosion. Replace hub if cracks or chipping of rims is evident. Refer to OVH Manual 040–21101.

Note Small nicks or gouges can be blended out and polished with fine aluminum oxide cloth (refer to Manufacturer's Overhaul Manual for definition and limitation of damage). Refer to OVH Manual 040–21101.

4) Inspect bearings and replace if damaged. Refer to OVH Manual 040–21101.

5) Inspect brake disc wheel bolts and nuts as per OVH 030–17000.

6) Install tire to main wheel and install main wheel on aircraft. (Refer to Para 2.B and Para 2.D).

Main Wheel (Mod S.O.O. 6048 and Mod 6/2183) – General data

1. General

On aircraft with Mod S.O.O. 6048 and Mod 6/2183 incorporated the main wheel hub is fitted with a 36 x 13 x 12 tire, a brake spacer installed between the hub and brake disc, a valve stem bushing and the wheel assembly bolts replaced by longer bolts to accommodate the spacer.

2. Removal/Installation

A. Remove Main Wheel

1) Remove the brake assembly and place it on a clean flat surface, as per 32-40-31, Wheel Brakes Unit (Mod 6/2002) – General Data.

2) Jack aircraft until wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data).

3) Deflate tire.

4) Remove snap ring and hubcap.

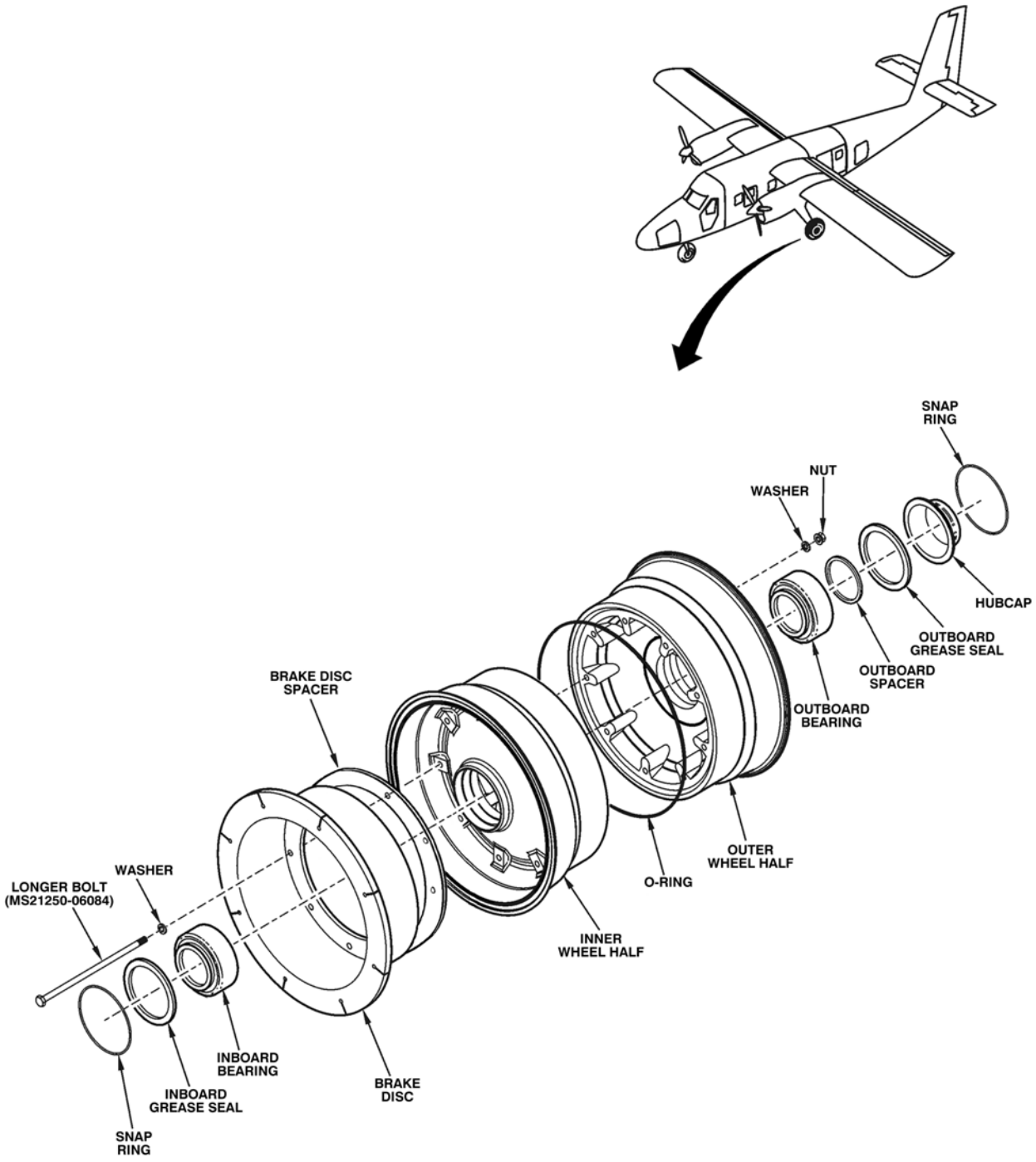
5) Remove two safety bolts and self locking nuts from axle nut. Remove axle nut using wrench SD5523.

6) Remove outboard grease seal, spacer and bearing.

Note Place wheel seals, spacer and bearings on a clean, flat surface to prevent any contamination.

7) Carefully slide wheel/tire assembly off of axle.

8) Remove inboard bearing, seal and retaining clip.



Main Wheel Assembly (Mod S.O.O. 6048 Mod 6/2183)
Figure 1

B. Install Main Wheel

1) Apply a thin coat of grease (MIL–PRF–81322) to axle and axle threads.

2) Install lubricated inboard bearing in wheel assembly with grease seal and retaining clip.

3) Carefully slide wheel/tire assembly onto axle.

4) Install lubricated outboard bearing into wheel assembly with spacer and grease seal.

a) Install axle nut using wrench SD5523 and torque to 150 in–lbs to 200 in–lbs while seating the bearing.

b) Back off axle nut.

c) Retighten to 30 in–lbs to 40 in–lbs while rotating.

d) Rotate axle nut to nearest axle slot and install two safety bolts and self locking nuts.

Note Wheel must rotate freely, without perceptible play.

5) Install hub cap and snap ring.

6) Inflate tire.

7) Reinstall brake assembly as per 32-40-31, Wheel Brakes Unit (Mod 6/2002) – General Data.

Note If installation of main wheel on aircraft is part of gear configuration change then do step as per Para 2.B.8).

8) Do procedure for gear configuration change as per 32-10-11, Main Gear Leg – General Data, Para 2.C.

C. Remove Main Wheel Tire (with tube) (Mod S.O.O. 6048 and Mod 6/2183)

1) Remove the wheel/tire assembly from aircraft as per Para 2.A.

2) Release all pressure from tire by slackening air valve core.

3) Remove the valve stem bushing from wheel assy and valve core.

4) Remove valve core completely from wheel assy.

5) Using tire demounters, wooden wedges, or an approved tire machine, separate edges of tire from wheel hub beading.

6) Remove bolts, self-locking nuts, washers and countersunk washers, holding the brake disc, brake disc spacer and halves of the hub together. Refer to Figure 1.

7) Carefully separate hub halves, and remove O-ring and tire. Check O-ring for serviceability and discard if damaged.

D. Install Main Wheel Tire (with tube) (Mod S.O.O. 6048 and Mod 6/2183)

1) Check hub for scratches, scores and cleanliness according to OVH Manual 040–21101. Clean tire area of hub with a dry cleaning solvent. Dust area with French chalk (talc).

2) Partially inflate tube and position in tire. Align yellow stripes on the tube base with the red dot on the tire sidewall.

3) Position hub half containing air valve hole on the bench and place serviceable tire and O-ring on the hub with red spot in tire aligned with control valve. Install position valve stem through valve hole and secure with the valve stem bushing. Ensure valve stem bushing is snug.

CAUTION **ENSURE BOLTS ARE MS21250–06084. DO NOT USE BOLTS THAT ARE PROVIDED WITH WHEEL ASSEMBLY FOR STANDARD GEAR, (MS21250–06080).**

4) Lubricate threads of bolts and bearing surfaces of nuts, bolt heads and washers with anti-seize compund (MIL–T–5544A).

5) Install O-ring, other half hub, brake disc spacer and brake disc. Install bolts and washers and self locking nuts, then tighten by equal increments in diagonal order to torque values stamped on wheel hub.

Note Install bolts with heads on inboard side of wheel with one washer under nut and one under bolt head.

WARNING **DO NOT INFLATE TIRE UNTIL ALL BOLTS HAVE BEEN PROPERLY TIGHTENED. IT IS RECOMMENDED THAT ALL MOUNTED TIRES BE INFLATED INSIDE A METAL CAGE TO AVOID POSSIBLE INJURY TO PERSONNEL.**

CAUTION **DO NOT USE IMPACT OR POWER WRENCHES TO TIGHTEN OR TORQUE WHEEL BOLTS OR NUTS.**

6) Inflate tire to correct pressure. (Refer to 12-10-20, Landing Gear – General Data).

7) Apply a 1” x 3” creep mark using a suitable contrasting coloured paint between tire and rim.

8) Install wheel on aircraft as per Para 2.B.

3. Inspection/Check

A. Inspect Main Wheel Hub

1) Remove main wheel and main wheel tire. (Refer to Para 2.A and Para 2.C).

2) Clean hub halves with dry cleaning solvent (P–D–680 or equivalent) and dry with filtered and dried compressed air. Refer to OVH Manual 040–21101.

3) Inspect hub halves for cracks (using the fluorescent penetrant method or an approved dye penetrant method), chipping of rims, damage and corrosion. Replace hub if cracks or chipping of rims is evident. Refer to OVH Manual 040–21101.

Note Small nicks or gouges can be blended out and polished with fine aluminum oxide cloth (refer to Manufacturer's Overhaul Manual for definition and limitation of damage). Refer to OHV Manual 040–21101.

4) Inspect bearings and replace if damaged. Refer to OVH Manual 040–21101.

5) Inspect brake disc spacer for damage or corrosion. Replace if damaged.

6) Inspect brake disc wheel bolts and nuts as per OVH 030–17000.

7) Install tire to main wheel and install main wheel on aircraft. (Refer to Para 2.B and Para 2.D).

Nose Wheel – General Data

1. General

The nose wheel is carried on the axle mounted in the fork of the nose gear, and is of the split hub type to facilitate removal and installation of tubeless tires. An O-ring is installed between the two halves of the hub to provide an airtight housing for the tire. The nose wheel has 6–ply rating 8.90 x 12.50, Type III, low pressure tubeless tire. When Mod S.O.O. 6048, Mod 6/1526 and Mod 6/2183 is embodied, a main hub is fitted with a 36 x 13.00 x 12 tire and tube is installed in the nose wheel position. (Refer to 32-40-62, Nose Wheel (Mod S.O.O. 6048 and Mod 6/2183) – Maintenance Practices).

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Nose Wheel

1) Jack aircraft until nose wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data).

2) Remove axle locating bolts from ends of fork.

3) Remove set screws from adjustable axle spacer and using wrenches SD12552 and SD12553, loosen off nut of adjustable spacer. Remove lock ring securing fixed spacer to axle.

4) Support nose wheel, remove towing spools, withdraw axle from nose wheel, axle spacers and fork. Lower and remove nose wheel.

5) Remove nose wheel discs from each side of nose wheel.

6) Remove bearing seals and cone bearings from both sides of nose wheel.

7) Clean cone bearings, inspect and repack with grease.

B. Install Nose Wheel

1) Ensure nose wheel bearing housings are clean, then install cone bearings and bearing seals.

2) Install nose wheel discs with plug opposite valve.

3) Lightly lubricate axle with grease (MIL–PRF–23827), support nose wheel in fork and, with locating bolt holes aligned with holes in fork, insert axle through fork bushings and nose wheel together with axle spacers. Ensure that the adjustable axle spacer is installed with the nut adjacent to the fork. Secure fixed spacer to axle with lock ring.

4) Install towing spools and axle locating bolts in each fork end and secure with washers and nuts.

5) Extend adjustable axle spacer, using wrenches SD12552 and SD12553 until wheel bearings begin to bind, then back off to nearest locking position. Insert set screws to lock adjustable spacer.

6) Check tire for correct inflation. (Refer to 12-10-20, Landing Gear – General Data).

7) Lower aircraft to ground.

Note If installation of nose wheel on aircraft is part of gear configuration change then.

8) Do procedure for airspeed programming instructions as per 32-10-11, Main Gear Leg – General Data, 2.C.

C. Remove Nose Wheel Tire

1) Remove wheel from aircraft.

2) Release all pressure from tire by slackening air valve core then finally remove core completely.

3) Using tire demounters, wooden wedges, or an approved tire machine, separate edges of tire from wheel hub beading.

4) Remove bolts, washers and self-locking nuts, holding hub halves, and remove O-ring and tire. Check O-ring for serviceability and discard if damage.

D. Install Nose Wheel Tire

1) Check hub for scratches, scores and cleanliness. Clean tire area of hub with a dry cleaning solvent. Dust area with french chalk (talc).

2) Position hub half containing air valve on a bench. Place serviceable tire and O-ring on hub, with red spot in tire aligned with air valve.

3) Lubricate threads of bolts and bearing surfaces of nuts, bolt heads, and washers, with an anti-seize compound (MIL–T–5544A).

4) Ensure O-ring is in position and install other half of hub.

Note Ensure that the “L” stamping (which indicates the light side) on the wheel halves are aligned opposite approximately 180 degrees apart.

5) Install bolts, washers and self-locking nuts, and tighten to torque value stamped on wheel.

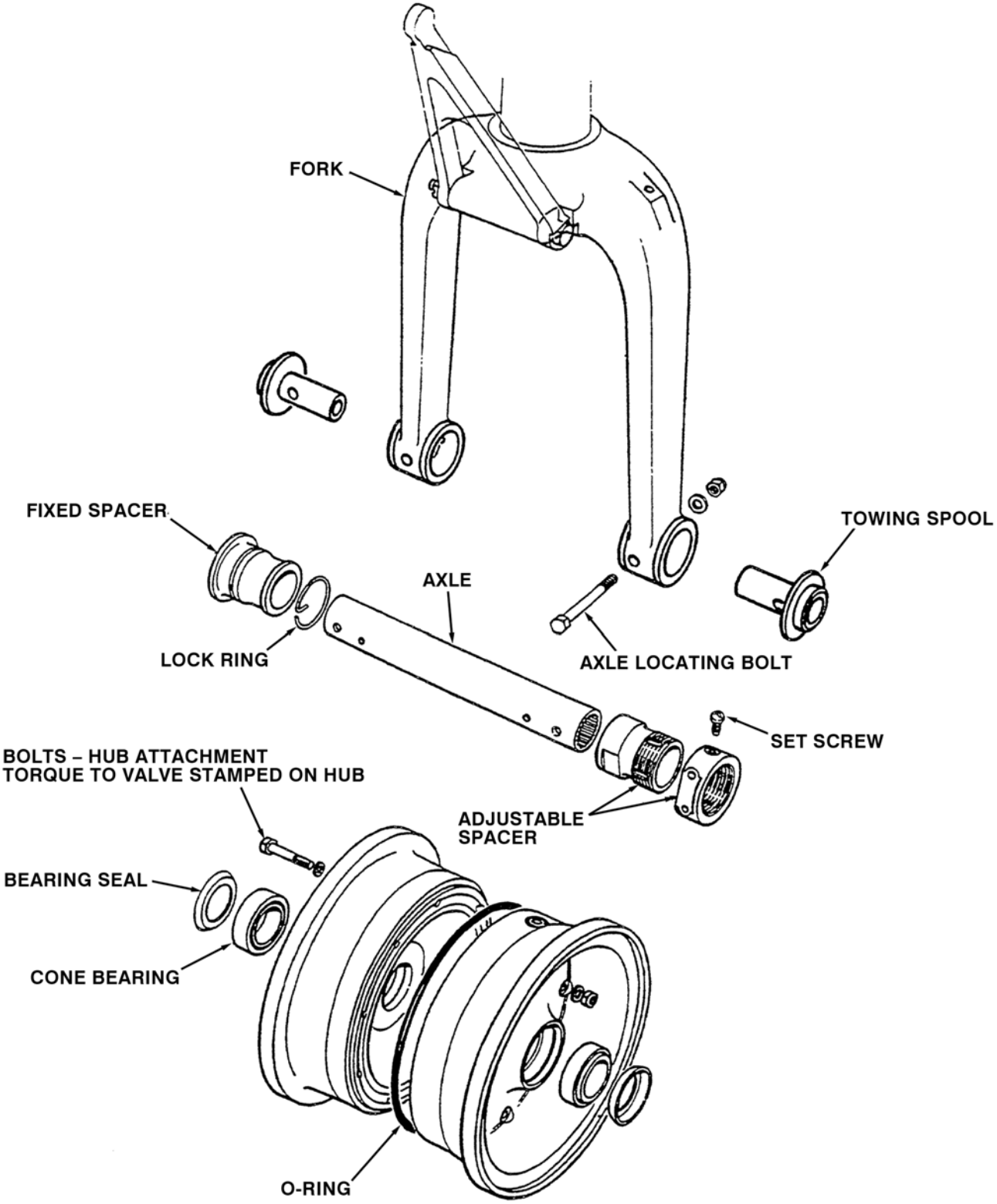
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6) Install core in air valve and inflate tire to correct pressure. (Refer to 12-10-20, Landing Gear – General Data).

Note It is recommended that wheel balance is checked when a new tire is installed on hub. The following method of balancing is recommended: check static balance, and add adhesive strip weight as necessary, applying 1/2 of required weight to each side of the hub.

7) Apply a 1” x 3” creep mark using a suitable contrasting coloured paint between tire and rim.

8) Install wheel on aircraft.



Nose Wheel Installation
Figure 1

3. Inspection/Check

A. Inspect Nose Wheel Hub

1) Remove nose wheel and nose wheel tire (refer to Para 2.A and Para 2.C).

2) Clean hub halves with dry cleaning solvent (P–S–661 or equivalent) and dry with compressed air or a lint-free cloth.

3) Inspect hub halves for cracks and breaks using the fluorescent penetrant or an approved dye penetrant method, noting in particular the bead seat, tubewell, and web at junction areas. Replace hub if cracks or chipping of rims is evident.

Note Scratches, nicks and other surface blemishes on hub halves can be removed by sanding with emery cloth removing as little metal as possible. Polish the repaired surface and repaint the repaired areas. Use one coat of Chromated Epoxy Primer to MIL–PRF–23377 on packing groove and mating surfaces of hub halves. Use two coats of Chromated Epoxy Primer to MIL–PRF–23377 and one finish coat of aluminum lacquer (MIL–L–7178) for other areas.

4) Inspect bearing cups and replace if damaged.

5) Install tire to nose wheel and install nose wheel (refer to Para 2.B and Para 2.D).

Nose Wheel (Mod S.O.O. 6048 and Mod 6/2183) – Maintenance Practices

1. General

The nose wheel is carried on the axle mounted in the fork of the nose gear, and is of the split hub type to facilitate removal and installation of tubeless tires. An O-ring is installed between the two halves of the hub to provide an airtight housing for the tire. The nose wheel is fitted with a 36 x 13.00 x 12 tire. (For STD refer to 32-40-61, Nose Wheel – General Data).

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Nose Wheel

1) Jack aircraft until nose wheel is clear of ground. (Refer to 07-00-00, Lifting and Shoring – General Data).

2) Remove and retain nuts, clamps, washers and bolts securing axle to fork.

3) Remove and retain the two lock screws locking retaining nut to adjustment spacer.

4) Hold adjustment spacer with wrench, and slacken retaining nut.

5) Support nose wheel and withdraw axle from fork, spacers and nose wheel. Lower nose wheel and retain spacers.

6) Remove snap ring bearing seals and cone bearings from inboard side, snap ring shim outboard grease bearing, spacer and bearing from nose wheel.

7) Suitably protect nose wheel bearings from contamination.

B. Install Nose Wheel

1) Ensure nose wheel bearing housings are clean, then install snap ring cone bearings, shim, spacer, bearing seals and snap ring, as per Para 2.A.6).

2) Install nose wheel discs with plug opposite valve.

3) Lightly lubricate axle with grease (MIL–PRF–81322), support nose wheel in fork, position spacer, and adjustment spacer with retaining nut, align locating bolt holes with holes in fork and insert axle through fork, wheel and spacers.

Note Ensure that adjustment spacer is installed so that retaining nut is adjacent to left leg of fork.

4) Install axle locating bolts, clamps, washers and nuts. Use washers to a maximum thickness of 0.130 inch between clamp and fork hub as required to mate flats. Coat bolts with corrosion preventive compound MIL–PRF–16173 Grade 2 (Shell CG929) prior to installation.

5) Extend adjustable spacer until wheel bearings begin to bind, then back off the next screw locating slot. Insert lock screws and lockwire.

6) Lower aircraft to ground.

7) Check tire for correct inflation. (Refer to 12-10-20, Landing Gear – General Data).

Note If installation of nose wheel on aircraft is part of gear configuration change then do step as per Para 2.B.8).

8) Do procedure for gear configuration change as per 32-10-11, Main Gear Leg – General Data, Para 2.C.

C. Remove Nose Wheel Tire (with tube)

1) Remove wheel/tire assembly from aircraft. (Refer to Para 2.A).

2) Release all pressure from tire by slackening air valve core.

3) Remove the valve stem bushing from wheel assembly and valve core.

4) Remove valve core completely from wheel assembly.

5) Using tire demounters, wooden wedges, or an approved tire machine, separate edges of tire from wheel hub beading.

6) Remove bolts, washers and self-locking nuts, holding hub halves, and remove O-ring and tire.

7) Carefully separate hub halves and remove o-ring and tire and tube. Check O-ring for serviceability and discard if damaged.

D. Install Nose Wheel Tire (with tube)

1) Check hub for scratches, scores and cleanliness. Clean tire area of hub with a dry cleaning solvent. Dust area with french chalk (talc).

2) Partially inflate tube and position in tire. Align yellow stripes on the tube base with red dot on the tire side wall.

3) Position hub half containing air valve on a bench. Place serviceable tire and O-ring on hub, with red spot in tire aligned with air valve. Position valve stem through valve hole and secure with valve stem bushing. Ensure valve stem bushing is snug.

4) Lubricate threads of bolts and bearing surfaces of nuts, bolt heads, and washers, with an anti-seize compound (MIL–T–5544A).

5) Ensure O-ring is in position and install other half of hub.

Note Ensure that the “L” stamping (which indicates the light side) on the wheel halves are aligned opposite approximately 180 degrees apart.

6) Install bolts, washers and self-locking nuts, to secure hub halves and tighten bolts to torque value stamped on wheel hub.

Note Ensure NAS1149F0663P washer is installed under counter sunk washer (MS2002C6) as per Figure 1.

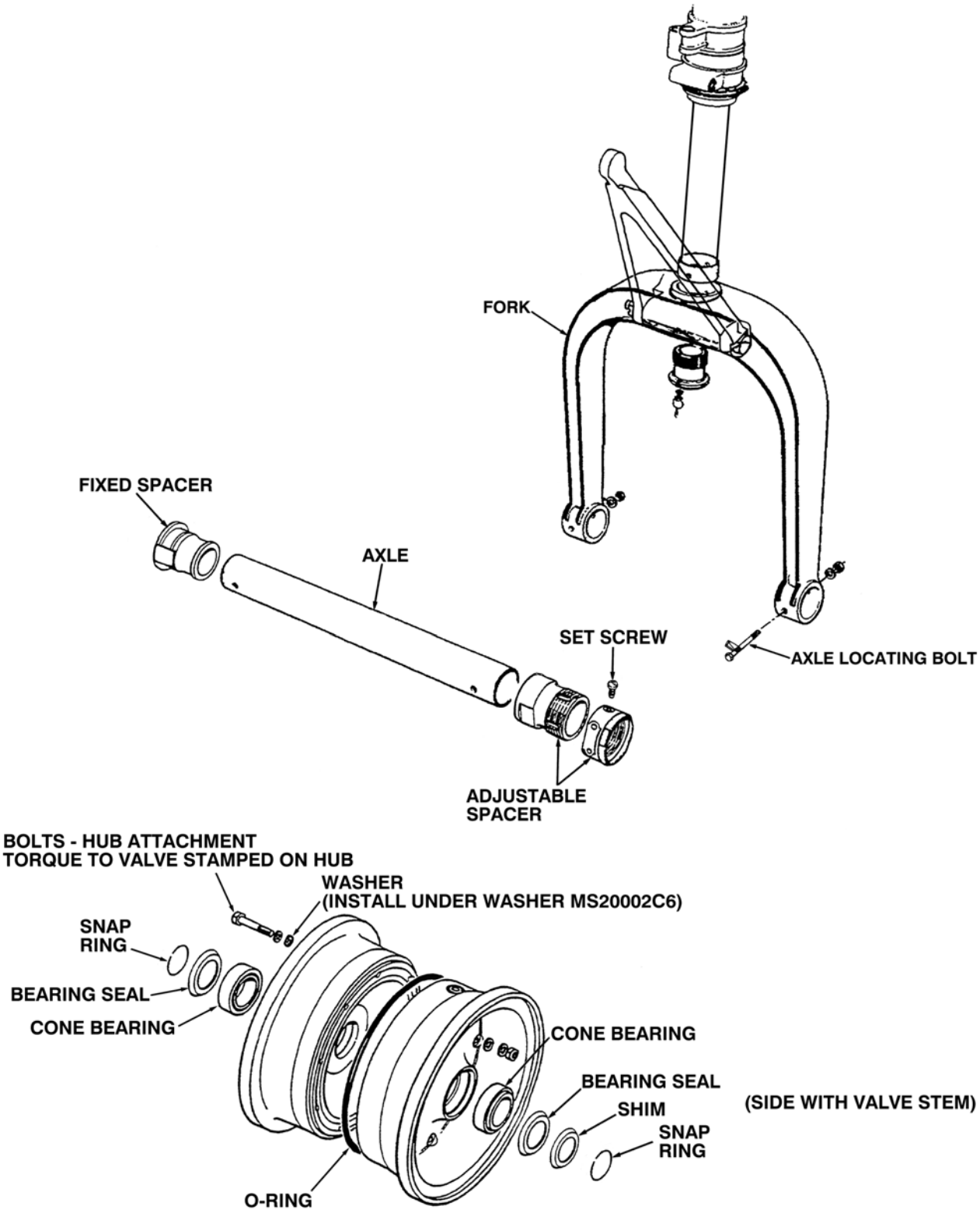
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7) Inflate tire to correct pressure. (Refer to 12-10-20, Landing Gear – General Data).

Note It is recommended that wheel balance is checked when a new tire is installed on hub. The following method of balancing is recommended: check static balance, and add adhesive strip weight as necessary, applying 1/2 of required weight to each side of the hub.

8) Apply a 1” x 3” creep mark using a suitable contrasting coloured paint between tire and rim.

9) Install wheel on aircraft.



Nose Wheel Installation (Mod S.O.O. 6048 and Mod 6/2183)
Figure 1

3. Inspection/Check

A. Inspect Nose Wheel Hub (Mod S.O.O. 6048 and Mod 6/2183)

1) Remove nose wheel and nose wheel tire (refer to Para 2.A and Para 2.C).

2) Clean hub halves with dry cleaning solvent (P–S–661 or equivalent) and dry with compressed air or a lint-free cloth.

3) Inspect hub halves for cracks and breaks using the fluorescent penetrant or an approved dye penetrant method, noting in particular the bead seat, tubewell, and web at junction areas. Replace hub if cracks or chipping of rims is evident.

Note Scratches, nicks and other surface blemishes on hub halves can be removed by sanding with emery cloth removing as little metal as possible. Polish the repaired surface and repaint the repaired areas. Use one coat of Chromated Epoxy Primer to MIL–PRF–23377 on packing groove and mating surfaces of hub halves. Use two coats of Chromated Epoxy Primer to MIL–PRF–23377 and one finish coat of aluminum lacquer (MIL–L–7178) for other areas.

4) Inspect bearing cups and replace if damaged.

5) Install tire to nose wheel and install nose wheel (refer to Para 2.D and Para 2.D).

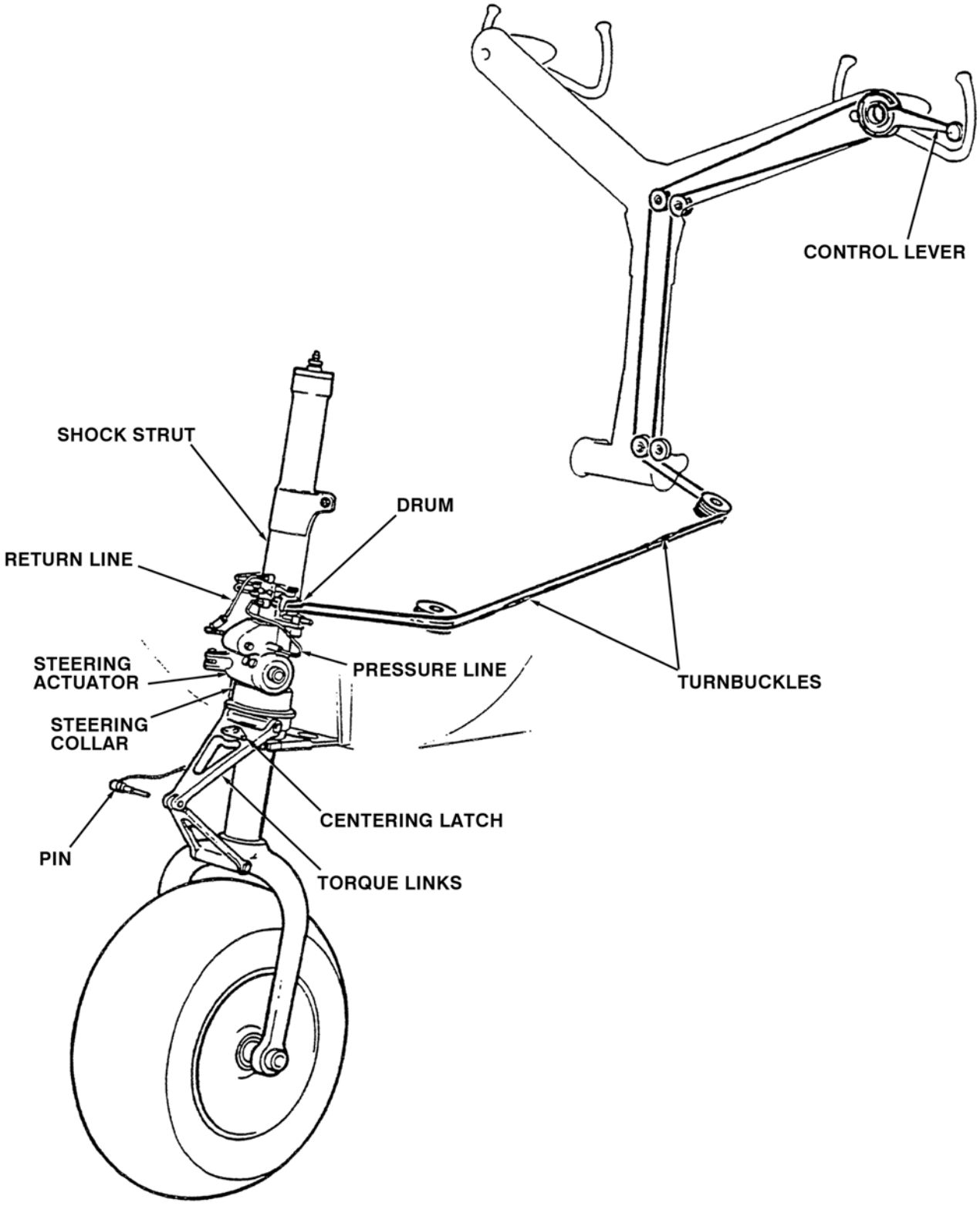
Nose Wheel Steering System – General Data

1. General

Refer to Figure 1 for details.

The nose wheel steering system consists of a hydraulic steering actuator and steering control system. The steering actuator is mounted on the nose gear shock strut and is connected by a link rod to the steering collar and torque links. Hydraulic lines from the steering actuator connect with the main hydraulic system. Steering control comprises a lever and quadrant, which operates a cable and pulley system to rotate a drum and turn a steering valve in the steering actuator. The steering system control lever is located on the pilot's control column. A latch assembly on the upper torque arm will retain the nose wheel in central position in event of hydraulic failure during flight. The latch assembly is disengaged on the ground on compression of nose gear strut.

A luminous tape nose wheel position indicator to indicate the nose wheel neutral reference position. The indicator consists of a luminous tape strip on the steering control lever hub and the control column cover.



Nose Wheel Steering System
Figure 1

2. Servicing

A. Bleed Nose Wheel Steering System

1) Bleeding the nose wheel steering system need only be accomplished if castoring is not smooth, a hydraulic line or component has been replaced, or if the hydraulic system has been dismantled.

2) Bleeding can be carried out with the aircraft on the ground or in the jacked-up position, as follows:

a) Aircraft on ground:

Position aircraft nose wheel on greased plates or similar arrangement. Ensure parking brake is on, or main wheels are securely chocked. Open nose baggage compartment.

Note Two smooth sheets of steel plate with grease applied between them, is a satisfactory platform for nose wheel steering tests. The upper plate should turn smoothly with wheel when steering is operated without any movement between the nose wheel tire and the upper surface of the plate.

b) Aircraft on jacks:

Remove quick-release pin connecting upper and lower torque links on shock strut. Move upper torque link as required, push center latch pin against spring so that latch pin is flush with lower surface and protrudes the upper surface; secure latch pin at upper surface using a hose clamp (jubilee clip). Install quick-release pin at upper and lower torque links. Open nose baggage compartment.

3) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Check wing flaps selector corresponds with position of wing flaps.

4) Set POWER SOURCE switch to EXTERNAL, MASTER switch to ON (Pre Mod 6/2314); or set EXTERNAL switch to ON (Post Mod 6/2314). Check electric motor-driven hydraulic pump charges damping accumulator and wheel brakes accumulator, and that motor cuts out when reading on pressure indicators is 1575 ± 50 psi.

5) Operate nose wheel steering fully left and right at least five times.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

7) Check hydraulic reservoir fluid level. (Refer to 12-10-15, Hydraulic System – General Data).

8) If bleeding was carried out with aircraft on jacks, remove hose clamp (jubilee clip) from latch pin at upper torque link. Lower aircraft to ground unless required on jacks for other work. Close nose baggage compartment.

9) If bleeding was carried out with aircraft on the ground, remove greased plates. Close nose baggage compartment.

3. Adjustment/Test

A. Rig Nose Wheel Steering System

1) Rigging of the nose wheel steering system can be accomplished, with the aircraft on the ground or in the jacked-up position, refer to Para 2.A.2) for preparation method. Ensure steering actuator is correctly adjusted (refer to 32-50-11, Nose Wheel Steering Actuator – General Data).

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Check wing flaps selector corresponds with position of wing flaps.

3) Set POWER SOURCE switch to EXTERNAL, MASTER switch to ON (Pre Mod 6/2314); or set EXTERNAL switch to ON (Post Mod 6/2314). Check electric motor-driven hydraulic pump charges damping accumulator and wheel brakes accumulator, and that motor cuts out when reading on pressure indicators is 1575 ± 50 psi.

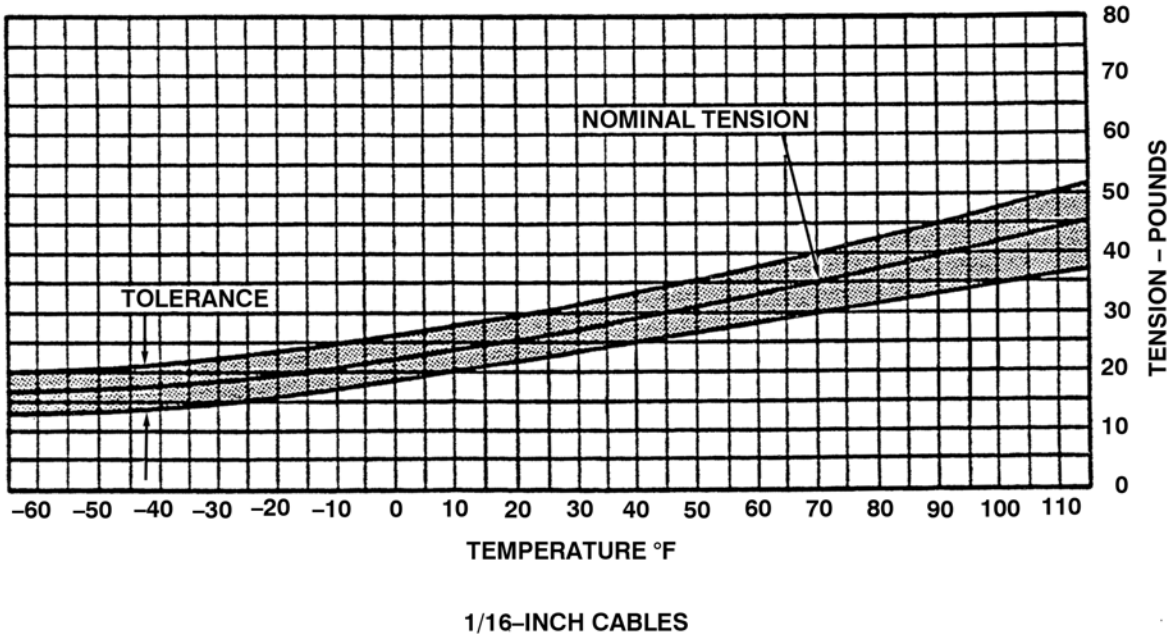
4) Set nose wheel neutral position by turning steering control lever until neutral marks on lever and control column are aligned.

5) Check nose wheel is centered (neutral). If not, adjust turnbuckles to obtain neutral position. Set cable tensions to correct tension given in Figure 2, and lock turnbuckles.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

7) If rigging was carried out with aircraft on jacks, remove hose clamp (jubilee clip) from latch pin at upper torque link. Lower aircraft to ground. Close nose baggage compartment door.

8) If rigging was carried out with aircraft on ground, remove greased plates. Close nose baggage compartment door.



Nose Wheel Steering Cable Tension Graph
Figure 2

B. Function Test Nose Wheel Steering System

1) Testing of the nose wheel steering system can be accomplished with the aircraft on the ground or in the jacked-up position, refer to Para 2.A.2) for preparation method. Ensure steering actuator is correctly adjusted. (Refer to 32-50-11, Nose Wheel Steering Actuator – General Data).

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.). Check wing flaps selector corresponds with position of wing flaps.

3) Set POWER SOURCE switch to EXTERNAL, MASTER switch to ON (Pre Mod 6/2314); or set EXTERNAL switch to ON (Post Mod 6/2314). Check electric motor-driven hydraulic pump charges damping accumulator and wheel brakes accumulator, and that motor cuts out when reading on pressure indicators is 1575 ± 50 psi.

4) Steer nose wheel left and right. From straight ahead position to full left or full right. Check operation time is 5.0 to 7.0 seconds.

5) Steer nose wheel to any intermediate position and apply torque manually to the lower leg assembly. Check there is no movement of the wheel from the selected position.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

7) Discharge hydraulic system pressure by operating wing flaps.

8) Apply torque manually to the lower leg assembly. Check that there is a steady resisting force throughout the entire steering range.

9) If function test was carried out with aircraft on jacks, remove hose clamp (jubilee clip) from latch pin at upper torque link. Lower aircraft to ground.

10) If function test was carried out with aircraft on ground, remove greased plates.

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Nose Wheel Steering Actuator – General Data

1. General

The nose wheel steering actuator consists of a steering valve, cylinder, piston and rod assembly. The actuator is hydraulically operated. Movement of the steering control lever in the flight compartment rotates the drum and changes the position of the steering valve. This action directs hydraulic pressure to one side of the piston and rod, moving the piston outboard and rotating the steering linkage and torque links which turn the nose wheel. Internal stops in the steering actuator limit the arc of travel. The actuator also acts as a shimmy damper during taxiing, take-off, and landing runs. A 10 micron in-line filter at the pressure inlet protects the internal components of the actuator against damage from contaminated hydraulic fluid.

2. Servicing

A. Service Hydraulic Filter

1) Discharge damping accumulator pressure by operating wing flaps.

2) Place container to receive spillage and remove pressure tube assembly.

3) Cut lockwire and remove filter, using Heroux special tool T–189.

4) Disassemble filter and examine filter element for damage.

5) Clean filter element by back-flushing with cleaning solvent and drying with clean compressed air.

Note Ultrasonic cleaning equipment may be used, if available, to clean filter element.

6) Examine O-rings for serviceability and replace as necessary. O-ring on filter element is P/N MS28775–010, and O-ring between end cap and housing is P/N MS28778–6.

7) Assemble filter element in housing. Install and tighten end cap.

8) Inspect packing for serviceability and replace as necessary with new packing P/N MS28778–4. Install packing and filter to actuator using special tool T–189. Ensure that direction of flow arrow on filter body points towards actuator.

9) Lockwire filter to actuator body using MS20995C20.

10) Install pressure tube assembly.

11) Bleed and function test nose wheel steering system. (Refer to 32-50-00, Nose Wheel Steering System – General Data).

3. Removal/Installation

Refer to Figure 1 for details.

A. Remove Nose Wheel Steering Actuator

Note Removal can be carried out with aircraft on the ground or in the jacked-up position.

1) Prepare aircraft as follows:

a) Aircraft on ground:

Position aircraft nose wheel on greased plates or similar arrangement. Ensure parking brake is on, or main wheels are securely chocked.

Note Two smooth sheets of steel plate with grease applied between them is a satisfactory platform for nose wheel actuator removal and tests. The upper plate should turn smoothly with wheel when steering is operated without any movement between the nose wheel tire and the upper surface of the plate.

b) Aircraft on jacks:

Remove quick-release pin connecting upper and lower torque links on shock strut. Move upper torque link as required, push center latch pin against spring so that latch pin is flush with lower surface and protrudes the upper surface; secure latch pin at upper surface using a hose clamp (jubilee clip). Install quick-release pin at upper and lower torque links.

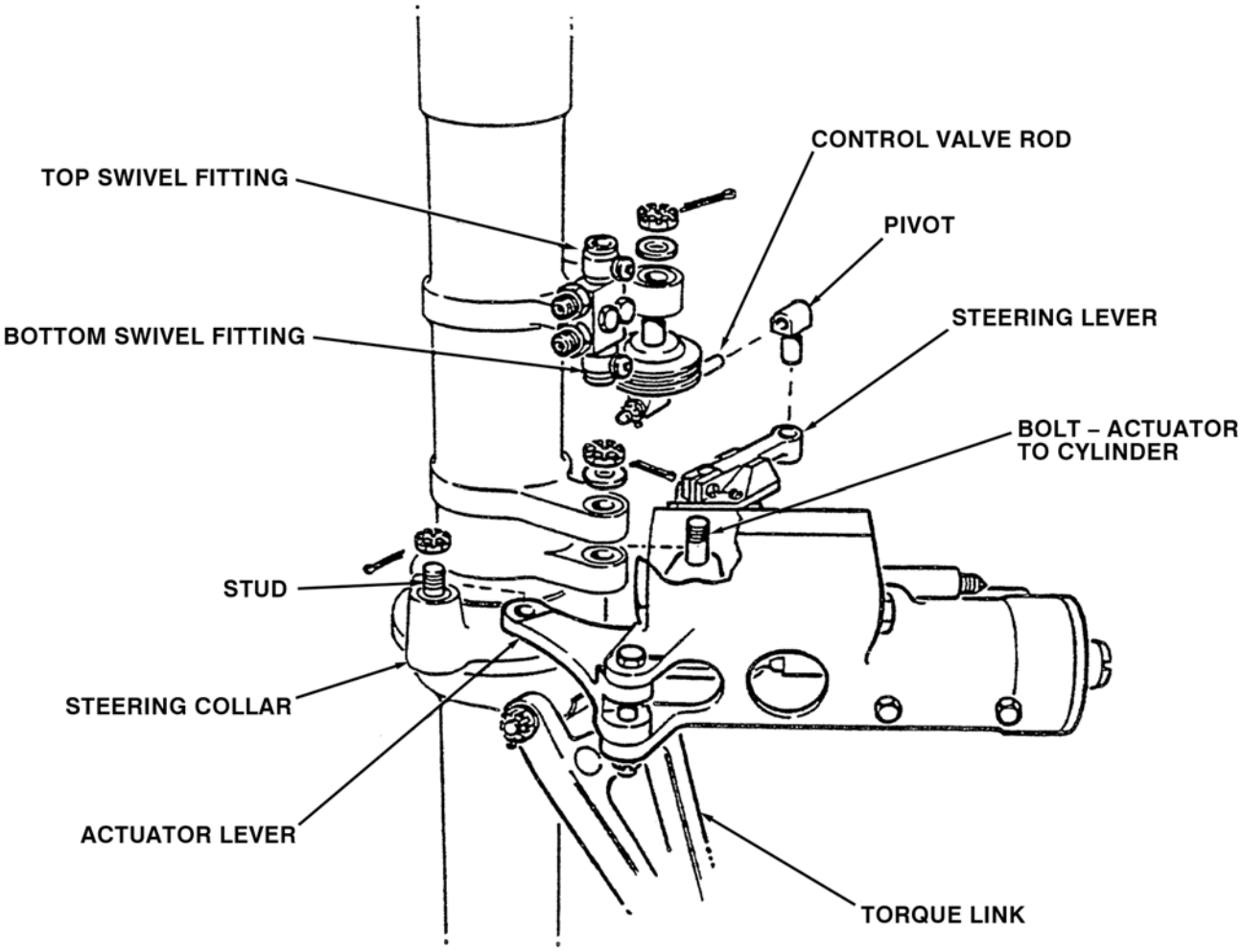
2) Discharge hydraulic system accumulator by operating wing flaps.

3) Open nose baggage compartment. Remove screens and covers as required.

4) Remove cover on underside of fuselage at nose gear leg.

5) Disconnect hydraulic return line at top swivel fitting and hydraulic pressure line at bottom swivel fitting on nose gear leg. Cap openings and lines.

6) Remove cotter pin, nut and washer attaching actuator lever to steering collar. Support actuator and remove cotter pin, nut and washer from bolt attaching actuator to cylinder. Turn leg to obtain clearance from steering collar and torque links and remove bolt. Turn leg fully to right, disengage pivot on steering lever from control valve rod, and lift actuator clear of stud on steering collar.



Nose Wheel Steering Actuator Installation
Figure 1

B. Install Nose Wheel Steering Actuator

1) Position and support actuator at nose gear leg. Turn leg fully to right, locate actuator lever on stud of steering collar, and engage pivot on steering lever with control valve rod. Align actuator with cylinder and insert bolt. Secure bolt with washer, nut and cotter pin.

2) Install washer, nut and cotter pin to stud on steering collar.

3) Connect hydraulic pressure line to bottom swivel fitting and hydraulic return line to top swivel fitting on nose gear leg.

4) Check actuator range of movement and adjust if necessary. (Refer to Para 4.B).

CAUTION **BEFORE CHECKING THE ACTUATOR FOR RANGE OF MOVEMENT, ENSURE CORRECT ADJUSTMENT OF THE STEERING ACTUATOR. INCORRECT ADJUSTMENT CAN CAUSE THE FRACTURE OF THE STEERING COLLAR DUE TO TOO MUCH RANGE OF MOVEMENT OF THE ACTUATOR LEVER (REFER TO Para 4.B).**

5) Bleed, function test, and check rigging of nose wheel steering system. (Refer to 32-50-00, Nose Wheel Steering System – General Data).

6) If installation was carried out with aircraft on jacks, remove hose clamp (jubilee clip) from latch pin at upper torque link. Lower aircraft to ground.

7) If installation was carried out with aircraft on ground, remove greased plates.

8) Install cover on underside of fuselage at nose gear leg.

9) Install screens and covers as necessary. Close nose baggage compartment.

4. Adjustment/Test

A. Bench Pressure Test

Refer to Figure 2 for details.

Equipment required:

- Hydraulic test rig, two pressure supplies 2250 psi, and flow rate 1.6 gpm (U.S.)
- Fluid MIL–H–5606
- Flowmeter to check flow rate of 1.2 to 1.6 gpm (U. S.)
- Torque wrench to 100 pound–inches

1) Filling and bleeding:

a) Remove two test port plugs, and connect test ports A and B together with shorting line.

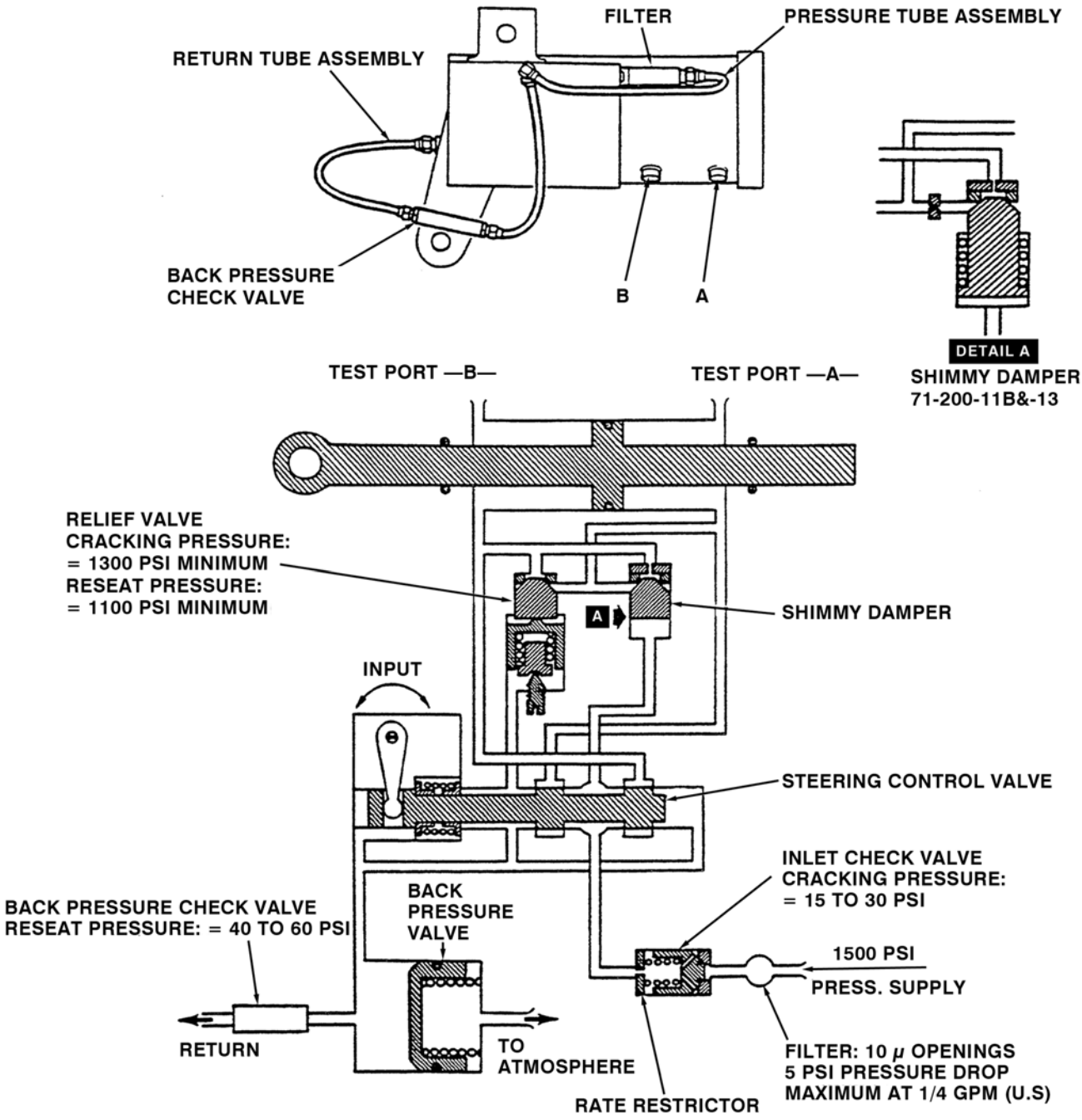
b) Connect hydraulic test rig supply line to pressure tube assembly and return line to the return tube assembly.

c) Place actuator so that input swivel assembly is facing down.

d) Apply pressure of 1500 psi, turn lever of input swivel assembly fully in one direction and hold for 10 seconds minimum. Turn lever fully in opposite direction and hold for 10 second minimum.

e) Repeat Para 4.A.1).d) a minimum of five times.

f) Place actuator so that input swivel assembly is facing up, then repeat Para 4.A.1).d), and Para 4.A.1).e).



Test Ports and Schematic Hydraulic Diagram
Figure 2

2) Cracking pressure, inlet check valve:

   a) Connect hydraulic test rig supply line to pressure tube assembly. Remove test port plugs and leave test ports open.

   b) Turn lever of input swivel assembly fully in either direction and gradually increase supply pressure until a small steady flow is obtained at test port. Check that supply pressure is 15 to 30 psi.

3) Reseat pressure, back pressure check valve:

   a) Open pressure tube assembly to atmosphere. Connect hydraulic test rig supply line to test port A.

   b) Place actuator so that input swivel assembly is facing up.

   c) Turn lever of input swivel assembly fully counterclockwise (looking from above). Raise supply pressure until a steady flow is obtained at return tube assembly. Decrease supply pressure gradually until flow stops. Check supply pressure is 40 to 60 psi at reseat.

   d) Maintain reseat pressure. After 3 minutes, external leakage must not exceed five drops per minute at each port.

4) Relief valve and shimmy damper:

   a) Place lever of input swivel assembly in neutral position (control valve centered). Connect hydraulic test rig supply line to pressure tube assembly and apply pressure of 1500 psi.

   b) Connect test port A to a regulated supply line. Raise regulated pressure until flow is increased over leakage flow at test port B. Cracking pressure must not be less than 1300 psi.

   c) Lower regulated pressure at test port A gradually until flow at test port B is reduced to leakage flow. Reseat pressure must not be less than 1100 psi.

   d) Connect flowmeter at test port B. Increase regulated pressure at test port A until 1.2 gpm (U.S.) is obtained at test port B. Check that pressure does not exceed 1800 psi.

   e) Connect regulated supply line to test port B and flowmeter to test port A. Repeat tests similar to Para 4.A.4).a) through Para 4.A.4).d).

   f) Release pressure supply at pressure tube assembly and disconnect supply line. Leave pressure tube assembly open to atmosphere to permit bypass through shimmy damper orifice.

   g) Apply regulated pressure of 1000 psi at test port B. Flow at test port A must be 1.4 to 1.6 gpm (U.S.).

5) Control valve:

   a) Connect test ports A and B together with shorting line. Connect hydraulic test rig supply line to pressure tube assembly. Place lever of input swivel assembly in neutral (control valve centered).

   b) Apply pressure of 1500 psi. Leakage flow at return tube assembly must be 12 to 20 cc per minute.

   c) Torque required at lever of input swivel assembly to crack control valve must not exceed 65 pound–inches, and the flow must stop on reduction to 40 pound–inches.

   d) Disconnect test ports A and B and install packing and plug at each port. Connect hydraulic test rig return line to return tube assembly.

   e) With supply pressure at 1500 psi, apply a torque of 75 pound–inches; extension or retraction rate must be 7 ± 1 seconds for a complete stroke.

f) Position piston at center of stroke. With control valve centered, piston creep rate must be less than 0.030 inch per minute.

6) Proof pressure:

a) With plugs installed at test ports A and B and return tube assembly capped, apply a pressure of 2250 psi at pressure tube assembly at a rate of not more than 25,000 psi per minute (400 psi per second).

b) Turn lever of input swivel assembly slowly clockwise and hold for two minutes.

c) Turn lever of input swivel assembly slowly counterclockwise and hold for two minutes.

d) There must be no leakage or permanent set.

Note A slight wetting insufficient to form a drop is acceptable.

e) Release pressure completely and then apply 40 psi at pressure tube assembly. There must be no leakage.

f) Release pressure and disconnect actuator from test rig. Cap pressure and return tube assemblies unless actuator is to be installed immediately.

B. Adjust Actuator Range of Movement

Equipment required:

- Heroux special tool T–186
- Heroux special tool T–194

1) Adjustment of actuator range of movement can be accomplished with the aircraft on the ground or in the jacked-up position; refer to Para 3.A.1) for preparation method.

2) Open nose baggage compartment. Remove screens and covers as required.

3) Remove lockwire from lock key of jam nut on piston rod end. Disengage lock key and loosen jam nut using special tool T–194.

4) Turn leg assembly fully to the right. Turn piston to adjust rod end so that with piston firmly against internal stop, a minimum clearance of 0.010–inch and a maximum clearance of 0.020–inch is obtained between the actuator lever and the outer cylinder of the shock strut. Tighten jam nut on piston rod end using special tool T–194. Engage lock key, and lockwire, using MS20995C20.

5) Turn leg assembly fully to the left. Check that with piston firmly against internal stop (end cap on manifold), a minimum clearance of 0.010–inch and a maximum clearance of 0.020–inch is obtained between stud lug on steering collar and lugs on outer cylinder. If necessary, cut lockwire, disengage lock key and nut from end cap, and adjust end cap using special tool T–186 to obtain clearance. Finger tighten nut on end cap, engage lock key and lockwire using MS20995C20.

6) If adjustment was carried out with aircraft on jacks, remove hose clamp (jubilee clip) from latch pin at upper torque link. Lower aircraft to ground.

7) If adjustment was carried out with aircraft on ground, remove greased plates.

8) Install screens and covers as necessary. Close nose baggage compartment.

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Wheel-Ski Indication (Mod S.O.O. 6001) – General data

1. General

Refer to Figure 1 for details.

Wheel-ski position indication is provided by a group of six lights mounted on the selector lever panel (refer to 32-30-00, Extension and Retraction (Wheel-Ski) (Mod S.O.O. 6001) – General Data) which are controlled by limit switches on each ski assembly. The three upper lights indicate skis "up" and the lower three skis "down". Each pair of lights are located on the panel in relation to their respective ski position (for example, the two center lights provide nose ski indication). Two limit switches installed on each ski are actuated by the ski latch lever, one when the ski is "down" and the other when the ski is "up". The power supply for the indicator lights is derived from the 28 Volt left DC busbar, through a 5-ampere GEAR IND (H7) circuit breaker located on the avionics circuit breaker panel, and a relay operated by the CAUTION LIGHTS DIM/BRIGHT/TEST switch, refer to 33-10-11, Caution Lights Dimming Control Box – General Data. When the skis are operated, the relevant position limit switches are actuated to provide a ground and complete the electrical circuit for their associated lights. When the CAUTION LIGHT dimmer switch is selected to DIM, the relay changes over to connect a resistance in series with the lights, which are connected in parallel. For electrical schematic refer to PSM 1-64-2W, Section 32-61-00.

2. Adjustment/Test

Note The access panels from part of the stressed structure of the ski and must be reinstalled as soon as adjustments are completed.

A. Adjust Ski Up Limit Switch

Refer to Figure 1 for details.

1) Remove forward access panel. Check that ski is in ski-up position and that latch pin is fully extended and latched.

2) Loosen up-lock microswitch mounting screws.

3) Move switch toward latch pin until an audible click indicates that switch is actuated. Move switch an additional 1/32 inch approximately (to provide overtravel for consistent switch actuation in service) but ensure that latch pin does not come in direct contact with switch body.

4) Tighten switch mounting screws.

5) Using continuity tester, function check switch actuation by operating the ski hydraulically in and out of lock. If ski is installed on aircraft, check indicator lights in flight compartment. Ski UP/LAND light should come on when latch pin engages locking hole and light should go out when latch pin comes out of lock.

6) Reinstall access panel.

B. Adjust Ski Down Limit Switch

Refer to Figure 1 for details.

1) Remove forward access panel.

2) Loosen microswitch mounting screws.

3) Actuate ski to ski down position.

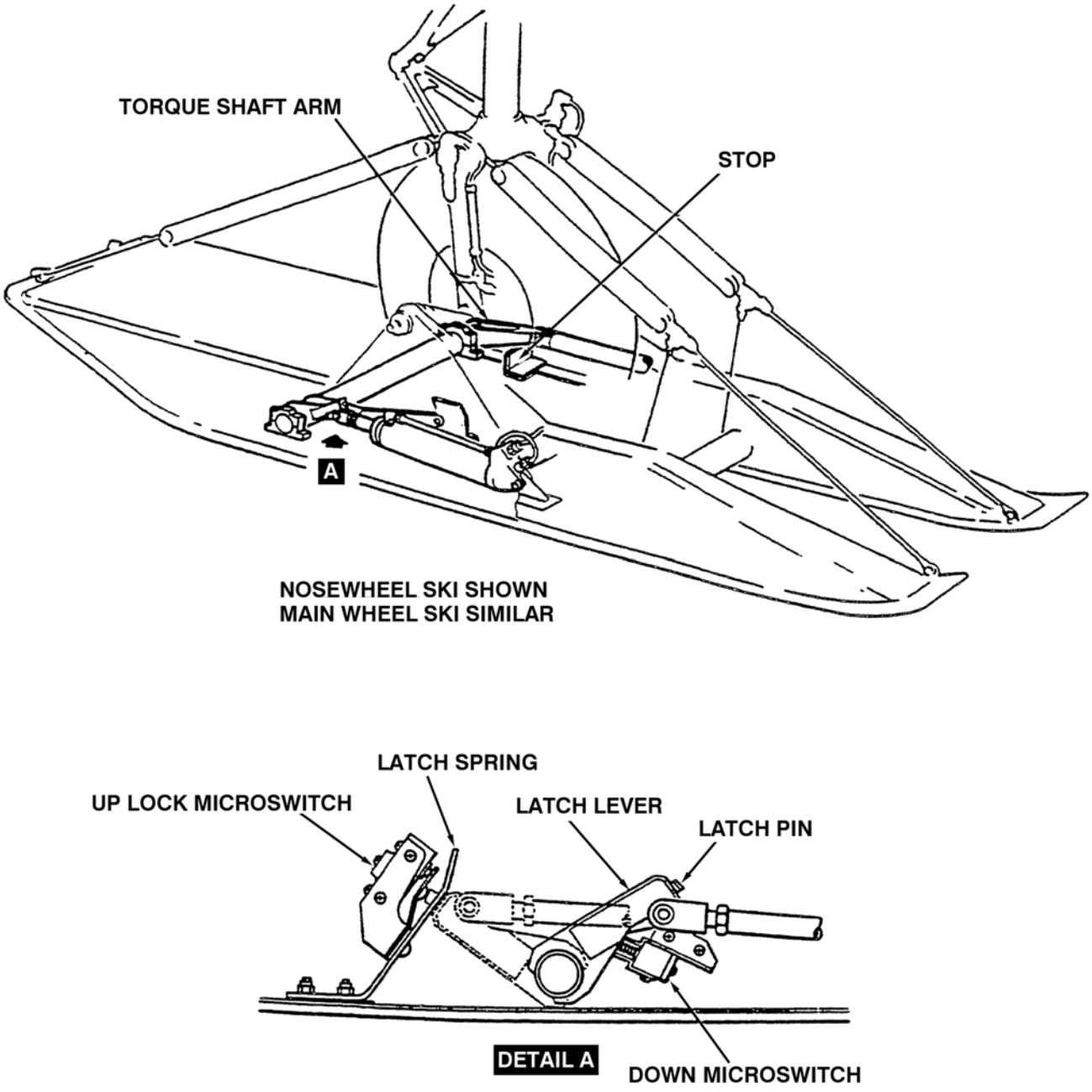
4) Move switch and actuator toward latch lever arm until an audible click indicates that switch is actuated.

5) Final adjustment of switch must result in actuation of switch when torque shaft arms are within 0.2 inch of their stops.

CAUTION **FAILURE TO COMPLY WITH THE 0.2 INCH ADJUSTMENT REQUIREMENT COULD RESULT IN PREMATURE OPERATION OF SKI POSITION INDICATOR LIGHT (BEFORE SKI HAS REACHED THE GEOMETRIC LOCK).**

6) Function check with continuity tester if skis are off aircraft. Function check by means of indicator lights in flight compartment if skis are on aircraft.

7) Reinstall access panel.



Limit Switch Adjustment
Figure 1

Supplementary Gear – General Data

1. General

The supplementary gear consists of a leaf spring type tail bumper. The tail bumper leaf spring is installed at the aft end of the rear fuselage, and is attached to the fuselage structure by an axle fitting. A rubber bumper pad, bonded to the center of the spring, together with the leaf spring, absorbs shock on contact with the ground. The front end of the leaf spring is shrouded by a cover.

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CHAPTER 33

LIGHTS

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CHAPTER 33 – LIGHTS

LIST OF EFFECTIVE PAGES

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Lights – General Data

1. General

The aircraft lighting system includes interior and exterior illumination. All lights operate from the aircraft 28 Volt DC power supply system. Individual circuits are protected by thermal push-pull circuit breakers and powered from a multiple bus network. (Refer to 24-30-00, DC Generation – General Data).

The landing lights are high intensity discharge (xenon) (Pre Mod 6/2285) or LED (Post Mod 6/2285). The wing inspection lights are incandescent (Pre Mod 6/2140). The majority of the other interior and exterior lighting is provided by LED (light emitting diode) lamps. All LED lamps are “high reliability” specification, which means that failure of a single LED in a lamp array will not result in failure of the lamp unit.

Pulse lighting capability is built into the landing light controller assembly. Aircraft equipped with optional TCAS I and TCAS II have a pulse light switch that enables manual or automatic (in response to a traffic advisory) activation of the pulse light system.

Controls for all instrument panel lighting are mounted on the control yoke, immediately forward of the reversion controller. There are two dimmers and one small toggle switch.

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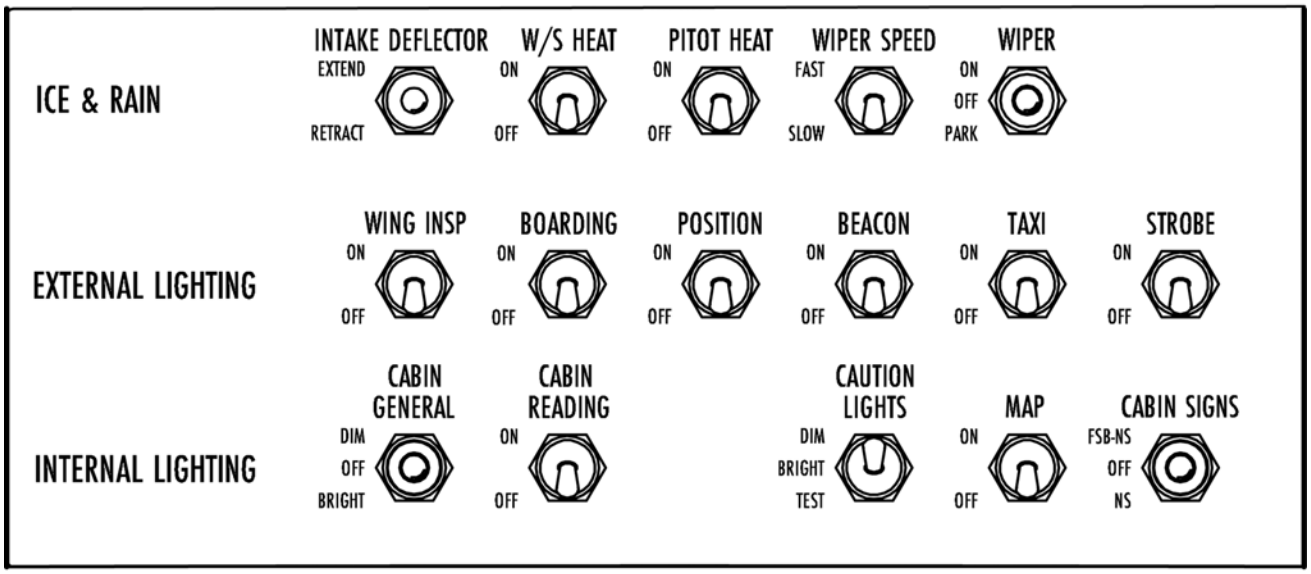
Flight Compartment Lights (Mod 6/2028) – General Data

1. General

The flight compartment lights include the dome light, aisle light on control column pedestal, pilot and co-pilot map lights, primary instrument panel lighting, secondary instrument panel flood lighting, foot well lighting and caution lighting.

The interior lighting system consists of six overhead general lights, twenty passenger reading lights, an entrance light, forward and aft baggage compartment lights, and cabin sign lights. All lights are LED (light-emitting diode).

All light switches are located on the middle row (for external lighting) or the bottom row (for internal lighting) of the lower left pilot sub-panel. The landing light switches are located on the overhead switch panel.



External and Internal Lighting Switches
Figure 1

2. Panel and Instrument Lighting

A. Panel Lighting

General instrument panel illumination is provided by LED arrays. Specific instruments and panels are lit internally or illuminated by LED post lights. For further detail refer to PSM 1-64-2W, Chapter 33.

B. Flight Compartment Dome and Aisle Lights

A flight compartment dome light is installed on the center aft overhead console and is supplied power from the hot battery bus. The dome light employs both clear and coloured LEDs to emit light at a colour temperature similar to daylight.

An additional aisle light is installed on the aft face of the control column pedestal, below the data entry keypad, for the purpose of illuminating the well below the two pilot seats.

Each of these lights have their own switches.

C. Flight Compartment Map Lights

Two flight compartment map lights are mounted on the center aft overhead console, just forward and to each side of the dome light. The map lights have a concentric control for dimming. A toggle switch labeled MAP, controlling master power to both map lights is provided on the left pilot sub-panel.

D. ESIS Bezel Lighting

The ESIS bezel light is dimmable through R2 INSTRUMENT dimmer control when power is on the aircraft, and the ESIS power switch is set to ARM. When power is removed from aircraft, and ESIS power switch is set to ARM, the ESIS power relay in the junction box is disengaged which causes the ESIS bezel light to remain at full brightness, and cannot be dimmed.

E. Control and Electrical Supply

1) Panel Lights

The LPS 99–1128 (PS1) power supply supplies 5V and 28V variable voltage to panel lights, avionics instruments, as well as the static valve lighting, and standby compass lighting. The unit is powered by a five amp breaker on the left bus, and dimming is controlled by a 10k Ω rheostat on the Control Column Dimming Panel. The PS1 supplies 0-5 volt variable voltage to the Fuel Selector Panel overlay, the Overhead Switch Panel overlay, the Left Hand Panel overlay, the Right Hand Panel overlay, the Cockpit Dimmer Panel overlay, and TB13. It supplies 0-28V variable voltage to avionics instruments, the standby compass, and TB13. It also supplies 0-28V variable voltage to the trim panel overlay, the flap selector post lights, and the static valve illumination through switch S107 TRIM PANEL LIGHT switch. The PS1 system also includes a 9-pin maintenance port, which is utilized by any PC computer equipped with an RS422 adapter unit maintenance or set up. The display revisionary panel is supplied by the 0-28V output, through R101, a 150 Ω drop resistor, so as to supply 14V max to panel.

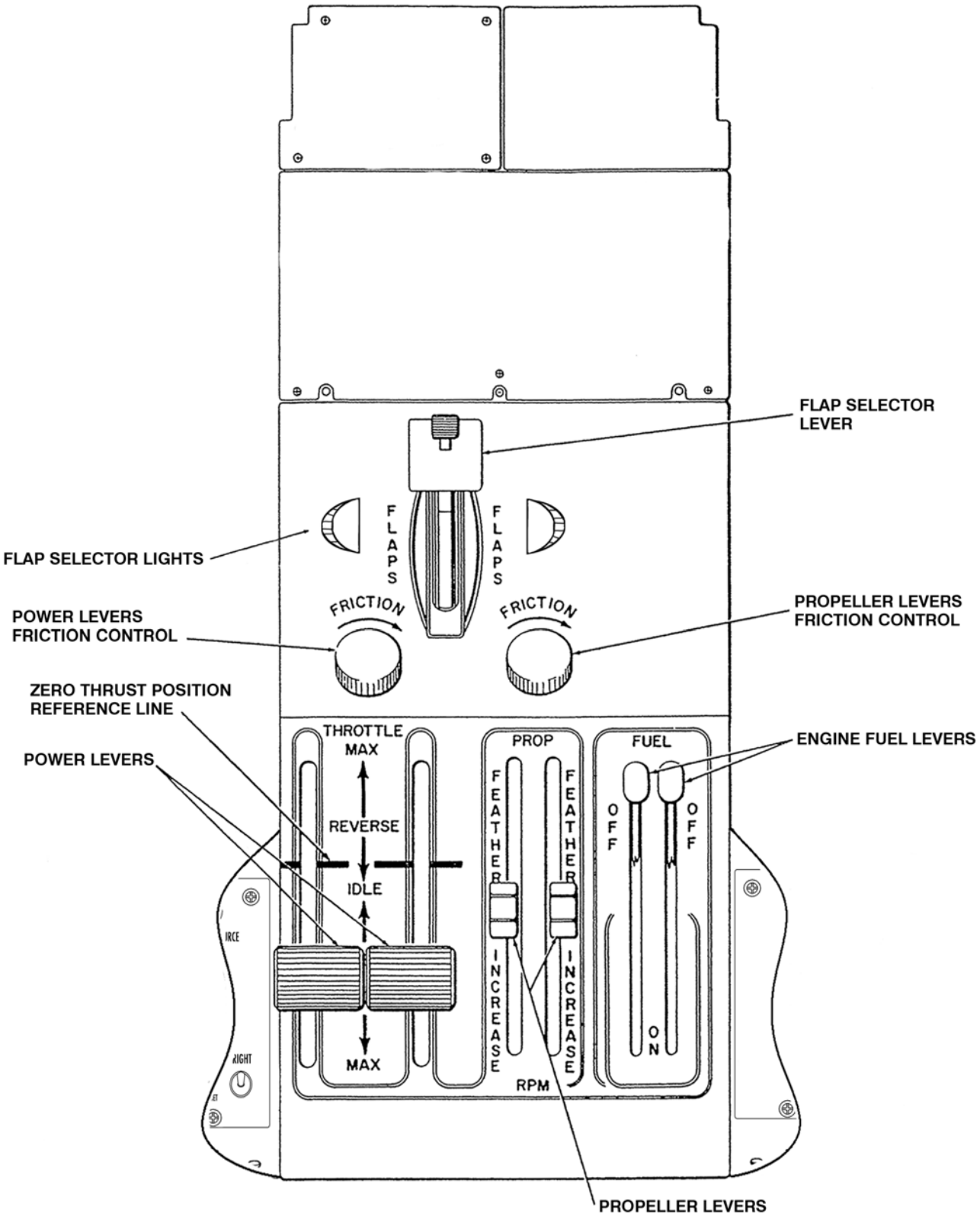
2) Flood/Thunderstorm Lights

The LT–50 () or LT–50(D) (PS2) power supply supplies 0-28V variable voltage to the avionic c/b panel illumination, eye-height reference lighting, foot lights, and instrument panel flood lights. The unit is powered by a five amp breaker on the left bus, the aisle light is powered by the same circuit breaker however, it is dimmed by its own switch (S40) AISLE LIGHT switch. Dimming of PS2 is controlled by R1 FLOOD, a 5k Ω rheostat on the Control Column Dimming Panel. Switch (S42) FOOTWELL activates foot lights and avionic circuit breaker panel illumination. Instrument panel flood lights, eyeheight reference device is activated with switch S41 OFF EYE REFERENCE light located on the window post.

3. Caution Lights Dimming/Test Switch

The Caution Lights Dimming Control Box forms part of the master warning and master caution system, and consists of a remote mounted light dimmer control box which is capable of controlling up to 12 independent lamps with full test functions.

A three-position switch labelled CAUTION LIGHTS with positions DIM, BRIGHT, and TEST is located on the lower left sub-panel. The momentary TEST position of the switch permits a simultaneous check of Autofeather, Master Caution, Master Warning, Stall, and the Fire Caution Lights. The Fire lights can also be tested with their own dedicated switch. The test and intensity circuit is protected by a CAUTION LTS NORMAL circuit breaker on the avionics circuit breaker panel. The DIM and BRIGHT positions provide alternative levels of lighting brilliance for the MASTER CAUTION and MASTER WARNING lights, the autofeather system SELECT and ARMED indicator lights, and the stall warning light, and the engine FIRE WARNING lights. In the event of failure of the control system used for caution light dimming, a reversionary, non-dimming mode can be selected by moving the CAUTION LIGHTS DIMMING switch on the lower right sub-panel from NORMAL to BYPASS. It is powered by the CAUTION LTS BYPASS circuit breaker on the avionics circuit breaker panel.



Overhead Console – Typical
Figure 2

4. Adjustment/Test

A. Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Pull the following circuit breakers:

1) MAP LIGHTS located on the main circuit breaker panel.

2) CLOCK AND DOME LIGHT on the rear circuit breaker panel.

3) PANEL LIGHTS located on the avionics circuit breaker panel.

4) FLOOD LIGHTS located on the avionics circuit breaker panel.

C. Test Map Lights

1) Reset the MAP LIGHTS circuit breaker in the main circuit breaker panel.

2) Select the MAP switch to ON.

3) Pilot MAP Light

a) Select the dimmer to MAX and verify the lamp illuminates.

b) Cycle the MAP switch to the ON and OFF positions. Verify the lamp goes ON/OFF with the switch position.

c) Operate across dimming range. Verify the brightness of lamp varies, smooth operation and no flickering.

d) Dimmer to OFF. Verify the lamp extinguishes.

4) Repeat Para 4.C.3) for the Co-pilot MAP light.

5) Select MAP switch to OFF.

D. Test Dome Light

1) Check that MASTER and POWER SOURCE switches have no effect on entrance lights operation.

Note Will not operate on External Power. Battery must be connected. Light is on the Hot Battery Bus.

2) Reset the CLOCK AND DOME LIGHT circuit breaker in the rear circuit breaker panel.

3) Operate the Dome Light switch on. Verify the lamp illuminates.

4) Operate the Dome Light switch off. Verify the lamp extinguishes.

E. Test Panel Lights

1) Select MASTER switch to OFF (Pre Mod 6/2314); or EXTERNAL and BATTERY switches to OFF (Post Mod 6/2314).

2) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Reset the PANEL LIGHTS circuit breakers on the avionics circuit breaker panel.

4) Select the INSTRUMENT dimmer to full bright (CW).

5) Verify the following lights are illuminated and illumination across the panel is adequately balanced:

- HF Control (if installed)
- Audio Panels (two)
- Flight Controller Panel
- PFD Control Panels (two)
- Display Bezels (four)
- ESIS Bezels (if installed)
- Compass
- Clock
- MFD Control Panel (one in pedestal)
- Reversionary Control Panel
- Dimmer Panel
- Fuel Selector Panel
- Overhead Panel
- LH Switch Panel
- RH Switch Panel

6) Select TRIM PANEL LT switch to ON.

7) Verify that the Trim Panel, Flap Selector post lights and the Static Valve post lights are illuminated.

8) Operate the INSTRUMENT dimmer across the dimming range. Verify the brightness of lamp varies, smooth operation and no flickering.

9) Cycle the TRIM PANEL LT switch ON/OFF. Verify the TRIM PANEL, FLAP SELECTOR post lights and the STATIC VALVE post lights illuminate accordingly.

10) Select the INSTRUMENT dimmer to dim (CCW) position. Verify the TRIM PANEL, FLAP SELECTOR post lights and the STATIC VALVE post lights dim.

11) Select TRIM PANEL LT switch to OFF.

F. Test Flood Lights

1) Reset FLOOD LIGHTS circuit breaker in the avionics circuit breaker panel.

2) Select the AISLE LT switch (Pedestal) to BRT (up). Verify the AISLE LT is bright.

3) Select the AISLE LT switch (Pedestal) to DIM (down). Verify the AISLE LT is dim.

4) Select the AISLE LT switch (Pedestal) to OFF (center). Verify the AISLE LT is extinguished.

5) Select the FLOOD light dimmer to full bright position.

6) Verify the instrument panel visor lights, six lights, are illuminated.

7) Select the FOOTWELL switch to ON/OFF. Verify the footwell lights (two) and the avionics circuit breaker post lights cycle on/off with switch position.

8) Select the FOOTWELL switch to ON.

9) Select the EYE REFERENCE LIGHT switch to ON/OFF. Verify the eye reference light cycles with the switch position.

10) Select the EYE REFERENCE LIGHT switch to ON.

11) Operate the FLOOD dimmer across the dimming range. Verify the brightness of Instrument Panel Visor, Footwell, Avionics Circuit Breaker Panel Post and Eye Reference Lamps varies, smooth operation and no flickering.

12) Select FOOTWELL switch to OFF.

13) Select the EYE REFERENCE LIGHT switch to OFF.

14) Select the FLOOD dimmer to full dim.

15) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

If there is a problem with the Panel or Footwell lights dimming function, please contact Viking Air Ltd., technical services.

G. Test Panel Light Power Supply

Note When first installing the LPS 99–1128, follow the manufacturer's installation instructions before testing the system. Without proper set-up with a computer through the maintenance port, the power supply will have the factory settings for the variable dimming.

Operating the LPS 99–1128 with factory settings and no configuration tables loaded will not damage any of the lighting fixtures. When operated with factory settings, the illumination balance will be poor across the instrument panel, most naturally, the clock and the reversionary panel will be noticeably brighter than after equipment when the LPS is set between medium and high settings.

The clock, compass and eyeheight trim panel have incandescent lamps and may appear to have a warmer colour than other cockpit lights at some brightness settings.

1) With power on bus, and CB277 PANEL LIGHTS circuit breakers pushed in, rotate R2 rheostat labeled INSTRUMENT on the control column dimming panel to bright. Ensure the fuel selector panel, overhead switch panel, left hand panel, right hand panel, standby compass, and avionics instruments are at full brightness.

2) Activate switch S107 TRIM PANEL LIGHT switch, to the ON position. Ensure the flap selector post lights, static valve illumination, and trim panel overlay illuminate at their brightest

3) Rotate R2 rheostat to dim. Ensure the fuel selector panel, overhead switch panel, left hand panel, right hand panel, standby compass, avionics instruments, flap selector post lights, static valve illumination, and trim panel overlay dim.

4) Reset all switches and controls to appropriate settings.

H. Test Flood/Thunderstorm Light Power Source

1) With power on bus, and CB279 FLOOD LIGHTS circuit breaker pushed in, rotate R1 rheostat labeled FLOOD on the control column dimming panel to bright. Ensure instrument panel flood lights are at full brightness.

2) Activate switch S42 FOOTWELL to the ON position. Ensure the foot lights, and avionic circuit breaker panel illuminations are at full brightness.

3) Activate switch S41 OFF EYE REFERENCE light to the ON position. Ensure the eyeheight reference light is at full brightness.

4) Rotate R1 rheostat to dim. Ensure the instrument panel flood lights, eyeheight reference light, foot lights, and avionic circuit breaker panel lighting has dimmed.

5) Activate S40 AISLE LIGHT switch to the BRIGHT position. Ensure the aisle light is at full brightness.

6) Activate S40 AISLE LIGHT switch to the DIM position. Ensure the aisle light has dimmed.

7) Reset all switches and controls to appropriate settings.

I. ESIS Bezel Light Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Set ESIS POWER switch to ARM.

3) Rotate R2 INSTRUMENT dimmer on control column dimming panel to full bright. Ensure bezel is at full brightness.

4) Rotate R2 INSTRUMENT dimmer control to fully dim. Ensure bezel fully dims.

5) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

6) Ensure ESIS bezel light is at full brightness, and does not dim when R2 INSTRUMENT dimmer control is rotated.

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Caution Lights Dimming Control Box – General Data

1. General

The Caution Lights Dimming Control Box forms part of the Master Warning and Master Caution system, and consists of a remote mounted light dimmer control box which is capable of controlling up to 12 independent lamps with full test functions.

2. Description

A three-position switch labelled CAUTION LIGHTS with positions DIM, BRIGHT, and TEST is located on the lower left sub-panel. The momentary TEST position of the switch permits a simultaneous check of Autofeather, Master Caution, Master Warning, Stall and Fire lights. The Fire Lights are tested using their own dedicated switch. The test and intensity circuit is protected by a CAUTION LTS NORMAL avionics breaker on the main circuit breaker panel. The DIM and BRIGHT positions provide alternative levels of lighting brilliance for the MASTER CAUTION and MASTER WARNING lights, the autofeather system SELECT and ARMED indicator lights, and the stall warning light, and the engine FIRE warning lights. In the event of failure of the control system used for caution light dimming, a reversionary, non-dimming mode can be selected by moving the CAUTION LIGHTS DIMMING switch on the lower right sub-panel from NORMAL to BYPASS. It is powered by the CAUTION LTS BYPASS circuit breaker on the avionics circuit breaker panel.

3. Adjustment/Test

A. Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Caution Lights (Bright)

1) Press the CAUTION LIGHTS to TEST and verify that the following lights illuminate:

a) AUTOFEATHER (L & R)

b) MASTER CAUTION (L & R)

c) MASTER WARNING (L & R)

d) STALL

e) FIRE (L & R)

C. Caution Lights (Bypass)

1) Pull the CAUTION LTS NORMAL circuit breaker (3N).

2) Select BYPASS on the CAUTION LIGHTS DIMMING switch.

3) Press the CAUTION LIGHTS to TEST and verify that the following lights illuminate:

a) MASTER CAUTION (L & R)

b) MASTER WARNING (L & R)

4) Operate the AUTOFEATHER switch to illuminate SELECT and verify illumination.

5) Operate the FIRE WARNING switch to illuminate FIRE (L & R) and verify illumination.

6) Operate the sensor on the wing to illuminate the STALL light.

7) Reset the CAUTION LTS NORMAL circuit breaker (3N).

D. Caution Lights (Dim)

1) Autofeather (L & R):

a) Set the CAUTION LIGHTS to BRIGHT.

b) Operate the AUTOFEATHER switch to illuminate SELECT.

c) Set the CAUTION LIGHTS to DIM and verify a reduction in brightness.

2) Master Caution (L & R) & Master Warning (L & R):

a) Set the CAUTION LIGHTS to BRIGHT.

b) Pull the DAU 2B circuit breaker to trigger the MASTER CAUTION (L & R) and MASTER WARNING (L & R).

c) Set the CAUTION LIGHTS to DIM and verify a reduction in brightness.

d) Reset the DAU 2B circuit breaker.

3) Fire (L & R):

a) Set the CAUTION LIGHTS to BRIGHT.

b) Operate the FIRE WARNING switch to illuminate FIRE (L & R).

c) Set the CAUTION LIGHTS to DIM and verify a reduction in brightness.

4) Stall:

a) Set the CAUTION LIGHTS to BRIGHT.

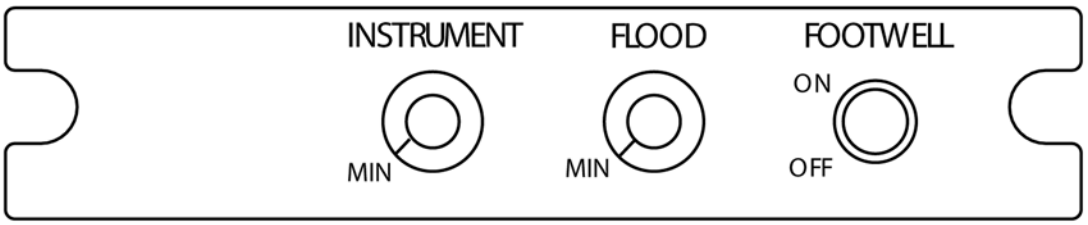
b) Operate the sensor on the wing to illuminate the STALL light.

c) Set the CAUTION LIGHTS to DIM and verify a reduction in brightness.

Instrument Panel Lighting System – General data

1. General

Instrument Panel Lighting system provides lighting to the instrument panel. The system consists of the dimming rheostat, the Lighting Power Supply (LPS), and the instrument panel lights. The instrument dimming rheostat is located on the dimming control panel mounted forward of the multi-function controller. Refer to Figure 1. The individual instrument panel lights are powered by the LPS through dedicated outputs. The output voltages of the LPS to the instrument lights are adjusted by the rheostat.



Control Column Dimming Panel
Figure 1

2. Description

A. Lighting Power Supply (LPS)

The LPS supplies 5V and 28V variable voltage to panel lights, avionics instruments, static valve, and the standby compass lighting. The dimming is controlled by a 10k Ω rheostat on the Control Column Dimming Panel. The LPS supplies 0-5 volt variable voltage to the Fuel Selector Panel overlay, Overhead Switch Panel overlay, Left Hand Panel overlay, Right Hand Panel overlay, Cockpit Dimmer Panel overlay, and TB13. The LPS supplies 0-28V variable voltage to the avionics instruments, standby compass, TB13, trim panel overlay, flap selector post lights, and the (ESIS) Alternate Static Valve. The LPS also includes a 9-pin maintenance port, which is utilized by any PC computer equipped with RS422 adapter unit maintenance or set up. The display revisionary panel is supplied by the 0-28V output through R101, a 150 Ω drop resistor, so as to supply 14V max to the panels.

The LPS output voltages are controlled by the instrument dimming rheostat. The lights that are illuminated and dimmed through the LPS are listed in Table 1 — LPS Lighting Interconnect. The LPS is located on the lower left hand tray of the forward avionics bay. The unit is accessible from the forward baggage compartment by removing the lower HIRF protection cover.

28 Volt DC power is protected by a 5 Amp PANEL LIGHTS L (P7) (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

Table 1 LPS Lighting Interconnect

LPS Pin	Signal Name	LPS Table Ref.	System (DHC–6–400) Mod 6/2028
1	Output 1 (0-28V)	Table 16	HF (S.O.O. 6215)
2	Output 2 (0-28V)	Table 17	Audio Panel, R/H
3	Output 3 (0-28V)	Table 18	Audio Panel, L/H
4	Output 4 (0-28V)	Table 19	FD/AP (S.O.O. 6221)
5	Output 5 (0-28V)	Table 20	FD/AP (S.O.O. 6221)

Table 1 LPS Lighting Interconnect (continued)

LPS Pin	Signal Name	LPS Table Ref.	System (DHC–6–400) Mod 6/2028
6	Output 6 (0-28V)	Table 21	Reversionary Panel
7	Output 7 (0-28V)	Table 22	Apex Panels
8	Output 8 (0-28V)	Table 23	Apex Panels
9	Output 9 (0-28V)	Table 24	Apex Panels
10	Output 10 (0-28V)	Table 25	PFD, L/H
11	Output 11 (0-28V)	Table 26	PFD, R/H
12	Output 12 (0-28V)	Table 27	PFD, Upper
13	Output 13 (0-28V)	Table 28	PFD, Lower
14	Output 14 (0-28V)	Table 29	ESIS (Except C-FDHT)
15	Output 15 (0-28V)	Table 30	Clock
16	Output 16 (0-28V)	Table 31	Trim Panel, Flap Sel, Alt Static
17	Output 17 (0-28V)	Table 32	Compass
18	Output 18 (0-28V)	Table 33	not used
19	Output 19 (0-28V)	Table 34	not used
20	Output 20 (0-28V)	Table 35	not used
21	Output 21 (0-28V)	Table 36	not used
22	Output 22 (0-28V)	Table 37	not used
23	Output 23 (0-28V)	Table 38	not used
24	Output 24 (0-28V)	Table 39	not used
25	Output 25 (0-28V)	Table 40	Spare 28 V
52	Output 15 (0-5V)	Table 15	Spare 5 V
53	Output 14 (0-5V)	Table 14	not used
54	Output 13 (0-5V)	Table 13	not used
55	Output 12 (0-5V)	Table 12	not used
56	Output 11 (0-5V)	Table 11	not used
57	Output 10 (0-5V)	Table 10	not used
58	Output 9 (0-5V)	Table 09	not used
59	Output 8 (0-5V)	Table 08	not used
60	Output 7 (0-5V)	Table 07	not used
61	Output 6 (0-5V)	Table 06	not used
62	Output 5 (0-5V)	Table 05	Dim Panel
63	Output 4 (0-5V)	Table 04	Right Switch Panel
64	Output 3 (0-5V)	Table 03	Left Switch Panel

Table 1 LPS Lighting Interconnect (continued)

LPS Pin	Signal Name	LPS Table Ref.	System (DHC–6–400) Mod 6/2028
65	Output 2 (0-5V)	Table 02	Upper Switch Panel
66	Output 1 (0-5V)	Table 01	Fuel Selector Panel

3. Adjustment/Test

A. Function Test Lighting Power Supply (LPS)

1) General

The unit does not require alignments or adjustments. However, if ramp test or maintenance check is required, the unit should be connected to a laptop via the maintenance port located on the maintenance panel in the forward avionics bay. The connection is RS422 type communication. Thus, a RS422 adapter may be used for connection to a laptop. EASY SYNC UNIT P/N ES–U–3001M is recommended for this purpose. Other applicable RS422 to USB adapters may be used as alternatives.

The example setup procedure using the recommended unit is listed below.

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Turn the Instrument dim control from the counterclockwise (MIN) position, to the fully clockwise position on the Control Column Dimming Panel. Verify that the instrument panel lights go from dark to full brightness.

d) Turn the Instrument dim control from the counterclockwise (MIN) position, to the fully clockwise position on the Reversion Control Panel. Verify that the instrument panel lights go from dark to full brightness.

e) Set the Instrument dim control from the full brightness position to a comfortable viewing dim level.

f) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

3) Fault Log Data Test

a) Assemble the RS422 adapter to a RS422 connector (M24308/4–259 preferred) as per PSM 1-64-2W, Chapter 33-10-10.

b) Acquire a USB type B to USB type A adaptor cable (regular commercial printer cable).

c) Acquire a computer equipped with USB port and install applicable RS422 virtual COM port driver based on the computer operating system.

d) Acquire a computer equipped with USB port and install applicable RS422 virtual COM port driver based on the computer operating system.

e) Connect USB type B port to the ES–U–3001M USB port.

f) Connect M24308/4–259 connector to LPS maintenance port.

g) Setup the communication via Windows Hyperterminal

1 Create a new connection in Windows Hyperterminal. Refer to Figure 2.

2 Confirm correct COM selected for USB adapter. Refer to Figure 3.

3 Set the communication protocol as 9600 bits per second, 8 data bits, none parity, 1 stop bit, and none flow control. Refer to Figure 4.

4 The Emulation should be set as Auto detect. Refer to Figure 5.

5 The ASC II should be set to send line ends with line feeds and echo typed characters locally. Refer to Figure 6.

h) Check if the connection is successfully established. Upon establishment of the connection, carriage return should show similar output from the LPS unit. Refer to Figure 7. The recommended RS422 adaptor ES–U–3001M may be ordered from Viking Air Ltd. The LPS unit must be powered for successful connection.

i) Once the connection has been successfully established, there are two commands that may be used for verify the correct operation of the unit. All commands entered into the Hyperterminal should be using UPPER CASE LETTERS ONLY.

j) Type DISP in the command window will output the current operating parameters of the PS 1 unit, including temperature switch status, and output voltage levels. Valid range of voltage outputs should be within the range from 00 to FF in hex number format.

k) Type DFLG in the command window will output the fault log data stored in the PS 1 unit internal memory.

l) A correctly functioning unit should not have any fault log output by entering this command.

4. Removal and Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the LPS

1) General

These procedures are for the removal and installation of the LPS. The unit is not considered ESD sensitive. However, a high density connector has been used for the power supply outputs. For removal and installation, extra care should be taken for assembling the cable assembly to the LPS unit. The pins may be easily damaged.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open PANEL LIGHTS L (P7) circuit breaker on the avionics circuit breaker panel and fit with safety clip.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Disconnect the electrical connectors from the front of the unit.

e) Loosen the four screws and nuts used to fasten the unit to the mounting bracket.

f) Remove unit from aircraft.

Note Stow connectors as required.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the PANEL LIGHTS (P7) circuit breaker on the avionics circuit breaker panel is opened and fitted with safety clip.

c) Slide the LPS unit into the mounting tray.

d) Tighten the four screws and nuts used to fasten the unit to the mounting bracket.

e) Reconnect the electrical connectors to the unit. Ensure connections are secure.

f) Remove safety clip and close PANEL LIGHTS L (P7) circuit breaker on the avionics circuit breaker panel.

B. Removal and Installation of the LPS Mounting Tray

1) General

These procedures are for the removal and installation of the LPS Mounting Tray.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove the LPS unit as per Para 4.A.2).

c) Remove screws securing the mounting tray to the instrument panel.

d) Remove mounting tray from aircraft.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Align the tray to the mounting holes.

c) Install the screws that secure the LPS mounting tray to the front avionics rack.

d) Install the LPS unit as per Para 4.A.3).

e) Test the LPS as per Para 3.A.2).

5. Troubleshooting

A. LPS

1) If the LPS is deemed inoperative check the following:

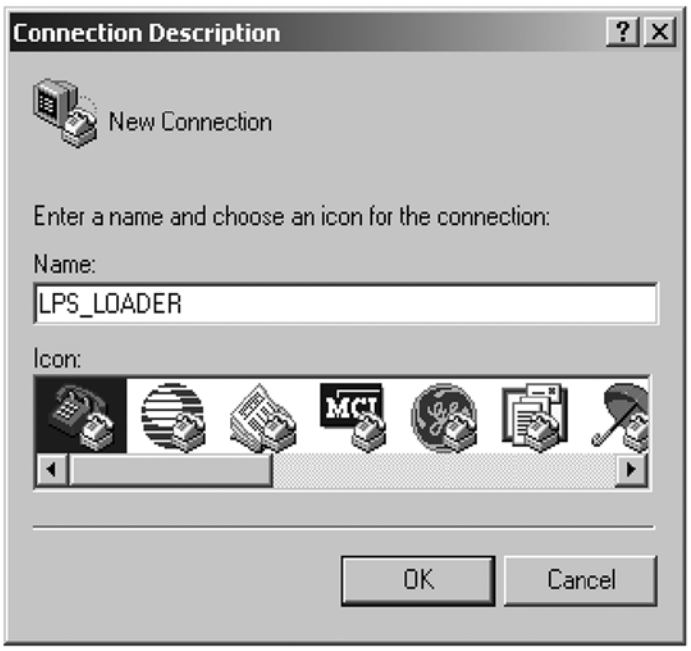
a) Ensure PANEL LIGHTS L (P7) circuit breaker is closed.

b) Check that the units are securely installed and the locking screw is securely tightened.

c) Visually inspect LPS unit connectors for damaged wires or contacts.

d) Check if the unit output voltage is valid at various rheostat locations.

e) If repair or service is required, send to an appropriate rated avionics repair facility.

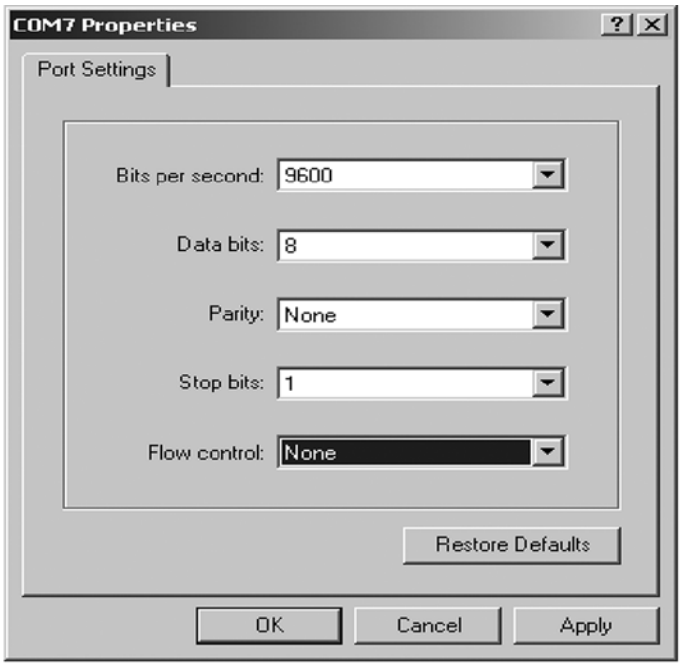


Connection Description
Figure 2

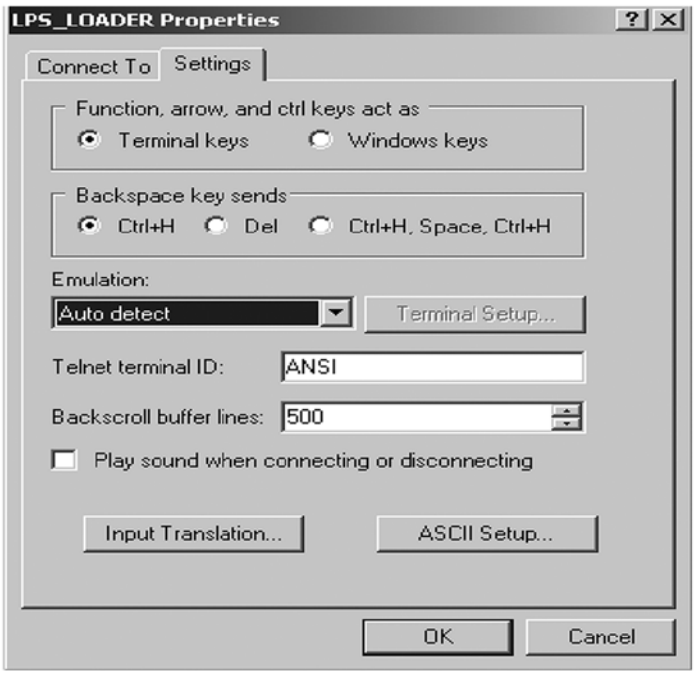


Confirm Correct COM Selected for USB Adaptor
Figure 3

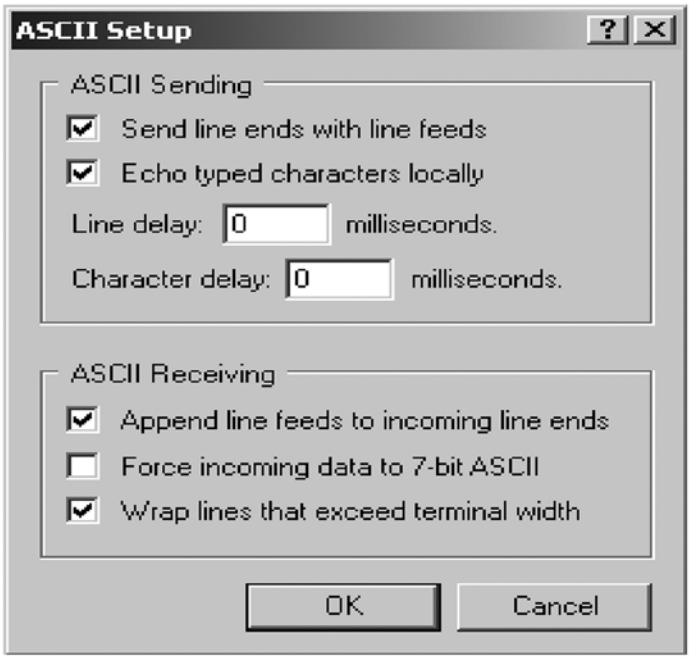
Note COM 7 shown for reference only actual port may be different.



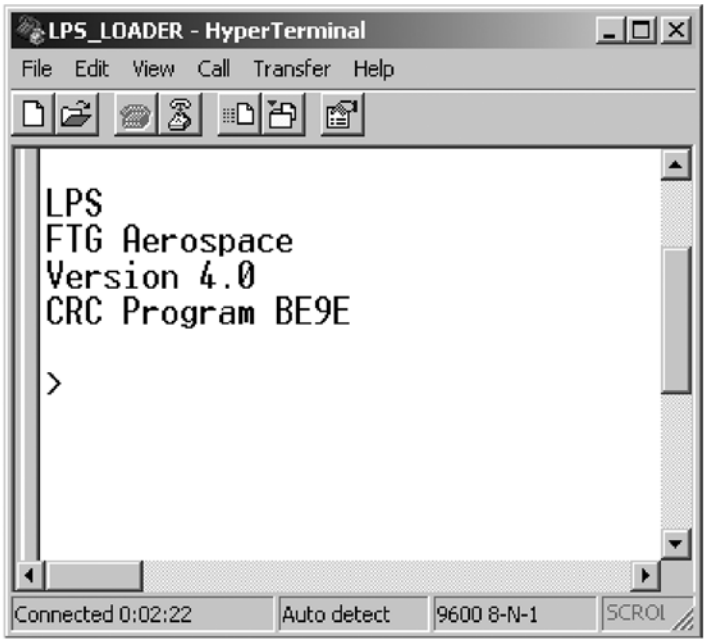
Set the Communication Protocol Window 1
Figure 4



Set the Communication Protocol Window 2
Figure 5



ASCII Setup
Figure 6



Carriage Return Response Upon Successful Connection
Figure 7

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TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR33-3**

Insert in 33-20-00 before Page 1

PASSENGER COMPARTMENT LIGHTS (MOD 6/2028)

REASON FOR ISSUE:

To introduce Mod 6/2414, Cabin Fresh Air and Distribution – O/H Ducting Sealing Improvements.

ACTION:

In Chapter 33, 33-20-00 ignore existing Para 2. and insert revised Para, as shown on Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

2. Description

A. Cabin Lighting

The cabin lighting system consists of six general cabin lights are installed along the centre line of the cabin roof. The lights are controlled from a three position switch labelled CABIN GENERAL on the lower left pilot sub-panel; the switch positions are labelled DIM, OFF and BRIGHT. The circuit is protected by the circuit breakers identified CABIN LIGHTS on the circuit breaker panel, and powered from the left DC bus. (Refer to 24-60-00, DC Electrical Load Distribution – General Data). When the CABIN GENERAL switch is set to the DIM position a 100 ohm dimming resistor is connected in series with the supply to the cabin lights, reducing the luminance of the cabin lights.

B. Passenger Entrance Light

The entrance light is a threshold floodlight recessed in the top of the main cabin door aperture. The light is controlled by either one of two switches; one on the lower left pilot sub-panel label BOARDING and the other (a rocker type switch) on the fascia immediately forward of the door and labelled BOARDING LIGHTS. The entrance light circuit is powered from the hot battery bus through the COMPT LTS circuit breaker on the rear bulkhead circuit breaker panel.

C. Passenger Reading Lights

Twenty passenger reading lights, six on the left and fourteen on the right, are integral with the cove ventilating ducts in the cabin walls. The lights are controlled from a two-position switch labelled CABIN READING on the lower left pilot sub-panel. A switch beside each light provides passenger control over individual lights. Circuit protection is provided by the READING LIGHTS circuit breakers on the main circuit breaker panel and is powered by the right bus. (Refer to 24-30-00, DC Generation – General Data).

On aircraft incorporating Mod 6/2414, there is a light and switch cover to improve air distribution in the cabin gasper vents. For removal and installation instructions refer to 25-20-02, Cabin Compartment – Cabin Upholstery (Mod 6/2013) – General data, Para 3.

D. Cabin Signs

Two cabin signs with pictograms indication NO SMOKING and FASTEN SEAT BELTS are installed in the cabin to the right of the cabin/flight compartment doorway station 111.00. They are illuminated by lights controlled by a three position switch labelled CABIN SIGNS FSB–NS, OFF and NS on the lower left pilot sub-panel. There are three possible switch positions; both off, no smoking on, and both no smoking and fasten seat belt on. (Refer to 24-30-00, DC Generation – General Data). Power is provided by the MAP LIGHTS circuit breaker on the left bus, and installed in the main circuit breaker panel.

NOTE: Please see the TEMPORARY REVISION that revises this page.

Passenger Compartment Lights (Mod 6/2028) – General Data

1. General

The passenger compartment lights include cabin lighting (commuter interior), passenger reading lights and warning sign lights.

2. Description

A. Cabin Lighting

The cabin lighting system consists of six general cabin lights are installed along the centre line of the cabin roof. The lights are controlled from a three position switch labelled CABIN GENERAL on the lower left pilot sub-panel; the switch positions are labelled DIM, OFF and BRIGHT. The circuit is protected by the circuit breakers identified CABIN LIGHTS on the circuit breaker panel, and powered from the left DC bus. (Refer to 24-60-00, DC Electrical Load Distribution – General Data). When the CABIN GENERAL switch is set to the DIM position a 100 ohm dimming resistor is connected in series with the supply to the cabin lights, reducing the luminance of the cabin lights.

B. Passenger Entrance Light

The entrance light is a threshold floodlight recessed in the top of the main cabin door aperture. The light is controlled by either one of two switches; one on the lower left pilot sub-panel label BOARDING and the other (a rocker type switch) on the fascia immediately forward of the door and labelled BOARDING LIGHTS. The entrance light circuit is powered from the hot battery bus through the COMPT LTS circuit breaker on the rear bulkhead circuit breaker panel.

C. Passenger Reading Lights

Twenty passenger reading lights, six on the left and fourteen on the right, are integral with the cove ventilating ducts in the cabin walls. The lights are controlled from a two-position switch labelled CABIN READING on the lower left pilot sub-panel. A switch beside each light provides passenger control over individual lights. Circuit protection is provided by the READING LIGHTS circuit breakers on the main circuit breaker panel and is powered by the right bus. (Refer to 24-30-00, DC Generation – General Data).

D. Cabin Signs

Two cabin signs with pictograms indication NO SMOKING and FASTEN SEAT BELTS are installed in the cabin to the right of the cabin/flight compartment doorway station 111.00. They are illuminated by lights controlled by a three position switch labelled CABIN SIGNS FSB–NS, OFF and NS on the lower left pilot sub-panel. There are three possible switch positions; both off, no smoking on, and both no smoking and fasten seat belt on. (Refer to 24-30-00, DC Generation – General Data). Power is provided by the MAP LIGHTS circuit breaker on the left bus, and installed in the main circuit breaker panel.

3. Adjustment/Test

A. Operational Test Passenger Boarding Light

1) Switch ON BOARDING switch forward of left cabin door and check that the threshold floodlight illuminates.

2) Switch OFF BOARDING switch on lower pilot sub-panel, and check that the threshold floodlight extinguishes.

3) Check that MASTER and POWER SOURCE switches (Pre Mod 6/2314); or EXTERNAL and BATTERY switches (Post Mod 6/2314) have no effect on entrance light operation.

B. Operational Test Cabin Lights

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Set CABIN GENERAL switch to BRIGHT and check that six cabin dome lights illuminate at full brightness; set switch to DIM and check that lights are dimmed; set switch OFF.

3) Switch ON CABIN READING switch and check each reading light by switching on integral switches.

4) Switch OFF CABIN READING switch.

C. Operational Test Cabin Sign Lights

1) With power ON aircraft buses switch CABIN SIGNS to NS and check that NO SMOKING sign comes on.

2) Switch OFF CABIN SIGNS.

3) Switch ON CABIN SIGNS to FSB–NS and check that FASTEN SEAT BELT/NO SMOKING lights come on.

Cargo and Service Compartment Lights (Mod 6/2028) – General Data

1. General

The cargo and service compartment lights comprise front and rear baggage compartment lights.

2. Forward Baggage Compartment Light

The forward baggage compartment light is operated by an integral switch on the light. The light is powered from the battery bus through the CABIN LIGHTS circuit breaker on the rear bulkhead hot battery bus circuit breaker panel.

3. Rear Baggage Compartment Lights

The rear baggage compartment is illuminated by two dome lights which are operated by a limit switch activated by the rear baggage door latching pins, or when the baggage door is closed, by a switch (labelled BAGGAGE COMP LT) on the aft face of the baggage compartment forward bulkhead. The lights are powered from the hot battery bus through the CABIN LIGHTS circuit breaker on the rear bulkhead (station 332.00) circuit breaker panel. When the rear baggage door is open, BAGGAGE COMP LT switch will not turn the lights off (refer to Table 1 — Operation of Rear Baggage Compartment Lights).

Table 1 Operation of Rear Baggage Compartment Lights

Door Position	Switch Position (BAGGAGE COMP LT)	Lights
Closed	OFF	OFF
Closed	ON	ON
Open	OFF	ON
Open	ON	ON

Note Aircraft with S.O.O. 6393 incorporated do not have a rear baggage compartment door, and the two dome lights in the rear baggage compartment are only operated by the switch (labelled BAGGAGE COMP LT) on the aft face of the baggage compartment forward bulkhead.

4. Adjustment/Test

A. Operational Test Front and Rear Baggage Compartment Lights

Note On aircraft with S.O.O. 6393 incorporated, there is no rear baggage compartment door. In the following test, any reference to close/open the rear baggage door is not applicable.

1) With the rear baggage compartment door closed, switch ON BAGGAGE COMP LT switch on bulkhead at station 332.00 and check that the rear baggage compartment lights come on. Switch OFF BAGGAGE COMP LT switch.

2) Open the rear baggage compartment door and check that lights come on. Close door.

3) Open the front baggage compartment door, check front light by operating integral switch on the light.

4) Check that MASTER and POWER SOURCE switches (Pre Mod 6/2314), or EXTERNAL and BATTERY switches (Post Mod 6/2314) have no effect on front and rear baggage compartment lights operation.

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Exterior Lights (Mod 6/2058) – General Data

1. General

The exterior lighting system consists of position lights, landing lights, a red beacon light, and white strobe lights. All exterior lights are powered from the left or right DC buses. A taxi light is supplied as standard equipment on all aircraft except floatplanes.

2. Position Lights

The position (navigation) lights consist of red and green wing tip lights and a white taillight. The lights are controlled by a switch labelled POSITION on the left sub-panel console. The circuit is powered from the left DC bus and protected by a POSITION LIGHTS (C17) circuit breaker on the main circuit breaker panel. (Refer to 24-30-00, DC Generation – General Data).

3. Landing Lights

Two high intensity discharge (xenon) landing lights are installed with independent power supplies, one in each wing leading edge outboard of the engine (Pre Mod 6/2285). Aircraft incorporating Post Mod 6/2285 have two LED Landing Lights installed, one in each wing leading edge outboard of the engine. The lights are controlled by two switches labelled LANDING LIGHTS that are located on the start panel directly above the left pilot windshield. The STEADY switch activates them continuously. The PULSE switch activates the lights alternately by manual selection, or triggered by the TCAS system (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data, and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General data). The circuits are powered from the left and right DC buses and protected by LANDING LIGHT L (C16) and LANDING LIGHT R (D16) circuit breakers on the main circuit breaker panel, capable of pulsing. (Refer to 24-30-00, DC Generation – General Data).

4. Wing Tip Strobe Lights and Beacon Light

Two white strobe type lights are part of the position light assembly installed in each wing tip. A red beacon light is mounted on the top of the vertical stabilizer. The white strobe lights are controlled by a switch labelled STROBE on the lower left pilot sub-panel; the circuit is powered from the left DC bus and is protected by the STROBE LIGHTS (D15) circuit breaker on the main circuit breaker panel. (Refer to 24-30-00, DC Generation – General Data).

The red beacon light is controlled by a switch labelled BEACON on the lower left pilot sub-panel; the circuit is powered from the left DC bus and is protected by the BEACON LIGHT (C15) circuit breaker on the main circuit breaker panel.

5. Taxi Light

A LED taxi light is installed on the nose wheel fork. The light is controlled by a switch labelled TAXI on the lower left pilot sub-panel and the circuit is powered from the right DC bus and protected by a TAXI LIGHT (B16) circuit breaker on the main circuit breaker panel. The taxi light is not installed on floatplanes. (Refer to 24-30-00, DC Generation – General Data).

6. Wing Inspection Lights (Mod 6/2028 or Mod 6/2140)

Two 40 watt incandescent wing inspection lights are provided (Pre Mod 6/2140), one on the outboard side of each engine nacelle. Aircraft incorporating Mod 6/2140 have two 6.5 watt LED wing inspection lights, one on the outboard side of the each engine nacelle. The lights illuminate the leading edges of the wings and are controlled by a switch labelled WING INSP on the lower left pilot sub-panel. The circuit is powered from the right DC bus and protected by a WING INSPECTION LIGHT (D17) circuit breaker on the main circuit breaker panel. (Refer to 24-30-00, DC Generation – General Data).

7. Adjustment/Test

A. Operational Test Landing Lights

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Switch LANDING LIGHTS STEADY switch to ON and check that left and right landing light comes on.

3) Switch the LANDING LIGHTS PULSE switch to PULSE and check that the left and right landing lights alternate illuminating. Switch the LANDING LIGHTS PULSE switch to OFF. Switch the LANDING LIGHTS STEADY switch to OFF.

Note For testing of the TCAS triggered pulse function refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General data.

B. Adjust the Landing Lights

To adjust the landing lights for normal operation, proceed as follows:

1) Position aircraft on level ground.

2) Adjust setting of three nuts on mounting ring until light beam is parallel with aircraft center line (line of flight).

3) Adjust setting of nuts until light beam focuses its center on the ground at a point 140 feet ahead of pilot.

C. Operational Test Position Lights

1) With power on the buses, switch POSITION switch to ON and check that the left and right wing, and tail lights come on.

2) Switch POSITION switch to OFF.

D. Operational Test Beacon Lights

1) With power on the buses, switch on the BEACON switch to ON and check that the upper beacon lights come on.

2) Switch BEACON switch to OFF.

E. Operational Test Taxi Light

1) With power on the buses, switch TAXI switch to ON and check that taxi light comes on.

2) Switch TAXI switch to OFF.

F. Operational Test Wing Inspection Lights

1) With power on the buses, switch the WING INSP switch ON and check that right and left wing inspection lights come on.

2) Switch WING INSP switch to OFF.

G. Operational Test Strobe (Mod 6/2028)

WARNING HIGH VOLTAGE SHOCK HAZARD PRESENT IN STROBE LIGHT SYSTEM (PRE MOD 6/2447 ONLY).

1) With power on the buses, switch STROBE switch to ON and check that left and right wing strobe lights come on.

2) Switch STROBE switch to OFF.

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Emergency Lighting (Mod S.O.O. 6205) – General Data

1. General

Emergency Lighting (Mod S.O.O. 6205) is available as a customer order option.

Emergency lighting is provided by using an emergency battery to operate the six cabin lights in the event of failure of the 28 Volt DC power supply. The dedicated battery is on continuous charge whenever the main 28 Volt DC system is energized.

A single switch labelled EMERGENCY LIGHTING has three positions marked ARM, TEST, and OFF. The ARM position (lever locked) is normally selected for flight to both arm the lights for automatic operation should a power failure occur, and to trickle charge the battery. The TEST position (a momentary position) interrupts the aircraft input voltage, testing the monitoring circuit, battery power. Select switch to OFF before selecting the MASTER switch to OFF.

The emergency lights circuit is powered from the left 28 Volt DC bus and is protected by a circuit breaker labelled EMERGENCY LIGHTING BATTERY on the rear circuit breaker panel station 332.00.

2. Adjustment/Test

A. Operational Test Emergency Lights

1) Select the EMERGENCY LIGHTING switch to TEST.

2) Verify 6 cabin center lights come on.

3) Release switch from TEST to OFF.

4) Select the MASTER switch to ON (Pre Mod 6/2314); or select the EXTERNAL and BATTERY switches to ON (Post Mod 6/2314).

5) Move to ARM.

6) Check that the cabin lights are OFF. Confirm CABIN GENERAL light switch is OFF.

7) Select the MASTER switch to OFF (Pre Mod 6/2314); or select the EXTERNAL and BATTERY switches to OFF (Post Mod 6/2314).

8) Verify cabin lights come on.

9) Move EMERGENCY LIGHTING switch to OFF.

10) Verify lights are off.

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Emergency Lighting Battery – General data

1. Removal/Installation

   A. Remove Emergency Lighting System Battery

      CAUTION **BEFORE DISCONNECTING THE BATTERY, ENSURE THAT THE MASTER SWITCH IS OFF AND PULL COMP LTS CIRCUIT BREAKER. ANY SUBSTANTIAL LOAD ON THE BATTERY COULD CAUSE DAMAGE TO THE CONNECTOR PINS AND SOCKETS WHEN BATTERY IS BEING DISCONNECTED.**

      Before removing the battery, refer to precautions in 24-00-00, Electrical Power – General Data.

      1) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      2) Remove the access panel in the rear of the baggage compartment floor. (Refer to 12-00-00, Servicing – General Data).

      3) Disconnect the battery connector.

      4) Remove the battery screws.

      5) Remove the battery.

   B. Install Emergency Lighting System Battery

      1) Inspect the battery for damage. Cracks in metal or plastic containers are not permitted. Dents in the metal containers that impinge on the interior plastic container are not acceptable.

      2) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

      3) Remove the access panel in the rear of the baggage compartment floor. (Refer to 12-00-00, Servicing – General Data).

      4) Install the (4) screws to secure the battery.

      5) Connect the battery connector.

      6) Install the access panel in the rear baggage compartment floor.

      7) Do operational test as per 33-50-00, Emergency Lighting – General Data, Para 2.A.

2. Adjustment/Test

   A. Capacity Test

      1) Stabilize the battery at 15°C or higher. The battery must be at this temperature for at least 24 hours.

      2) Discharge the battery at the C1 rate on the label to the end point voltage (EPV) of 20 volts.

      3) Record the time to the end point voltage (EPV).

      4) The battery is acceptable for continued use if the ampere hour capacity (actual hours if discharge X ampere rate of discharge) is greater than 85% of the normal capacity (C1) shown on the battery label (i.e., 51 minutes or more).

      5) If the battery passes capacity test, it can be charged as per and return to service.

      6) If the battery fails the capacity test, condition battery as per Para 5.D.

7) After battery has been conditioned, repeat the capacity test.

Note If the battery gets very hot (external case temperature greater than 55°C (130°F) during charge, it should be replaced.

- If battery passes, refer to Para 2.A.5).
- If the battery fails, replace it.

3. Inspection/Check

A. Inspect Emergency Lighting Battery

A visual inspection in the aircraft should be made of the battery connector, receptacle and should be checked for contact failure or damage and the Allen head screws or nuts should be torqued, as per manufactures' data.

For further information refer to the manufacturers' data as follows:

1) Charge the Battery

a) Specials Tools:

- Advanced Power Products Beta D–50 Aircraft Battery Analyzer P/N 4126, or equal
- Advanced Power Products Alpha C–25 Battery Charger P/N 4142, or equal
- Advanced Power Products CA15–50 Charger/Analyzer P/N 4159, or equal

WARNING CONTACT CONCORDE FOR DETERMINATION OF EQUAL TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY THE BATTERY.

b) Depending on the type of charger available, charge the battery Constant Potential (CP). Charge at 2.35 volts/cell or 14.1 volts for 12 volt batteries, or 28.2 volts for 24 volt batteries until the charge current stabilizes for 1 hour.

2) Return to Service

a) Charge the battery as above. If the battery gets very hot (external case temperature greater than 55°C (130°F) during charging, it should be replaced.

4. Troubleshooting

Symptom	Probable Cause	Corrective Action
Low voltage/no voltage	Battery partially or fully discharged	Charge in accordance with Para 5.
	Battery fuse blown or circuit breaker tripped	Inspect and replace fuse Reset circuit breaker
Battery does not hold charge	Battery beyond serviceable life	Replace battery
Battery gets hot while recharging	Battery beyond serviceable life	Replace battery

5. Servicing Discharged Batteries

A. Recharging dual (parallel) batteries on the aircraft should be done individually.

B. Recharging a single battery on the aircraft:

1) Advanced Power Products Activator 282 Battery Charger, P/N 4105, or equal constant potential charger.

2) The Activator 282 is fully automatic. Connect to battery and initiate charging. If using a constant potential charger, follow instructions below.

CAUTION **ONLY CONSTANT POTENTIAL CHARGING MAY BE DONE ON THE AIRCRAFT. ENSURE THE CHARGING AREA IS WELL VENTILATED. DO NOT CONSTANT CURRENT CHARGE A BATTERY ON THE AIRCRAFT. THERE MAY BE A SERIOUS RISK OF INJURY TO PERSONNEL AND/OR DAMAGE TO THE AIRCRAFT AND AIRCRAFT SYSTEMS.**

WARNING **CONTACT CONCORDE FOR DETERMINATION OF EQUAL TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY THE BATTERY.**

- **SET THE CHARGER FOR 2.35 VOLTS PER CELL (14.1 VOLTS FOR A 12 VOLT BATTERY, 28.2 VOLTS FOR A 24 VOLT BATTERY).**
- **SET THE CURRENT LIMIT TO THE C1 RATE.**

Note If the current on external power unit is not adjustable, do not charge the battery in the aircraft.

- **INITIATE CHARGING AND CONTINUE TO CHARGE UNTIL THE CHARGE CURRENT STABILIZES FOR 1 HOUR.**

C. Uninstalled Recharging

1) Remove the battery from the aircraft.

2) Special Tools:

- Advanced Power Products Alpha C–25 Battery Charger, P/N 4142 or equivalent; Advanced Power Products CA15–50 Charger/Analyzer P/N 4159; or equal.

WARNING **CONTACT CONCORDE FOR DETERMINATION OF EQUAL TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY THE BATTERY.**

3) Depending on the type of charger available, charge the battery Constant Potential (CP). Charge at 14.1 volts for 12 volt batteries or 28.2 volts for 24 volt batteries until the charge current stabilizes for 1 hour.

Note Batteries that have been allowed to stand in a deeply discharged state may not accept a CP recharge. See conditioning procedure in Component Maintenance Manual PSM 1-6-6A for handling those batteries.

D. Conditioning Procedure:

WARNING **THE BATTERY MUST BE REMOVED FROM THE AIRCRAFT PRIOR TO DOING A CONDITIONING CHARGE.**

1) Special Tools:

- Advanced Power Products Beta D–50 Aircraft Battery Analyzer P/N 4146 or 4159, or equal
- Advanced Power Products Alpha C–25 Battery Charger P/N 4142 or 4159, or equal

WARNING **CONTACT CONCORDE FOR DETERMINATION OF EQUAL TEST EQUIPMENT. SOME BRANDS OF BATTERY CHARGERS WILL DESTROY THE BATTERY.**

2) Procedure:

a) Discharge the battery at the C1 rate to an end point voltage of 9 volts for 12 volt batteries and 18 volts for 24 volt batteries.

b) Constant current charge at the C1/10 rate for 16 hours.

WARNING **THIS PROCEDURE MAY DAMAGE THE BATTERY IF DONE ON A REPETITIVE BASIS.**

c) Allow the battery to cool down for 8 hours.

d) Retest the battery following the capacity test, refer to Para 2.A.

CHAPTER 34

NAVIGATION

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CHAPTER 34 – NAVIGATION

LIST OF EFFECTIVE PAGES

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Navigation – General Data

1. General

The navigation information required for aircraft operations is provided by the following systems; flight environmental data, attitude and direction, landing and taxiing aids independent position determining.

For more information on the VHF communications system refer to 23-00-00, Communications – General Data.

The standard (Basic) aircraft is equipped with the following Avionics which are suitable for day VFR certification.

The IFR Operational Bundle (Mod S.O.O. 6209) equips the aircraft for day or night VFR or IFR certification.

The Public Commercial Operations Bundle (Mod S.O.O. 6210) equips the aircraft with the most commonly required equipment for public commercial operations. The IFR Operations bundle (Mod S.O.O. 6209) is a prerequisite for this bundle.

2. Antennas

The location for the navigation system antennas refer to Figure 1, Figure 2, Figure 3 and Figure 4 for locations.

A. The VOR antenna used, Mod 6/1473, installed on each side of the vertical stabilizer and is the half loop type (DM N4–4, Dorne and Margolin) VOR antenna.

Note On aircraft with Mod 6/1599 incorporated, structural provisions in the basic aircraft permits installation of antennas only for specified products of VHF, transponder and DME, marker beacon and glide slope (refer to Chapter 23).

B. The GS (Glide Slope) antenna (DM N25–2, Dorne and Margolin) is located on the nose bulkhead behind the nose cap.

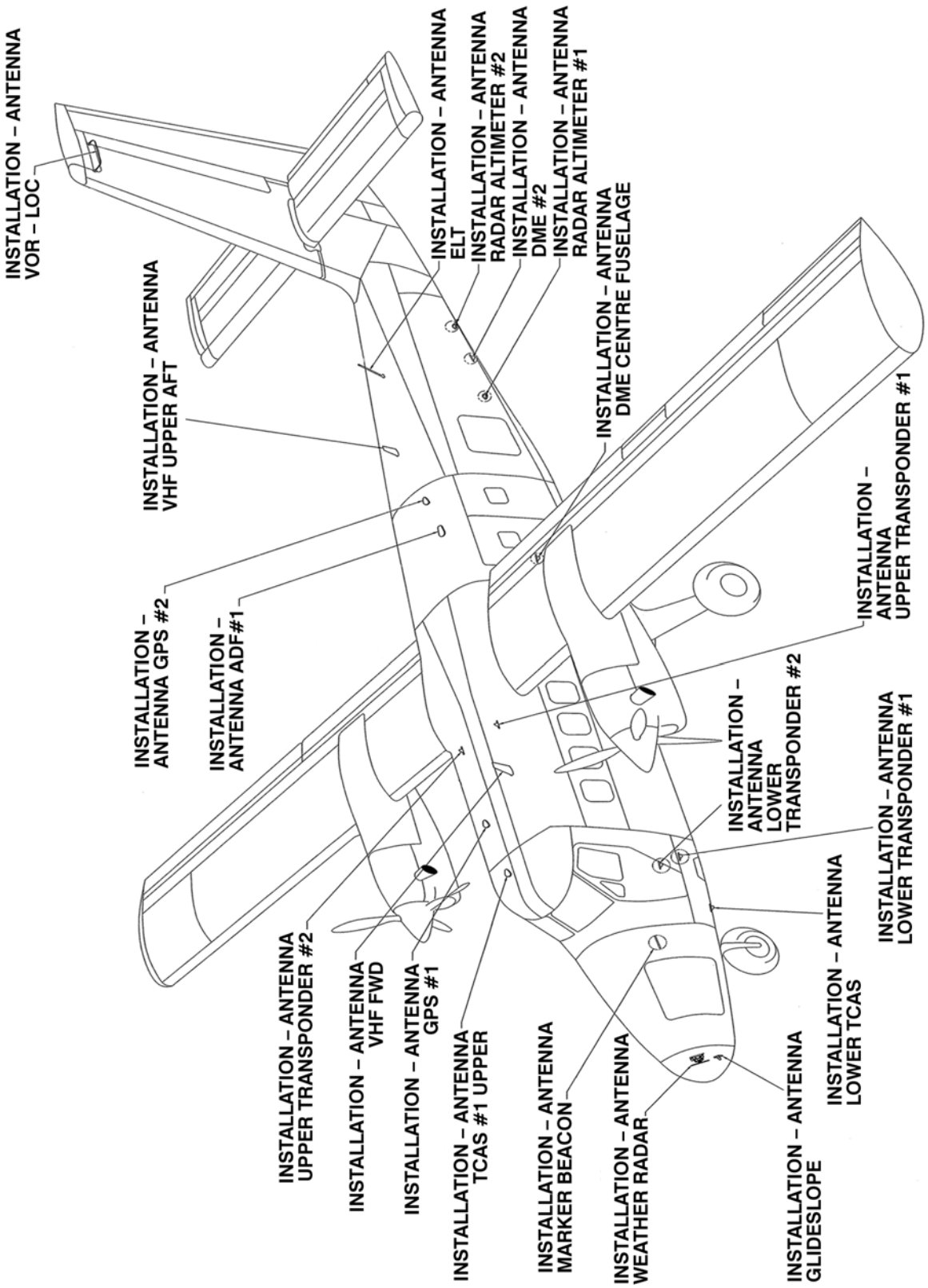
C. The MB (Marker Beacon) antenna (CI–118, Comant Industries) is located on the underside of the fuselage.

D. The ADF antennas, No. 1 and No. 2 (071–01234–0000, Honeywell) are installed in the cabin roof, in line with each other.

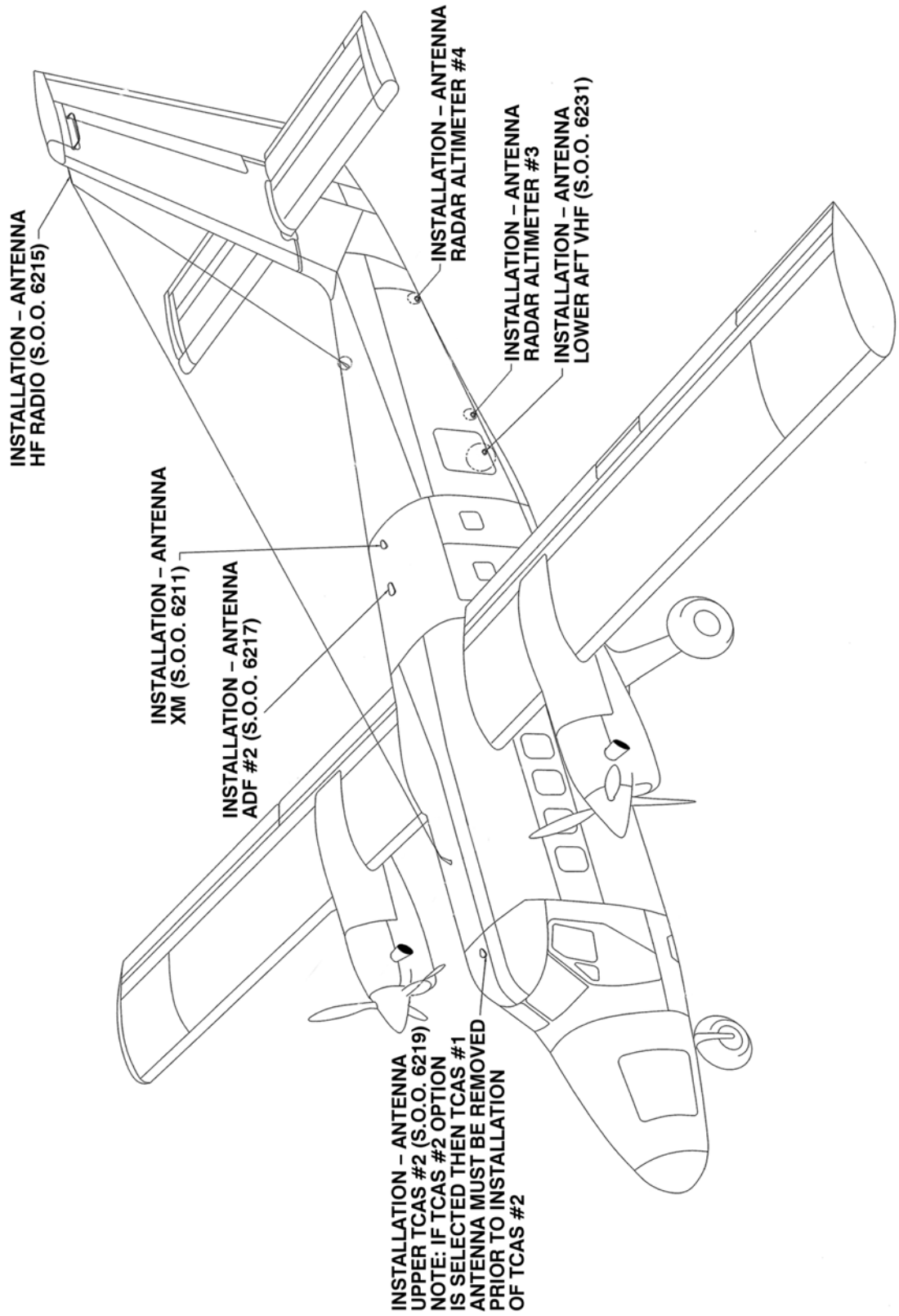
E. The weather radar antenna is part of the Honeywell ART 2000 radar receiver/transmitter located in the nose radome.

F. The DME antennas (CI–110–41–30, Comant Industries) are installed on the underside of the rear fuselage.

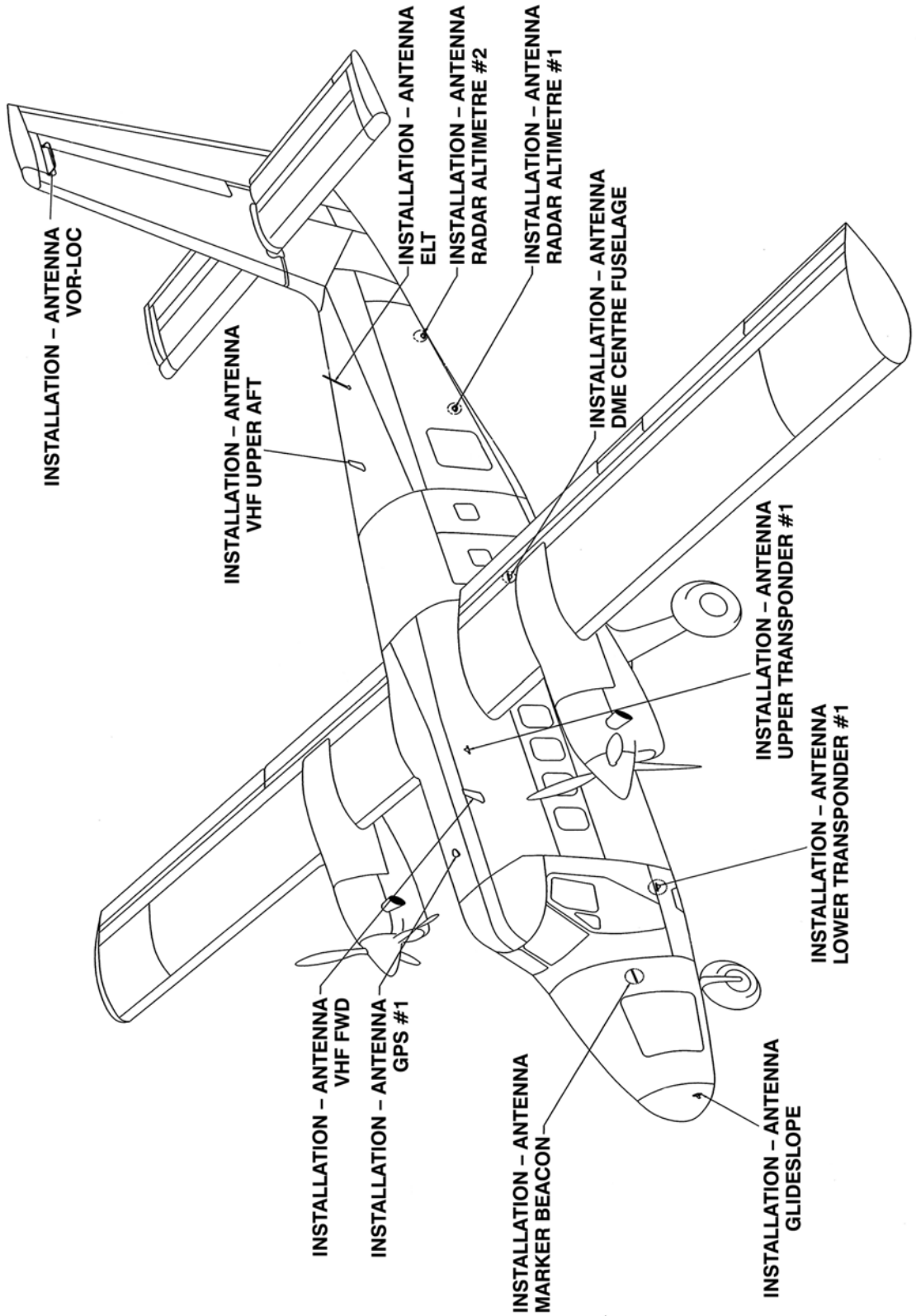
G. The Transponder antennas (CI–110–41–30 Comant Industries) are installed as per Figure 1, Figure 3 and Figure 4.



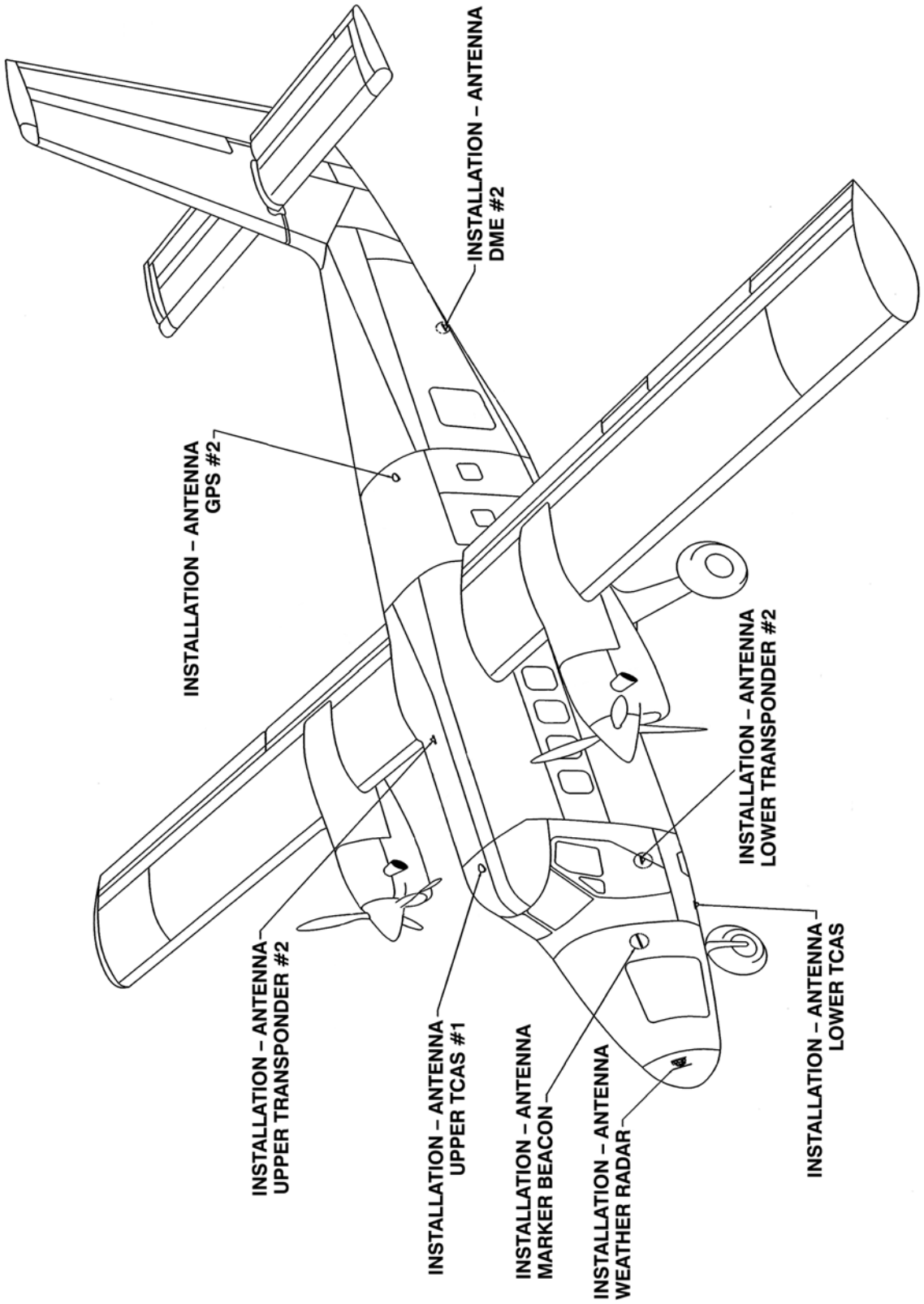
Navigation System Antennas Locations
Figure 1



Custom Standard Options Locations
Figure 2



Standard Navigation System Antennas Locations with S.O.O. 6388 installed
Figure 3



Provisional Navigation System Antenna Locations with S.O.O. 6388 installed
Figure 4

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Flight Environment Data – General Data

1. General

The flight environment data comprises of the pitot-static system.

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Pitot-Static System (Mod 6/2033) – General Data

1. General

The pitot-static system provides the air pressure input for the Air Data, Attitude and Heading Reference Systems, ADAHRS 1 and ADAHRS 2, refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data and to the Electronic Standby Instrument System, ESIS, refer to 31-20-21, Electronic Standby Instrument System (ESIS) (Mod S.O.O. 6209, Mod 6/2049) – General data. Refer to Figure 1.

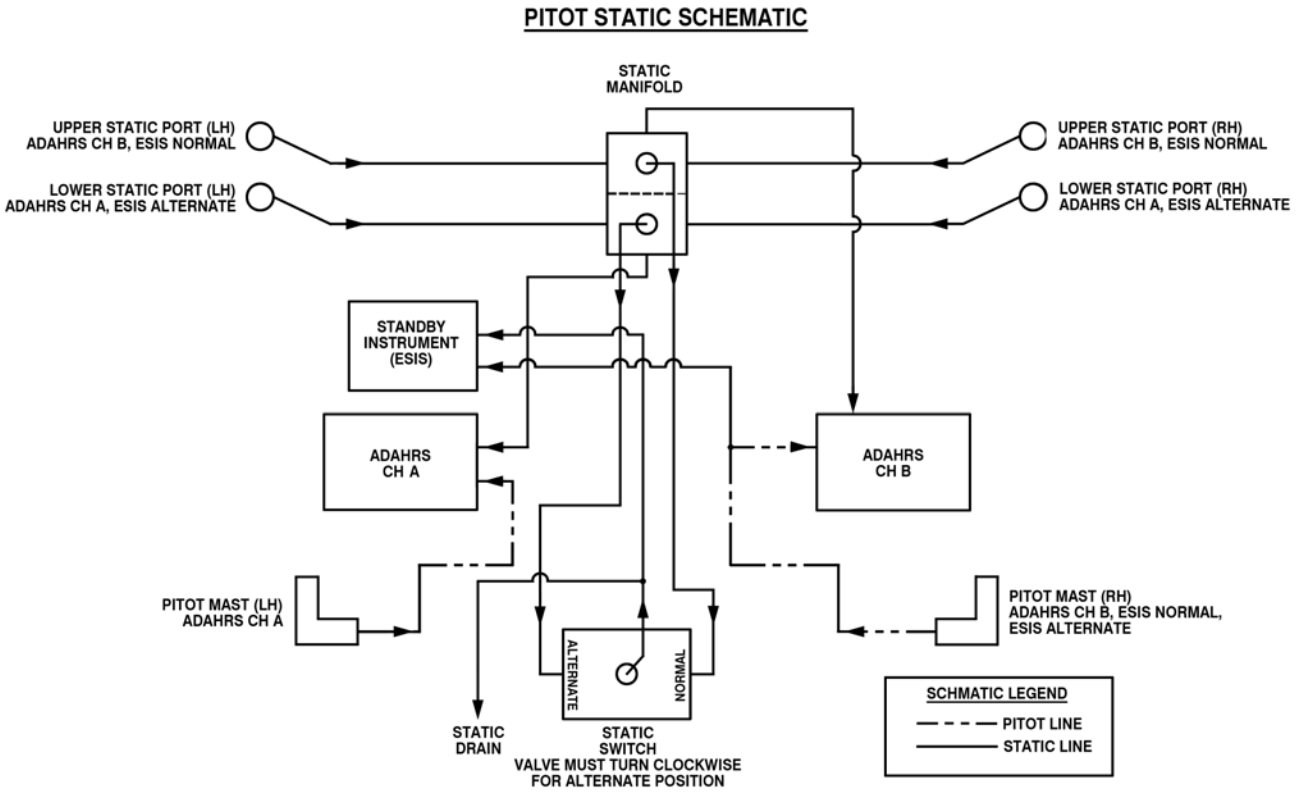
2. Description

A. Pitot Heads

There are two electrically heated pitot heads, each installed on a mast projecting from the left and right hand sides of the aircraft nose. The left hand pitot head supplies air pressure to the Air Data, Attitude and Heading Reference Systems, ADAHRS 1, the right pitot head supplies air pressure to for the Air Data, Attitude and Heading Reference Systems, ADAHRS 2 and to the Electronic Standby Instrument Systems, (ESIS).

B. (ESIS) Static Source Valve

The (ESIS) STATIC SOURCE valve is located on the dash panel, directly above the avionics circuit breakers. The valve can be selected to the NORMAL or ALTERNATE static source position. With the valve selected to the NORMAL position, the ESIS static lines will be connected to the upper static vents. With the valve selector in the ALTERNATE position the ESIS static lines will be connected to the lower static vents.



Pitot Static Schematic
Figure 1

3. System Service

A. Service Pitot Heads and Static Ports

1) Keep pitot heads and static ports clean and free from foreign matter. Do not use polishing compounds.

2) Ensure that drain holes in pitot heads are kept open.

3) Function test pitot heaters.

CAUTION **DO NOT LEAVE PITOT HEATERS SWITCHED ON FOR LONGER THAN ONE MINUTE. DO NOT BLOW INTO PITOT HEADS WHILE THEY ARE CONNECTED TO INSTRUMENTS.**

4. Adjustment/Test

A. Connecting Air Data Test Set

CAUTION **DO NOT REVERSE PITOT AND STATIC CONNECTIONS ON THE AIR DATA TEST SET, OR DAMAGE MAY OCCUR.**

Note Connection to the air data test set must be accomplished in accordance with the air data test set manufacturer's instructions, if available.

1) Ensure that the PITOT HEAT switch is off and the PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers, on the main circuit breaker panel, are pulled and fitted with safety clips.

2) Connect the PITOT port of the air data test set to the desired aircraft pitot head(s) (refer to Para 2. for pitot head definition).

3) Seal drain hole(s) on the back side of any connected pitot head(s).

4) Connect the STATIC port of the air data test set to the desired static port(s) on either side of aircraft (refer to Para 2. for static port definition).

5) Seal the appropriate static port(s) on the unused side of aircraft.

B. Disconnecting Air Data Test Set

1) Ensure the air data test set is vented to ambient conditions.

2) Disconnect the STATIC port of the air data test set from any connected static port(s).

3) Remove seals from any unused static port(s).

4) Disconnect the PITOT port of the air data test set from any connected pitot head(s).

5) Remove seals from any connected pitot head(s).

6) Remove safety clips and reset the PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers on the main circuit breaker panel.

C. Leak Tests

Note For the following tests, ensure only the listed pitot head or static port is connected to the air data test set, all others must remain vented to ambient conditions.

1) Pitot Leak Test – Channel A

a) Connect the air data test set to LH pitot head (refer to Para 4.A).

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait several minutes for the aircraft electrical and avionics systems to initialize.

d) Ensure the pilot's PFD is displaying ADAHRS Channel A information (default).

e) Adjust the air data test set to provide 150 knots on the pilot's Primary Flight Display (PFD) airspeed indication.

f) With the air data test set sealed, or in leak test mode, verify that the pilot's PFD airspeed indication does not drop more than 2 knots in one minute.

g) Adjust the air data test set to provide 35 knots on the pilot's PFD airspeed indication.

h) With the air data test set sealed, or in leak test mode, remove the seal from the LH pitot head drain, ensure airspeed on the pilot PFD blanks.

i) Vent the air data test set to ambient conditions.

j) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

k) Disconnect the air data test set from LH pitot head (refer to Para 4.B).

2) Pitot Leak Test – Channel B, ESIS

a) Connect the air data test set to RH pitot head (refer to Para 4.A).

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Set the ESIS POWER switch to ARM.

d) Wait several minutes for the aircraft electrical and avionics systems to initialize.

e) Ensure the co-pilot's PFD is displaying ADAHRS Channel B information (default).

f) Adjust the air data test set to provide 150 knots on the co-pilot's PFD airspeed indication.

g) Verify the airspeed is increasing on the ESIS and co-pilot's PFD.

h) With the air data test set sealed, or in leak test mode, verify that the co-pilot's PFD airspeed indication does not drop more than 2 knots in one minute.

i) Adjust the air data test set to provide 35 knots on the co-pilot's PFD airspeed indication.

j) With the air data test set sealed, or in leak test mode, remove the seal from the RH pitot head drain, ensure the airspeed on the co-pilot's PFD blanks.

k) Vent the air data test set to ambient conditions.

l) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

m) Set the ESIS POWER switch to OFF.

n) Disconnect the air data test set from RH pitot head (refer to Para 4.B).

3) Static Leak Test – Channel A, ESIS Alternate

a) Connect air data test set to lower static port (refer to Para 4.A).

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Set the ESIS POWER switch to ARM.

d) Wait several minutes for the aircraft electrical and avionics systems to initialize.

e) Ensure the pilot's PFD is displaying ADAHRS Channel A information (default).

f) Set the STANDBY INSTRUMENT STATIC SOURCE valve to ALTERNATE.

g) Press the BARO knob on the pilot's PFD controller and ESIS to set baro to 29.92 inHg (STD).

h) Adjust the air data test set to provide an altitude 1000 feet above the ambient elevation.

i) Verify altitude is increasing on the ESIS and pilot's PFD.

j) With the air data test set sealed, or in leak test mode, check that the pilot's PFD altimeter indication does not drop more than 50 feet in one minute.

k) Vent the air data test set to ambient conditions.

l) Return the STANDBY INSTRUMENT STATIC SOURCE valve to NORMAL.

m) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

n) Set the ESIS POWER switch to OFF.

o) Disconnect the air data test set from lower static port (refer to Para 4.B).

4) Static Leak Test – Channel B, ESIS Normal

a) Connect the air data test set to the upper static port (refer to Para 4.A).

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Set the ESIS POWER switch to ARM.

d) Wait several minutes for the aircraft electrical and avionics systems to initialize.

e) Ensure the co-pilot's PFD is displaying ADAHRS Channel B information (default).

f) Ensure the STANDBY INSTRUMENT STATIC SOURCE valve is set to NORMAL.

g) Press the BARO knob on the co-pilot's PFD controller and ESIS to set baro to 29.92 inHg (STD).

h) Adjust the air data test set to provide an altitude 1000 feet above the ambient elevation.

i) Verify altitude is increasing on the ESIS and co-pilot's PFD.

j) With the air data test set sealed, or in leak test mode, check that the co-pilot's PFD altimeter indication does not drop more than 50 feet in one minute.

k) Vent the air data test set to ambient conditions.

l) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

m) Set the ESIS POWER switch to OFF.

n) Disconnect air data test set from the upper static port (refer to Para 4.B).

D. Function Test Pitot Heater

WARNING **USE CAUTION WHEN TOUCHING THE PITOT HEADS WHILE THE SYSTEM IS OPERATIONAL AS THE PITOT HEADS QUICKLY BECOME HOT ENOUGH TO CAUSE SEVERE BURNS.**

(Refer to 30-30-00, Pitot Heating (Mod 6/2033) – General Data).

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Select the PITOT HEAT switch to ON.

CAUTION **DO NOT LEAVE THE PITOT HEATER SWITCHED ON FOR MORE THAN ONE MINUTE.**

3) Confirm that the pitot heads warm up.

4) Select PITOT HEAT switch to OFF.

5) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

5. Cleaning

A. Clean Pitot and Static Pressure Lines

1) Disconnect pitot and static lines from instruments and accessories. Cap instrument and accessory ports.

2) Remove, clean and dry all moisture traps.

3) Using pressure source of 25 psi nitrogen or clean dry air, blow through each pitot and static line (from inside of aircraft to outside) until clear of dust and moisture.

4) Reinstall moisture traps and reconnect lines to instruments.

5) Carry out pitot and static leak test. (Refer to Para 4.).

6. Removal/Installation

A. Removal and Installation of Pitot Static Components

1) Ensure the pitot static lines are correctly identified.

2) Cap lines and component ports.

3) Carry out leak test on the reconnection of the lines to components as per Para 4.

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Attitude and Direction – General Data

1. General

The attitude and direction system is comprised of the standby magnetic compass, the Electronic Standby Instrument System (ESIS), the Air Data, Attitude and Heading Reference Systems (ADAHRS 1 and ADAHRS 2) and is displayed through the Primary Flight Display, (PFD) and the Multi-Function Displays, (MFDs).

2. Description

A. Standby Magnetic Compass

Refer to 34-25-00, Standby Magnetic Compass – General Data for system details.

The standby magnetic compass gives the aircraft magnetic heading.

B. Electronic Standby Instrument System (ESIS)

Refer to 31-20-21, Electronic Standby Instrument System (ESIS) (Mod S.O.O. 6209, Mod 6/2049 and Mod 6/2170) – General Data for system details.

The Electronic Standby Instrument System (ESIS) gives displays for attitude, heading, altitude and airspeed in case of primary display or systems failures. The ESIS is powered by its own dedicated PS–850B backup battery, and is entirely independent of the Apex system.

C. Air Data, Attitude and Heading Reference Systems (ADAHRS)

Refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data for system details.

Each channel of the ADAHRS contains a solid-state micro-electro-mechanical system (MEMS) technology sensor block, which contains three rate sensors and three accelerometers in an orthogonal triad configuration. The triad in Channel B is skewed relative to Channel A. Each channel has a connection for an Outside Air Temperature (OAT) probe, a magnetometer, and two isolated absolute pressure sensors (one for pitot and one for static pressure). Channel A receives inputs from the left pitot tube, lower static ports, and one channel of the temperature probe and magnetometer. Channel B receives inputs from the right pitot tube, upper static ports, and the second channel of the temperature probe and magnetometer. Each channel of the ADAHRS has a Central Processing Unit (CPU). The ADAHRS is installed in the nose avionics rack. A Rosemount (Goodrich) dual element OAT sensor is mounted just forward and below the ram air scoop on the left side of the fuselage to supply the ADAHRS with OAT data. Two KMG 7010 magnetometers are mounted near the outboard end of the right wing (Pre Mod 6/2171), or in the vertical stabilizer (Post Mod 6/2171), one for each channel of the ADAHRS. During normal operation, the left pilot's PFD receives ADAHRS source data from the left pitot tube and lower static ports and ADAHRS Channel A. The right pilot's PFD receives ADAHRS source data from the right pitot tube, upper static ports, and ADAHRS Channel B.

Each ADAHRS channel CPU receives air data, temperature and heading information from that channel's sensor block and passes it to the other channel. Both CPUs then compare the data to verify sensor integrity. Verified ADAHRS information is sent to each channel of the MAU for display on the Apex System.

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Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data

1. General

The aircraft is equipped with a dual channel KSG 7200 Air Data, Attitude and Heading Reference System (ADAHRS). The system provides primary attitude, heading and air data parameters from each channel to the Modular Avionics Unit (MAU).

2. Description

A. Air Data, Attitude and Heading Reference System (KSG 7200)

Each channel of the ADAHRS contains a solid-state micro-electro-mechanical system (MEMS) technology sensor block, which contains three rate sensors and three accelerometers in an orthogonal triad configuration. The triad in Channel B is skewed relative to Channel A. Each channel has a connection for an Outside Air Temperature (OAT) probe, a magnetometer, and two isolated absolute pressure sensors (one for pitot and one for static pressure). Channel A receives inputs from the left pitot tube, lower static ports, and one channel of the temperature probe and magnetometer. Channel B receives inputs from the right pitot tube, upper static ports, and the second channel of the temperature probe and magnetometer. Each channel of the ADAHRS has a Central Processing Unit (CPU). The ADAHRS is installed in the nose avionics rack. A Rosemount (Goodrich) dual element OAT sensor is mounted just forward and below the ram air scoop on the left side of the fuselage to supply the ADAHRS with OAT data. Two KMG 7010 magnetometers are mounted near the outboard end of the right wing (Pre Mod 6/2171), or in the vertical stabilizer (Post Mod 6/2171), one for each channel of the ADAHRS. During normal operation, the left pilot's PFD receives ADAHRS source data from the left pitot tube and lower static ports and ADAHRS Channel A. The right co-pilot's PFD receives ADAHRS source data from the right pitot tube, upper static ports, and ADAHRS Channel B. Each ADAHRS channel has a KCM 200 configuration module installed into the wiring harness near the KSG 7200 connector, which stores configuration and calibration data for the ADAHRS system.

Each ADAHRS channel CPU receives air data, temperature and heading information from that channel's sensor block and passes it to the other channel. Both CPUs then compare the data to verify sensor integrity. Verified ADAHRS information is sent to each channel of the MAU for display on the Apex System.

If the data from a sensor does not pass the verification check the data is discarded and will not be used. A fault signal will be sent to the MAU and a caution will be posted on the CAS. In this case the ADAHRS button on the PFD Controller of the failed side can be pressed to change the ADAHRS source channel to the opposite side. An amber text message will appear in the HSI window when both PFDs are using the same ADAHRS source channel.

28 Volt DC power is protected by a 3 Amp ADAHRS 1A (1K) (Left Bus) and a 3 Amp ADAHRS 1B (2K) (Right Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The ADAHRS interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

2) Traffic Alert and Collision Avoidance System (TCAS) (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General Data), if installed.

3) Enhanced Ground Proximity Warning System (EGPWS) (refer to 34-44-00, Enhance Ground Proximity Warning System (TAWS) (Mod 6/2049) – General Data), if installed.

4) Global Positioning System (GPS) (refer to 34-56-00, Global Positioning System (GPS) (Mod 6/2049) – General Data).

5) Weather Radar System (WXR) (refer to 34-41-00, Weather Radar System (Mod 6/2049) – General Data), if installed.

3. Adjustment/Test

Note The procedures in Para 3. Adjustment/Test are written to be stand-alone sub-sections. Each individual procedure includes aircraft and maintenance laptop computer set up and close out procedures. In cases where multiple procedures will be carried out in succession, it is acceptable to proceed to the subsequent procedure without the intermediate close out and set up procedure.

A. KSG 7200 ADAHRS Certification File Loading

1) General

The following procedures are for the loading of the KSG 7200 ADAHRS Certification File.

The KSG 7200 ADAHRS certification file change is required when the aircraft landing gear configuration is changed. Reloading the certification file will be required if the ADAHRS configuration modules are required to be replaced due to defect.

2) ADAHRS Certification File Loading

Note The KSG 7200 has two distinct channels, A and B. The following procedures must be done on both channels. The certification file loading procedure must be completed on both channels independently. Do the procedure on Channel A, then repeat for Channel B.

a) Ensure ADAHRS 1A (1K) and ADAHRS (1B) (2K) circuit breakers on the avionics circuit breaker panel are closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait for the aircraft electrical systems to initialize and the attitude flags to clear.

d) Turn on the maintenance laptop computer and boot up to the desktop.

e) Connect the maintenance laptop computer to the ADAHRS using an RS-232 maintenance cable from an available computer COM port to the ADAHRS 1A maintenance port on the aircraft maintenance panel located in the forward avionics bay. Start the KSG 7200 Maintenance program on the laptop.

f) Select appropriate COM port.

g) Click ‘Start Maintenance'.

h) Click ‘Aircraft Installation Setup'.

i) Click ‘Load KSG 7200 Certification File'. The field maintenance program should close and the ADAHRS loader program should open.

j) Insert the KSG 7200 Certification files CD-ROM into the laptop maintenance computer CD-ROM drive.

Note In some cases the certification file may be located on other media type such as local computer hard drive or removable media such as USB memory stick. In this case during the following steps select the correct file location as applicable.

k) Select the KSG 7200 located at the top of the program.

l) Select the Certification File button and click OK. The file select screen will appear.

m) Select the CD-ROM drive or alternate media containing the certification file(s).

CAUTION **IF THIS OPERATION IS CANCELLED, THE UNIT MUST BE RESTARTED BY REMOVING POWER FOR 5 SECONDS AND THEN REAPPLYING POWER.**

n) Select the configuration file (.cmf) and click open. An upload status bar will appear.

o) Click on OK when the certification file upload is complete.

p) Close the ADAHRS LOADER program by selecting EXIT.

q) Reset the ADAHRS computer by cycling the ADAHRS 1A (1K) and ADAHRS 1B (2K) circuit breakers on the avionics circuit breaker panel.

r) Wait for the KSG 7200 to re-initialize and the attitude flags to clear.

s) Start the KSG 7200 Maintenance program on the maintenance laptop computer.

t) Select appropriate COM port.

u) Click ‘Start Maintenance'.

v) Click ‘Part Numbers'.

w) Verify the “Certification File ID Number” displayed in the part number window matches the part number of the file loaded in previous steps.

x) Close the maintenance program on the maintenance laptop computer.

CAUTION **ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.**

y) Switch the RS-232 maintenance cable to the ADAHRS 1B Maintenance Port on the aircraft maintenance panel and repeat the above certification file loading procedure for Channel B.

z) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

aa) Disconnect and remove the maintenance laptop computer and RS-232 maintenance cable from the aircraft.

B. ADAHRS Alignment

1) General

The following procedures are for the attitude alignment of the ADAHRS system.

2) KSG 7200 ADAHRS Alignment

Note The KSG 7200 has two distinct channels, A and B. The following procedure must be done on both channels. The maintenance laptop computer can be connected to ADAHRS Channel A first and the procedure repeated for Channel B. Alternately a second RS-232 maintenance cable with a USB to RS-232 converter can be utilized for the second serial input to the maintenance laptop. Two separate instances of the KSG 7200 maintenance program must be running in Windows.

a) Ensure ADAHRS 1A (1K) and ADAHRS 1B (2K) circuit breakers on the avionics circuit breaker panel are closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait for the aircraft electrical systems to initialize and the attitude flags to clear.

d) Turn on the maintenance laptop computer and boot up to the desktop.

e) Connect the maintenance laptop computer to the ADAHRS using an ADAHRS 1A and/or ADAHRS 1B maintenance port on the aircraft maintenance panel located in the forward avionics bay.

f) Start the KSG 7200 Maintenance program on the maintenance laptop computer.

g) Select appropriate COM port.

h) Click ‘Start Maintenance'.

i) Click 'Aircraft Installation Setup'.

j) Click 'ADAHRS Alignment'.

Table 1 ADAHRS Pitch and Roll Offsets

Parameter		Setting
KSG 7200 ADAHRS Channel A and Channel B	Pitch Offset	+1.2
	Roll Offset	0
	Heading Offset	0
	Mounting Direction	ADAHRS Connector pointing towards Right Wing of Aircraft

k) Enter the Pitch Offset Value per Table 1 — ADAHRS Pitch and Roll Offsets.

l) Click ‘Set ADAHRS Pitch Offset'.

m) Enter the Roll Offset Value per Table 1 — ADAHRS Pitch and Roll Offsets.

n) Click ‘Set ADAHRS Roll Offset'.

o) Enter the Heading Offset Value per Table 1 — ADAHRS Pitch and Roll Offsets.

p) Click ‘Set ADAHRS Heading Offset'.

q) Click ‘Store ADAHRS Offsets'.

r) When ‘SUCCESS' is displayed, click ‘Close'.

s) Select the ADAHRS Mounting Direction number per Table 1 — ADAHRS Pitch and Roll Offsets.

t) Click ‘Set ADAHRS Mounting Direction'.

u) Click ‘Store ADAHRS Mounting Direction'.

v) When ‘SUCCESS' is displayed, click ‘Close' on the ADAHRS Mounting Installation screen.

w) Click ‘Exit Maintenance'.

x) Wait for the KSG 7200 to re-initialize and the attitude flags to clear.

y) Click ‘Start Maintenance'.

z) Click 'Aircraft Installation Setup'.

aa) Click 'ADAHRS Alignment'.

ab) Confirm Pitch Roll and Heading Offsets and Mounting Direction match the values entered above.

ac) Click 'Close' on the ADAHRS Mounting Installation screen.

ad) Click 'Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

ae) Close the Maintenance Program on the maintenance laptop computer.

CAUTION **ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.**

af) If Channel B alignment procedure was not completed simultaneously with Channel A above, switch the RS-232 maintenance cable to the ADAHRS 1B Maintenance Port on the aircraft maintenance panel and repeat the above KSG 7200 alignment procedure for Channel B.

ag) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

ah) Disconnect and remove the maintenance laptop computer and RS-232 maintenance cables from the aircraft.

3) KMG 7010 Magnetometer Alignment (Pre Mod 6/2171)

a) Ensure that ADAHRS 1A (1K) and ADAHRS 1B (2K) circuit breakers on the avionics circuit breaker panel are closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait for the aircraft electrical systems to initialize and the attitude flags to clear.

d) Turn on maintenance laptop computer and boot up to the desktop.

e) Connect the maintenance laptop computer to the ADAHRS using an RS-232 maintenance cable from an available computer COM port to the ADAHRS 1A and/or ADAHRS 1B maintenance port on the aircraft maintenance panel located in the forward avionics bay.

f) Start the KSG 7200 Maintenance program on the laptop.

g) Select appropriate COM port.

h) Click 'Start Maintenance'.

i) Click 'Aircraft Installation Setup'.

j) Click 'Magnetometer Alignment'.

Table 2 Magnetometer Pitch and Roll Offsets

Parameter		Setting
KMG 7010 Magnetometers A and B	Pitch Offset	+2.2
	Roll Offset Normal Flight Operation	–3.3
	Roll Offset On-Ground Compass Calibration	–2.3
	Mounting Direction	Option "7" Bottom of magnetometer (mounting side of the unit) towards the top of aircraft. Front of magnetometer (connector side of the unit) towards tail of the aircraft.

k) Enter the Magnetometer Pitch Offset Value per Table 2 — Magnetometer Pitch and Roll Offsets.

l) Click 'Set Magnetometer Pitch Offset'.

m) Enter the Magnetometer Roll Offset Value per Table 2 — Magnetometer Pitch and Roll Offsets.

Note If setting up for the compass calibration procedure, enter the Roll Offset for 'On-Ground Compass Calibration'.

If setting up for flight, enter the Roll Offset for 'Normal Flight Operation'.

n) Click 'Set Magnetometer Roll Offset'.

o) Click 'Store Magnetometer Offsets'.

p) When 'SUCCESS' is displayed, click 'Close'.

q) Select the Magnetometer Mounting Direction number per Table 2 — Magnetometer Pitch and Roll Offsets.

r) Click 'Set Magnetometer Mounting Direction'.

s) Click 'Store Magnetometer Mounting Direction'.

t) When 'SUCCESS' is displayed, click 'Close'.

u) Click 'Close' on the Magnetometer Mounting Installation Screen.

v) Click 'Close' on the Installation Setup Screen.

w) Click 'Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

x) Wait for the KSG 7200 to re-initialize and the attitude flags to clear.

y) Click 'Start Maintenance'.

z) Click 'Aircraft Installation Setup'.

aa) Click 'Magnetometer Alignment'.

ab) Confirm pitch and roll offsets and Mounting Direction match the values entered above.

ac) Click ‘Close' on the Magnetometer Mounting Installation screen.

ad) Click ‘Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

ae) Close the Maintenance Program on the maintenance laptop computer.

CAUTION **ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.**

af) If Channel B alignment procedure was not completed simultaneously with Channel A above, switch the RS-232 maintenance cable to the ADAHRS 1B Maintenance Port on the aircraft maintenance panel and repeat the above Magnetometer alignment procedure for Channel B.

ag) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

ah) Disconnect and remove the maintenance laptop computer and RS-232 maintenance cables from the aircraft.

4) KMG 7010 Magnetometer Alignment (Post Mod 6/2171)

a) Ensure that ADAHRS 1A (1K) and ADAHRS 1B (2K) circuit breakers on the avionics circuit breaker panel are closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait for the aircraft electrical systems to initialize and the attitude flags to clear.

d) Turn on maintenance laptop computer and boot up to the desktop.

e) Connect the maintenance laptop computer to the ADAHRS using an RS-232 maintenance cable from an available computer COM port to the ADAHRS 1A and/or ADAHRS 1B maintenance port on the aircraft maintenance panel located in the forward avionics bay.

f) Start the KSG 7200 Maintenance program on the laptop.

g) Select appropriate COM port.

h) Click ‘Start Maintenance'.

i) Click ‘Aircraft Installation Setup'.

j) Click ‘Magnetometer Alignment'.

Table 3 Magnetometer Pitch and Roll Offsets

Parameter		Setting
KMG 7010 Magnetometers A and B	Pitch Offset	0.0
	Roll Offset Normal Flight Operation	0.0
	Mounting Direction	Option “7” Bottom of magnetometer (mounting side of unit) towards the top of aircracft. Front of magnetometer (connector side of the unit) towards tail of the aircraft.

k) Enter the Magnetometer Pitch Offset Value per Table 3 — Magnetometer Pitch and Roll Offsets.

l) Click ‘Set Magnetometer Pitch Offset'.

m) Enter the Magnetometer Roll Offset Value per Table 3 — Magnetometer Pitch and Roll Offsets.

n) Click ‘Set Magnetometer Roll Offset'.

o) Click ‘Store Magnetometer Offsets'.

p) When ‘SUCCESS' is displayed, click ‘Close'.

q) Select the Magnetometer Mounting Direction number per Table 3 — Magnetometer Pitch and Roll Offsets.

r) Click ‘Set Magnetometer Mounting Direction'.

s) Click ‘Store Magnetometer Mounting Direction'.

t) When ‘SUCCESS' is displayed, click ‘Close'.

u) Click ‘Close' on the Magnetometer Mounting Installation Screen.

v) Click ‘Close' on the Installation Setup Screen.

w) Click ‘Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

x) Wait for the KSG 7200 to re-initialize and the attitude flags to clear.

y) Click ‘Start Maintenance'.

z) Click 'Aircraft Installation Setup'.

aa) Click ‘Magnetometer Alignment'.

ab) Confirm pitch and roll offsets and Mounting Direction match the values entered above.

ac) Click ‘Close' on the Magnetometer Mounting Installation screen.

ad) Click ‘Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

ae) Close the Maintenance Program on the maintenance laptop computer.

CAUTION **ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.**

af) If Channel B alignment procedure was not completed simultaneously with Channel A above, switch the RS-232 maintenance cable to the ADAHRS 1B Maintenance Port on the aircraft maintenance panel and repeat the above Magnetometer alignment procedure for Channel B.

ag) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

ah) Disconnect and remove the maintenance laptop computer and RS-232 maintenance cables from the aircraft.

C. ADAHRS System Function Test

1) General

The following procedures are for the function test of the ADAHRS System.

2) ADAHRS Function Test

Note The KSG 7200 has two distinct channels, A and B. The following procedure must be done on both channels. The maintenance laptop computer can be connected to ADAHRS Channel A first and the procedure repeated for Channel B. Alternately a second RS-232 maintenance cable with a USB to RS-232 converter can be utilized for the second serial input to the maintenance laptop. Two separate instances of the KSG 7200 maintenance program must be running in Windows.

a) Ensure ADAHRS 1A (1K) and ADAHRS 1B (2K) circuit breakers on the avionics circuit breaker panel are closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait for the aircraft electrical systems to initialize and the attitude flags to clear.

d) Verify correct display of ADAHRS parameters on pilot and co-pilot PFD's, including attitude, heading, airspeed, altitude, vertical speed, and OAT.

e) Turn on the maintenance laptop computer and boot up to the desktop.

f) Connect the maintenance laptop computer to the ADAHRS using an RS-232 maintenance cable from an available computer COM port to the ADAHRS 1A and/or ADAHRS 1B maintenance port on the aircraft maintenance panel located in the forward avionics bay.

g) Start the KSG 7200 Maintenance program on the maintenance laptop computer.

h) Select the appropriate COM port.

i) Click ‘Start Maintenance'.

j) Click ‘Built in Test Results'.

k) Verify all parameters indicate ‘Passed'.

l) Click ‘Close'.

m) Click ‘Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

n) Close the Maintenance Program on the laptop maintenance computer.

CAUTION ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.

o) If Channel B ADAHRS System Function Test procedure was not completed simultaneously with Channel A above, switch the RS-232 maintenance cable to the ADAHRS 1B Maintenance Port on the aircraft maintenance panel and repeat the above ADAHRS Function Test procedure for Channel B.

p) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

q) Remove the maintenance laptop computer and RS-232 maintenance cables from the aircraft.

D. ADAHRS (KSG 7200) Air Data System Verification

1) General

The following procedures are for the Air Data System Verification of the ADAHRS System.

2) ADAHRS (KSG 7200) Air Data System Verification

a) Ensure that ADAHRS 1A (1K) and ADAHRS 1B (2K) circuit breakers on the avionics circuit breaker panel are closed.

b) Ensure that the 'PITOT HEAT' switch is off and the PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers on the main circuit breaker panel are pulled and fitted with safety clips.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Wait for the aircraft electrical systems to initialize and the attitude flags to clear.

e) Leak test the Pitot-Static System. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Adjustment/Test).

f) Ensure that the (ESIS) STATIC SOURCE valve is in the NORMAL position.

g) Connect the air data test set to the pitot heads and static ports on the aircraft. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Adjustment/Test).

h) Complete the Altitude Calibration Table. Refer to Table 4 — Altitude Calibration Table.

Note Additional maintenance and inspection requirements for the Altimeter System may be mandated by the regulations of the jurisdiction under which the aircraft is registered. In the event of disagreement between the PSM 1-64-2 and these additional requirements, the more restrictive requirements shall take precedence unless an alternative program of maintenance has been approved by the regulatory authority.

Table 4 Altitude Calibration Table

Test Set Altitude (feet)	Tolerance (± feet)	Record Error		REMARKS	INITIALS
		CH A	CH B		
-1,000	20				
0	20				
500	20				
1,000	20				
1,500	25				
2,000	25				
3,000	25				
4,000	25				
5,000	25				
6,000	25				
8,000	30				
10,000	30				

Table 4 Altitude Calibration Table (continued)

Test Set Altitude (feet)	Tolerance (± feet)	Record Error		REMARKS	INITIALS
		CH A	CH B		
12,000	35				
14,000	40				
16,000	40				
18,000	45				
20,000	50				
22,000	50				
25,000	50				

i) Complete the Airspeed Calibration Table. Refer to Table 5 — Airspeed Calibration Table

Note Additional maintenance and inspection requirements for the Airspeed System may be mandated by the regulations of the jurisdiction under which the aircraft is registered. In the event of disagreement between the PSM 1-64-2 and these additional requirements, the more restrictive requirements shall take precedence unless an alternative program of maintenance has been approved by the regulatory authority.

Table 5 Airspeed Calibration Table

Test Set Airspeed (knots)	Tolerance (± knots)	Record Error		REMARKS	INITIALS
		CH A	CH B		
30	7.0				
50	5.0				
80	3.5				
100	2.0				
120	2.0				
150	2.0				
200	2.0				

j) Disconnect the Pitot/Static Test Set from the pitot and static ports on the aircraft.

k) Ensure any tape has been removed and pitot and static ports are clear.

l) Remove the safety clips and reset the PITOT HEAT L (A6) and PITOT HEAT R (B6) circuit breakers on the main circuit breaker panel.

m) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

E. ADAHRS (KSG 7200) Compass Calibration & Verification

1) General

The following procedures are for the Compass Calibration and verification of the ADAHRS system.

2) ADAHRS System Compass Calibration (Pre Mod 6/2171)

Note The procedure below includes steps to adjust the magnetometer roll offset to compensate for wing flex arising from steady-state non-maneuvering flight. This wing flex can induce long-term compass heading errors, particularly in geographic areas with higher magnetic field dip angles (i.e., lower horizontal magnetic field strength). Heading errors induced by wing flex are reduced and considered negligible in geographic areas with small magnetic field dip angles as is the case for aircraft located in more equatorial regions. These considerations apply equally to northern and southern latitudes.

For best results the compass calibration procedure should be done on level ground without obstructions nearby. It is also recommended the calibration be done on a calm day. Excessive wind can rock the aircraft and result in compass errors, particularly in geographic locations with high magnetic field dip angles.

The compass calibration may be completed one channel at a time. However, more accurate results can be achieved by connecting both channels simultaneously. A second RS-232 maintenance cable with a USB to RS-232 converter can be utilized for the second serial input to the maintenance laptop. Two separate instances of the KSG 7200 maintenance program must be running in Windows.

a) Prior to compass calibration, ensure Magnetometer 'On-Ground Compass Calibration Procedure' Roll Offset is entered in the KMG 7010 Magnetometer Alignment screen. Refer to Para 3.B.3) KMG 7010 Magnetometer Alignment (Pre Mod 6/2171).

b) Connect the maintenance laptop computer to the ADAHRS using an RS-232 maintenance cable from an available computer COM port to the ADAHRS 1A and/or ADAHRS 1B maintenance port on the aircraft maintenance panel located in the forward avionics bay.

Note The RS-232 maintenance cables for channel A and B can be routed from the maintenance panel to the flight compartment area, for convenience, for the calibration procedure.

c) Position aircraft on compass rose.

Note A ground support technician using a land sight compass with an accuracy of ± 2 degrees can be used if a compass rose is not available.

Note Less desirable and only if required due to circumstances compass calibration may also be done against a known calibrated on board compass source such as the offside ADAHRS Channel or ESIS instrument on condition the compass source has been verified by external means for accuracy prior to commencement of the following ADAHRS compass calibration procedures.

d) Ensure engines, avionics, and all normal electrical flight loads are operating for the duration of the compass calibration.

e) Turn on maintenance laptop computer and boot up to the desktop.

f) If not already completed, apply power to the ADAHRS. Wait for a minimum of 2 minutes and until the attitudes stabilize.

g) Start the KSG 7200 Maintenance program on the laptop.

h) Select appropriate COM port.

i) Click ‘Start Maintenance'.

j) Click ‘Aircraft Installation Setup'.

k) Click ‘Compass Calibration'.

l) Click ‘Perform Compass Calibration using 12 points on the Compass Rose'.

m) Align the aircraft to the heading requested in the compass rose calibration window.

n) **IMPORTANT** Wait a minimum of 1 minute to let the ADAHRS system stabilize at each new heading.

o) Click ‘Take Magnetometer Reading' when the aircraft is aligned to the requested heading.

p) Repeat the compass calibration at all 12 points observing the minimum 1 minute to stabilize at all headings.

q) Upon completion of all 12 heading calibrations the ‘Store Compass Rose Calibration Data' button becomes active.

r) Click ‘Store Compass Rose Calibration Data'.

s) When ‘SUCCESS' is displayed, click ‘Close'.

t) Click ‘Close'.

u) Click ‘Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

v) Close the Maintenance Program on the maintenance laptop computer.

w) Keep aircraft stationary for a minimum of 2 minutes before proceeding.

CAUTION **ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.**

x) If Channel B alignment procedure was not completed simultaneously with Channel A above, switch the RS-232 maintenance cable to the ADAHRS 1B Maintenance Port on the aircraft maintenance panel and repeat the above compass calibration procedure for Channel B.

y) Do the compass accuracy verification, outlined in Para 3.E.4).

3) ADAHRS System Compass Calibration (Post Mod 6/2171)

Note For best results the compass calibration procedure should be done on level ground without obstructions nearby. It is also recommended the calibration be done on a calm day. Excessive wind can rock the aircraft and result in compass errors, particularly in geographic locations with high magnetic field dip angles.

The compass calibration may be completed one channel at a time. However, more accurate results can be achieved by connecting both channels simultaneously. A second RS-232 maintenance cable with a USB to RS-232 converter can be utilized for the second serial input to the maintenance laptop. Two separate instances of the KSG 7200 maintenance program must be running in Windows.

a) Connect the maintenance laptop computer to the ADAHRS using an RS-232 maintenance cable from an available computer COM port to the ADAHRS 1A and/or ADAHRS 1B maintenance port on the aircraft maintenance panel located in the forward avionics bay.

Note The RS-232 maintenance cables for channel A and B can be routed from the maintenance panel to the flight compartment area, for convenience, for the calibration procedure.

b) Position aircraft on compass rose.

Note A ground support technician using a land sight compass with an accuracy of ± 2 degrees can be used if a compass rose is not available.

Note Less desirable and only if required due to circumstances compass calibration may also be done against a known calibrated on board compass source such as the offside ADAHRS Channel or ESIS instrument on condition the compass source has been verified by external means for accuracy prior to commencement of the following ADAHRS compass calibration procedures.

c) Ensure engines, avionics, and all normal electrical flight loads are operating for the duration of the compass calibration.

d) Turn on maintenance laptop computer and boot up to the desktop.

e) If not already completed, apply power to the ADAHRS. Wait for a minimum of 2 minutes and until the attitudes stabilize.

f) Start the KSG 7200 Maintenance program on the laptop.

g) Select appropriate COM port.

h) Click ‘Start Maintenance'.

i) Click ‘Aircraft Installation Setup'.

j) Click ‘Compass Calibration'.

k) Click ‘Perform Compass Calibration using 12 points on the Compass Rose'.

l) Align the aircraft to the heading requested in the compass rose calibration window.

m) **IMPORTANT** Wait a minimum of 1 minute to let the ADAHRS system stabilize at each new heading.

n) Click ‘Take Magnetometer Reading' when the aircraft is aligned to the requested heading.

o) Repeat the compass calibration at all 12 points observing the minimum 1 minute to stabilize at all headings.

p) Upon completion of all 12 heading calibrations the ‘Store Compass Rose Calibration Data' button becomes active.

q) Click ‘Store Compass Rose Calibration Data'.

r) When ‘SUCCESS' is displayed, click ‘Close'.

s) Click ‘Close'.

t) Click ‘Exit Maintenance'. The ADAHRS will restart when the maintenance program is exited.

u) Close the Maintenance Program on the maintenance laptop computer.

v) Keep aircraft stationary for a minimum of 2 minutes before proceeding.

CAUTION **ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.**

w) If Channel B alignment procedure was not completed simultaneously with Channel A above, switch the RS-232 maintenance cable to the ADAHRS 1B Maintenance Port on the aircraft maintenance panel and repeat the above compass calibration procedure for Channel B.

x) Do the compass accuracy verification, outlined in Para 3.E.4).

4) Compass Accuracy Verification

a) Prior to compass accuracy verification (Pre Mod 6/2171), ensure Magnetometer On-Ground Compass Calibration Procedure Roll Offset is entered in the KMG 7010 Magnetometer Alignment maintenance screen. Refer to Para 3.B.3) KMG 7010 Magnetometer Alignment (Pre Mod 6/2171).

b) Ensure engines, avionics and all normal electrical flight loads are operating for the duration of the compass verification.

c) Align the aircraft to each of the four cardinal headings on the compass rose and verify compass accuracy ± 2.0 degrees to actual aircraft heading.

Note Additional maintenance and inspection requirements for the Compass System may be mandated by the regulations of the jurisdiction under which the aircraft is registered. In the event of disagreement between the PSM 1-64-2 and these additional requirements, the more restrictive requirements shall take precedence unless an alternative program of maintenance has been approved by the regulatory authority.

d) If any heading falls outside the listed specifications, do the compass calibration procedure listed in Para 3.E.2) (Pre Mod 6/2171) or Para 3.E.3) (Post Mod 6/2171).

5) Wing Flex Compensation (Pre Mod 6/2171)

a) Following compass accuracy verification procedure, compensate for in-flight wing flex by entering the Magnetometer Normal Flight Operation Roll Offset into the KMG 7010 Magnetometer Alignment maintenance screen. Refer to Para 3.B.3) KMG 7010 Magnetometer Alignment (Pre Mod 6/2171).

Note Compensating for in-flight wing flex by adjusting the Magnetometer Roll Offset following compass calibration will have varying effect on calibrated compass headings based on geographic location and magnetic field dip angle. Most effect will be noticed in Northern or Southern latitudes above/below 45 degrees latitude with high magnetic field dip angles indicated by up to 2 or 3 degree change in ground compass calibration readings. but this is required to ensure in-flight compass accuracy. Wing flex will have minimal effect in equatorial geographic areas with low magnetic field dip angles, but the wing flex compensation procedure should still be completed to ensure best in-flight compass performance.

F. Configuration Module Programming

1) General

The following procedures are for programming a new configuration module.

2) Configuration Module Programming

Note The following procedure can only be accomplished if the ADAHRS was previously installed and operating in the aircraft.

a) Ensure ADAHRS 1A (1K) and ADAHRS 1B (2K) circuit breakers on the avionics circuit breaker panel are closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait for the aircraft electrical systems to initialize and the attitude flags to clear.

d) Turn on maintenance laptop computer and boot up to the desktop.

e) Connect the maintenance laptop computer to the ADAHRS using an RS-232 maintenance cable from an available computer COM port to the ADAHRS 1A or ADAHRS 1B (as required) maintenance port on the aircraft.

f) Start the KSG 7200 Maintenance program on the maintenance laptop computer.

g) Select appropriate COM port.

h) Click ‘Start Maintenance'.

i) Click ‘NVM/Configuration Module Data Transfer'.

j) Click ‘Copy NVM Data to Configuration Module'.

k) Wait 60 seconds to assure transfer is complete.

l) Close the maintenance program on the maintenance laptop computer.

m) Wait for the ADAHRS to initialize and the attitude flags to clear.

n) Start the KSG 7200 Maintenance program on the maintenance laptop computer.

o) Select appropriate COM port.

p) Click ‘Start Maintenance'.

q) Click ‘Part Numbers'.

r) Verify valid data is present.

s) Close the maintenance program on the maintenance laptop computer.

CAUTION **ALWAYS ENSURE THE ADAHRS MAINTENANCE PROGRAM ON THE MAINTENANCE LAPTOP COMPUTER IS CLOSED PRIOR TO DISCONNECTING THE RS-232 MAINTENANCE CABLES BETWEEN THE COMPUTER AND ADAHRS.**

t) If the other configuration module also requires programming, switch the RS-232 maintenance cable to the ADAHRS 1A or ADAHRS 1B Maintenance Port (as applicable) on the aircraft maintenance panel and repeat the above configuration module programming procedure.

u) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

v) Remove the maintenance laptop computer and RS-232 maintenance cables from the aircraft.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the ADAHRS (KSG 7200)

1) General

The following procedures are for the removal and installation of the ADAHRS (KSG 7200).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Remove middle cover attached to bulkhead station 37.21.

e) Disconnect the Pitot and Static lines from the front of the unit.

f) Disconnect the electrical connectors from the front of the unit.

g) Loosen retaining screw clamp from the front and slide the unit out of the mounting tray.

h) Remove unit from aircraft.

Note Stow connectors of the removed components. Install the pitot and static line covers.

i) Visually inspect mounting tray. If damaged replace as per Para 4.B.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Slide the unit into the mounting tray. Tighten the two retaining screw clamps at the front lip of the ADAHRS unit.

d) Reconnect the electrical connector to the unit. Ensure connections are secure. Use lockwire .025 to secure the two retaining screws on channel A and B ADAHRS connectors.

e) Ensure Pitot and Static lines are free of debris. Reconnect the Pitot and Static lines. Ensure connections are secure.

f) Do a leak check on the Pitot and Static system as per 34-11-00, Pitot-Static System (Mod 6/2033) – General Data.

g) Install middle cover attached to bulkhead station 37.21.

WARNING **ACCESS COVER IS REQUIRED TO MAINTAIN PROTECTION FEATURES THAT ARE ESSENTIAL TO MINIMIZING ADVERSE EFFECTS FROM HIGH INTENSITY RADIATED FIELDS (HIRF).**

h) Remove safety clips and close ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel.

i) Test the ADAHRS as per Para 3.C and Para 3.D.

B. Removal and Installation of the ADAHRS Mounting Tray

1) Removal

a) Remove ADAHRS as per Para 4.A.2).

b) Remove the four screws that secure mounting tray to shelf.

c) Remove mounting tray from aircraft.

2) Installation

a) Locate mounting tray on avionic shelf, align mounting holes. Ensure the tray in oriented properly on the shelf, hold down bolts are inboard.

b) Install the four screws and ensure tray is securely held on shelf.

C. Removal and Installation of the Magnetometers (KMG 7010)

1) General

The following procedures are for the removal and installation of the magnetometers (KMG 7010).

CAUTION **ONLY NON-MAGNETIC TOOLS ARE PERMITTED WITHIN 12 INCHES OF THE MAGNETOMETER.**

2) Removal (Pre Mod 6/2171)

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Locate the panel on the starboard (RH) wing between station 272.5 and station 285.0.

d) Remove the (8) screws securing the panel to the wing with non-magnetic tools.

Note ADAHRS channel A magnetometer is mounted inboard, channel B outboard.

e) Remove the four mounting screws with non-magnetic tools.

f) Disconnect the electrical connector from the aft side of the unit.

g) Remove the unit from the aircraft.

Note Stow connectors of the removed components.

Note Store the magnetometer in a secure area, away from magnetic fields. Typically, a distance of 12 inches is considered safe for most magnetic fields but actual safe distance varies with the strength of the magnetic source. Exposing the magnetometer to any magnetic fields during transport, storage, re-packaging, or installation may disrupt its calibration. To minimize exposure of the magnetometer to strong magnetic fields, keep it away from the following devices or objects:

1 High Current Carrying Cables

2 Magnetic Tools or Fasteners

3 Permanent Magnets

4 Cathode Ray Tubes / Computer Monitors

5 Magnetic Surfaces

6 Motors & Servos

3) Installation (Pre Mod 6/2171)

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Reconnect the electrical connector to the unit. Ensure connections are secure. Use lockwire .025 to secure the two retaining screws on the connector.

d) Set unit in position and install the four mounting screws with non-magnetic tools.

e) Verify the electrical conductivity between the magnetometer and airframe is < 2.5 milliohms.

f) Install the panel screws (8) and secure to the wing with non-magnetic tools.

g) Remove safety clips and close ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel.

h) Do a function test of the ADAHRS as per Para 3.C.

i) Do Compass Calibration & Verification procedure per Para 3.E.2) and Para 3.E.4).

j) Do Wing Flex Compensation per Para 3.E.5).

4) Removal (Post Mod 6/2171)

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Remove fairings enclosing vertical stabilizer and horizontal stabilizer to rear fuselage. Refer to 12-00-00, Servicing – General Data, Figure 1, Access and Inspection Provisions.

Note ADAHRS channel A magnetometer is mounted on top, directly to mounting tray, channel B magnetometer is mounted below, using spacers.

d) Remove the four mounting screws, and four spacers with non-magnetic tools.

e) Disconnect the electrical connector from the aft side of the each unit.

f) Remove the units from the aircraft.

Note Stow connectors of the removed components.

Note Store the magnetometer in a secure area, away from magnetic fields. Typically, a distance of 12 inches is considered safe for most magnetic fields but actual safe distance varies with the strength of the magnetic source. Exposing the magnetometer to any magnetic fields during transport, storage, re-packaging, or installation may disrupt its calibration. To minimize exposure of the magnetometer to strong magnetic fields, keep it away from the following devices or objects:

1 High Current Carrying Cables

2 Magnetic Tools or Fasteners

3 Permanent Magnets

4 Cathode Ray Tubes / Computer Monitors

5 Magnetic Surfaces

6 Motors & Servos

5) Installation (Post Mod 6/2171)

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Reconnect the electrical connector to the unit. Ensure connections are secure. Use lockwire .025 to secure the two retaining screws on the connector.

d) Hold units in position on mounting tray.

Note ADAHRS channel A magnetometer is mounted on top, directly to mounting tray, channel B magnetometer is mounted below, using spacers.

e) Install four spacers, and four mounting screws with non-magnetic tools.

f) Verify the electrical conductivity between the magnetometer and airframe is < 2.5 milliohms.

g) Install fairings enclosing vertical stabilizer and horizontal stabilizer to rear fuselage.

h) Remove safety clips and close ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel.

i) Do a function test of the ADAHRS as per Para 3.C.

j) Do Compass Calibration & Verification procedure per Para 3.E.3) and Para 3.E.4).

D. Removal and Installation of the Outside Air Temperature (OAT) Probe

1) General

The following procedures are for the removal and installation of the Outside Air Temperature (OAT) probe.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the probe.

d) Remove the screws (6) from the base of the probe.

e) Carefully lift the probe clear of the fuselage until there is access to the electrical connector.

f) Disconnect the electrical connector.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Reconnect the electrical connector to the unit. Ensure connection is secure.

d) Install the screws (6) securing the probe to the fuselage.

e) Solvent clean the exposed area of the airframe around the probe (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

f) Touch up the exposed area of airframe around the probe if required with F19 primer by brush within 24 hours of the probe installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

g) Use PR 1422 sealant to both fillet seal around the probe base and dome seal the securing screws. The aircraft will be ready to fly approximately 24 hours after sealing.

h) Remove safety clips and close ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel.

i) Do the ADAHRS System Function Test procedure as per Para 3.C.

E. Removal and Installation of the Outside Air Temperature (OAT) Probe (S.O.O. 6304)

1) General

The following procedures are for the removal and installation of the Outside Air Temperature (OAT) probe.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the probe.

d) Remove the screws (6) from the base of the probe.

e) Carefully cut fay seal between base of probe and aircraft skin and lift the probe clear of the fuselage until there is access to the electrical connector.

f) Disconnect the electrical connector.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Ensure leftover sealant is carefully removed from the base of the probe and the aircraft skin.

d) Reconnect the electrical connector to the unit. Ensure connection is secure.

e) Apply PR-1422 sealant to faying surface of the probe base and aircraft skin.

f) Install the screws (6) securing the probe to the fuselage.

g) Do a bonding check between the probe and aircraft structure (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

h) Touch up the exposed area of airframe used for a bonding check with alodine and F19 primer, by brush, within 24 hours of the probe installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

i) Use PR-1422 sealant to both fillet seal around the probe base and dome seal the securing screws. The aircraft will be ready to fly approximately 24 hours after sealing.

j) Remove safety clips and close ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel.

k) Do the ADAHRS System Function Test procedure as per Para 3.C.

F. Removal and Installation of the KCM 200 Configuration Modules

1) General

The following procedures are for the removal and installation of the KCM 200 Configuration Modules.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Remove middle cover attached to bulkhead station 37.21.

e) Remove electrical connector from the applicable channel of the ADAHRS.

f) Remove backshell from electrical connector and extract the configuration module pins from connector. Refer to ADAHRS wiring diagram for pinout information.

g) Cut cable ties as required to free the configuration module from aircraft harness.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the ADAHRS 1A (1K) (Left Bus) and ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Prepare the configuration module harness as required to attach to the ADAHRS connector harness. Insert the configuration module pins into the connector. Refer to ADAHRS wiring diagram.

d) Install backshell onto connector and reconnect electrical connector to ADAHRS

e) Secure the configuration module into aircraft harness using cable ties, as required.

f) If a new KCM 200 Configuration Module is installed, programming is required:

1 If the original ADAHRS unit is installed, the configuration module can be programmed by the ADAHRS, refer to Para 3.F.

2 If the original ADAHRS is not installed, the configuration module will need to be programmed per Para 3.A, Para 3.B, and Para 3.E. Once programming is complete, do a function test per Para 3.C.

5. Troubleshooting

A. Air Data and Attitude Heading Reference System (KSG 7200)

1) If the Air Data and Attitude Heading Reference System (KSG 7200) is deemed inoperative check the following:

a) Ensure ADAHRS 1A (1K) (Left Bus) and the ADAHRS 1B (2K) (Right Bus) circuit breakers on the avionics circuit breaker panel are closed.

b) Check that the units are securely installed and the locking screw is securely tightened.

c) Visually inspect ADAHRS system connectors for damaged wires or contacts.

d) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

e) If repair or service is required, send to appropriate rated avionics repair facility.

6. Servicing

It is recommended that a compass accuracy verification be done (refer to Para 3.E.4)) whenever any of the following occur:

- Accuracy of the compass is suspected
- A modification or major replacement involving ferrous metals
- After lightning strike
- Whenever a modification or new installation is made to the aircraft electrical system

- When an aircraft operation is changed to a different geographical location with a major change in magnetic deviation and/or magnetic dip angle
- When magnetometers are replaced

Note Additional maintenance and inspection requirements for the installation of the ADAHRS may be mandated by the regulations of the jurisdiction under which the aircraft is registered. In the event of disagreement between the PSM 1-64-2 and these additional requirements, the more restrictive requirements shall take precedence unless an alternative program of maintenance has been approved by the regulatory authority.

A. Magnetometers (KMG 7010)

1) Magnetometer operation is tested as per Para 3.E.4) – Compass Accuracy Verification.

B. Outside Air Temperature (OAT) Probe

1) Outside Air Temperature (OAT) probe operation is tested as per Para 3.C – ADAHRS System Function Test.

Standby Magnetic Compass – General Data

1. General

The vertical card magnetic compass (PAI–700) is used to indicate relative bearing to the earths magnetic poles.

2. Description

A. Vertical Card Magnetic Compass (PAI–700)

Unlike a “wet” compass the PAI–700 utilizes eddy current magnetic damping, no fluid is inside the instrument. The rotation of the sensing magnet is transferred to the azimuth compass card which indicates the relative bearing of the aircraft. Instrument lighting is controlled by the INSTRUMENT PANEL dim potentiometer.

3. Adjustment/Test

A. Function Test

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Start both engines. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

3) Wait several minutes for the aircraft electrical and avionics systems to initialize.

4) Move the prop levers to the cruise prop rpm position, (full aft) not feathered.

5) All in flight systems should be turned on.

6) Rotate the aircraft to a known magnetic NORTH heading. Note the degrees of deviation.

7) Rotate the aircraft to a known magnetic SOUTH heading. Note the degrees of deviation.

8) Verify the NORTH and SOUTH degrees of deviation is the same. If the degrees of deviation are not the same, do a compensation procedure, refer to Para 3.B.

9) Rotate the aircraft to a known magnetic EAST heading. Note the degrees of deviation.

10) Rotate the aircraft to a known magnetic WEST heading. Note the degrees of deviation.

11) Verify the EAST and WEST degrees of deviation is the same. If the degrees of deviation are not the same, do a compensation procedure, refer to Para 3.B.

12) Turn off all in flight systems that were turned on for the test.

13) Turn the engines OFF. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

14) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Error Compensation

CAUTION **USE A NON-MAGNETIC SCREW DRIVER FOR ADJUSTMENTS.**

Note Instrument friction during the heading checks can be minimized by tapping the top right hand portion of the case three to four times with a wooden pencil at each heading or by slightly tapping the upper right hand portion of the case with your forefinger.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Start both engines. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

3) Wait several minutes for the aircraft electrical and avionics systems to initialize.

4) Move the prop levers to the cruise prop rpm position, (full aft) not feathered.

5) All in flight systems should be turned on.

6) Position the aircraft to a known magnetic NORTH heading.

7) Use the N-S (LH screw) adjustment screw to remove all deviation. Verify the compass indicates N.

8) Rotate the aircraft to a known magnetic EAST heading.

9) Use the E-W (RH screw) adjustment screw to remove all deviation. Verify the compass indicates E.

10) Rotate the aircraft to a known magnetic SOUTH heading. Note the degrees of deviation.

11) Remove half of the total deviation from the SOUTH heading.

12) Rotate the aircraft to a known magnetic WEST heading. Note the degrees of deviation.

13) Remove half of the total deviation from the WEST heading.

14) Rotate the aircraft around the compass card (0 deg. to 330 deg.) at every 30 degree increment. Note the deviation.

15) Record the 30 degree deviation error on a compass correction card. Maximum tolerance of the compass deviation is 10 degrees.

Note Additional maintenance and inspection requirements for the installation of the Magnetic Compass may be mandated by the regulations of the jurisdiction under which the aircraft is registered. In the event of disagreement between the PSM 1-64-2 and these additional requirements, the more restrictive requirements shall take precedence unless an alternative program of maintenance has been approved by the regulatory authority.

16) Turn off all in flight systems that were turned on for the test.

17) Turn the engines off. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

18) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal and Installation

A. Removal and Installation Vertical Card Magnetic Compass (PAI–700)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove the screws (4) securing the compass unit to the windshield center post cover.

c) Disconnect the molex lighting connector.

d) Remove compass from aircraft.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Connect the molex lighting connector.

c) Install the screws (4) securing the compass unit to the windshield center post cover.

d) Do a function test, refer to Para 3.A.

5. Troubleshooting

A. Vertical Card Magnetic Compass (PAI–700)

1) Visually inspect the electrical connectors for damaged wiring or contacts. Ensure connectors are fully seated and tight.

2) If troubleshooting at a wire level is required, refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

3) If repair or service is required, send to appropriately rated avionics repair facility.

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Marker Beacon – General Data

1. General

The Marker Beacon (MKR) uses visual and audio indicators to alert the crew for when the aircraft passes over a 75 MHz transmitter. The Marker Beacon receiver is contained in the audio control panel (KMA 29) (refer to 23-53-00, Audio Control Panel – General Data).

2. Description

A. Marker Beacon

Marker Beacon receive signals are sent to the Modular Avionics Unit (MAU) and then displayed on the Primary Flight Display (PFD). The Outer Marker has an associated 400-Hertz ‘dash' tone. The tone will be keyed at a rate of two tones per second when the aircraft is in the range of the Outer Marker Beacon. The Middle Marker is coupled with a 1300-Hertz tone. It is keyed alternately with short ‘dot' and long ‘dash' bursts at 95 combinations per minute. The Inner Marker has a 3000-Hertz ‘dot' tone. The tone will be keyed at a rate of six times per second.

The audio from the Marker Beacon receiver can be heard by selecting the “MKR” push-button switch. To adjust the volume level, there is a service adjustment located on the unit. The Marker Beacon CI 118 antenna is installed on the bottom of the fuselage, right of the centre line, between station 73.00 and 86.00.

3. Adjustment/Test

A. Function Test

Required Equipment:

- IFR 4000 or equivalent
- Milliohm or Micro-Ohmmeter (AEMC Micro-Ohmmeter Model 6250 or similar – required accuracy: 0.0005 Ω)

1) General

These procedures are for the function test of the Marker Beacon.

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that PILOT AUDIO (3E) and COPILOT AUDIO (4E) circuit breakers are closed.

d) Press and hold the MKR MUTE/TEST button on the pilot audio control panel.

e) Verify the inner marker indication (white boxed I) is displayed on the PFD.

f) Set the test set to generate each of the following:

- 400 Hz tone for Outer Marker (blue boxed ‘O’)
- 1300 Hz tone for Middle Marker (amber boxed ‘M’)
- 3000 Hz tone for Inner Marker (white boxed ‘I’)

Note For each generated tone, make sure the respective marker indication is displayed on both PFD’s.

g) Remove test set.

h) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Audio Control Panel Marker Beacon Audio Adjustment

1) General

These procedures are for Marker Beacon audio level adjustment.

2) Adjustment

a) Ensure that PILOT AUDIO (3E) circuit breaker is closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait several minutes for the aircraft electrical systems to initialize.

d) Set up the test equipment to generate a 400, 1300 or 3000 Hertz tone.

e) Connect a headset to the pilot position.

f) Select MKR push button switch on the pilot audio control panel and ensure the 400, 1300, or 3000 Hertz tone is heard.

g) Evaluate the audio level.

h) Remove the pilot's audio control panel as per 23-53-00, Audio Control Panel – General Data, Para 4.A.2).

i) Adjust the pilot's audio panel MRK potentiometer adjustment on the top of the unit.

j) Install the pilot's audio control panel as per 23-53-00, Audio Control Panel – General Data, Para 4.A.3).

k) Test the pilot's audio control panel as per 23-53-00, Audio Control Panel – General Data, Para 3.A.

l) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Electrical Bonding Test – Marker Beacon Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to right-hand Access and Inspection Provision 15 panel. (Refer to 12-00-00, Servicing – General Data).

3) Turn DZUS fastener 1/4 turn and swing open access panel.

4) Locate Marker Beacon Antenna aft of station 73.0.

5) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the Marker Beacon Antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be 3.0 milliohms unless stated otherwise.

6) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

7) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

8) Close right-hand Access and Inspection Provision 15 and tighten the DZUS fasteners.

4. Removal and Installation

A. Removal and Installation of the Marker Beacon

The Marker Beacon receiver is contained in the Audio Control Panel (KMA 29). Refer to 23-53-00, Audio Control Panel – General Data Para 4. for Removal/Installation of the Audio Control Panel (KMA 29).

B. Removal and Installation of the Marker Beacon Antenna (CI 118)

1) General

These procedures are for the removal and installation of the Marker Beacon antenna (CI 118).

2) Removal

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Pull the PILOT AUDIO (3E) and/or the COPILOT AUDIO (4E) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the screws (6) from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the PILOT AUDIO (3E) and/or the COPILOT AUDIO (4E) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clips.

c) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Install the screws (6) securing the antenna to the fuselage.

f) Check the antenna electrical bonding as per Para 3.C.

g) Remove safety clips and close PILOT AUDIO (3E) and/or the COPILOT AUDIO (4E) circuit breaker on the avionics circuit breaker panel.

h) Test the Marker Beacon as per Para 3.A.

i) Touch up the exposed area of airframe around the antenna (if required) with F19 primer by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

j) Use PR-1422 sealant to both fillet seal around the antenna base and dome seal the securing screws. The aircraft will be ready to fly approximately 24 hours after sealing.

5. Troubleshooting

The Marker Beacon receiver is contained in the Audio Control Panel (KMA 29). Refer to 23-53-00, Audio Control Panel – General Data, Para 5. for Troubleshooting of the Audio Control Panel (KMA 29).

Independent Position Determining – General Data

1. General

The independent position determining system has the following sub-systems:

A. Weather Radar System

Refer to 34-41-00, Weather Radar System (Mod 6/2049) – General Data for system details.

The RDR 2000 weather radar (Radio Detecting and Ranging) system gives the pilot a selectable horizontal display of thunderstorms or high density precipitation in front of the aircraft. When a pulse intercepts a target, the energy is reflected as an echo, or return signal, back to the antenna. From the antenna, the returned signals are transferred to the receiver/transmitter unit. The returned signals, are then displayed.

For aircraft incorporating customer Standard Order Option S.O.O. 6388 the Weather Radar System is not installed.

For aircraft with S.O.O. 6388 incorporated, the Weather Radar System and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6383.

B. Stormscope System

Refer to 34-41-10, Stormscope – General data for system details.

The Stormscope (WX–500) System maps electromagnetic signals (lightning) 360 degrees around the aircraft and to a maximum radius of 200 nautical miles.

The Stormscope system consists of a receiver installed in the rear avionics rack and a top mounted antenna installed at FS400. A receiver inhibit line is interfaced to the aircraft audio control panel ensuring that erroneous strike signals are not displayed when a radio transmitter is keyed.

C. Radar Altimeter System

Refer to 34-42-00, Radar Altimeter System (Mod 6/2049) – General Data for system details.

The KRA 405B radar altimeter transceiver is installed in the empennage aft of the rear baggage compartment. A second radar altimeter can be fitted if desired. The transceiver sends a signal to the transmit antenna and gets the return signal from the receive antenna. The maximum operating height AGL used by the system is 2,500 ft.

The radar altimeter system measures the aircraft height Above Ground level (AGL) electronically and sends the height AGL data to the Modular Avionics Unit (MAU) for display in the ADI window of the left and right PFDs. The digital readout for radio altitude is displayed in white text in a black box in the lower center part of the attitude display on the PFD. The radar altitude display is removed at altitudes greater than 2,500 ft. When altitude is less than 550 feet, the lower portion of the PFD altitude tape will show a yellow cross hatched box to indicate the ground proximity.

If the radar altitude data becomes invalid, the digital readout will be replaced with RAD in white in an amber box. The radar altimeter data is (in addition to indicated airspeed data from the ADAHRS) is used by the aircraft air-ground monitor logic system to determine on ground or in-flight status.

The radar altimeter is an integral component of the Class A TAWS (Terrain Awareness and Warning System) and must be installed in aircraft that are ordered with TAWS.

D. Traffic Alert and Collision Avoidance Systems

Refer to 34-43-00, Traffic Alert and Collision Avoidance System (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System (TCAS II) (Mod S.O.O. 6219) – General data for system details.

A KTA 910 TCAS I or TPU–67A TCAS II can be fitted. The TCAS display is integrated into the Apex system, and TCAS alerts are voiced through the aural warning system.

Both TCAS systems use a processor, two directional antennas (upper and lower) and a configuration module. Aural alerts are provided through the audio panel and from there onward to the crew headsets.

TCAS I provides traffic advisories (TA), but does not provide resolution advisories (RA). TCAS II provides both traffic advisories (TA) and resolution advisories (RA).

For aircraft with S.O.O. 6388 incorporated. the Traffic Alert and Collision Avoidance System (TCAS I) is not installed.

For aircraft with S.O.O. 6388 incorporated, the Traffic Alert and Collision Avoidance System (TCAS I) and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6382.

E. Enhanced Ground Proximity Warning System (TAWS)

Refer to 34-44-00, Enhance Ground Proximity Warning System (TAWS) (Mod 6/2049) – General Data for system details.

The Enhanced Ground Proximity Warning System (EGPWS) (TAWS) provides alerts and warnings to prevent controlled flight into terrain. This is part of the Public Commercial Operation Bundle customer option order 6210.

Enhanced Ground Proximity Warning System (EGPWS) The Honeywell MK VI Enhanced Ground Proximity Warning System (EGPWS) consists of an TAWS computer installed in the nose avionics bay and a configuration module mounted as part of the P2 connector backshell. TAWS provides an enhanced capability of reducing accidents cause by controlled flight into terrain. The system achieves this by receiving a variety of aircraft parameters as inputs, then applying alerting algorithms to provide the flight crew with aural messages and visual annunciations and display.

The Terrain Avoidance Warning System (TAWS) unit receives GPS position via the MAU for accurate position determination in conjunction with a regional database. One of three databases may be installed in the system (Atlantic, Americas or Pacific). The databases also contain the locations of all runways longer than 2000 feet (600 m) that have a published instrument approach. A visual depiction of the terrain proximate to the aircraft can be displayed on the PFD and the MFD.

For aircraft with S.O.O. 6388 incorporated, the Enhanced Ground Proximity Warning System (EGPWS) (TAWS) are not installed.

For aircraft with S.O.O. 6388 incorporated, the Enhanced Ground Proximity Warning System (EGPWS) (TAWS) and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6387.

Weather Radar System (Mod 6/2049) – General Data

1. General

The RDR 2000 weather radar (Radio Detecting and Ranging) system gives the pilot a selectable horizontal display of thunderstorms or high density precipitation in front of the aircraft. When a pulse intercepts a target, the energy is reflected as an echo, or return signal, back to the antenna. From the antenna, the returned signals are transferred to the receiver/transmitter unit. The returned signals, are then displayed.

The radar system can also be used for ground mapping (GMAP). In GMAP mode, prominent landmarks are displayed which enable the pilot to identify coastlines, hilly or mountainous regions, cities, or in some cases even large structures.

For aircraft incorporating customer Standard Order Option S.O.O. 6388 there is no Weather Radar System installed on the aircraft.

For aircraft with S.O.O. 6388 incorporated, the Weather Radar System and associated systems are installed by incorporating customer Standard Order Option S.O.O. 6383.

2. Description

A. WX RADAR Transceiver (RDR 2000)

The RDR 2000 installation consists of a 4kW radar transmitter, configuration module, a receiver, and an integral processing unit, all installed in the nose radome. The processing unit receives pitch and roll signals from the ADAHRS via ARINC 429 and processes this data to stabilize the 12 inch flat panel radar antenna. The sensor unit transmits a beam of pulsed microwave energy. When a pulse reaches visible moisture, the energy is reflected back to the antenna. The return signals are processed and sent to the Modular Avionics Unit (MAU). The weather radar system is connected to the air/ground monitor system, and is automatically switched to STANDBY mode when the aircraft is on the ground. The configuration module stores the configuration and calibration data for the RDR 2000 system.

The antenna pedestal elevation assembly permits the radar beam to be tilted up or down 15 degrees. Tilt may be controlled manually on the multi-function controller or may be determined automatically. In the case of automatic tilt, the manual tilt control provides a trim capability around the automatically set position. Stabilization capability of up to 30 degrees in pitch and roll is possible. Combined antenna stabilization plus tilt angle is functional over a range of ±30 degrees. The azimuth motor drives the antenna up to 50 degrees left and right for a total maximum scan angle of 100 degrees, although in practice a 90 degree image is scanned. The scan rate is 25 degrees per second, providing 16.7 “looks per minute”.

The system displays storm intensity levels in colour against a black background. Rainfall intensity is presented in four colours: green, yellow, red, and magenta, with magenta representing rainfall of the highest intensity. GMAP mode uses green, yellow and magenta colours to display the different reflectivities. Magenta represents the most highly reflective target. Separate sensitivity timing control (STC) compensation is used for weather and GMAP modes.

Weather radar information can be displayed as overlays on the PFDs and the navigation map display of the MFD. The PFD weather radar overlay can be assessed by pressing the soft key on the side of the HSI display. The soft key identifier OVRLY then appears in white. Pressing the OVRLY soft key displays the overlay selection menu. Selecting WX RDR will enable the weather radar overlay to be displayed on the HSI. There is also an OFF section to remove the overlay. The controls for the weather radar are located on the MF Controller. A WX RADAR dual concentric knob control with a push select function is used to control the mode and the tilt, gain settings.

The outer control is a four position rotary knob with the position OFF/STBY/TEST/WX. The inner control is rotary click control knob and is pressed to modify the tilt or gain setting.

Weather radar annunciations for ALERT, MODE and TILT are located on the left side of the HSI. The ALERT annunciations are TX ON GND in amber when WX is selected on the MF Controller and the aircraft is on the ground. TGT ALRT is given in amber when there are potentially hazardous targets directly in front of the aircraft that are outside of the selected range. Longer ranges should then be selected to view the questionable target. TGT is given in white when WX is selected and the aircraft is in the air. The MODE annunciation is that set by the WX RADAR outer control knob. The TILT annunciation value is a three digit number preceded by an arrow, up for positive value and down for negative value. Faults are annunciated WX FAULT in white on the right lower part of the weather radar overlay and failures are annunciated WX FAIL in amber. The Avionics window of the systems MFD contains WX/LX/TAWS setup pages. The WX setup tab is selected via the page menu of the multi-functional window. The RDR 2000 Weather Radar, when installed in combination with the Honeywell Primus Apex® integrated avionics suite, is not capable of displaying vertical depiction of weather. Refer to the Honeywell Primus Apex® Integrated Avionics System for the Viking Twin Otter Series 400 – Pilot Guide for complete information on the description and operation of the weather radar system.

The weather radar antenna and configuration module is mounted on the nose assembly radome bulkhead at FS-6.

28 Volt DC power is provided from the left bus through a 5 amp WX RADAR circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The RDR 2000 weather radar system interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

2) Air Data Attitude and Heading Reference System (ADAHRS) (refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data).

3. Adjustment/Test

WARNING **TESTS INVOLVING RADIATION OF RADIO FREQUENCY ENERGY BY THE RADAR ANTENNA MUST NOT BE MADE IN THE VICINITY OF REFUELLING OPERATIONS, OR WHEN PERSONNEL ARE STANDING DIRECTLY IN FRONT OF THE RADAR ANTENNA.**

RADIO FREQUENCY ENERGY IS GENERATED IN ALL MODES EXCEPT STANDBY AND TEST. DO NOT ALLOW PERSONNEL WITH THE MPEL (MAXIMUM PERMISSIBLE EXPOSURE LEVEL) BOUNDARY BEING SCANNED BY THE ANTENNA WHEN THE SYSTEM IS TRANSMITTING.

A SAFE DISTANCE OF 15 FEET FROM THE ANTENNA RADOME IS RECOMMENDED.

A. WX RADAR Transceiver

1) Testing of WX RADAR Transceiver (RDR 2000)

a) Ensure the aircraft is outside, away from all hangers and personnel.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait several minutes for the aircraft electrical and avionics systems to initialize.

d) Ensure that WX RADAR (P3) circuit breaker on the avionics circuit breaker panel is closed.

e) Warm-up the weather radar by placing the unit in STBY for at least 60 seconds.

f) Set the weather radar control to TEST.

g) Adjust LH/RH PFDs range to 100 Nm. Ensure the test pattern shows the green, yellow, red and magenta colour bands on the LH/RH PFDs and MFD.

h) In the MFD WX tab page on the MFD, press NEXT to show the ANT STABILIZE and WX ROLL TRIM adjust page.

i) Ensure ANT STABILIZE is selected.

j) Gain access to the forward avionics shelves through the forward baggage compartment and locate the ADAHRS (refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data).

k) Loosen retaining screw clamp from the front lip of the ADAHRS mounting tray.

l) Tilt the ADAHRS to simulate a right hand turn. Visually confirm that the weather radar compensates by tilting to the left.

m) Tilt the ADAHRS to simulate a left hand turn. Visually confirm that the weather radar compensates by tilting to the right.

n) Tilt the ADAHRS to simulate nose down. Visually confirm that the weather radar compensates by tilting up.

o) Tilt the ADAHRS to simulate nose up. Visually confirm that the weather radar compensates by tilting down.

p) Return ADAHRS unit to its mounting tray. Tighten screw clamp at the front lip of the ADAHRS mounting tray.

q) Enable the transmit on ground function by selecting the WX tab in the WX/LX/TAWS sensors page.

r) Align the aircraft such that a target that local cloud conditions are viewed on the weather radar display.

s) Place the radar in WX mode.

t) Turn the multi-function control (MFC) weather radar TILT/GAIN control. Ensure that the GAIN on the display changes.

u) Press the multi-function control (MFC) weather radar TILT/GAIN control. Ensure the TILT value changes from +15 to –15 degrees when the control is turned fully clockwise and than fully counterclockwise.

v) Enable the GMAP function by selecting the WX tab in the WX/LX TAWS sensors page. Ensure the GMAP displays a picture of the local conditions.

w) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Configure Module Setup (CM 2000) (Pre Mod 6/2200)

a) Connect the KPA–900 configuration module programming kit in conjunction with a personal computer (PC).

b) Ensure that the 9V radio battery in the KPA–900 is fully charged and that the KPA–900 is connected to your PC printer port.

c) The DOS program will only work with Windows up to 98 version, it will not work with later windows software.

d) Boot up your PC in DOS and work solely in DOS using the 3.5” floppy.

e) When you have entered the configuration page, go to Hardware, and select write. This puts default settings into the configuration module.

f) Go to the Memory and select View. Select each item and ensure a check mark appears to the left of the item as shown in Table 1 — CM 2000 Configuration Module Settings.

g) When you have selected all items as shown in Table 1 — CM 2000 Configuration Module Settings, depress the ESC button on PC.

h) Go to Hardware and select write. When the program has finished writing and verified itself, select Exit.

i) Remove the KPA–900 configuration module programming kit from the CM 2000 configuration module and from the PC.

3) Configure Module Setup (CM 2000) (Post Mod 6/2200)

a) Connect the KPA–900 configuration module programming kit in conjunction with a personal computer (PC) to the printer port.

b) Wait at least 2 minutes for the computer to reconfigure its own printer port. If self configuration is not complete before the KPA–900 maintenance program is started, the program will not detect a low battery status correctly, and the CM 2000 will not program correctly. (When program starts it checks the battery).

c) The Windows program will only work with Windows 95 up to Windows XP, it has not been tested for Windows VISTA or higher.

d) Start the KPA–900 software.

e) Select 'Unit Type' – 'ART 2000'.

f) Ensure the correct Radial Button is selected to the left of each item, as shown in Table 1 — CM 2000 Configuration Module Settings.

g) Select 'Tools – Write to CM 2000'.

h) Verify the activity log.

i) Select 'File – Save Log File'. (Optional).

j) Close program.

k) Remove the CM 2000 configuration module from the KPA–900 configuration module programming kit, and the KPA–900 kit from the PC.

Table 1 CM 2000 Configuration Module Settings

CM 2000 Configuration Module Settings	
Configuration Selection	**Setting**
Gyro Source	429 Bus
Analog Gyro Type	Gyro Source set to 429 bus will cause the Analog Gyro Type option to be ignored
Gyro Compensation	Gyro Source set to 429 bus will cause the Analog Gyro Type option to be ignored
Mechanical Scan Limit	–
Connector Position	Bottom
Attitude Source	–

Table 1 CM 2000 Configuration Module Settings (continued)

CM 2000 Configuration Module Settings	
Configuration Selection	**Setting**
Desired Antenna Sweep	100 degrees
Wx Gain	–
Map Gain	Gain Change Accepted
EFIS Strut Switch	Override Disallowed
ARL	–
Auto Step Scan	–
Sector Scan	–
Auto Tilt	–
Target Alert	Enabled

B. Electrical Bonding Test – Weather Radar Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to the Weather Radar Antenna which is mounted on the forward bulkhead inside the radome nose assembly.

3) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the Weather Radar mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

4) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

5) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

6) Reattach radome nose assembly and tighten retaining screws. Refer to 53-10-02, Forward Baggage Compartment – General Data.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. WX RADAR Transceiver/Antenna (RDR 2000)

1) Removal

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Pull the WX RADAR (P3) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the transceiver which is mounted on the forward bulkhead behind the radome nose assembly.

d) Disconnect the electrical connector on the transceiver unit.

e) Remove the four screws that secure the transceiver unit to the radome bulkhead.

f) Remove unit from aircraft.

Note Stow connectors of the removed components.

2) Installation

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the WX RADAR (P3) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clips.

c) Gain access to the weather radar transceiver, mounted on the forward bulkhead behind the radome nose assembly.

d) Align the weather radar transceiver mounting holes to the radome bulkhead. Ensure correct vertically polarized orientation, connector faces down.

e) Reconnect the electrical connector to the radar transceiver unit. Ensure connections are secure, and sound.

f) Remove the screw that secures the cover from the Data Interrupt Program (DIP) switches, located to the right of the electrical connector.

g) Ensure that the Data Interrupt Program (DIP) switch settings are in the correct position (refer to Table 2 — Data Interrupt Program (DIP) Switch Positions).

Note A setting is in the ON position when the switch is facing the center of the unit.

Table 2 Data Interrupt Program (DIP) Switch Positions

DIP Switch Number	Switch Position
1	ON
2	ON
3	OFF
4	OFF
5	ON
6	OFF
7	ON
8	OFF

h) Install the screw that secures the Data Interrupt Program (DIP) switch cover to the transceiver unit.

i) Remove safety clips and close WX RADAR (P3) circuit breaker on the avionics circuit breaker panel.

j) Test the weather radar system as per Para 3.A.1).

B. Configuration Module (CM 2000)

1) Removal

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Pull the WX RADAR (P3) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the configuration module which is mounted on the forward bulkhead inside the radome nose assembly.

d) Remove the two screws that secure the configuration module to the radome bulkhead.

e) Remove configuration module from aircraft.

Note Stow connectors of the removed components.

2) Installation

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the WX RADAR (P3) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Gain access to the configuration module which is mounted on the forward bulkhead inside the radome nose assembly.

d) Align the configuration module mounting holes to the radome bulkhead. Ensure correct orientation.

e) Reconnect the electrical connector to the configuration module. Ensure connections are secure, and sound.

f) Remove safety clip and close WX RADAR (P3) circuit breaker on the avionics circuit breaker panel.

g) Verify the configuration module setup as per Para 3.A.2).

5. Troubleshooting

A. WX RADAR Transceiver (RDR 2000)

1) If the WX RADAR is deemed inoperative check the following:

a) Ensure WX RADAR (P3) circuit breaker is closed.

b) Visually inspect WX RADAR connector for damaged wires or contacts.

c) Visually inspect WX RADAR antenna for corrosion and contaminants.

d) Ensure that the configuration module (CM 2000) and WX RADAR connector is properly mated and secure.

e) Inspect antenna external surfaces for damage.

f) Check that the antenna is properly mounted and screws are tight.

g) Confirm configuration module settings as per Table 1 — CM 2000 Configuration Module Settings.

h) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

i) If repair or service is required, send to appropriate rated avionics repair facility.

Stormscope – General data

1. General

The Stormscope (WX–500) is a passive sensor that uses a receive antenna to detect the bearing and range of electromagnetic signals (lightning).

2. Description

A. Stormscope (WX–500)

The Stormscope (WX–500) System maps electromagnetic signals (lightning) 360 degrees around the aircraft and to a maximum radius of 200 nautical miles.

When selected in the horizontal situation indicator (HSI) overlay, the partial compass HSI format displays information from the Lightning Sensor System. There are two components of the lightning display on the HSI, mode/fault annunciations/Strike Rate and lightning cell strike data. Mode/fault and Strike Rate annunciations are placed outside of the HSI Display area while lightning strike signals are placed inside the HSI. LX CLR in the HSI overlay clears all strike signals that have been received.

The Stormscope system consists of a receiver installed in the rear avionics rack and a top mounted antenna installed at FS400. A receiver inhibit line is interfaced to the aircraft audio control panel ensuring that erroneous strike signals are not displayed when a radio transmitter is keyed.

The WX–500 contains three levels of self test. Power-on self test, continuous self test and pilot initiated self test. The results are communicated to the host through System Status messages and Faults Code messages. Pilot initiated self test takes approximately 30 seconds during which time data is not updated.

28 Volt DC power is protected by a 3 Amp STORMSCOPE (M4) (Right Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The Stormscope (WX–500) interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

2) Audio Control Panel System (refer to 23-53-00, Audio Control Panel – General Data).

3. Adjustment/Test

A. Function Test Stormscope (WX–500)

1) General

These procedures are for the function test of the Stormscope (WX–500).

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that STORMSCOPE (M4) (Right Bus) circuit breaker on the avionics circuit breaker panel is closed.

d) On the lower MFD, select the WX/TAWS/LX page, then select LX.

e) Press the bezel button next to LX SELF TEST and make sure that the HSI LX MODE shows a white TEST message.

f) Ensure that the HSI LX MODE shows a white STRIKE message after the self test is complete.

g) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of Stormscope (WX–500)

1) General

These procedures are for the removal and installation of the Stormscope (WX–500).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the STORMSCOPE (M4) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the rear avionics shelves through the rear baggage compartment.

d) Disconnect the electrical connector on the Stormscope unit.

e) Loosen retaining screw clamp from the front and slide the unit out of the mounting tray.

f) Remove unit from aircraft.

g) Visually inspect mounting tray. If damaged replace as per Para 4.C.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the STORMSCOPE (M4) circuit breaker is pulled and fitted with safety clip.

c) Slide the unit onto the mounting tray. Tighten the retaining screw clamp at the front lip of the STORMSCOPE unit.

d) Reconnect the electrical connector to the Stormscope unit. Ensure connections are secure.

e) Remove safety clip and close STORMSCOPE (M4) circuit breaker on the avionics circuit breaker panel.

f) Test the Stormscope as per Para 3.A.2).

B. Stormscope Antenna (NY–163)

1) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the STORMSCOPE (M4) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the two screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

Note Stow connectors of the removed components.

2) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the STORMSCOPE (M4) circuit breaker is pulled and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with the two mounting screws.

f) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primer by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Use MIL–PRF–81733 sealant to both fillet seal around the antenna base and seal the securing screws. The aircraft will be ready to fly approximately 24 hours after sealing.

j) Remove safety clip and close STORMSCOPE (M4) circuit breaker on the avionics circuit breaker panel.

k) Check the antenna electrical bonding according to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

l) Test the Stormscope system as per Para 3.A.2).

C. Removal and Installation of Stormscope (WX–500) Mounting Tray

1) General

These procedures are for the removal and installation of the Stormscope mounting tray.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the rear avionics shelves through the rear baggage compartment.

c) Loosen retaining screw clamp from the front lip of the transceiver and slide the unit out of the tray mount.

d) Remove the four screws that secure the mounting tray to the bottom of the avionics rack.

e) Remove mounting tray from aircraft.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Align the tray to the mounting holes on the rear avionics rack.

c) Install the four screws that secure the mounting tray to the avionics rack.

5. Troubleshooting

A. Stormscope (WX–500)

1) If the Stormscope (WX–500) is deemed inoperative check the following:

a) Ensure STORMSCOPE (M4) circuit breaker is closed.

b) Check that the unit is securely installed and the retaining mechanism is securely tightened.

c) Visually inspect Stormscope connector for damaged wires or contacts.

d) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM1-64-2W.

e) If repair or service is required, send to appropriate rated avionics repair facility.

Radar Altimeter System (Mod 6/2049) – General Data

1. General

The radar altimeter system RAD ALT 1 is part of the basic aircraft avionics suite, (Mod 6/2049). Provisions for a second radar altimeter system RAD ALT 2 have been made and is installed as customer option order 6218.

2. Description

The KRA 405B Radar Altimeter (RAD ALT) transceiver is installed in the empennage aft of the rear baggage compartment. A second radar altimeter can be fitted if desired. The transceiver sends a signal to the transmit antenna and gets the return signal from the receive antenna. The maximum operating height Above Ground Level (AGL) used by the system is 2,500 feet. The radar altimeter system measures the aircraft height AGL electronically and sends the height AGL data to the Modular Avionics Unit (MAU) for display in the ADI window of the left and right PFDs. The digital readout for radio altitude is displayed in white text in a black box in the lower center part of the attitude display on the PFD. The radar altitude display is removed at altitudes greater than 2,500 feet. When altitude is less than 550 feet, the lower portion of the PFD altitude tape will show a yellow cross hatched box to indicate the ground proximity. If the radar altitude data becomes invalid, the digital readout will be replaced with RAD in white in an amber box. The radar altimeter data (in addition to indicated airspeed data from the ADAHRS) is used by the aircraft air-ground monitor logic system to determine on ground or in-flight status.

The radar altimeter is an integral component of the Class A TAWS (Terrain Awareness and Warning System) and must be installed in aircraft that are ordered with TAWS.

RAD ALT 1 receive antenna is installed on the bottom of the rear fuselage at FS416 and the transmit is located at FS431. RAD ALT 2 wire integration, circuit breaker, and mounting tray location are provisioned in the aircraft.

28 Volt DC power is protected by a 3 Amp RAD ALT 1 (Left Bus) and a provisioned RAD ALT 2 (Right Bus) circuit breaker, located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The KRA 405B Radar Altimeter System interfaces to the following aircraft system:

A. Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

B. MK VI Enhanced Ground Proximity Warning System (EGPWS) (refer to 34-44-00, Enhance Ground Proximity Warning System (TAWS) (Mod 6/2049) – General Data), if installed.

C. Terrain Collision Avoidance System (TCAS) (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General data), if installed.

3. Adjustment/Test

A. Radar Altimeter Transceiver (KR 405B)

1) General

These procedures are for the adjustment and test, and the height calibration procedures for RAD ALT 1 Transceiver and for RAD ALT 2 Transceiver, if installed.

2) Testing of RAD ALT Transceiver (KR 405B)

a) Ensure that RAD ALT 1 (J5) or RAD ALT 2 (J6) circuit breaker on the avionics circuit breaker panel is closed.

b) Inject a DC variable power supply (0 to 25 Volt DC) signal into the transceiver test input. Ensure correct polarity before turning the power supply on.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Wait several minutes for the aircraft electrical and avionics systems to initialize.

e) On the Flight Control Panel, use the MINIMUMS knob to set Radar Altimeter (RA) Decision Height (DH) to 50 ft.

f) Slowly increase the DC variable power supply voltage until 150 feet is displayed.

g) Connect the Air Data Test set to the aircraft pitot ports and increase airspeed to > 55 Knots.

h) Ensure aircraft changes from “Ground” to “Air” mode.

i) Wait 60 seconds.

j) On the Flight Control Panel, increase the MINIMUMS knob and set Radar Altimeter (RA) Decision Height (DH) to 100 ft.

k) Verify a semi-transparent window is displayed in the upper corner of the LH/RH PFD.

l) On the Flight Control Panel, increase the MINIMUMS knob and set Radar Altimeter (RA) Decision Height (DH) to 150 ft.

m) Verify a MIN annunciation is now displayed in the semi-transparent window in the upper corner of the LH/RH PFD. Verify the word MINIMUMS is annunciated through the aural alerting system.

n) Decrease the Air Data Test set airspeed to 0 Knots. Ensure aircraft changes from “Air” to “Ground” mode.

o) Remove Air Data Test set from the aircraft pitot ports.

p) Turn aircraft power OFF. Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

q) Turn off and remove the DC variable power supply signal from the transceiver test input.

3) Height Calibration of RAD ALT (KR 405B)

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that RAD ALT 1 (J5) or RAD ALT 2 (J6) circuit breaker on the avionics circuit breaker panel is closed.

d) Wait two minutes to allow the RAD ALT to stabilize.

e) Verify that the radar altitude is shown on the LH/RH PFD. Make a note of the altitude displayed.

f) Locate the internal calibration switch hole on the connector side of the KR 405B transceiver.

g) Use a insert/extract tool (P/N M15570–20) to press the internal calibration switch. Hold the switch for 10 sec.

h) Verify on the LH/RH PFDs that the radar altitude displayed is 0 feet.

i) Turn aircraft power OFF. Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Radar Altimeter Antenna (DMPN 3–3A Pre Mod 6/2258, 071–01501–0000 Post Mod 6/2258)

1) General

These procedures are for the verification of the electrical bonding of the RAD ALT 1 antenna and RAD ALT 2 antenna, if installed.

2) Electrical Bonding Testing – Radar Altimeter Antenna

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to empennage through rear baggage compartment.

c) Locate Radar Altimeter Antenna forward of station 421.00 on the floor of the aircraft.

d) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the Rad Alt antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

e) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

f) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal/Installation RAD ALT Transceiver (KR 405B)

1) General

These procedures are for the removal and installation of the RAD ALT 1 Transceiver and RAD ALT 2 Transceiver, if installed.

2) Removal

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Pull the RAD ALT 1 (J5) or RAD ALT 2 (J6) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the rear avionic shelves through the rear baggage compartment.

d) Disconnect the electrical and coaxial cables on the radar altimeter transceiver unit.

e) Loosen retaining screw clamp from the front lip of the transceiver and slide the unit out of the rack mount.

f) Remove unit from aircraft.

Note Stow connectors as required.

3) Installation

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Ensure the RAD ALT 1 (J5) or RAD ALT 2 (J6) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Align the radar altimeter transceiver mounting holes to the bottom of the middle avionics shelf. Ensure correct orientation, connectors facing forward.

d) Slide the unit onto the rack mount. Tighten the retaining screw clamp at the front lip of the transceiver.

e) Reconnect the electrical and coaxial cable to the radar altimeter transceiver unit. Ensure connections are secure.

f) Remove safety clip and close RAD ALT 1 (J5) circuit breaker on the avionics circuit breaker panel.

g) Test the radar altimeter system as per Para 3.A.2).

B. Radar Altimeter Antenna (DMPN 3–3A Pre Mod 6/2258, 071–01501–0000 Post Mod 6/2258)

1) General

These procedures are for the removal and installation of the RAD ALT 1 Antenna and RAD ALT 2 Antenna, if installed.

2) Removal

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Pull the RAD ALT 1 (J5) or RAD ALT 2 (J6) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the eight screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

Note Stow connectors as required.

3) Installation

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Ensure the RAD ALT 1 (J5) or RAD ALT 2 (J6) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with the eight mounting screws.

f) Solvent clean the exposed area of airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primer by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Use MIL–PRF–81733 sealant to both fillet seal around the antenna base and seal the securing screws. The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.B.2).

k) Remove safety clip and close RAD ALT 1 (J5) circuit breaker or RAD ALT 2 (J6) on the avionics circuit breaker panel.

l) Test the radar altimeter system as per Para 3.A.2).

C. Removal/Installation RAD ALT Mounting Tray

1) General

These procedures are for the removal and installation of the RAD ALT 1 Mounting Tray and RAD ALT 2 Mounting Tray, if installed.

2) Removal

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the rear avionic shelves through the rear baggage compartment.

c) Loosen retaining screw clamp from the front lip of the transceiver and slide the unit out of the tray mount.

d) Remove the four screws that secure the mounting rack to the bottom of the rear avionics rack.

e) Remove mounting tray from aircraft.

3) Installation

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the rear avionic shelves through the rear baggage compartment.

c) Align the mounting holes to the avionics shelf. Ensure correct orientation, retaining screw clamp facing forward.

d) Install the four screws and secure the mounting rack to the bottom of the rear avionics rack.

5. Troubleshooting

A. RAD ALT 1 and RAD ALT 2 Transceiver

1) If the RAD ALT is deemed inoperative check the following:

a) Ensure RAD ALT 1 (J5) or RAD ALT 2 (J6) circuit breaker is closed.

b) Visually inspect RAD ALT connector for damaged wires or contacts.

c) Visually inspect RAD ALT antenna and cable connection for corrosion and contaminants.

d) Ensure that the RAD ALT antenna coaxial and cable connector are properly mated and secure.

e) Inspect antenna external surfaces for damage.

f) Check that the antenna is properly mounted and screws are tight.

g) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

h) If repair or service is required, send to appropriate rated avionics repair facility.

Traffic Alert and Collision Avoidance System I (TCAS I) – General Data

1. General

Provision has been made for TCAS I to be installed as part of the commercial IFR bundle S.O.O. 6210.

A KTA 910 TCAS I system audible and visual Traffic Advisories (TA) to alert the flight crew of the proximity of intruding aircraft with the ability of replying to ATCRBS Mode C interrogations.

For aircraft with S.O.O. 6388 incorporated, the Traffic Alert and Collision Avoidance System (TCAS I) is not installed.

For aircraft with S.O.O. 6388 incorporated, the Traffic Alert and Collision Avoidance System (TCAS I) and associated systems are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6382.

2. Description

A. Traffic Alert and Collision Avoidance System I (TCAS I)

TCAS I (KTA 910) is an independent airborne system that does not rely on air traffic control (ATC) for control or coordination for traffic separation. The TCAS is designed to act as a backup to the ATC system and the see-and-avoid principle. The TCAS detects traffic conflicts with other transponder-equipped aircraft and assists the flight crew in avoiding intruders inside a protected airspace. This detection is done by interrogating surrounding aircraft Mode A, Mode C, and Mode S transponders, tracking responses, and issuing advisories to the flight crew of the vertical separation from intruders.

Generated TCAS I processor audio signals are sent to the audio isolation amplifier (refer to 23-54-00, Audio System Isolation Amplifier – General data) and then to the unmuted audio control panel line. The TCAS voice messages are suppressed when the EGPWS sends voice messages to the audio control panel. Traffic Advisories (TA) from the TCAS trigger pulse light operation when the LANDING LIGHT switch is selected to the upper TCAS position.

The KTA 910 TCAS I processor unit and configuration module (KCM 805) are mounted in the nose avionics rack. A directional antenna mounted on the top of the fuselage at FS110 is used to decipher intruder bearing. A omni-directional antenna is mounted on the bottom of the fuselage at FS67.4. During TCAS transmission on the directional or omni-directional antenna, the TCAS generates a suppression pulse to inhibit other L-band installed equipment transmission.

28 Volt DC power is protected by a 5 Amp TCAS (H5) (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The TCAS I interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

2) Enhanced Ground Proximity Warning System (EGPWS) (refer to 34-44-00, Enhance Ground Proximity Warning System (TAWS) (Mod 6/2049) – General Data), if installed.

3) Audio Isolation Amplifier (refer to 23-54-00, Audio System Isolation Amplifier – General data).

4) Lights (refer to 33-00-00, Lights – General Data).

5) A suppression pulse to/from the Single or Dual (if installed) Transponder System (refer to 34-53-00, KXP 2290 Transponder System – General Data).

6) A suppression pulse to/from the Single or Dual (if installed) DME System (refer to 34-54-00, Distance Measuring Equipment (DME) – General Data).

3. Adjustment/Test

A. Function Test TCAS I (KTA 910)

1) General

These procedures are for the function test of the TCAS I (KTA 910).

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that TCAS (H5) circuit breaker is closed.

d) Initiate a TCAS self test in the XPDR/TCAS detail page.

e) Verify the following:

- A white TEST caption is displayed on the lower left corner of the PFDs.
- A solid yellow circle TA (TRAFFIC ADVISORY) illuminates at 9 o'clock, range 2 miles, 200 feet below and climbing.
- A solid cyan diamond (PROXIMITY TRAFFIC) illuminates at 1 o'clock, range 3.6 miles, 1000 feet below and descending.
- A hollow cyan diamond (NON THREAT TRAFFIC) illuminates at 11 o'clock, range 3.6 miles, flying level 1000 feet above.

f) Confirm that “TCAS SYSTEM TEST OK” is audible after the test. Should a failure be detected during self test, the audio message says, “TCAS SYSTEM TEST FAIL”

g) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – Upper TCAS I Antenna (KA 815)

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to the cockpit Temperature Sensor MTG. Over Head Console panel, in the flight compartment.

3) Remove screws and open panel.

4) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the TCAS Upper Antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

5) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

6) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

7) Close Temperature Sensor MTG. Over Head Console Panel and tighten the retaining screws.

C. Electrical Bonding Test – Lower TCAS I Antenna (CL–110–41–30)

1) Ensure the electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to left-hand Access and Inspection Provision panel 15. (Refer to 12-00-00, Servicing – General Data).

3) Turn DZUS fasteners 1/4 turn and swing open access panel.

4) Locate TCAS Lower Antenna forward of station 73.0.

5) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the TCAS Lower Antenna mounting screw head. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

6) Check the electrical conductivity between several random mounting screws within the electrical within the electrical bond contact area to ensure consistency.

7) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

8) Close left-hand Access and Inspection Provision panel 15 and tighten DZUS fasteners.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND LIGHTNING. ENSURE INTEGRITY OF THE HIRF/LIGHTNING PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – PANEL I PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the TCAS I (KTA 910)

1) General

These procedures are for the removal and installation of the TCAS I (KTA 910).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Disconnect the electrical connectors from the front of the unit.

e) Loosen the two retaining screw clamps from the front and slide the unit out of the mounting tray.

f) Visually inspect mounting tray, if damaged replace as per Para 4.B.

g) Remove unit from aircraft.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Slide the unit into the mounting tray.

d) Tighten the two retaining screw clamps at the front lip of the TCAS unit.

e) Reconnect the electrical connector to the unit. Ensure connections are secure.

f) Install top and middle covers attached to bulkhead station 37.21.

WARNING ACCESS COVERS ARE REQUIRED TO MAINTAIN PROTECTION FEATURES THAT ARE ESSENTIAL TO MINIMIZING ADVERSE EFFECTS FROM HIGH INTENSITY RADIATED FIELDS (HIRF).

g) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

h) Test the TCAS system as per Para 3.A.2).

B. Removal and Installation TCAS I Mounting Tray

1) General

These procedures are for the removal and installation of the TCAS I mounting tray.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove the KTA 910 TCAS I unit as per Para 4.A.2).

c) Remove screws securing the mounting tray to the instrument panel.

d) Remove mounting tray from aircraft.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Align the tray to the mounting holes on the instrument panel.

c) Install the screws that secure the mounting tray to the front avionics rack.

d) Install the KTA 910 TCAS I unit as per Para 4.A.3).

e) Test the TCAS I system as per Para 3.A.2).

C. Removal and Installation Configuration Module (KMC 805)

1) Removal

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the configuration module which is mounted in the forward avionics area (underside of right hand middle shelf).

d) Disconnect the electrical connectors from the unit.

e) Remove the (2) screws that secure the configuration module.

f) Remove configuration module from aircraft.

Note Stow connectors of the removed components.

2) Installation

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Gain access to the configuration module which is mounted in the forward avionics area (underside of right hand middle shelf).

d) Reconnect the electrical connector to the configuration module. Ensure connections are secure.

e) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

f) Verify the configuration module setup.

D. Removal and Installation of the Top Directional Antenna (KA 815)

1) General

These procedures are for the removal and installation of the top directional TCAS I antenna (KA815).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the (4) screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connections.

f) Before disconnecting the antenna cables, ensure that the four antenna cables are labelled correctly and correspond to the correct post on the antenna.

g) Disconnect the coaxial connectors.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage.

e) Ensure that the four antenna cables are labelled correctly and correspond to the correct post on the antenna.

f) Ensure MS29513–148 O-Ring is seated securely in O-Ring Seal Groove located around coaxial connections. (Refer to 20-50-02, Sealant and Adhesives – General Data).

g) Connect the coaxial connectors.

h) Secure the antenna assembly on fuselage with four mounting screws, wet with PR1422B 1/2. When installed, clean antenna fastener septum with solvent and pot installed fasteners (in antenna fastener septum) with RTV 105 sealant.

i) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

j) Touch up the exposed area of airframe around the antenna if required with primer.

k) If the affected area requires paint touch up, paint the area with the top-coat paint system matching the existing.

l) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

m) Check the antenna electrical bonding as per Para 3.B.

n) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

o) Do a function test on the TCAS system as per Para 3.A.2).

E. Removal and Installation of the Bottom Omni-Directional Antenna (CL–110–41–30)

1) General

These procedures are for the removal and installation of the bottom omni-directional (CL–110–41–30) TCAS I antenna.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the four screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connectors.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with four mounting screws.

f) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.C.

k) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

l) Do a function test on the TCAS system as per Para 3.A.2).

5. Troubleshooting

A. TCAS I (KTA 910)

1) If the TCAS I (KTA 910) is deemed inoperative check the following:

a) Ensure TCAS (H5) circuit breaker is closed.

b) Check that the units are securely installed and the locking screw is securely tightened.

c) Visually inspect TCAS unit connectors for damaged wires or contacts.

d) Ensure that the DME (refer to 34-54-00, Distance Measuring Equipment (DME) (Mod 6/2049) – General Data) and Transponder (refer to 34-53-00, KXP 2290 Transponder System (Mod 6/2049) – General Data) suppression lines are functioning correctly.

e) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

f) Ensure that the antenna coaxial cable connector are properly mated and secure.

g) Inspect antenna external surfaces for damage.

h) Check that the antenna is properly mounted and screws are tight.

i) If repair or service is required, send to an appropriate rated avionics repair facility.

Note The TCAS I (KTA 910) contains BITE (Built-In Test Equipment) to enable the operational health of the unit to be constantly monitored. When a critical fault is detected, the KTA 910 will notify the Apex maintenance system, which will record the fault. The KTA 910 also stores detected failures in non volatile memory for later review.

Note If appropriately rated for avionics repair, shop fault isolation procedures are contained in the KTA 910 CMM.

Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General data

1. General

Provision for TCAS II is made and can be installed as customer order option 6219.

A CAS 67A TCAS II provides both Traffic Advisories (TA) and Resolution Advisories (RA) to the flight crew for appropriate vertical avoidance maneuvers when a possible collision is threatened with an intruding aircraft.

2. Description

A. Traffic Alert and Collision Avoidance System II (TCAS II)

The CAS 67A TCAS II includes all the functionality of TCAS I, but also includes pitch targets (avoidance and fly-to zones) and aural advisories, known as Resolution Advisories (RAs). There are two types of RAs:

1 Corrective – Pilot action is required. Corrective RAs instruct the pilot to take avoidance action such as, "DESCEND, DESCEND", "CLIMB, CLIMB", or "ADJUST VERTICAL SPEED".

2 Preventative – Current path and speed must be maintained. Preventative RAs warn the pilot not to deviate from the current path and speed by instructing, "MONITOR VERTICAL SPEED" or "MAINTAIN VERTICAL SPEED".

TCAS II systems coordinate the resolution advisories between aircraft before issuing commands to the pilots, so that when one aircraft is instructed to descent, the other is typically told to climb. Traffic symbols including intruder symbol, data tag, and vertical speed arrow are consistent between the Multi-Function Display (MFD) and Horizontal Situation Indicator (HSI) overlay. The number of intruders displayed on the HSI overlay is limited to 32.

Intruder numbers assigned by the TCAS processor are used to associate intruder parameters with the correct intruders. A pattern of intruders, as transmitted by the TCAS unit, is displayed on the traffic overlay, when the TCAS is in test mode and traffic is selected for display. When a TCAS resolution advisory is displayed, the TCAS pitch command is displayed on the Attitude Director Indicator (ADI). The red goal posts indicate the do-not-fly zone (or zones, if both pitch-up and pitch-down avoidance zones are displayed). The green box indicates the fly-to zone. When the aircraft symbol is inside the red do-not-fly zones, it turns red. If the aircraft symbol is inside the green fly-to box, it turns green. Otherwise, it remains white. Traffic has display priority over weather (WX) radar and flight plan symbols (Terrain has priority over TCAS).

Generated TCAS II processor audio signals are sent to the audio isolation amplifier (refer to 23-54-00, Audio System Isolation Amplifier – General data) and then to the unmuted audio control panel line. The TCAS voice messages are suppressed when the EGPWS sends voice messages to the audio control panel. Traffic Advisories (TA) and Resolution Advisories (RA) from the TCAS II processor trigger pulse light operation when the landing light switch is selected to the upper TCAS position.

The CAS 67A TCAS II processor unit and configuration module (CM 2000) are mounted in the nose avionics rack. A directional antenna mounted on the top of the fuselage at station FS110 is used to determine intruder bearing. A omni-directional antenna is mounted on the bottom of the fuselage at station FS67.4. During TCAS transmission on the directional or omni-directional antenna, the TCAS generates a suppression pulse to inhibit other L-band installed equipment transmission.

28 Volt DC power is protected by a 5 Amp TCAS (H5) (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The TCAS II interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

2) Enhanced Ground Proximity Warning System (EGPWS) (refer to 34-44-00, Enhance Ground Proximity Warning System (TAWS) (Mod 6/2049) – General Data).

3) Audio Isolation Amplifier (refer to 23-54-00, Audio System Isolation Amplifier – General data).

4) Lights (refer to 33-00-00, Lights – General Data).

5) A suppression pulse to/from the Dual KXP Transponder system (refer to 34-53-00, KXP 2290 Transponder System (Mod 6/2049) – General Data).

6) A suppression pulse to/from the Distance Measuring Equipment (DME) (refer to 34-54-00, Distance Measuring Equipment (DME) (Mod 6/2049) – General Data).

3. Adjustment/Test

A. Function Test TCAS II (CAS 67A)

1) General

These procedures are for the function test of the TCAS II (CAS 67A).

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that TCAS (H5) circuit breaker is closed.

d) Initiate a TCAS self test in the XPDR/TCAS detail page.

e) Verify the following:

- A white TEST caption is displayed on the lower left corner of the PFDs.
- Resolution Advisory (solid red square) will appear at 3 o'clock, range of 2 nm, at 200 feet relative altitude, above (+02) with no VS arrow.
- Traffic Advisory (solid yellow circle) will appear at 9 o'clock, range of 2 miles, at 200 feet relative altitude, below (–02) with ascending arrow.
- Proximity traffic (solid white diamond, or Honeywell – Cyan) will appear at 1 o'clock, range 3.6 miles, at 1000 feet relative altitude, below (–10) with descending arrow.
- Non-Threat traffic (open white diamond, or Honeywell – Cyan) will appear at 11 o'clock, range of 3.6 miles, at 1000 feet relative altitude, above (+10) with no arrow.

f) Confirm that “TCAS SYSTEM TEST OK” is audible after the test. Should a failure be detected during self test, the audio message says, “TCAS SYSTEM TEST FAIL”

g) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – Upper TCAS II Antenna (ANT 67A)

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to the cockpit Temperature Sensor MTG. Over Head Console panel, in the flight compartment.

3) Remove screws and open panel.

4) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the TCAS Upper Antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

5) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

6) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

7) Close Temperature Sensor MTG. Over Head Console Panel and tighten the retaining screws.

C. Electrical Bonding Test – Lower TCAS II Antenna (CL–110–41–30)

1) Ensure the electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to left-hand Access and Inspection Provision panel 15. (Refer to 12-00-00, Servicing – General Data).

3) Turn DZUS fasteners 1/4 turn and swing open access panel.

4) Locate TCAS Lower Antenna forward of station 73.0.

5) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the TCAS Lower Antenna mounting screw head. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

6) Check the electrical conductivity between several random mounting screws within the electrical within the electrical bond contact area to ensure consistency.

7) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

8) Close left-hand Access and Inspection Provision 15 panel and tighten DZUS fasteners.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND LIGHTNING. ENSURE INTEGRITY OF THE HIRF/LIGHTNING PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – PANEL / PEDESTAL MOUNT UNITS CANNOT BE RELIABLY GROUNDED TO THE AIRFRAME VIA THE PANEL MOUNTING SCREWS OR DZUS FASTENERS. GROUNDING OF THESE UNITS IS ACCOMPLISHED BY USING THE SHIELD GROUNDING BRAID OR JUMPER. VERIFY PROPER ELECTRICAL BONDING AS DETAILED IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE CONNECTOR AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the TCAS II (CAS 67A)

1) General

These procedures are for the removal and installation of the TCAS II (CAS 67A).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Loosen the two retaining screw clamps from the front and slide the unit out of the mounting tray.

e) Visually inspect mounting tray, if damaged replace as per Para 4.B.

f) Remove unit from aircraft.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Slide the unit into the mounting tray.

d) Tighten the two retaining screw clamps at the front lip of the TCAS unit.

e) Reconnect the electrical connector to the unit. Ensure connections are secure.

f) Install top and middle covers attached to bulkhead station 37.21.

WARNING **ACCESS COVERS ARE REQUIRED TO MAINTAIN PROTECTION FEATURES THAT ARE ESSENTIAL TO MINIMIZING ADVERSE EFFECTS FROM HIGH INTENSITY RADIATED FIELDS (HIRF).**

g) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

h) Test the TCAS system as per Para 3.A.2).

B. Removal and Installation TCAS II Mounting Tray

1) General

These procedures are for the removal and installation of the TCAS II mounting tray.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove the CAS 67A TCAS II unit as per Para 4.A.2).

c) Remove screws securing the mounting tray to the instrument panel.

d) Remove mounting tray from aircraft.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Align the tray to the mounting holes on the instrument panel.

c) Install the screws that secure the mounting tray to the front avionics rack.

d) Install the CAS 67A TCAS II unit as per Para 4.A.3).

e) Test the TCAS II system as per Para 3.A.2).

C. Removal and Installation Configuration Module (CM 2000)

1) Removal

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the configuration module which is mounted in the forward avionics compartment (underside of middle shelf).

d) Disconnect the electrical connectors from the unit.

e) Remove the two screws that secure the configuration module.

f) Remove configuration module from aircraft.

Note Stow connectors of the removed components.

2) Installation

a) Ensure the aircraft power is OFF (refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Gain access to the configuration module which is mounted in the forward avionics compartment (underside of middle shelf).

d) Reconnect the electrical connector to the configuration module. Ensure connections are secure.

e) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

f) Verify the configuration module setup.

D. Removal and Installation of the Top Directional Antenna (ANT 67A)

1) General

These procedures are for the removal and installation of the top directional TCAS II antenna (ANT 67A).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the four screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connections.

f) Before disconnecting the antenna cables, ensure that the four antenna cables are labelled correctly and correspond to the correct post on the antenna.

g) Disconnect the coaxial connectors.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage.

e) Ensure that the four antenna cables are labelled correctly and correspond to the correct post on the antenna.

f) Connect the coaxial connectors.

g) Secure the antenna assembly on fuselage with four mounting screws.

h) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

i) Touch up the exposed area of airframe around the antenna if required with primer.

j) If the affected area requires paint touch up, paint the area with the top-coat paint system matching the existing.

k) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

l) Check the antenna electrical bonding as per Para 3.B.

m) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

n) Do a function test on the TCAS system as per Para 3.A.2).

E. Removal and Installation of the Bottom Omni-Directional Antenna (CL–110–41–30)

1) General

These procedures are for the removal and installation of the bottom omni-directional (CL–110–41–30) TCAS II antenna.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the TCAS (H5) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the four screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connectors.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the TCAS (H5) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with four mounting screws.

f) Solvent clean the exposed area of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.C.

k) Remove safety clip and close TCAS (H5) circuit breaker on the avionics circuit breaker panel.

l) Do a function test on the TCAS system as per Para 3.A.2).

5. Troubleshooting

A. TCAS II (CAS 67A)

1) If the TCAS II (CAS 67A) is deemed inoperative check the following:

a) Ensure TCAS (H5) circuit breaker is closed.

b) Check that the units are securely installed and the locking screw is securely tightened.

c) Visually inspect TCAS unit connectors for damaged wires or contacts.

d) Ensure that the DME (refer to 34-54-00, Distance Measuring Equipment (DME) (Mod 6/2049) – General Data) and Transponder (refer to 34-53-00, KXP 2290 Transponder System (Mod 6/2049) – General Data) suppression lines are functioning correctly.

e) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

f) Ensure that the antenna coaxial cable connector are properly mated and secure.

g) Inspect antenna external surfaces for damage.

h) Check that the antenna is properly mounted and screws are tight.

i) If repair or service is required, send to an appropriate rated avionics repair facility.

Note The TCAS II (CAS 67A) contains BITE (Built-In Test Equipment) to enable the operational health of the unit to be constantly monitored. When a critical fault is detected, the CAS 67A will notify the Apex maintenance system, which will record the fault. The CAS 67A also stores detected failures in non volatile memory for later review.

Note If appropriately rated for avionics repair, shop fault isolation procedures are contained in the CAS 67A CMM.

Enhance Ground Proximity Warning System (TAWS) (Mod 6/2049) – General Data

1. General

The Enhanced Ground Proximity Warning System (EGPWS) (TAWS) provides alerts and warnings to prevent controlled flight into terrain. This is part of the Public Commercial Operation Bundle customer option order 6210.

For aircraft with S.O.O. 6388 incorporated, there is no Enhanced Ground Proximity Warning System (EGPWS) (TAWS) installed on the aircraft.

For aircraft with S.O.O. 6388 incorporated, the Enhanced Ground Proximity Warning System (EGPWS) (TAWS) and associated systems are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6387.

2. Description

A. Enhanced Ground Proximity Warning System (EGPWS)

The Honeywell MK VI Enhanced Ground Proximity Warning System (EGPWS) consists of a TAWS computer installed in the nose avionics bay and a configuration module mounted as part of the P2 connector backshell. TAWS provides an enhanced capability of reducing accidents cause by controlled flight into terrain. The system achieves this by receiving a variety of aircraft parameters as inputs, then applying alerting algorithms to provide the flight crew with aural messages and visual annunciations and display.

The Terrain Avoidance Warning System (TAWS) unit receives GPS position via the MAU for accurate position determination in conjunction with a regional database. One of three databases may be installed in the system (Atlantic, Americas or Pacific). The databases also contain the locations of all runways longer than 2000 feet (600 m) that have a published instrument approach. A visual depiction of the terrain proximate to the aircraft can be displayed on the PFD and the MFD.

The terrain alerting algorithms continuously compute the terrain clearance envelopes ahead of the aircraft. If the boundaries of these envelopes conflict with terrain elevation data in the terrain database, then alerts are issued. Two envelopes are computed, one corresponding to a terrain caution alert and the other to a terrain warning alert. Terrain awareness caution and warning alerts are inhibited below 30 feet of radio altitude, within 1 nautical mile of the runway, and below 30 knots groundspeed.

When the conditions exist to generate a terrain or obstacle caution alert, the terrain image on the PFD TAWS Overlay is enhanced to highlight the threatening terrain as solid yellow for caution threats and the appropriate aural alert is given. When the required conditions have been met to generate a terrain or obstacle warning alert, the display image on the PFD TAWS Overlay is enhanced to highlight the terrain as solid red and the appropriate aural alert is given.

The TAWS sends aural alert messages, when necessary, to the audio control panel and from there onward to the headsets. At the same time annunciations are displayed on the PFD ADI in an amber box for GND PROX or red for PULL UP. The annunciations flash in reverse video for 5 seconds and then remain on until the condition is no longer detected. If the terrain overlay is not selected for display and a TAWS alert is generated, the terrain overlay will be displayed (automatic pop-up) on the HSI in the partial compass mode. The TAWS sends a suppression signal to the TCAS to inhibit voice messages from the TCAS when the TAWS is generating voice messages.

28 Volt DC power is protected by a 5 Amp TAWS (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The EGPWS (MK VI) interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) – General Data).

2) Global Positioning System (GPS) (refer to 34-56-00, Global Positioning System (GPS) (Mod 6/2049) – General Data).

3) Air Data, Attitude and Heading Reference System (ADAHRS) (refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data).

4) Traffic Alert and Collision Avoidance System (TCAS) (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General Data), if installed.

5) Radar Altimeter System (refer to 34-42-00, Radar Altimeter System (Mod 6/2049) – General Data).

3. Adjustment/Test

A. Function Test EGPWS (MK VI) – (TAWS)

Note The aircraft must be in a position to receive signals from the GPS satellites prior to doing the following function tests.

1) General

These procedures are for Levels 1 to 6 of Self Test (ST) that describe the current condition and configuration of the EGPWS (MK VI) – (TAWS).

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait for the aircraft electrical and avionics systems to initialize.

c) Ensure that TAWS (G5) (Left Bus) circuit breaker on the avionics circuit breaker panel is closed and the EGPWS Computer has finished its boot sequence.

d) Ensure that the TERRAIN INHIB message is not displayed on the HSI Indicator. If displayed, deactivate Terrain Inhibit via the TERR INHIB switch on the MF Controller.

e) Push the soft key self test button for the TAWS, which is found on MFD 2 display in the Systems Information Section (lower left of screen), following the sequence below:

1 Use the 'Arrow Keys' to navigate the selection over the 'Systems Info' section of the MFD 2.

2 Push the 'Page' key on the MF Controller.

3 Select 'WX/LX/TAWS' from the drop down menu.

4 Push the 'Enter' key on the MF Controller.

5 Use the Joystick on the MF Controller to select the 'TAWS' tab.

6 Push the 'Enter' key on the MF Controller.

7 Use the Joystick on the MF Controller to select the 'TAWS SELF TEST'.

8 Push the 'Enter' key on the MF Controller.

f) Verify the following Level 1 test sequence:

- The red PULL UP annunciator is displayed on the PFDs.
- Aural message EGPWS SYSTEM OK is heard.
- The PFDs show a terrain test pattern sweep.
- The red PULL UP annunciator turns off.
- Amber GND PROX annunciator turns off.
- The PFD terrain test pattern turns off.

- Amber GND PROX annunciator turns off.

g) Within 2 seconds, select the TAWS SELF TEST softkey to continue to Level 2.

h) Verify no faults for Level 2.

i) When ‘press to continue' is heard, select the TAWS SELF TEST softkey within 2 seconds to continue to Level 3.

j) Verify correct System Configuration for Level 3.

k) When ‘press to continue' is heard, select the TAWS SELF TEST softkey within 2 seconds to continue to Level 4.

l) Level 4 lists Fault History record.

m) When ‘press to continue' is heard, select the TAWS SELF TEST softkey within 2 seconds to continue to Level 5.

n) Level 5 lists Warning History record of flights.

o) When ‘press to continue' is heard, select the TAWS SELF TEST softkey within 2 seconds to continue to Level 6.

p) Level 6 initiates a Discrete Input Test. Voice audio is annunciated when the state of a discrete input is changed.

q) Select the TAWS SELF TEST sofkey to disable the Discrete Input Test.

r) Verify ‘End of Self Test' is annunciated.

s) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Loading a Database onto the EGPWS MK VI – (TAWS)

1) General

These procedures are for loading a Terrain Database onto the EGPWS Computer. The EGPWS Computer has a database preloaded from factory. Verification of the Terrain Database should be done first, to confirm the desired database is not already installed.

The MK VI EGPWS, 965–1180–0XX, is shipped from the factory loaded with the Regional Terrain Database, requested when the purchase order was placed. Use (operation) of a MK VI EGPWS outside of the loaded Regional Terrain Database will result in the Terrain Awareness function being unavailable. Aircraft operating outside the loaded Regional Terrain Database region will have to load one of the other two Regional Terrain Databases before beginning the ground test.

The EGPWS Computer must be powered up and must have finished its boot sequence. Access to the EGPWS computer, located on the lower right avionics shelf, is required. The Smart Cable (Honeywell PN 951–0386–001) and a PCMCIA data storage card with the desired database on it is required.

2) Function Test

Note Precautionary notes on the PCMCIA card, regarding insertion and/or removal while power is applied, should be ignored since the EGPWC automatically handles the application and removal of PCMCIA card power.

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait for the aircraft electrical and avionics systems to initialize.

c) Ensure that the TAWS (G5) circuit breaker on the left bus is pushed in and the EGPWS Computer has finished its boot sequence.

d) Connect the Smart Cable (Honeywell PN 951–0386–001) to EGPWC J3 connector.

e) Insert the PCMCIA card into the Smart Cable PCMCIA card slot.

f) While the loading is in progress the IN PROG LED on the Smart Card remains ON and the COMPUTER OK LED on the EGPWC is OFF.

g) When loading is complete the XFER COMP LED on the Smart Card turns ON.

h) Remove the PCMCIA card from the Smart Card slot.

i) After approximately 15 seconds the COMPUTER OK LED comes ON to indicated that the contents of the PCMCIA card were successfully transferred.

j) Remove the Smart Card connector from the EGPWC front panel J3 connector.

k) Verify the Terrain Database Version to make sure the new Database has been loaded correctly.

l) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Verification of Terrain Database EGPWS (MK VI) – (TAWS)

Note The aircraft must be in a position to receive signals from the GPS satellites prior to doing the following function test.

1) General

These procedures are for the verification of the Terrain Database loaded into the EGPWS (MK VI) – (TAWS).

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait for the aircraft electrical and avionics systems to initialize.

c) Ensure that TAWS (G5) (Left Bus) circuit breaker on the avionics circuit breaker panel is closed and that the COMPUTER OK LED on the EGPWC front panel is on.

d) Ensure that the TERRAIN INHIBIT message is not displayed on the HSI Indicator. If displayed, deactivate Terrain Inhibit via the TERR INHIB switch on the MF Controller.

e) Select the TAWS from the pull down menu, which is activated via the OVRLY soft key located on the outboard side on either PFD.

f) Select a Range on the HSI display between 300 and 2.5 miles.

g) Push the soft key self test button for the TAWS, which is found on MFD 2 display in the Systems Information Section (lower left of screen), following the sequence below:

1 Use the 'Arrow Keys' to navigate the selection over the 'Systems Info' section of the MFD 2.

2 Push the 'Page' key on the MF Controller.

3 Select 'WX/LX/TAWS' from the drop down menu.

4 Push the 'Enter' key on the MF Controller.

5 Use the Joystick on the MF Controller to select the ‘TAWS' tab.

6 Push the ‘Enter' key on the MF Controller.

7 Use the Joystick on the MF Controller to select the ‘TAWS SELF TEST'.

8 Push the ‘Enter' key on the MF Controller.

h) Verify the new Database Version, which is displayed on the test patter on the HSI Indicator on either PFD.

i) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

D. Configuration Module Programming

1) General

These procedures are provided to program the configuration module.

2) Program the Configuration Module

a) Gain access to the forward avionic shelves through the forward baggage compartment.

b) Use a D-sub cable to connect a personal computer (PC) with WinViews software installed to the maintenance connector (J3) on the front of the EGPWS unit.

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Wait for the aircraft electrical and avionics systems to initialize.

e) Ensure that TAWS (G5) (Left Bus) circuit breaker on the avionics circuit breaker panel is closed.

f) Start the WinViews software on the PC.

g) Select COM 1 and then CONNECT. Ensure that the Terminal Mode Page is shown.

h) At the C: prompt type CFG and then press ENTER.

i) Type the following text:

CUW 0/14 3 5 4 1 2 9 93 2 3 5 2 0 1 2/

j) Press ENTER.

k) The cursor will flash with Y or N displayed.

l) Type Y. The EGPWS unit will reboot. Ensure this is complete before continuing.

m) Type CTRL Z.

n) Type PS and press ENTER.

o) Ensure that the correct configuration has been programmed. Ensure the display reads:

CUW 0/14 3 5 4 1 2 9 93 2 3 5 2 0 1 2/

p) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

q) Remove the D-sub cable connected to the maintenance connector (J3) on the front of the EGPWS unit and turn off the PC.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECT OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES - GENERAL DATA).

SHIELDS GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELLS/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of EGPWS (MK VI) – (TAWS)

1) General

These procedures are for the removal and installation of the EGPWS (MK VI) – (TAWS).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull circuit breaker TAWS (G5) (Left Bus) on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the forward avionic shelves through the forward baggage compartment.

d) Disconnect the electrical connector on the radar EGPWS unit.

e) Loosen retaining screw clamp from the front and slide the unit out of the mounting tray.

f) Visually inspect mounting tray for condition, if damaged replace as per Para 4.B.

g) Remove unit from aircraft.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the TAWS (G5) (Left Bus) circuit breaker is pulled and fitted with safety clip.

c) Slide the unit onto the mounting tray. Tighten the retaining screw clamp at the front lip of the EGPWS unit.

d) Reconnect the electrical connector to the EGPWS unit. Ensure connections are secure.

e) Install top and middle covers attached to bulkhead station 37.21.

WARNING **ACCESS COVERS ARE REQUIRED TO MAINTAIN PROTECTION FEATURES THAT ARE ESSENTIAL TO MINIMIZING ADVERSE EFFECTS FROM HIGH INTENSITY RADIATED FIELDS (HIRF).**

f) Remove safety clip and close TAWS (G5) (Left Bus) circuit breaker on the avionics circuit breaker panel.

g) Test the EGPWS as per Para 3.A.2).

B. Removal and Installation of EGPWS (MK VI) – (TAWS) Mounting Tray

1) General

These procedures are for the removal and installation of the EGPWS (MK VI) – (TAWS) mounting tray.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the forward avionic shelves through the forward baggage compartment.

c) Loosen retaining screw clamp from the front lip of the transceiver and slide the unit out of the tray mount.

d) Remove the four screws that secure the mounting tray to the bottom of the avionics rack.

e) Remove mounting tray from aircraft.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Align the tray to the mounting holes on the instrument panel.

c) Install the four screws that secure the mounting tray to the avionics rack.

5. Troubleshooting

A. EGPWS (MK VI) – (TAWS)

1) If the EGPWS (MK VI) – (TAWS) is deemed inoperative check the following:

a) Ensure TAWS (G5) circuit breaker is closed.

b) Check that the unit is securely installed and the retaining mechanism is securely tightened.

c) Visually inspect EGPWS connector for damaged wires or contacts.

d) Ensure that the Configuration Module has been programmed with the correct aircraft specific values as per Para 3.D.2).

e) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

f) If repair or service is required, send to appropriate rated avionics repair facility.

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Dependent Position Determining – General Data

1. General

The dependent position determining aids can be comprised of Automatic Direction Finding System (ADF), Distance Measuring Equipment (DME), VHF Omni-Range (VOR) and transponder systems, dependent on customer requirements.

A. VHF Navigation

Refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General Data and 34-52-00, VHF Navigation System – General data for system details.

The VHF navigation systems 1 and 2 receive VHF Omni-Range (VOR), Localizer (LOC) and Glide Slope (GS) signals. The systems process:

- The VOR signals to give course navigation data to a VOR ground station
- The LOC and GS signals to show deviation along the flightpath of the instrument landing systems.

The VHF navigation systems 1 and 2 have these components:

- VHF/NAV 1/GS 1 receivers (integrated in the No. 1 MMDR system)
- VHF/NAV 2/GS 2 receivers (integrated in the No. 2 MMDR system)
- A NAV antenna with antenna coupler (splitter)
- A Glideslope antenna

B. Distance Measuring Equipment (DME)

Refer to 34-54-00, Distance Measuring Equipment (DME) (Mod 6/2049) – General Data for system details.

The Distance Measuring Equipment (DME) gives the distance from the aircraft to a DME ground station. The dual DME consists of two transceivers (DME 1 and DME 2) and one antenna per DME transceiver. DME 1 antenna and DME 2 antenna. Provision has been made for the dual DMEs to be installed on the aircraft, this is part of the IFR Operations Bundle customer Order Option, S.O.O. 6209.

For aircraft incorporating customer Standard Order Option S.O.O. 6388 there is one DME transceiver (DME 1) and one antenna installed on the aircraft.

For aircraft with S.O.O. 6388 incorporated, the second DME transceiver (DME 2) and associated antenna are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6386.

C. Automatic Direction Finding System (ADF)

Refer to 34-55-00, Automatic Direction Finding System (ADF) – General Data for system details.

The Automatic Direction Finder (ADF) is used to display the aircrafts relative bearing to a tuned ground station.

The ADF tunes frequencies from 190 kHz to 1,799 kHz and 2,180 kHz to 2,189 kHz. Functions consist of ADF, ANT, and BFO tuning modes. ADF 1 and optional ADF 2 antennas include both a loop and sense component and are installed at FS297.7 on the top of the fuselage.

The ADF system is an integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR) (refer to 34-51-00, Multi-Mode Digital Radio – General Data).

For aircraft with S.O.O. 6388 incorporated, the Automatic Direction Finding (ADF) system is not installed.

For aircraft with S.O.O. 6388 incorporated, the single Automatic Direction Finding (ADF) and associated antenna are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6405 and dual

Automatic Direction Finding (ADF) and associated antennas are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6217.

D. KXP 2290 Transponder System

Refer to 34-53-00, KXP 2290 Transponder System – General Data for system details.

The Mode S Transponder (KXP 2290) (Pre Mod 6/2303) replies to Air Traffic Control Radar Beacon System (ATCRBS) Mode A and C, Intermode, and Mode S interrogations. Provisions have been made for the KXP 2290 Transponder System, this is part of the IFR operational bundle customer order option, S.O.O. 6209.

Aircraft incorporating Mod 6/2354 upgrade the transponders from KXP 2290 to KXP 2290A, with optional Automatic Dependent Surveillance – Broadcast (ADS-B) Out capability. The KXP 2290A transponders are direct replacements to the KXP 2290 LRU's, installed behind each of the pilot's PFD Controller.

The ADS-B out functionality is enabled through the incorporation of customer Standard Order Option, S.O.O. 6347. SBAS GPS Receivers (Mod 6/2304) is a pre-requisite for S.O.O. 6347.

Each transponder system includes the KXP 2290 or KXP 2290A LRU located behind the associated PFD Controller and bottom and top (diversity) antennas.

For aircraft incorporating customer Standard Order Option S.O.O. 6388 there is one KXP 2290A Transponder installed on the aircraft.

For aircraft with S.O.O. 6388 incorporated, the second KXP 2290A Transponder and associated antennas are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6381.

E. Global Positioning System (GPS)

Refer to 34-56-00, Global Positioning System (GPS) – General Data for system details.

Two independent KGS 200 Global Navigation System Sensor Unit (GNSSU) receivers are installed in the nose avionics bay. The receiver use 24 channels, grouped in 3 hardware groups, each consisting of 8 channels. One group of 8 channels is used for navigation at any given time. The GPS sensor unit calculates and outputs navigation data, satellite measurement data, Receiver Autonomous Integrity Monitoring (RAIM) and Predictive RAIM (PRAIM). The processed output data is sent to the CSIO module in the MAU, via ARINC 429, for use by the rest of the avionic system. The sensor data unit also manages Sign Status Matrix (SSM), satellite status and does Built-In Test Equipment (BITE) functions. An Apex maintenance function interfaces with the GPS sensor. The GPS data page can be accessed from the Sensors page. The Sensors page can be accessed with the systems MFD lower left window in focus and selecting the Sensors page menu. The TAWS unit receives GPS position via the MAU for accurate position determination.

Aircraft incorporating Mod 6/2304 have upgraded the KGS 200 Global Navigation System Sensor Unit (GNSSU) receivers for Satellite Based Augmentation System (SBAS) capable GPS receivers.

Provisions have been made for SBAS GPS approach capability to allow SBAS-based GPS approaches, such as LPV approaches, through the incorporation of customer Standard Order Option, S.O.O. 6225.

For aircraft incorporating customer Standard Order Option S.O.O. 6388 there is one upgraded KGS 200 Global Navigation System Sensor Unit (GNSSU) receiver installed on the aircraft.

For aircraft with S.O.O. 6388 incorporated, the second upgraded KGS 200 Global Navigation System Sensor Unit (GNSSU) receiver and associated antenna is installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6380.

F. XM Satellite Weather System

Refer to 34-57-00, XM Satellite Weather System – General data for system details.

The XM Satellite Weather System is a streaming weather data receiver.

The XM Weather receiver is mounted on the rear avionics rack. The XM Satellite Weather system sends received weather data to the Modular Avionics Unit (MAU) Advanced Graphics Unit (AGU), which is then displayed on the Multi-Function Display (MFD).

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Multi-Mode Digital Radio (MMDR) – General data

1. General

The speech communication system is part of the Very High Frequency (VHF) Communication systems 1 and 2 (refer to 23-12-00, VHF Communication System – General Data).

The VHF Communication systems 1 and 2 give voice communication between aircraft, or between the aircraft and a ground station. The two systems (VHF 1 and VHF 2) are the same.

The VHF Communication system 1 and 2 have the following components:

- Two Multi-Mode Digital Radio (MMDR) transceivers
- Two VHF antennas

The MMDR transceiver is a VHF NAV/COM transceiver with a self contained Automatic Direction Finder (ADF) receiver.

The description and operation of the navigation part of the MMDR transceiver is given in this section.

The description and operation of the ADF part of the MMDR transceiver is given in 34-55-00, Automatic Direction Finding System (ADF) – General Data.

The KTR Multi-Mode Digital Radio (MMDR) is a blind rack-mounted integrated transceiver consisting of one transmitter and six receivers.

2. Description

A. Multi-Mode Digital Radio (KTR 2280)

Two KTR 2280 Multi-Mode Digital Radio (MMDR) integrated transceivers are installed in the nose avionics bay. The MMDR comprises of one 16W COM transmitter and six receivers. The receivers consist of a VHF Omni Range (VOR), Glide Slope (GS), Localizer (LOC), and three simultaneously operating COM receivers. Primary controls for the MMDR are on the MF controller and the PF controllers, with display of the selected information on the PFD screen.

28 Volt DC power is protected by a 7.5 Amp MMDR 1 (F3) (Left Bus) and a 7.5 Amp MMDR 2 (F4) (Right Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The Multi-Mode Digital Radio is integrated to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

2) VHF Navigation (LOC/GS/VOR) (refer to 34-52-00, VHF Navigation System – General Data).

3) VHF Communication (refer to 23-12-00, VHF Communication System – General Data).

4) Automatic Direction Finder (ADF) (refer to 34-55-00, Automatic Direction Finding System (ADF) – General Data), if installed.

3. Adjustment/Test

A. Multi-Mode Digital Radio (KTR 2280)

To function test the VHF Communication, VHF Navigation, and the ADF components of the MMDR refer to the following chapters:

1) VHF Navigation (LOC/GS/VOR) (refer to 34-52-00, VHF Navigation System – General Data, Para 3.A.2)).

2) VHF Communication (refer to 23-12-00, VHF Communication System – General Data, Para 3.A.2)).

3) Automatic Direction Finder (ADF) (refer to 34-55-00, Automatic Direction Finding System (ADF) – General Data, Para 3.A.2)).

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECT OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELDS GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELLS/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal/Installation Multi-Mode Digital Radio (KTR 2280)

1) General

These procedures are for the removal and installation of the Multi-Mode Digital Radio (KTR 2280). MMDR 1 and MMDR 2 (if installed).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Using an Allen wrench, turn the securing rod counterclockwise until the unit disengages from the mounting tray.

d) Remove the MMDR unit from the mounting rack.

e) Remove unit from aircraft.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Using an Allen wrench, turn the securing rod clockwise until it has drawn the unit into the rack and is tight.

d) Remove safety clips and close MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel.

e) Test the MMDR as per Para 3.A.

5. Troubleshooting

A. Multi-Mode Digital Radio (KTR 2280)

1) If the Multi-Mode Digital Radio (KTR 2280) is deemed inoperative check the following:

a) Ensure the MMDR 1 (F3) and MMDR (F4) circuit breakers are closed.

b) Visually inspect the MMDR units for damaged wires or contacts.

c) Ensure that the MMDR connectors are properly mated and secure.

d) Ensure antenna connections are properly mated and secure.

e) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

f) If repair or service is required, send to appropriate rated avionics repair facility.

Note The MMDR (KTR 2280) contains BITE (Built-In Test Equipment) to enable the operational health of the unit to be constantly monitored. When a critical fault is detected, the KTR 2280 will notify the Apex maintenance system, which will record the fault. The KTR 2280 also stores detected failures in non volatile memory for later review.

Note If appropriately rated for avionics repair, shop fault isolation procedures are contained in the KTR 2280 CMM.

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VHF Navigation System – General Data

1. General

The VHF Navigation (VHF NAV) system receives VHF Omni Range (VOR), Localizer (LOC) and Glide Slope (GS) signals.

2. Description

A. VHF Navigation

The VHF NAV system consists of a 200 channel VHF Navigation and a 40 channel Glide Slope receiver. The VHF NAV signals give the aircraft course navigation information received from a VOR ground station. Localizer and Glide Slope flight path deviation signals received from the instrument landing system (ILS) is then displayed on the PFD.

The VOR/LOC antenna is installed on the vertical stabilizer. A dual NAV coupler (CI–502) is installed to allow simultaneous use of two VOR/LOC receivers from a single antenna. The dual output GS antenna is installed on the nose bulkhead at FS6.

The VHF NAV system is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR) (refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General Data).

3. Adjustment/Test

A. Function Test VHF Navigation (VOR, LOC, GS) System

1) General

These procedures are for the function test of the VHF NAV system NAV 1 and NAV 2.

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus), circuit breakers on the avionics circuit breaker panel are closed.

d) Set up VHF Navigation ramp test unit, in accordance with the instructions in the test set manual.

e) Enable a MOD TONE on the ramp test unit. Verify a tone is heard when NAV 1 and NAV 2 are selected on the audio control panel.

f) Tune NAV 1 and NAV 2 frequency to a corresponding LOC ramp test unit frequency. Verify correct LOC/GS 1 and LOC/GS 2 indication on the HSI.

g) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – VOR/LOC Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Rotate rudder to gain sufficient access to Access and Inspection Provision 6a panel. (Refer to 12-00-00, Servicing – General Data).

3) Remove screws and then remove access panel.

4) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the VOR/LOC antenna mounting screw. Place the second probe on exposed surface of access panel doubler. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

5) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

6) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

7) Place Access and Inspection Provision 6a panel onto access cavity and tighten the retaining screws.

8) Return rudder to neutral position.

C. Electrical Bonding Test – Glide Slope Antenna (DM N25–2)

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to the Glide Slope antenna which is mounted on the forward bulkhead inside the radome nose assembly. (Refer to 53-10-02, Forward Baggage Compartment – General Data).

3) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the head of the Glide Slope mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

4) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

5) If the electrical conductivity is no consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

6) Reattach radome nose assembly and tighten retaining screws. (Refer to 53-10-02, Forward Baggage Compartment – General Data).

4. Removal/Installation

The VHF Navigation system is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR), refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data, Para 4. for the MMDR Removal/Installation procedure.

A. Removal/Installation of the VOR/LOC Antenna (DM N4–4)

1) General

These procedures are for the removal and installation of the VOR/LOC antenna.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the screws (24) from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with the mounting screws (24).

f) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.B.

k) Remove safety clips and close MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel.

l) Do a function test on the NAV system as per Para 3.A.2).

B. Removal/Installation of GS Antenna (DM N25–2)

1) General

These procedures are for the removal and installation of the GS antenna (DM N25–2).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the screws (6) from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Solvent clean the electrical contact bond areas immediately before the installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with the mounting screws (6).

f) Solvent clean the exposed area of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.C.

k) Remove safety clips and close MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel.

l) Do a function test on the NAV system as per Para 3.A.2).

5. Troubleshooting

The VHF Navigation system is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR), refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data, Para 5. for the MMDR Troubleshooting procedure.

KXP 2290 Transponder System – General Data

1. General

The Mode S Transponder (KXP 2290) (Mod 6/2049) (Pre Mod 6/2354) replies to Air Traffic Control Radar Beacon System (ATCRBS) Mode A and C, Intermode, and Mode S interrogations. Provisions have been made for the KXP 2290 Transponder System, this is part of the IFR operational bundle customer order option 6209.

Mod 6/2354 upgrades the dual KXP 2290 Transponders to dual KXP 2290A Transponders with ADS-B Out capability. S.O.O. 6347 enables the ADS-B Out functionality.

Each transponder system includes the KXP 2290 or KXP 2290A LRU located behind the associated PFD Controller and bottom and top (diversity) antennas.

For aircraft with S.O.O. 6388 incorporated, there is one KXP 2290A Transponder installed.

For aircraft with S.O.O. 6388 incorporated, a second KXP 2290A Transponder and associated systems are installed on the aircraft by incorporating S.O.O. 6381.

2. Description

A. KXP 2290 Transponder System

1) Transponder System (KXP 2290) (Mod 6/2049) (Pre Mod 6/2354)

A KXP 2290 Mode S transponder is installed behind each pilot's PFD controller. The KXP 2290 has the ability to transmit an 8 digit alphanumeric Flight ID code. The flight crew may enter Flight ID information into the system via the bezel buttons, PFD controller, or MF controller. If the Flight ID field is left blank, the aircraft registration will automatically be transmitted. The aircraft ICAO address is also always transmitted regardless if Flight ID is manually entered by the pilot or not.

The XPDR detail window is displayed in the radio tuning window by pressing the adjacent bezel button or XPDR button on MF Controller and then pressing the DETAIL button on either the PFD or MF Controller. The detail window of the transponder contains soft keys for the selection of XPDR MODE, TCAS functionality and default VFR CODE. The XPDR SEL soft key is used for active transponder selection. The non-selected transponder will be in a standby mode.

The transponders receive input from the aircraft air ground monitor and use this input to automatically switch from ground to flight mode and vice-versa. Standby Mode is normally only used for maintenance purposes. In the event of failure of the air-ground monitor, automatic switching of the transponder will not be provided. When in GND mode, the transponder will inhibit ATCRBS, ATCRBS/Mode S All Call and Mode S-only All Call replies. However, the unit will continue to generate Mode S transmissions when replying to discretely addressed Mode S interrogations.

The transponders support Mode S enhanced surveillance (EHS) requirements by broadcasting additional downlink aircraft parameters (DAP) using a 1090 MHz extended squitter. This functionality is automatic and transparent to the pilot. No control is given or needed. DAP information transmitted by the transponder includes flight ID, magnetic heading, indicated airspeed, true airspeed, groundspeed, vertical speed, bank angle, selected altitude, and altimeter setting.

The transponders use ARINC 735A TCAS interface protocol to interface with the (optional) TCAS II system. As a peripheral to the TCAS II processor, the transponder receives and replies to short and long air to air surveillance and TCAS coordination interrogations, and receives and replies to ground to air surveillance and Comm A interrogations directed to TCAS. When the transponder system transmits, a suppression pulse is sent to the DME and the TCAS to prevent operation of these systems at the same time, which may cause interference.

TCAS traffic can be viewed by the pilot while the aircraft is on the ground when TA or TA/RA mode has been selected (TA/RA mode may be selected, but only TA will be displayed on the ground and during

fight below 1100 feet AGL). The aircraft does not reply to interrogations from other aircraft when in ground mode, but does display information received from aircraft broadcasting Mode C or Mode S transmissions.

XPDR 1 and XPDR 2 (if installed) bottom antenna(s) are installed on the bottom of the fuselage at stations FS103.8 and the top antenna(s) are installed on the top of the fuselage at stations FS186.5. The transponder antennas are a blade shape and have a low contour with a connector on the base.

28 Volt power is protected by a 5 Amp XPDR 1 (Left Bus) and XPDR 2 (Right Bus) (if installed) circuit breaker located on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

The KXP 2290 transponder system interfaces to the following aircraft systems:

a) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

b) Global Positioning System (GPS) (refer to 34-56-00, Global Positioning System (GPS) – General Data).

c) A suppression pulse to/from the KN 63 DME system (refer to 34-54-00, Distance Measuring Equipment (DME) (Mod 6/2049) – General Data).

d) A suppression pulse to/from the TCAS (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General Data).

2) Mode S Transponder System (KXP 2290A) with ADS-B Out (Post Mod 6/2354)

Aircraft incorporating Mod 6/2354, have the KXP 2290A Mode S transponder with Automatic Dependent Surveillance – Broadcast (ADS-B) Out functionality. The KXP 2290A transponder is a direct replacement for the KXP 2290 LRU.

The ADS-B out functionality is enabled through the incorporation of customer standard option S.O.O. 6347. Mode 6/2304 SBAS GPS Receivers is a pre-requisite for S.O.O. 6347.

For aircraft with S.O.O. 6388 incorporated, there is one KXP 2290A Transponder installed.

For aircraft with S.O.O. 6388 incorporated, a second KXP 2290A Transponder and associated systems are installed on the aircraft by incorporating S.O.O. 6381.

When the aircraft is on the ground and **GND** is displayed, as long as the transponder is not in STBY mode, the transponder replies to interrogations that are specifically addressed to the aircraft by name. Otherwise, it does not emit transmissions. ADS-B ground stations and ground-based multilateration systems, which are used to monitor aircraft movement on taxiways, are examples of new ATC technologies that have the ability to address interrogations to specific aircraft while they are on the ground.

ADS-B Out status is displayed in the XPDR radio window. The ADS-B Out status displayed in white indicates normal ADS-B Out operation. The ADS-B Out status dispalyed in amber indicates ADS-B Out is failed. The ADS-B Out status not displayed at all indicates ADS-B Out is disabled by Apex configuration option, or the transponder indicates ADS-B Out is selected off, or the transponder is failed.

3. Adjustment/Test

A. Transponder System (KXP 2290A) ADS-B Out Conformity Verification (Post Mod 6/2354 with S.O.O. 6347 for ADS-B Out Enabled)

1) General

The following procedures are for the ADS-B Out conformity verification of the transponder, GPS and ADAHRS systems.

2) Transponder System (KXP 2290A) ADS-B Out Conformity Verification

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the forward avionic shelves, through the forward baggage compartment.

c) Remove top, middle and bottom covers attached to bulkhead station 44.0 for full access.

d) Verify the part number of GPS 1, GPS 2 (if installed) and ADAHRS as follows:

KGS 200 SBAS GPS – P/N 066–01201–0105
KSG 7200 ADAHRS – P/N 065–00188–5103 or 065–00188–5104

e) Reinstall top, middle and bottom covers attached to bulkhead station 44.0 at the aft wall of the baggage compartment. Ensure covers are mounted correctly and secure.

f) Gain access to XPDR 1 and XPDR 2 (if installed) dataplates by removing pilot and co-pilot Primary Flight Display controllers, respectively. (Refer to 31-13-00, Apex Controllers – General Data, Para 4., Removal and Installation).

g) Verify part number of the XPDR 1 and XPDR 2 (if installed) as follows:

KXP 2290A – P/N 066–01198–0102

h) Reinstall Primary Flight Display controllers that were removed to gain access to the transponders unit. (Refer to 31-13-00, Apex Controllers – General Data, Para 4., Removal and Installation).

i) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

j) Wait for the aircraft electrical systems to initialize and the system flags to clear.

k) Verify Primus Apex® system is utilizing Software Load 4.4 indicated by Top Level System P/N EB60002734–0104, displayed in the System Config ID Window on the Systems MFD.

l) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Functional Test Transponder System

1) General

Note It is recommended to coordinate with local ATC before broadcasting with transponder at simulated altitudes to prevent being a source of interference to ATC or TCAS or ADS-B IN equipped aircraft in the area.

Note Transponder ramp test units must have capability to test all parameters listed in the procedure below. For aircraft incorporating S.O.O. 6347, a transponder ramp test unit with the capability to test ADS-B Out parameters is required.

The following procedures are for the functional test of the Transponder 1 and Transponder 2 (if installed) systems.

2) Function Test (KXP 2290 (Mod 6/2049, Pre Mod 6/2354) or KXP 2290A (Post Mod 6/2354) without S.O.O. 6347 for ADS-B Out Not Enabled)

a) Verify pitot heat is OFF and connect the Pitot-Static System. (Refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test.)

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait several minutes for the aircraft electrical and avionics systems to initialize.

d) Ensure that XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus), circuit breaker on the avionics circuit breaker panel are closed.

e) Ensure L/R selection on Flight Controller set to “L” .

f) Ensure ADAHRS 1 is displayed on LH PFD and ADAHRS 2 is displayed on RH PFD.

g) Set standard barometric pressure (29.92 inHg or 1013 HPa) on PFDs.

h) Set Mode A code to the appropriate VFR code. For example, 1200 (Canada/U.S.) or 7000 (Europe).

i) Ensure transponder set to ALT or TA to enable Modes A, C and S.

j) Set up transponder ramp test unit to interrogate the bottom transponder antenna, in accordance with the instructions in the test set manual.

k) Install antenna shield boot on the transponder antenna not under test. For example: If testing XPDR 1 bottom antenna, ensure antenna shield boot is on the XPDR 1 top antenna. If testing XPDR 1 top antenna, ensure antenna shield boot is on the XPDR 1 bottom antenna.

l) Increase airspeed on the Pitot-Static Test set to 60 Knots.

m) Verify in the XPDR radio window and on the transponder ramp test unit, that the transponder switches from ground to air mode.

n) Using a transponder ramp test unit, do a transponder Mode S Auto Test, or equivalent.

o) Ensure the result of the transponder auto test is PASS. Verify the following:

Table 1 Top and Bottom Antenna Parameters

Parameter	Required Value
Effective Radiated Power (ERP)	51.0 – 57.0 dBm or 125 – 500 W or 21.0 – 27.0 dBw
Minimum Threshold Level (MTL)	–74.0 dBm (±3 dBm)
Radio Reply Frequency	1090 Mhz (±1 Mhz)
All-call Address (AA) (24–bit ICAO Address)	Matches Aircraft Address
Tail Number	Matches Aircraft Tail Number
Flight ID	As entered in FMS Init Tab
A CODE	Matches Mode A code set in Para 3.B.2).h)

Table 1 Top and Bottom Antenna Parameters (continued)

Parameter	Required Value
C ALT	Matches LH PFD, rounded to the nearest 100 ft
S ALT	Matches LH PFD, rounded to the nearest 25 ft
Vertical Status (VS)	IN AIR
Squitter	Verify transponder is transmitting squitters
Diversity Isolation	>25 dB

Table 2 Enhanced Surveillance Parameters

Parameter	Required Value
Roll Angle	Matches LH PFD (±0.5°)
Ground Speed	0 kts (±2 kts)
Indicated Airspeed (IAS)	Matches LH PFD (±2 kts)
True Airspeed (TAS)	Matches LH PFD (±2 kts)
Magnetic Heading	Matches LH PFD (±1.0°)
Baro Alt rate	Matches LH PFD (±50 fpm)

p) Ensure the following Common Usage GICB Capability Reports are shown under the Elementary Surveillance BDS 1,7 Test:

Table 3 BDS 1,7 Common Usage GICB Capability Reports

2,0 (Flight ID)
4,0 (Aircraft Vertical Intention)
5,0 (Track and Turn)
6,0 (Heading and Speed Report)

q) Relocate antenna shield boot from top antenna to bottom antenna.

r) Repeat Para 3.B.2).n) and Para 3.B.2).o), Table 1 — Top and Bottom Antenna Parameters for the top antenna. Table 2 — Enhanced Surveillance Parameters is not required to be repeated for the top antenna.

s) Vent Pitot Static Test set to ambient conditions.

t) Verify in the XPDR radio window and on the transponder ramp test unit that the transponder switches from air to ground mode.

u) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

v) Remove antenna shield boot and all test equipment form aircraft.

w) Disconnect the Pitot-Static System (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test).

3) Function Test (KXP 2290A (Post Mod 6/2354) with S.O.O. 6347 for ADS-B Out Enabled)

Note For procedures involving verification of Transponder ADS-B GPS input, ensure aircraft is located such that the GPS receivers can acquire satellite signals.

a) Verify pitot heat is OFF and connect the Pitot-Static System (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test.)

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait several minutes for the aircraft electrical and avionics systems to initialize.

d) Ensure that XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus), circuit breaker on the avionics circuit breaker panel are closed.

e) Ensure L/R selection on Flight Controller set to “L” .

f) Ensure ADAHRS 1 is displayed on LH PFD and ADAHRS 2 is displayed on RH PFD.

g) Set standard barometric pressure (29.92 inHg or 1013 HPa) on PFDs.

h) Set Mode A code to the appropriate VFR code. For example, 1200 (Canada/U.S.) or 7000 (Europe).

i) Ensure transponder set to ALT or TA to enable Modes A, C and S.

j) Ensure ADS-B Status indication in the XPDR radio window is white.

k) Set up transponder ramp test unit to interrogate the bottom transponder antenna, in accordance with the instructions in the test set manual.

l) Install antenna shield boot on the transponder antenna not under test. For example: If testing XPDR 1 bottom antenna, ensure antenna shield boot is on the XPDR 1 top antenna. If testing XPDR 1 top antenna, ensure antenna shield boot is on the XPDR 1 bottom antenna.

m) Increase airspeed on the Pitot-Static Test set to 60 Knots.

n) Verify in the XPDR radio window and on the transponder ramp test unit, that the transponder switches from ground to air mode.

o) Using a transponder ramp test unit, do a transponder Mode S auto test, or equivalent.

p) Ensure the result of the transponder auto test is PASS. Verify the following:

Table 4 Top and Bottom Antenna Parameters

Parameter	Required Value
Effective Radiated Power (ERP)	51.0 – 57.0 dBm or 125 – 500 W or 21.0 – 27.0 dBw
Minimum Threshold Level (MTL)	–74.0 dBm (±3 dBm)
Radio Reply Frequency	1090 Mhz (±1 Mhz)
All-call Address (AA) (24–bit ICAO Address)	Matches ICAO address assigned to aircraft
Tail Number	Matches Aircraft Tail Number
Flight ID	As entered in FMS Init Tab

Table 4 Top and Bottom Antenna Parameters (continued)

Parameter	Required Value
A CODE	Matches Mode A code set in Para 3.B.3).h)
C ALT	Matches LH PFD, rounded to the nearest 100 ft
S ALT	Matches LH PFD, rounded to the nearest 25 ft
Vertical Status (VS)	IN AIR
Squitter	Verify transponder is transmitting squitters
Diversity Isolation	>25 dB

Table 5 Enhanced Surveillance Parameters

Parameter	Required Value
Roll Angle	Matches LH PFD (±0.5°)
Ground Speed	0 kts (±2 kts)
Indicated Airspeed (IAS)	Matches LH PFD (±2 kts)
True Airspeed (TAS)	Matches LH PFD (±2 kts)
Magnetic Heading	Matches LH PFD (±1.0°)
Baro Alt rate	Matches LH PFD (±50 fpm)

q) Ensure the following Common Usage GICB Capability Reports are shown under the Elementary Surveillance BDS 1,7 Test:

Table 6 BDS 1,7 Common Usage GICB Capability Reports

0,5 (Airborne Position)
0,6 (Surface Position)
0,7 (Status)
0,8 (Type and Identification)
0,9 (Airborne Velocity Information)
2,0 (Flight ID)
4,0 (Aircraft Vertical Intention)
5,0 (Track and Turn)
6,0 (Heading and Speed Report)

r) Using a transponder ramp test unit, do ADS-B AC 20-165 auto tests, or equivalent.

s) Ensure the result of the ADS-B auto test is Pass. Verify the following:

Table 7 ADS-B Parameters

Parameter	Required Value
Aircraft Position (Lat/Long)	Matches FMS Lat/Long
Aircraft Position Error	Within NACP limits
NACP	≥ 8 EPU < 92.6m (0.05 nm)
NIC	≥ 7 Rc < 370.4m (0.2 nm)
NACV	≥ 1 (<10 m/s) Note Due to the way the KGS 200 GPS units process and filter the NACV value, it is not uncommon for the NACV parameter to display '0' during ground test. This is not considered a fail of the test procedure provided the other GPS derived parameters meet or exceed the required values shown.
E-W Velocity	0
N-S Velocity	0
ADS-B Baro Altitude	Matches LH PFD, rounded to the nearest 25ft
XPDR Baro Altitude	Matches LH PFD, rounded to the nearest 25ft
Geometric Altitude	Matches airfield height ± 100 feet
SIL	≥ 3 (≤$1x10^{-7}$ per hour)
SDA	≥ 2 (≤$1x10^{-5}$ per hour)
TCAS Operational (This parameter is applicable for TCAS II equipped aircraft only)	0 (TCAS II in TA Only Mode) 1 (TCAS II in TA/RA Mode)
RA Active (This parameter is applicable for TCAS II equipped aircraft only)	Matches TCAS II RA status (0 = No RA alert; 1 = Active RA alert)
Mode 3/A Code	As entered in XPDR radio window
Ident	Matches status of Ident (Yes = active)
ICAO Address	Matches ICAO address assigned to aircraft
Emerg/Prior Code	0
Emitter Category	Light
ADS-B In Capability	0 = Not capable
Flight ID	As entered in FMS Init Tab
Aircraft Length/Width Note Available during AC 20–165 Surface Test only.	2 = < 25 m, < 28.5 m

t) Relocate antenna shield boot from top antenna to bottom antenna.

u) Repeat Para 3.B.3).o) and Para 3.B.3).p), Table 4 — Top and Bottom Antenna Parameters for the top antenna. Table 5 — Enhanced Surveillance Parameters is not required to be repeated for the top antenna.

v) Open GPS 1 (J3) (Left Bus) circuit breaker, wait a few seconds for the GPS 1 Fail CAS message to be displayed. Ensure ADS-B status indication remains white, leave GPS 1 circuit breaker open for the next step (dual GPS). Ensure ADS-B status indication turns amber, skip the next step (single GPS).

w) Open GPS 2 (J4) (Right Bus) circuit breaker, wait a few seconds for the GPS 2 Fail CAS message to be displayed, ensure ADS-B status indication turns amber.

x) Close GPS 1 (J3) (Left Bus) circuit breaker, allow enough time for GPS 1 to initialize, ensure ADS-B status indicator returns to white.

y) Close GPS 2 (J4) (Right Bus) circuit breaker (dual GPS).

z) Turn ADS-B Out OFF in the XPDR detail window, ensure ADS-B status indicator clears.

aa) Vent Pitot Static Test set to ambient conditions.

ab) Verify in the XPDR radio window and on the transponder ramp test unit that the transponder switches from air to ground mode.

ac) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

ad) Disconnect the Pitot-Static System (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test.).

ae) Remove antenna shield boot and all remaining test equipment from aircraft.

C. Transponder Air Data Input Verification (KXP 2290 (Mod 6/2049) (Pre Mod 6/2354) or KXP 2290A (Post Mod 6/2354))

1) General

The following procedures verify the air data inputs to the Transponder 1 and Transponder 2 (if installed) systems.

2) Transponder Air Data Input Verification

Note It is recommended to coordinate with local ATC before broadcasting with transponder at simulated altitudes to prevent being a source of interference to ATC or TCAS or ADS-B IN equipped aircraft in the area.

a) Ensure that the (ESIS) STATIC SOURCE valve is in the NORMAL position.

b) Verify pitot heat is OFF and connect a calibrated Pitot/Static Test Set to the Static ports on the aircraft (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test.).

c) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

d) Ensure that XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus), circuit breaker on the avionics circuit breaker panel are closed.

e) Wait for the aircraft electrical system to initialize and the attitude flags to clear.

f) Ensure L/R selection on Flight Controller set to “L” (Post Mod 6/2303).

g) Ensure ADAHRS 1 is displayed on LH PFD and ADAHRS 2 is displayed on RH PFD.

h) Set standard barometric pressure (29.92 inHg or 1013 HPa) on PFDs.

i) Set Mode A code to the appropriate VFR code. For example, 1200 (Canada/U.S.) or 7000 (Europe).

j) Select XPDR 1 for use (dual transponder).

k) Ensure transponder set to ALT or TA to enable Modes A, C and S.

l) Leak test the Static System (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test).

m) Increase airspeed on the Pitot-Static Test set to 60 Knots.

n) Complete the Air Data Verification. Refer to Table 8 — Air Data Verification Table.

Table 8 Air Data Verification Table

LH PFD Altitude (feet)	Mode C Altitude	Mode S Altitude
–1,000	Matches LH PFD Altitude, rounded to the nearest 100 ft.	Matches LH PFD Altitude, rounded to the nearest 25 ft.
0		
525		
1,050		
1,575		
2,100		
3,000		
4,000		
5,000		
6,000		
8,000		
10,000		
12,000		
14,000		
16,000		
18,000		
20,000		
22,000		
25,000		

o) Select XPDR 2 for use (dual transponder).

p) Select ADAHRS 2 on LH PFD and repeat above procedure for Transponder 2 (dual transponder).

q) Vent Pitot Static Test set to ambient conditions.

r) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Part 2.).

s) Disconnect the Pitot/Static Test Set from the static ports on the aircraft (refer to 34-11-00, Pitot-Static System (Mod 6/2033) – General Data, Para 4., Adjustment/Test).

t) Ensure any tape has been removed and pitot and static ports are clear.

D. Electrical Bonding Test – Upper Transponder 1 and 2 (if installed) Antennas

1) General

The following procedures are for the electrical bonding for the Upper Transponder 1 and 2 (if installed) Antennas.

2) Electrical Bonding Test

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the ceiling compartment aft of station 180.75. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data.

c) Verify that the electrical conductivity is consistent over the antenna electrical bond contact area by placing a probe on the Upper Transponder antenna mounting surface area between the fuselage thru hole and the antenna connector. Place the second probe on the nearest airframe ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise. If necessary, remove antenna connector for additional access to the antenna bonding surface.

Note If disconnecting antenna connectors, do a functional test on the Transponder System, refer to Para 3.B.

d) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

e) Replace the ceiling compartment interior aft of station 180.75. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

E. Electrical Bonding Test – Lower Transponder 1 and 2 (if installed) Antennas

1) General

The following procedures are for the electrical bonding for the Lower Transponder 1 and 2 (if installed) Antennas.

2) Electrical Bonding Test

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to left-hand Access and Inspection Provisions 15 panel. (Refer to 12-00-00, Servicing – General Data).

c) Release fasteners and swing open access panel.

d) Locate Lower Transponder antenna aft of station 101.0.

e) Verify that the electrical conductivity is consistent over the antenna electrical bond contact area by placing a probe on the Lower Transponder antenna mounting surface area between the fuselage thru hole and the antenna connector. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise. If necessary, remove antenna connector for additional access to the antenna bonding surface.

Note If disconnecting antenna connectors, do a functional test on the Transponder System, refer to Para 3.B.

f) If the electrical conductivity does not meet requirements, check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

g) Close left-hand Access and Inspection Provision 15 panel and tighten the fasteners.

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECT OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELDS GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELLS/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of Transponder System (KXP 2290 (Mod 6/2049) (Pre Mod 6/2354) or KXP 2290A (Post Mod 6/2354))

1) General

These procedures are for the removal and installation of the XPDR 1 and/or the XPDR 2 (if installed) system.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Gain access to XPDR 1, by removing pilot's Primary Flight Display controller. To gain access to XPDR 2 (if installed), remove co-pilot's Primary Flight Display controller. (Refer to 31-13-00, Apex Controllers – General Data, Para 4.).

d) Using an Allen wrench, turn the securing rod counterclockwise until the unit disengages from the mounting tray.

e) Remove the transponder unit from the mounting rack.

f) Visually inspect mounting tray for condition, if damaged replace as per Para 4.B.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus) circuit breakers are pulled and fitted with safety clips.

c) Gain access to XPDR 1, by removing pilot's Primary Flight Display controller. To gain access to XPDR 2 (if installed), remove co-pilot's Primary Flight Display controller. (Refer to 31-13-00, Apex Controllers – General Data, Para 4.).

d) Slide the transponder unit into the tray until the securing rod engages the nut at the back of the rack.

e) Using an Allen wrench, turn the securing rod clockwise until it has drawn the unit into the rack and is tight.

f) Reinstall Primary Flight Display controller that was removed to gain access to the transponder unit. (Refer to 31-00-00, Instruments – General Data).

g) Remove safety clips and reset XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus) circuit breakers on the avionics circuit breaker panel.

h) Do a function test of the Transponder System, refer to Para 3.B.

B. Removal and Installation of Transponder System Mounting Tray

1) General

These procedures are for the removal and installation of the XPDR 1 mounting tray and the XPDR 2 mounting tray, if installed. The KXP 2290 Transponder shares a mounting tray with the KMC 2210 controller and is part of the APEX cockpit panel configuration.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Remove the desired PFD Controller and transponder as per Para 4.A.2).

c) Remove screws securing the mounting tray to the instrument panel.

d) Remove electrical connector and coaxial cable from mounting tray.

e) Remove mounting tray from aircraft.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Install electrical connector and coaxial cable to mounting tray.

c) Align the tray to the mounting holes on the instrument panel.

d) Install the screws that secure the mounting tray to the instrument panel.

e) Install the PFD Controller and transponder, refer to Para 4.A.3).

C. Removal and Installation of Transponder Antennas (CI 110–41)

1) General

These procedures are for the removal and installation of the XPDR 1 and XPDR 2 (if installed) antennas.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the antenna and mounting screws.

d) Remove the four screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

g) Remove antenna from fuselage.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus) circuit breakers are pulled and fitted with safety clips.

c) Solvent clean the electrical contact bond areas immediately before the installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with four mounting screws.

f) Solvent clean the exposed area of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data).

j) Remove safety clips and reset XPDR 1 (G3) (Left Bus) and/or XPDR 2 (G4) (Right Bus) circuit breakers.

k) Check the antenna electrical bonding as per Para 3.D and/or Para 3.E.

l) Do a function test of the Transponder System, refer to Para 3.B.

5. Troubleshooting

A. Transponder System (KXP 2290 (Mod 6/2049) (Pre Mod 6/2354) or KXP 2290A (Post Mod 6/2354))

1) If the XPDR 1 and/or XPDR 2 is deemed inoperative check the following:

a) Ensure XPDR 1 (G3) and/or XPDR 2 (G4) circuit breaker is closed.

b) Check that the unit is securely installed and the retaining mechanism is securely tightened.

c) Visually inspect XPDR 1 and/or XPDR 2 connectors for damaged wires or contacts.

d) Visually inspect XPDR 1 and/or XPDR 2 antennas for corrosion and contaminants.

e) Inspect antenna external surfaces for damage.

f) Check that the antennas are properly mounted and screws are tight.

g) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

h) If repair or service is required, send to appropriate rated avionics repair facility.

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Distance Measuring Equipment (DME) (Mod 6/2049) – General Data

1. General

The Distance Measuring Equipment (DME) gives the distance from the aircraft to a DME ground station. The DME consists of two transceivers (DME 1 and DME 2) and one antenna per DME transceiver. DME 1 antenna and DME 2 antenna. Provision has been made for the DMEs, DME 1 and DME 2 to be installed on the aircraft, this is part of the IFR Operations Bundle customer order option 6209.

For aircraft incorporating customer Standard Order Option S.O.O. 6388 there is one DME transceiver (DME 1) and one antenna installed on the aircraft.

For aircraft with S.O.O. 6388 incorporated, the second DME transceiver (DME 2) and associated antenna are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6386.

2. Description

KN 63 Distance Measuring Equipment (DME) transceivers may be installed in the rear avionics rack. The transceiver has 200 different channels. The transmitter processes signals between 1,025 MHz and 1,150 MHz and the receiver processes signals between 962 MHz and 1,213 MHz. Range to station are output to the MAU as digital serial data. The maximum displayed range of the DME transceiver is 300 nautical miles. The DME detail window can be displayed in the radio tuning window by pressing the DME button on the PFD Controller. An alternative means to accessing the DME window is by using the DME Detail soft key in the NAV detail window. The DME detail window contains soft keys ‘DME PAIR' to select the association of the DME to NAV 1 or NAV 2, and ‘DME HOLD' to select DME hold ON or OFF. When the DME HOLD is selected to ON, an H adjacent to the DME distance is displayed on the PFD HSI display.

DME 1 antenna is installed on the bottom of the fuselage, left of the centre line, between station 291.00 and 301.82. DME 2 is located on the lower empennage FS431 between the two Radar Altimeter antennas. The DME antenna is a blade shape and has a low contour with an antenna connector on the base.

When the system transmits, a suppression pulse is sent to the ATC transponder system and the TCAS to prevent operation of these systems at the same time. The DME receives a signal with an identification code from the ground station. The signal is sent to the Audio Control Panel, which generates a tone to identify the ground station.

28 Volt DC power is protected by a 3 Amp DME 1 (Left Bus) and DME 2 (Right Bus) circuit breaker is located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The KN 63 Distance Measuring Equipment System interfaces to the following aircraft systems:

A. Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

B. Dual KMA 29 Audio Control Panel (refer to 23-53-00, Audio Control Panel – General Data).

C. A suppression pulse to/from the Single or Dual (if installed) Transponder System (refer to 34-53-00, KXP 2290 Transponder System – General Data).

D. A suppression pulse to/from the TCAS (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data and 34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General data), if installed.

3. Adjustment/Tests

A. Function Test of Distance Measuring Equipment

1) General

These procedures are for the function test of DME 1 and DME 2 transceivers (KN 63).

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that DME 1 (H3) (Left Bus) and/or DME 2 (H4) (Right Bus), circuit breaker on the avionics circuit breaker panel are closed.

d) In the VHF NAV detail window select the DME detail page. The DME detail page can also be accessed by pressing the DME button on the PFD/Radio controller.

e) Channel the NAV 1 and NAV 2 frequency to a local ground station. Station ident will be heard on the aircraft audio panel.

f) Ensure a valid DME 1 and/or DME 2 indication is on the LH PFD and RH PFD.

g) With a valid DME 1 and/or DME 2 indication; select DME hold to ON and verify an amber H is displayed on both the LH PFD and RH PFD.

h) Channel the associated NAV 1 and NAV 2 to a invalid DME frequency. Verify DME 1 and/or DME 2 distance continues to be displayed.

i) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – DME 1 Antenna

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00-00, Electrical Power – General Data).

2) Gain access to Access and Inspection Provision 33 panel. (Refer to 12-00-00, Servicing – General Data).

3) Remove screws and remove access panel.

4) Carefully cut the sealant from around the base of the antenna and mounting screws.

5) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the DME antenna mounting screw head. Place the second probe on an anchor nut from the Access and Inspection Provision 33 panel. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

6) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

7) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

8) Place Access and Inspection Provision 33 panel onto access cavity and tighten the retaining screws.

9) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data).

C. Electrical Bonding Test – DME 2 Antenna (if installed)

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to empennage through rear baggage compartment.

3) Carefully cut the sealant from around the base of the antenna and mounting screws.

4) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the DME antenna mounting screw head. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

5) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

6) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

7) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data).

4. Removal/Installation

A. Removal/Installation of DME Transceiver (KN 63)

1) General

These procedures are for the removal and installation of the DME 1 transceiver and DME 2 transceiver.

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00-00, Electrical Power – General Data).

b) Pull circuit breaker DME 1 (H3) (Left Bus), DME 2 (H4) (Right Bus) respectively on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the aft avionic shelf through the rear baggage compartment.

d) Disconnect the electrical and coaxial cables at the front of the DME transceiver unit.

e) Loosen the hold down clamp at the front of unit.

f) Remove unit from mounting tray assembly.

Note Stow connectors of the removed components.

g) Visually inspect mounting tray for condition, if damaged replace as per Para 4.B.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00-00, Electrical Power – General Data).

b) Ensure the DME 1 (H3) (Left Bus), DME 2 (H4) (Right Bus) circuit breaker on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Gain access to the aft avionic shelf through the rear baggage compartment.

d) Place unit on the mounting tray assembly.

e) Slide unit onto mounting tray, ensure unit is correctly oriented on the mounting tray; electrical connections facing forward until the unit is snug and secure under the mounting tray lip.

f) Tighten the hold down clamps on the DME mounting tray. Ensure unit is securely held on mounting tray.

g) Reconnect the electrical and coaxial cables. Ensure connections are sound and secure.

h) Remove safety clips and replace circuit breaker DME 1 (H3) (Left Bus), DME 2 (H4) (Right Bus) respectively on the avionics circuit breaker panel.

i) Do a function test on the DME as per Para 3.A.2).

B. Removal/Installation of DME Transceiver Mounting Tray

1) General

These procedures are for the removal and installation of the DME 1 transceiver mounting tray and the DME 2 transceiver mounting tray.

2) Removal

a) Remove DME transceiver as Para 4.A.2).

b) Remove the four lock nuts and washers from the bottom of the shelf that secure mounting tray to shelf.

c) Remove mounting tray from aircraft.

3) Installation

a) Locate mounting tray on avionic shelf, align mounting holes, ensure the hold down clamp is forward.

b) Install the four lock washers and nuts from the bottom of shelf and ensure tray is securely held on shelf.

C. Removal/Installation DME 1 Antenna and DME 2 Antenna (if installed)

1) General

These procedures are for the removal and installation of DME 1 and DME 2 antenna.

2) Removal

a) Ensure the aircraft power is OFF. (Refer to 24-00-00-00, Electrical Power – General Data).

b) Pull the DME 1 (H3) (Left Bus) and DME 2 (H4) (Right Bus) circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the antenna and mounting screws.

d) Support the antenna and remove the four screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

g) Remove antenna from Aircraft.

Note Stow connectors of the removed components.

3) Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

a) Ensure the aircraft power is OFF. (Refer to 24-00-00-00, Electrical Power – General Data).

b) Ensure that the DME 1 (H3) (Left Bus) and DME 2 (H4) (Left Bus) circuit breakers on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Solvent clean the electrical contact bond areas immediately before the installation. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with four mounting screws.

f) Solvent clean the exposed area of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.B for DME 1 and Para 3.C for DME 2.

k) Remove the safety clips and replace DME 1 (H3) (Left Bus) and DME 2 (H4) (Right Bus) circuit breakers on the avionics circuit breaker panel.

l) Do a function test on the DME system as per Para 3.A.2).

5. Troubleshooting

A. DME 1 and DME 2

1) If the DME is deemed inoperative check the following:

a) Ensure DME 1 (H3) and/or DME 2 (H4) circuit breaker are closed.

b) Visually inspect DME connector for damaged wires or contacts.

c) Visually inspect DME antenna for corrosion and contaminants.

d) Ensure that the TCAS (refer to 34-43-00, Traffic Alert and Collision Avoidance System I (TCAS I) – General Data) and Transponder (34-43-01, Traffic Alert and Collision Avoidance System II (TCAS II) (Mod S.O.O. 6219) – General data) suppression lines are functioning correctly, if installed.

e) Check DME antenna and cable connection for corrosion and contaminants. DME range can be affected with having a dirty (oil covered) antenna.

f) Ensure that the antenna coaxial cable connector are properly mated and secure.

g) Inspect antenna external surfaces for damage.

h) Check that the antenna is properly mounted and screws are tight.

i) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

j) If repair or service is required, send to appropriate rated avionics repair facility.

Automatic Direction Finding System (ADF) – General Data

1. General

The Automatic Direction Finder (ADF) is used to display the aircrafts relative bearing to a tuned ground station.

For aircraft with S.O.O. 6388 incorporated, the Automatic Direction Finding (ADF) system is not installed.

For aircraft with S.O.O. 6388 incorporated, a single ADF and associated antenna are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6405 or a dual ADF and associated antennas are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6217.

2. Description

A. Automatic Direction Finder (ADF)

The ADF tunes frequencies from 19 kHz to 1,799 kHz and 2,180 kHz to 2,189 kHz. Functions consist of ADF, ANT, and BFO tuning modes. ADF 1 and optional ADF 2 antennas include both a loop and sense component and are installed at FS297.7 on the top of the fuselage.

The ADF system is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR) (refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data).

3. Adjustment/Test

A. ADF Function Test

1) General

These procedures are for the function test of ADF 1 and ADF 2.

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus), circuit breakers on the avionics circuit breaker panel are closed.

d) Within the ADF detail window, tune the ADF to local AM broadcast or a NDB station.

e) Select ANT mode on the ADF. Verify station audio is heard when ADF is selected on the audio control panel.

f) Select BFO mode on the ADF. Verify a 1 Khz tone is heard when the ADF is selected on the audio control panel.

g) Using the aircraft directional gyro or compass, turn the aircraft to the left 45 degrees. Verify ADF bearing error is not more than what is described in the manufacturers manual.

h) Continue to turn the aircraft stopping at 90 deg., 135 deg., and 270 deg. Verify ADF bearing error is not more than what is described in the manufacturers manual.

i) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – ADF Antenna

1) General

These procedures are for the electrical bonding testing for ADF 1 antenna and ADF 2 antenna.

2) Electrical Bonding Test

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the ceiling compartment forward of station 305.0.

c) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the ADF antenna backing plate. Place the second probe on the ADF antenna mounting screw anchor nut. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

d) Check the electrical conductivity between several random mounting screw anchor nuts within the electrical bond contact area to ensure consistency.

e) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

4. Removal/Installation

The ADF is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR), refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data, Para 4. for the MMDR Removal/Installation procedure.

A. Removal/Installation of the ADF 1 and ADF 2 Antenna (KA 44B)

1) General

These procedures are for the removal and installation ADF 1 and ADF 2 antenna (KA 44B).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the antenna.

d) Remove the screws (2) from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the electrical connector.

f) Disconnect the electrical connector.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the MMDR 1 (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel are pulled and fitted with safety clips.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data). Note that the gasket is non-conductive and the skin does not need to be stripped. Bonding is through the grounding ring and mounting screws.

d) Bring the antenna assembly close to the fuselage, connect the electrical connector.

e) Secure the antenna assembly on fuselage with the mounting screws (2).

f) Solvent clean the exposed area of the airframe around the antenna. (Refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.B.

k) Remove safety clips and close MMDR (F3) (Left Bus) and/or MMDR 2 (F4) (Right Bus) circuit breaker on the avionics circuit breaker panel.

l) Do a function test on the NAV system as per Para 3.A.2).

5. Troubleshooting

The ADF is a integral component of the KTR 2280 Multi-Mode Digital Radio (MMDR), refer to 34-51-00, Multi-Mode Digital Radio (MMDR) – General data, Para 5. for the MMDR Troubleshooting procedure.

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Global Positioning System (GPS) – General Data

1. General

The Global Positioning System (GPS) consists of one or two independent KGS 200 Global Navigation System Sensor Unit (GNSSU) receivers, and one or two GPS antennas. GPS 1 and GPS 2 (if installed), collect and compile signals from satellites and provide the aircraft avionics systems with position, velocity, and time. GPS 1 is part of the basic aircraft avionics suite. Provisions for GPS 2 have been made, and can be installed as part of the IFR Operations Bundle customer option order 6209.

Mod 6/2304 upgrades the KGS 200 GPS receivers to KGS 200 SBAS GPS receivers.

For aircraft with S.O.O. 6388 incorporated, there is one Post Mod 6/2304 KGS 200 Global Navigation System Sensor Unit (GNSSU) receiver installed.

For aircraft with S.O.O. 6388 incorporated, a second Post Mod 6/2304 KGS 200 Global Navigation System Sensor Unit (GNSSU) and associated systems are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6380.

2. Description

A. KGS 200 GNSSU System

1) Global Positioning System (KGS 200) (Mod 6/2049) (Pre Mod 6/2304)

One or two independent KGS 200 Global Navigation System Sensor Unit (GNSSU) receivers are installed in the nose avionics bay. The receiver uses 24 channels grouped in 3 hardware groups each consisting of 8 channels. One group of 8 channels is used for navigation at any given time. The GPS sensor unit calculates and outputs navigation data, satellite measurement data, Receiver Autonomous Integrity Monitoring (RAIM) and Predictive RAIM (PRAIM). The processed output data is sent to the CSIO module in the MAU via ARINC 429 for use by the rest of the avionics systems. The sensor data unit also manages Sign Status Matrix (SSM), satellite status and does Built-In Test Equipment (BITE) functions. An Apex maintenance function interfaces with the GPS sensor. The GPS data page can be accessed from the Sensors page. The Sensors page can be accessed with the systems MFD lower left window in focus and selecting the Sensors page menu. The No. 1 KGS 200 GNSSU provides GPS position and GPS altitude to the EGPWS.

28 Volt DC power is protected by a 3 Amp GPS 1 (J3) (Left Bus) and GPS 2 (J4) (Right Bus) (if installed) circuit breaker located on the avionics circuit breaker panel (refer to 24-60-00, DC Electrical Load Distribution – General Data).

There are one or two GPS antennas mounted on the top of the fuselage. GPS 1 antenna is mounted just off the centerline to the starboard side at FS140 and GPS 2 antenna, if installed, is mounted off the centerline to the port side at FS315. The GPS antenna is a low profile, tear drop type and has a single coaxial connector.

The KGS 200 GNSSU interfaces to the following aircraft systems:

a) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

1 SmartView Synthetic Vision System (SVS) (S.O.O. 6222) (refer to 31-10-00, Instrument and Control Panels – General Data).

2 Flight Management System (FMS) (Mod 6/2303) (refer to 34-60-00, Flight Management System (FMS) – General Data).

b) Single or Dual (if installed) Transponder System (refer to 34-53-00, KXP 2290 Transponder System – General Data).

c) Enhanced Ground Proximity Warning System (TAWS) (Mod 6/2049) (refer to 34-44-00, Enhanced Ground Proximity Warning System (TAWS) – General Data), if installed.

d) Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) (refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data).

2) Global Positioning System (KGS 200) with SBAS Capability (Post Mod 6/2304)

Various GNSS systems around the world use a correction signal generated by ground sensors, transmitted via geosynchronous satellites to SBAS capable GPS receivers. This facilitates the conduct of more accurate GPS-based navigation and approaches.

Accurate aircraft positioning is aided by the Wide Area Augmentation System (WAAS) that was developed to augment GPS. WAAS utilizes ground-based reference stations and geostationary WAAS satellites in a system known as the Satellite Based Augmentation System (SBAS). WAAS is capable of achieving comparable Category 1 Instrument Landing System (ILS) precision approach accuracy.

Aircraft incorporating Mod 6/2304 have the KGS 200 GPS receivers with Satellite-based Augmentation System (SBAS) capability. The KGS 200 SBAS GPS receivers are direct replacements for the non-SBAS KGS 200 GPS receiver LRU's installed on the forward avionics shelf. No changes are required to the KA 96 GPS antennas or coaxial cables. GPS 1 and GPS 2 (if installed) circuit breaker size and location are not affected.

Installation of Mod 6/2304 (SBAS GPS Receivers) is a prerequisite for S.O.O. 6225, SBAS navigation and approach capability, as well as S.O.O. 6347, ADS-B Out functionality, refer to 34-53-00, KXP 2290 Transponder System – General Data.

The SBAS navigation and approach capability is enabled through the incorporation of customer standard order option S.O.O. 6225. When two GPS receivers are installed, both must be SBAS receivers. SBAS GPS approach capability allows SBAS-based GPS approaches such as LPV approaches.

For aircraft with S.O.O. 6388 incorporated, there is one Post Mod 6/2304 KGS 200 Global Navigation System Sensor Unit (GNSSU) receiver installed.

For aircraft with S.O.O. 6388 incorporated, a second Post Mod 6/2304 KGS 200 Global Navigation System Sensor Unit (GNSSU) and associated systems are installed on the aircraft by incorporating customer Standard Order Option S.O.O. 6380.

3. Adjustment/Test

A. Function Test Global Positioning System

1) General

These procedures are for the function test of global positioning systems GPS 1 and GPS 2 (if installed).

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) In the Apex system Sensors page, verify GPS 1 and GPS 2 (if installed) receive data is valid.

d) Ensure there are no GPS Fail CAS messages or PFD annunciations.

e) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. ADS-B GPS Input Verification (Mod 6/2354 KXP 2290A with S.O.O. 6347 ADS-B Out)

1) General

These procedures are for the ADS-B Out GPS input verification test procedures for GPS 1 and GPS 2 (if installed).

2) ADS-B GPS Input Verification

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Verify ADS-B status indication is white in the Transponder Radio Tuning window.

d) Open GPS 1 (J3) (Left Bus) circuit breaker, wait a few seconds for the GPS 1 Fail CAS message to be displayed. Ensure ADS-B status indication remains white, leave GPS 1 circuit breaker open for the next step (dual GPS). Ensure ADS-B status indication turns amber, skip the next step (single GPS).

e) Open GPS 2 (J4) (Right Bus) circuit breaker, wait a few seconds for the GPS 2 Fail CAS message to be displayed, ensure ADS-B status indication turns amber.

f) Close GPS 1 (J3) (Left Bus) circuit breaker, allow enough time for GPS 1 to initialize, ensure ADS-B status indicator returns to white.

g) Close GPS 2 (J4) (Right Bus) circuit breaker (dual GPS).

h) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Electrical Bonding Test – GPS 1 and GPS 2 (if installed) Antenna

1) General

These procedures are for the electrical bonding test procedures for GPS 1 antenna and GPS 2 antenna (if installed).

2) Electrical Bonding Test

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Gain access to the ceiling compartment forward of station 143.95 for GPS 1 antenna and forward of station 321.6 for GPS 2 antenna (if installed). (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

c) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the GPS antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

d) If the electrical conductivity does not meet requirements, check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

e) Reinstall the ceiling compartment interior. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1–64–2W.

A. Removal/Installation of GPS Receiver (KGS 200 GNSSU)

1) General

These procedures are for the removal and installation of the GPS 1 receiver and GPS 2 receiver (if installed).

2) Removal

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the GPS 1 (J3) (Left Bus) and/or GPS 2 (J4) (Right Bus), circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Remove top, middle and bottom covers attached to bulkhead station 44.0 for full access.

e) Disconnect the electrical and coaxial cables at the port side rear of the GPS receiver unit.

f) Remove the four screws that hold the GPS receiver unit to the bottom of the middle avionics shelf.

g) Remove unit from aircraft.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that GPS 1 (J3) (Left Bus) and/or GPS 2 (J4) (Right Bus), circuit breakers on the avionics circuit breaker panel are open and fitted with safety clips.

c) Align the GPS receiver mounting holes to the bottom of the middle avionics shelf. Ensure correct orientation, connectors facing inward.

d) Install the four mounting screws and ensure receiver unit is securely held.

e) Reconnect the electrical and coaxial cables at the port side rear of the GPS receiver unit. Ensure connections are sound and secure.

f) Reinstall top, middle and bottom covers attached to bulkhead station 44.0 at the aft wall of the baggage compartment. Ensure covers are mounted correctly and secure.

WARNING **ACCESS COVERS ARE REQUIRED TO MAINTAIN PROTECTION FEATURES THAT ARE ESSENTIAL TO MINIMIZING ADVERSE EFFECTS FROM HIGH INTENSITY RADIATED FIELDS (HIRF).**

g) Remove safety clips and reset circuit breakers GPS 1 (J3) (Left Bus) and/or GPS 2 (J4) (Right Bus), on the avionics circuit breaker panel.

h) Do a function test on the Global Positioning System, refer to Para 3.A.

i) Do an ADS-B GPS Input Verification, refer to Para 3.B (applicable for Mod 6/2354 with S.O.O. 6347).

B. Removal/Installation of GPS Antenna (KA96)

1) General

These procedures are for the removal and installation of the GPS 1 and GPS 2 (if installed) antennas.

2) Removal

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the GPS 1 (J3) (Left Bus) and/or GPS 2 (J4) (Right Bus), circuit breakers on the avionics circuit breaker panel and fit safety clips.

c) Carefully cut the sealant from around the base of the antenna.

d) Support the antenna and remove the four screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

g) Remove antenna from aircraft.

Note Stow connectors of the removed components.

3) Installation

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that GPS 1 (J3) (Left Bus) and/or GPS 2 (J4) (Right Bus), circuit breaker on the avionics circuit breaker panel are open and fitted with safety clips.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with the four mounting screws.

f) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Use sealant to both fillet seal around the antenna base and dome seal the securing screws. The aircraft will be ready to fly approximately 24 hours after sealing (refer to 20-80-00, Installation of Aircraft Antennas – General data).

j) Check the antenna electrical bonding, refer to Para 3.C.

k) Remove safety clips and reset circuit breakers GPS 1 (J3) (Left Bus) and/or GPS 2 (J4) (Right Bus) on the avionics circuit breaker panel.

l) Do a function test on the Global Positioning System, refer to Para 3.A.

5. Troubleshooting

A. Global Positioning System (KGS 200 GNSSU)

If the GPS is deemed inoperative check the following:

1) Ensure GPS 1 (J3) and/or GPS 2 (J4) circuit breaker are closed.

2) Visually inspect GPS connector for damaged wires or contacts.

3) Visually inspect GPS antenna for corrosion and contaminants.

4) Ensure that the antenna coaxial cable connector is properly mated and secure.

5) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

6) If repair or service is required, send to appropriate rated avionics repair facility.

XM Satellite Weather System – General data

1. General

Provision for the XM Satellite Weather System to be installed as a customer standard option order 6211.

The XM Satellite Weather System is a streaming weather data receiver.

2. Description

A. XM Satellite Weather Receiver (XMD157–01)

The XM Weather receiver is mounted on the rear avionics rack. The XM Satellite Weather system sends received weather data to the Modular Avionics Unit (MAU) Advanced Graphics Unit (AGU), which is then displayed on the Multi-Function Display (MFD).

Basic functionality received by the XM Satellite and displayed on the MFD includes:

- Next Generation Radar (NEXRAD)
- Temporary Flight Restrictions (TFR)
- Airman's Meteorological Advisories (AIRMET)
- Significant Meteorological Information (SIGMET)
- Storm tops
- Lightning
- Winds
- Satellite cloud heights
- Turbulence
- METAR
- Continental United States (CONUS) radar coverage mask (part of NEXRAD)
- Echo tops

The XM Satellite Weather antenna is installed on the top of the fuselage at FS315.

28 Volt DC power is provided from the left bus through a 5 amp SATELLITE WX (K4) (Right Bus) circuit breaker on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The XM Weather receiver interfaces to the following aircraft systems:

1) Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

3. Adjustment/Test

A. Function Test XM Satellite Weather Receiver (XMD157–01)

1) General

These procedures are for the function test of the XM Satellite Weather System.

2) Function Test

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Wait several minutes for the aircraft electrical and avionics systems to initialize.

c) Ensure that SATELLITE WX (K4) (Right Bus) circuit breaker on the avionics circuit breaker panel is closed.

d) Select XM Satellite weather to be displayed on the lateral map display.

e) Verify XM WX annunciator flashes in reverse video until the initial streaming data is received. The XM WX annunciator turns amber if no streaming data has been received for 10 minutes.

f) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Electrical Bonding Test – XM Satellite Antenna (XMRANT–01)

1) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

2) Gain access to the ceiling compartment forward of station 321.6. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

3) Verify that the electrical conductivity is consistent over the electrical bond contact area by placing a probe on the XM antenna mounting screw. Place the second probe on the nearest ground stud. The resistance between the two points must be < 3.0 milliohms unless stated otherwise.

4) Check the electrical conductivity between several random mounting screws within the electrical bond contact area to ensure consistency.

5) If the electrical conductivity is not consistent over the electrical bond contact area check the antenna electrical bonding in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

6) Replace ceiling compartment. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTEGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT / REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal/Installation of XM Satellite Weather Receiver (XMD157–01)

1) General

These procedures are for the removal and installation of the XM Satellite Weather receiver, if installed.

2) Removal

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the SATELLITE WX (K4) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Gain access to the aft avionic shelf through the rear baggage compartment.

d) Disconnect the electrical and coaxial cables at the front of the XM Satellite weather receiver.

e) Remove the four screws that secure XM Weather receiver to the avionics shelf.

f) Remove unit from aircraft.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the SATELLITE WX (K4) (Right Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Gain access to the aft avionic shelf through the rear baggage compartment.

d) Align the configuration module mounting holes to the avionics shelf.

e) Reconnect connectors to the XM Satellite weather receiver. Ensure connections are secure.

f) Remove safety clip and close SATELLITE WX (K4) circuit breaker on the avionics circuit breaker panel.

g) Test the XM Satellite Weather receiver as per Para 3.A.2).

B. Removal/Installation of XM Satellite Weather Antenna (XMRANT–01)

1) General

These procedures are for the removal and installation of the XM SATELLITE antenna, if installed.

2) Removal

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Pull the SATELLITE WX (K4) (Right Bus) circuit breaker on the avionics circuit breaker panel and fit safety clip.

c) Carefully cut the sealant from around the base of the antenna.

d) Support the antenna and remove the four screws from the base of the antenna.

e) Carefully lift the antenna clear of the fuselage until there is access to the coaxial connection.

f) Disconnect the coaxial connector.

g) Remove antenna from aircraft.

Note Stow the connectors of the removed components.

3) Installation

a) Ensure the aircraft power is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure that the SATELLITE WX (K4) (Right Bus) circuit breaker on the avionics circuit breaker panel is pulled and fitted with safety clip.

c) Solvent clean the electrical contact bond areas immediately before the installation (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

d) Bring the antenna assembly close to the fuselage, connect the coaxial connector.

e) Secure the antenna assembly on fuselage with four mounting screws.

f) Solvent clean the exposed area of the airframe around the antenna (refer to 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data).

g) Touch up the exposed area of airframe around the antenna if required with F19 primers by brush within 24 hours of the antenna installation. Ensure that the primer re-finishing provides complete coverage of the affected area.

h) If the affected area requires paint touch up, brush coat the area with the top-coat paint system matching the existing.

i) Seal around the antenna base and dome seal the securing screws. (Refer to 20-80-00, Installation of Aircraft Antennas – General data). The aircraft will be ready to fly approximately 24 hours after sealing.

j) Check the antenna electrical bonding as per Para 3.B.

k) Remove the safety clip and replace SATELLITE WX (K4) (Right Bus) circuit breaker on the avionics circuit breaker panel.

l) Do a function test on the XM Satellite receiver as per Para 3.A.2).

5. Troubleshooting

A. XM Satellite Weather (XMD157–01)

1) If the XM Satellite weather is deemed inoperative check the following:

a) Ensure SATELLITE WX (K4) circuit breaker is closed.

b) Visually inspect XM Satellite Weather connector for damaged wires or contacts.

c) Visually inspect XM Satellite Weather antenna for corrosion and contaminants.

d) Ensure that the XM Satellite Weather connector is properly mated and secure.

e) Inspect antenna external surfaces for damage.

f) Check that the antenna is properly mounted and screws are tight.

g) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

h) If repair or service is required, send to appropriate rated avionics repair facility.

Flight Management System (FMS) – General Data

1. General

A Flight Management System (FMS) is an integrated system designed to give the pilot centralized control for the navigation sensors, computer based flight planning, fuel and radio management. The FMS functions are provided by the two Advanced Graphics Modules (AGM), located inside the Modular Avionics Unit (MAU) assembly (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data). The FMS uses two navigation databases, housed in the MAU, containing worldwide or regional data (refer to 45-00-00, Central Maintenance Computer – General Data). A custom database contains flight plans and pilot defined waypoints and an aircraft database contains the aircraft specific configuration for FMS performance calculations.

Using the databases, along with GPS and ADAHRS information, the FMS calculates and displays navigational and flight planning information to the flight crew. The navigation database is commercially produced and updated and released every 28 days.

The FMS provides lateral and vertical control output to the Automatic Flight Control System (AFCS), when S.O.O. 6221 is incorporated.

Provisions have been made by Mod 6/2303 for standard customer option S.O.O. 6313 for a second FMS, providing dual FMS functionality.

Provision have been made by Mod 6/2304 for upgrade of the dual KGS 200 GPS receivers to include SBAS capable units. SBAS Navigation and Approach Capability is provided by standard customer order option S.O.O. 6225.

The FMS interfaces to the following components:

A. MF Controller (KMC 2220) (refer to 31-13-00, Apex Controllers – General Data).

B. Modular Avionics Unit (MAU) (refer to 31-43-00, Modular Avionics Unit (MAU) (Mod 6/2049) – General Data).

C. Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) (refer to 34-23-00, Air Data, Attitude and Heading Reference System (ADAHRS) (KSG 7200) – General Data).

D. Global Positioning System (GPS) (refer to 34-56-00, Global Positioning System (GPS) – General Data).

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CHAPTER 36

PNEUMATIC

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CHAPTER 36 – PNEUMATIC

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Pneumatic – General Data

1. General

The pneumatic system consists of a low pressure pneumatic package which is required only when a customer option airframe de-icing is installed.

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Distribution (Mods S.O.O. 6004) – General Data

1. General

Refer to Figure 1 for details.

The pneumatic package is located on the left side of the cabin roof at station 177.00 approximately. The pressure supply for the pneumatic package is tapped from the bleed air system line on the left side of the cabin roof. (Refer to 21-00-00, Air Conditioning – General Data).

A. Pneumatic Package

The pneumatic package consists of a heat exchanger, a pressure regulator, a strainer, low pressure switch, and connecting pipe lines.

B. Heat Exchanger

The heat exchanger is a rearward facing airscoop mounted on top of the fuselage at station 176.50. The double walls of the airscoop and the internal baffle system form the heat exchanger. Bleed air enters the heat exchanger through the outboard pipe, and is directed through the baffle system which exposes the air to the entire surface area of the inner and outer walls of the scoop, where cooling air passing over these surfaces dissipates the heat from the bleed air. The cooled air leaves the heat exchanger through the inboard pipe.

C. Pressure Regulator

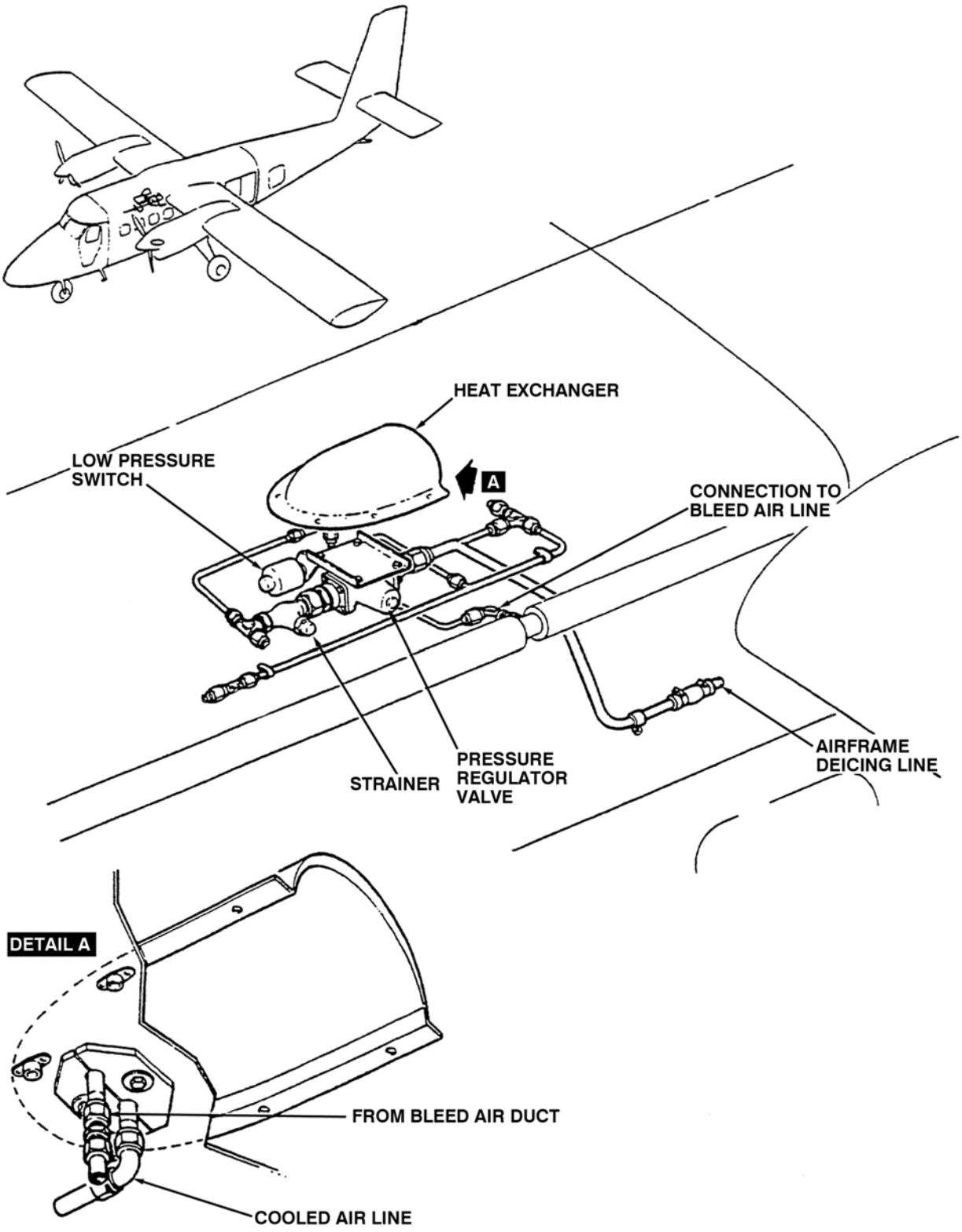
The pressure regulator is mounted on a plate which is secured to the cabin roof. It is set to regulate the pneumatic system at 18 psi. A safety relief valve on the pressure regulator is set to relieve pressure at 25 psi should the regulator section fail.

D. Strainer

The strainer incorporates a 60–mesh wire element and is fitted to the intake port of the pressure regulator in the cooled air line from the heat exchanger to prevent impurities in the bleed air from entering the pneumatic system.

E. Low Pressure Switch

The low pressure switch is set to illuminate a caution CAS message should the system pressure drop to between 13 and 15 psi. When pressure increases to between 16 and 18 psi, the switch contacts open to extinguish caution CAS message. If the aircraft is on the ground, a pneumatic pressure low advisory message will activate.



Pneumatic Package
Figure 1

2. Servicing

A. Strainer

1) Remove cabin ceiling panels. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

2) Detach upholstery in cabin roof.

3) Cut lockwire and unscrew strainer. Remove wire element.

4) Clean wire element and strainer.

5) Position wire element in strainer, install strainer and secure with lockwire.

6) Install cabin ceiling panels. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

3. Adjustment/Test

A. Function Test Pneumatic Package

Refer to Figure 1 for details.

Equipment required:

- Source of compressed air 100 psi
- Pressure gauge 0 to 30 psi
- Pressure test rig consisting of a pressure tank capable of withstanding 160 psi and having a volume of 10 cu ft, a 0 to 100 psi pressure gauge, and two shut-off valves to isolate the rig from the aircraft and the compressed air source

1) Remove cabin ceiling panels (refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data) in location of station 177.00 to provide access to heat exchanger.

2) Disconnect bleed air line at inlet to heat exchanger. Cap end of line, and connect pressure rig to union on heat exchanger.

3) On aircraft with airframe de-icing, disconnect the pressure supply line at the ejector located in the rear fuselage, station 468.00. Cap the fitting on ejector.

4) Carry out pressure test as follows:

a) Remove cap on pneumatic package and install 0 to 30 psi pressure gauge.

b) Set pressure rig to 80 psi. Check that low pressure caution CAS message goes out when pressure is between 16 and 18 psi. Check pressure gauge at forward tee reads 18 ± 2 psi. Adjust regulator valve to obtain correct pressure, if necessary.

c) Lower rig pressure gradually to 25 psi. Check that reading on pressure gauge is maintained at 18 ± 2 psi.

d) Lower rig pressure gradually and check that low pressure caution CAS message illuminates when pressure gauge reads between 13 and 15 psi.

e) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

f) Shut off rig pressure. Remove pressure gauge from forward tee and install cap.

g) Disconnect pressure rig. Connect bleed air line to union on heat exchanger.

h) Install cabin ceiling panels. (Refer to 25-20-00, Cabin Compartment – Commuter Version (Mod 6/2050) – General Data).

Pressure Regulator Valve – Maintenance Practices

1. Adjustment/Test

A. Bench Pressure Test

Equipment required:

- Pneumatic test rig
- Two pressure gauges 0 to 50 psi
- 1–inch gate valve
- Flowmeter

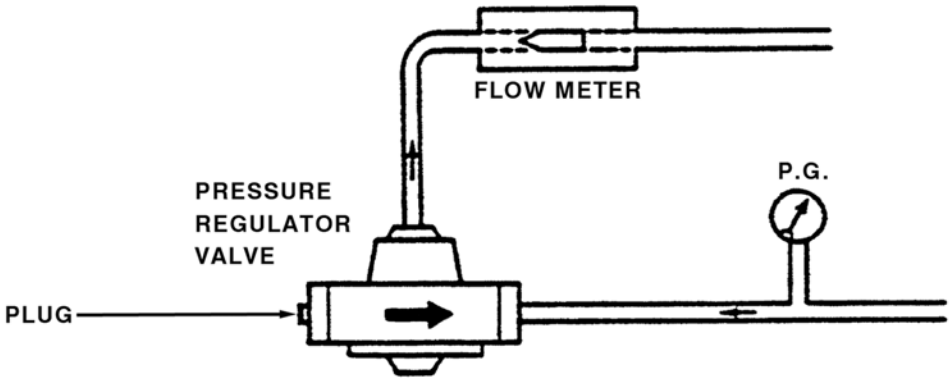
1) Crack and Reseat Pressure

Refer to Figure 1 for details.

a) Install pressure regulator valve as shown in Figure 1.

b) Gradually increase supply pressure to 22 psi; check that relieving flow is 12 cfh minimum.

c) Gradually decrease supply pressure to 18 psi; check that leakage flow is 5 cfh maximum.



Crack and Reseat Pressure – Test Set Up
Figure 1

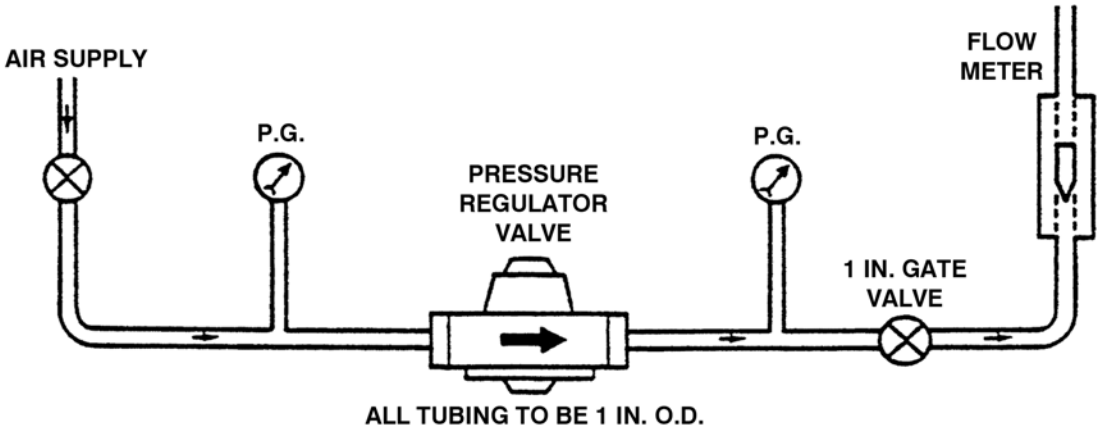
2) Function Test

Refer to Figure 2 for details.

a) Install pressure regulator valve as shown in Figure 2.

b) Apply supply pressure of 30 ± 1 psi and air flow of 5 ± 1/2 cfm: check that regulated air pressure is 18 ± 1 psi.

c) Release air pressure and remove pressure regulator valve. Cap pressure regulator valve connections unless it is to be installed immediately.



Function Test – Test Set Up
Figure 2

Indicating – General Data

1. General

The pneumatic indicating system is basically a low pressure warning displayed through CAS. The system consists of a Pneumatic Press Low caution message which is displayed when a low pressure switch actuates if regulated pressure drops to between 13 and 15 psi. On increasing pressure, the pressure switch actuates between 16 and 18 psi to extinguish the caution CAS message. The pressure switch is connected into the pressure line from the pressure regulator.

The Pneumatic Press Low caution message is driven by the following logic:

A. If the SURFACE DE-ICE switch is in the BOOTS AUTO–FAST or BOOTS AUTO–SLOW position (in other words, the switch is not in the OFF/MANUAL position), and;

B. If the airframe pressure sensing system indicates a low pneumatic pressure state, and;

C. If the aircraft is not on the ground, the Pneumatic Press Low caution message will activate. If the aircraft is on the ground, the Pneumatic Press Low advisory message will activate.

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CHAPTER 44

CABIN SYSTEMS

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CHAPTER 44 – CABIN SYSTEMS

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Cabin Systems – General Data

1. General

The Cabin Passenger Address System, (if installed) is part of the audio intergration system, refer to 23-50-00, Audio Integration – General Data. The Cabin Passenger Address System allows the flight compartment crew to communicate with the passengers in the cabin.

The Cabin Passenger Address System consists of a dedicated NAT 245 speaker amplifier used to drive six speakers mounted in the cabin roof.

2. Description

A. Cabin Passenger Address System (NAT 245)

When the audio control panel PA is selected and keyed, the NAT 245 drives six 8 ohm cabin speakers. The NAT 245 speaker amplifier is mounted on the middle shelf in the forward avionics rack. With the pilot and co-pilot's audio controls on PA, the pilot's microphone has priority if both the pilot and co-pilot's PTT switches are pushed at the same time.

28 Volt DC power is protected by a 5 amp CABIN PA (5M) (Left Bus) circuit breaker located on the avionics circuit breaker panel (refer to 24-00-00, Electrical Power – General Data).

The Cabin Passenger Address System interfaces to the following aircraft systems:

1) Audio Control Panel System (refer to 23-53-00, Audio Control Panel – General Data).

3. Adjustment/Test

A. Function Test Cabin Passenger Address System (NAT 245)

1) General

These procedures are for the function test of the Cabin Passenger Address System (NAT 245).

2) Function Test

a) Ensure that CABIN PA (5M) circuit breaker is closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait several minutes for the aircraft electrical systems to initialize.

d) On the audio control panel, adjust SPKR/PA switch to select PA mode.

e) Key the PTT from the pilot's position. Verify audio is heard in the cabin.

f) Key the PTT from the co-pilot's position. Verify audio is heard in the cabin.

g) Key the PTT from the pilot and co-pilot's position. Verify that the audio from the pilot is heard in the cabin.

h) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Audio Control Panel PA Adjustment

1) General

These procedures are for the adjustment of the audio level of the cabin passenger address speakers.

2) Adjustment

a) Ensure that CABIN PA (5M) circuit breaker is closed.

b) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

c) Wait several minutes for the aircraft electrical systems to initialize.

d) On the audio control panel, adjust SPKR/PA switch to select PA mode.

e) Key the PTT from the pilot's position. Verify audio is heard in the cabin.

f) Evaluate the audio level.

g) Remove the pilot's audio control panel as per 23-53-00, Audio Control Panel – General Data, Para 4.A.2).

h) Adjust the pilot's audio panel PA potentiometer adjustment on the top of the unit.

i) Install the pilot's audio control panel as per 23-53-00, Audio Control Panel – General Data, Para 4.A.2).

j) Test the pilot's audio control panel as per 23-53-00, Audio Control Panel – General Data, Para 3.A.2).

k) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

4. Removal/Installation

WARNING **INSTALLATION OF THIS COMPONENT CONTAINS PROTECTION FEATURES THAT ARE CRITICAL TO PREVENTING DEGRADATION FROM THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF) AND INDIRECT EFFECTS OF LIGHTNING (IEL). ENSURE INTERGRITY OF THE HIRF/IEL PROTECTION FEATURES ARE MAINTAINED IN ACCORDANCE WITH THE FOLLOWING:**

CHASSIS GROUNDING – ALL RACK MOUNT/REMOTE MOUNT UNITS SHALL BE ELECTRICALLY BONDED TO THE AIRFRAME. ENSURE THAT THE MATING SURFACES BETWEEN THE LRU MOUNTING TRAY (OR LRU MOUNTING FEET IF A TRAY IS NOT USED) PROVIDE A LOW IMPEDANCE ELECTRICAL PATH. ALL MATING SURFACES MUST BE FREE OF ALL PAINT AND OTHER NONCONDUCTIVE ELEMENTS. REFER TO BONDING REQUIREMENTS IN 20-75-00, ELECTRICAL BONDING AND GROUNDING OF AIRCRAFT STRUCTURES – GENERAL DATA.

SHIELD GROUNDS – VERIFY SHIELD GROUNDING AT CONNECTOR BACKSHELLS AS PER REQUIREMENTS IN PSM 1-64-2W.

BACKSHELLS OR STRAIN RELIEFS – VERIFY PROPER ELECTRICAL BONDING BETWEEN THE BACKSHELL/STRAIN RELIEF AND THE MOUNTING TRAY OR LRU CASE AS PER REQUIREMENTS IN PSM 1-64-2W.

A. Removal and Installation of the Cabin Passenger Address System (NAT 245)

1) General

These procedures are for the removal and installation of the Cabin Passenger Address System (NAT 245).

2) Removal

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Open CABIN PA (5M) circuit breaker on the avionics circuit breaker panel and fit with a safety clip.

c) Gain access to the forward avionics shelves through the forward baggage compartment.

d) Remove top and middle cover attached to bulkhead station 37.21.

e) Disconnect the electrical connector on the front of the unit.

f) Remove the (4) screws securing the unit to the forward avionics shelf.

g) Remove unit from aircraft.

Note Stow connectors as required.

3) Installation

a) Ensure the aircraft electrical system is OFF. (Refer to 24-00-00, Electrical Power – General Data).

b) Ensure the CABIN PA (5M) circuit breaker on the avionics circuit breaker panel is opened and fitted with a safety clip.

c) Align the units mounting holes to the forward avionics shelf.

d) Secure the unit to the avionics shelf with mounting (4) screws.

e) Reconnect connector to the unit. Ensure connections are secure.

f) Remove the safety clips and close CABIN PA (5M) circuit breaker on the avionics circuit breaker panel.

g) Test the Cabin Passenger Address System as per Para 3.A.

5. Troubleshooting

A. Cabin Passenger Address System (NAT 245)

1) If the Cabin Passenger Address System (NAT 245) is deemed inoperative check the following:

a) Ensure CABIN PA (5M) circuit breaker is closed.

b) Visually inspect the NAT 245 unit for damaged wires or contacts.

c) Visually inspect the cabin speakers for damaged wires or contacts.

d) Ensure that the audio NAT 245 and cabin speaker connectors are properly mated and secure.

e) If troubleshooting at the wire level is required refer to the applicable wiring diagrams in Wiring Manual PSM 1-64-2W.

f) If repair or service is required, send to appropriate rated avionics repair facility.

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CHAPTER 45

CENTRAL MAINTENANCE SYSTEM (CMS)

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CHAPTER 45 – CENTRAL MAINTENANCE SYSTEM (CMS)

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Central Maintenance Computer – General data

1. General

The Central Maintenance Computer (CMC) is used to:

- Verify the configuration of the Apex system
- Determine the status of an LRU or function
- Execute supported LRU or function test capabilities
- Diagnose a current Apex problem
- Use historical data to diagnose past Apex problems
- Download the CMC fault history database
- Verify the configuration of the CMC

A. Central Maintenance Computer Function (CMCF)

A software application hosted on the MAU advanced graphics module (AGM1). This software provides the real-time collection of member system fault reports and the subsequent storage of maintenance messages and any associated correlations to flight deck effects. Additionally, this application provides the maintenance system central point of control on the aircraft.

The Apex CMC is constrained to provide maintenance support for only the Honeywell Apex Avionics System. It does not provide maintenance capability for the other aircraft systems (for example, engine, hydraulics, etc.).

B. Loadable Diagnostic Interface (LDI)

A database that contains specific information pertaining to the maintenance characteristics of the Aircraft Diagnostics and Maintenance System (ADMS). The LDI provides the necessary operational data to permit the generic CMCF to operate within the particular aircraft.

C. Fault History Database (FHDB)

The storage repository for all fault data that is output from the CMCF. This is the data that is accessed by the maintenance technician, using the remote terminal, to assess the past and present operating condition of the aircraft.

D. Remote Terminal (RT)

A PC-based application, consisting of a PC, an operating system, and an ADMS application. This system connects to the CMCF via the aircraft local area network (LAN). The RT provides the means by which the user gains access to the aircraft-resident CMCF. The primary function of the RT is to provide the user interface, through which the capabilities provided by the ADMS are selected and controlled.

E. Software Loading

The CMC Remote Terminal data loading system will automatically load the software and data bases on the appropriate MAU module(s) if full load is selected.

Note Depending if Aircraft has Honeywell Apex Phase 1A or Apex Phase 2A software installed, the software required is listed in Table 1 — MAU Module Software/Database Locations – Phase 1A Load 1.3 or Table 2 — MAU Module Software/Database Locations – Phase 2A Load 4.4.

1) Specific MAU Module – Software Loading Procedures

For an individual MAU module replacement, the correct software loading sequence must be followed.

Note For ECL load procedures refer to 46-20-01, Apex Electronic Checklist File – Load Procedures.

2) Multiple MAU Modules – Software Loading Procedures

For multiple or complete MAU modules replacement, the correct software loading sequence must be followed.

3) The MAU Module Software Loading sequence is as follows:

a) APEX MAU Software via the 'CMC Remote Terminal' (refer to 45-00-02, Apex Modular Avionics Unit Software – Load Procedures). Media CD to be used PN: xx60000469–xxx – Apex 12 Load 1.3 (Pre Mod 6/2303); or PN: xx60002735–xxx – Apex 12 Load 4.4 (Post Mod 6/2303).

b) Load the APM Settings File via the 'CMC Remote Terminal' (refer to 45-00-01, Apex Options and Settings File – Load Procedures). Media CD to be used PN: VSP25–xxxxx.

c) Load the APM Options File via the 'CMC Remote Terminal' (refer to 45-00-01, Apex Options and Settings File – Load Procedures). Media CD to be used PN: xx60000053–xxxxx (Pre Mod 6/2303); or PN: xx600002657–xxxxx (Post Mod 6/2303) – Options File password required (on CD).

d) Load the MAU LDI (Loadable Diagnostics Interface) via 'CMC Remote Terminal'. Media CD to be used PN: VD6–LDI_1_1_20 or higher version (Pre Mod 6/2303); or PN: VD6_LDI_4_1_4_20141017 or higher version (Post Mod 6/2303).

e) FMS Performance Database via the 'CMC Remote Terminal' (refer to 45-00-03, Apex FMS Aircraft Performance Database Files – Load Procedures). Media CD to be used PN: xx60000461–xxx – APM FMS Performance Database.

f) All navigation database CD's (NAV Database (Purple), Charts Database (Blue), Terrain Database (Green), etc) via the 'CMC Remote Terminal' (refer to 45-00-04, Apex Database Files – Load Procedures). No special loading sequence is required between databases.

g) Electronic checklist CD ROM via the 'CMC Remote Terminal' (refer to 46-20-01, Apex Electronic Checklist File – Load Procedures).

Table 1 MAU Module Software/Database Locations – Phase 1A Load 1.3

		SOFTWARE RESIDING ON CARD								
		APEX OPERATING SOFTWARE	**APM SETTINGS FILE** (NOTE 2)	**APM OPTIONS FILE** (NOTE 2)	**LDI LOADABLE DIAGNOSTIC INTERFACE**	**FMS PERFORM. DATABASE**	**NAV DATABASE (PURPLE)** (NOTE 1)	**CHARTS DATABASE (BLUE)**	**TERRAIN DATABASE (GREEN)**	**ELECT. CHECKLIST DATABASE**
MAU MODULE	NIC NO. 1	YES	YES	YES						
	NIC NO. 2	YES	YES	YES						
	AGM NO. 1	YES			YES		YES	YES	YES	
	AGM NO. 2	YES					YES	YES	YES	
	AIOP a	YES				YES	YES			
	AIOP b	YES				YES	YES			YES
	DGIO	YES								
	CSIO	YES								
	NO. 1 CONFIG. MODULE		YES	YES						
	NO. 2 CONFIG. MODULE		YES	YES						

Note

1 The NAV data base (purple) holds two individual files, CRH (Cities, Roads, Highways), and the INAV-16M, airspace communication and navigational data. Both files have to be individually loaded, using the CMC Remote Terminal Software (Maintenance Laptop).

2 The APM Option and the APM settings data is stored in the Aircraft Personally Modules, located within the backshells of the NIC No. 1 and NIC No. 2 cards.

Table 2 MAU Module Software/Database Locations – Phase 2A Load 4.4

		SOFTWARE RESIDING ON CARD								
		APEX OPERATING SOFTWARE	**APM SETTINGS FILE** (NOTE 2)	**APM OPTIONS FILE** (NOTE 2)	**LDI LOADABLE DIAGNOSTIC INTERFACE**	**FMS PERFORM. DATABASE**	**NAV DATABASE (PURPLE)** (NOTE 1)	**CHARTS DATABASE (BLUE)**	**TERRAIN DATABASE (GREEN)**	**ELECT. CHECKLIST DATABASE**
MAU MODULE	NIC NO. 1	YES	YES	YES						
	NIC NO. 2	YES	YES	YES						
	AGM NO. 1	YES					YES	YES	YES	
	AGM NO. 2	YES					YES	YES	YES	
	AIOP a	YES			YES	YES	YES			
	AIOP b	YES				YES	YES			YES
	DGIO	YES								
	CSIO	YES								
	NO. 1 CONFIG. MODULE		YES	YES						
	NO. 2 CONFIG. MODULE		YES	YES						

Note

1 The NAV data base (purple) holds two individual files, CRH (Cities, Roads, Highways), and the INAV-16M, airspace communication and navigational data. Both files have to be individually loaded, using the CMC Remote Terminal Software (Maintenance Laptop).

2 The APM Option and the APM settings data is stored in the Aircraft Personally Modules, located within the backshells of the NIC No. 1 and NIC No. 2 cards.

F. General Laptop CMC Remote Terminal Setup

1) The following procedure is required to load:

a) APEX MAU Operating Software

b) Settings File (refer to 45-00-01, Apex Options and Settings File – Load Procedures)

c) Options File (refer to refer to 45-00-01, Apex Options and Settings File – Load Procedures)

d) MAU LDI (refer to 45-00-05, Loadable Diagnostic Interface File – Load Procedures)

e) FMS Performance Database (refer to 45-00-03, Apex FMS Aircraft Performance Database Files – Load Procedure)

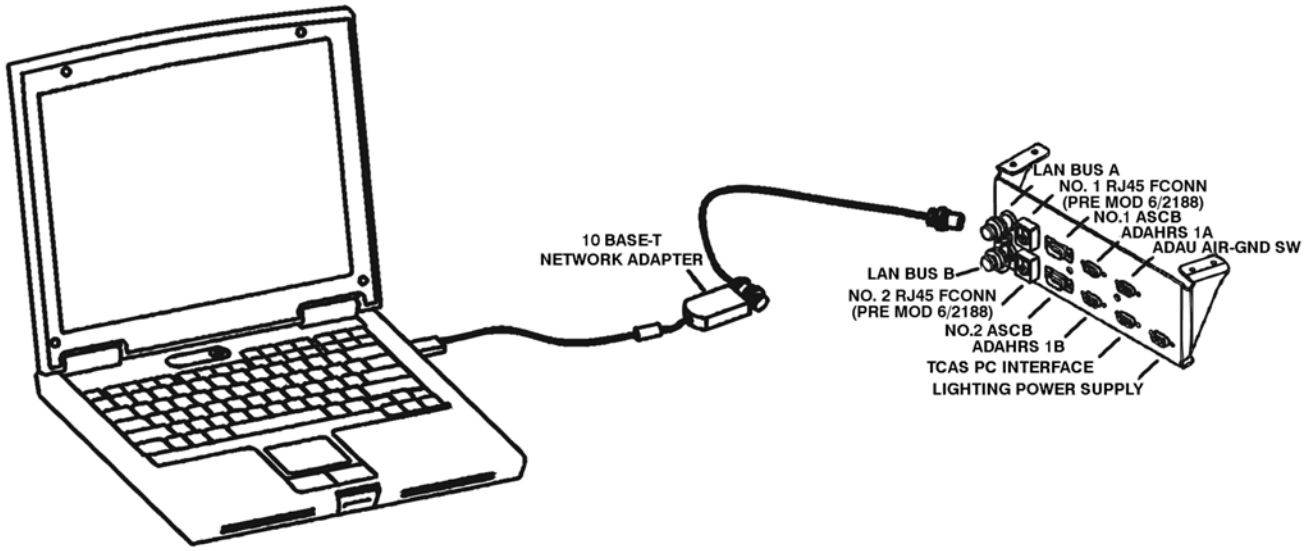
f) NAV Database (Purple) (refer to 45-00-04, Apex Database Files – Load Procedures)

g) CHART Database (Blue) (refer to 45-00-04, Apex Database Files – Load Procedures)

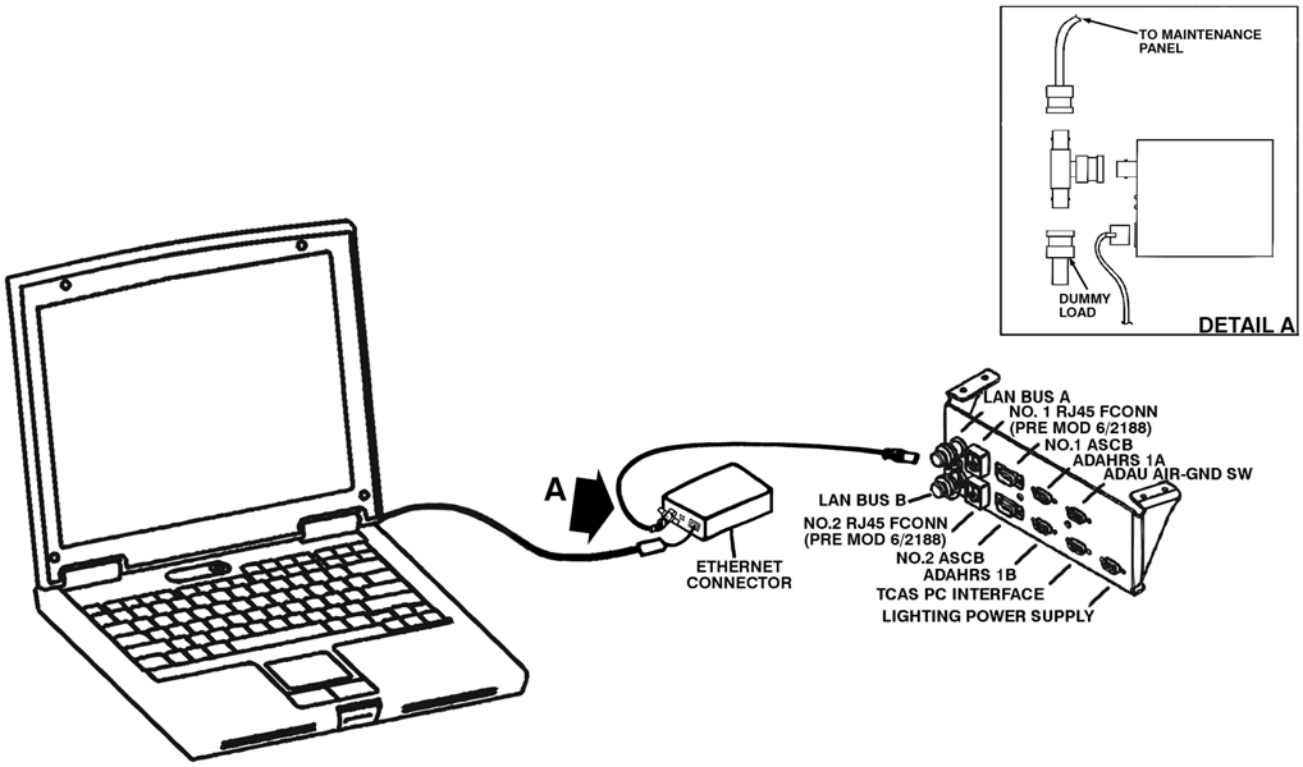
h) TERRAIN Database for MAU (Green) (refer to 45-00-04, Apex Database Files – Load Procedures)

2) Also required for:

Standard Maintenance features offered through the CMC remote tool.



Forward Maintenance Panel – PCMCIA Card
Figure 1



Forward Maintenance Panel – Ethernet to LAN Adapter Kit
Figure 2

G. Setup CMC Remote Terminal

1) This procedure requires the following:

a) Laptop computer with PCMCIA Card and 10BASE–T Network Adapter with LAN interface cable (refer to Figure 1), or Ethernet to LAN adaptor kit (refer to Figure 2).

b) Proprietary Honeywell CMC Remote Terminal Software RT 2.0 & DLS 5.0.2 or later applicable version (Pre Mod 6/2303); or RT 5.3 & DLS 7.3.1 or later applicable version (Post Mod 6/2303).

Note

- During loading, aircraft power must remain ON for the MAU system and for the Ethernet to LAN adapter.
- Ensure all MAU (4), ADAHRS (2), and MFD CTRL circuit breakers are closed. The MAU power must be on for a minimum of 15 minutes before beginning the loading procedure.
- Pitot/Static test sets may not be used during this procedure. Airspeed must be less than 40 knots and Rad Alt must be less than 10 feet.
- The MAU system must not be disturbed by other maintenance activities, which may affect the MAU system.
- APEX displays will be “X’d out” during this process. Do not attempt other activity on APEX during loading procedures.
- If loading has been disturbed in any way, there is a risk that loading will have to be repeated.
- This procedure requires the COM1 port of the laptop computer to be available. If problems are encountered disable other applications that may be using the COM1 port.

c) Gain access to the Avionics Maintenance Panel in the forward avionics area (refer to 12-00-00, Servicing – General Data).

d) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

e) Wait several minutes for the aircraft electrical and avionics systems to initialize.

f) Plug 10BASE–T Network Adaptor into laptop PCMCIA card (refer to Figure 1). Alternatively plug in Ethernet to LAN Adaptor Kit into Ethernet card on the laptop (refer to Figure 2).

g) On Maintenance Panel, on the Nose Avionics Rack, remove LAN terminator on LAN BUS A or LAN BUS B. (Refer to 12-00-00, Servicing – General Data).

Note Only remove one terminator at a time.

h) Connect LAN cable from adaptor to LAN BUS A connection on maintenance panel.

i) Insert the applicable CD or USB memory stick, which is holding the data to be loaded. Refer to applicable chapter for required files.

Note If the GIO card has been replaced, and the CMC Remote Terminal appears to not connect to the aircraft MAU, do the following to set the MAU into ground mode, to enable loading:

1 On the Maintenance Laptop open the Command Prompt:

WinXP:

- Click the ‘START’ button
- Click the ‘RUN’ button
- Type ‘cmd’
- Click ‘OK’

Win7:

- Click the ‘START’ button
- Type ‘RUN’ in the search field
- Type ‘cmd’
- Click ‘OK’

2 At the command prompt type each of the following commands followed by pushing the <ENTER> key:

- ftpshell 192.168.1.1
- quote site attrib 11 DEOS

3 Push <CTRL>-<C> on the keyboard

4 At the command prompt type each of the following commands followed by pushing the <ENTER> key:

- ftpshell 192.168.33.1
- quote site attrib 11 DEOS

5 Push <CTRL>-<C> on the keyboard

j) On laptop computer open the CMC Remote Terminal Program.

1 Open shortcut folder MAINT from desktop

2 Open MAU icon

3 Double click MAU CMC Remote Terminal

Note The shortcut may be placed at a different location on the laptop.

k) Select DATA LOADER from the CMC MAIN MENU. Select DATA LOADER again from the DATA LOADER MENU.

Note A 'Connection Failed' message will be witnessed prior to loading the LDI. CMC MAIN MENU will not be available, and pressing DATA LOADER will advance to the DLS INSTALLATION FUNCTION. It is possible for a 'Connection Failed' message to be displayed with the LDI already loaded. If the connection has not been established within 2 minutes, check cables for correct connection. Check LED's on network converter. LED's flash when operating correctly. Ensure Ethernet to LAN converter has a power source.

l) Once PRE-LOADING DMU/CDROM DRIVES show's 'DONE', select FULL LOAD.

H. Loading File/Data:

1) Navigate to the data file to be loaded:

a) Arrow down and select 'AVAILABLE DRIVES'.

Note To navigate with the arrow keys, the mouse cursor must be moved over the selection window. The window is highlighted in green when active.

b) Open up applicable folders by pushing the <ENTER> button on the displayed folder.

c) Select the file which is prefixed by [DR-PN] for the current DATA to be loaded. To select, highlight file and press <ENTER>. Selected file will be highlighted in green.

Note Non - [DR-PN] files are not displayed. The program will respond with an error if the connection to the MAU is lost or if the file is not for the DHC-6 Series 400 Twin Otter.

d) Click 'SELECT FILE' in the menu. 'CONFIGURATION CHECK IN PROGRESS' will be displayed. Configuration check may take several minutes.

Note When loading the options file, the following additional steps are required before configuration check is performed:

1 When 'SELECT FILE' is selected, the Options File Licensing Agreement is displayed.

2 Review agreement using the arrow keys to scroll to bottom.

3 Highlight 'I ACCEPT' and press <ENTER> to select, when review is complete.

4 Select 'ENTER CODE' in the menu.

5 Enter code from CD into the AUTHORIZATION CODE window.

6 Select 'ENTER' in the menu. Confirm red password accepted message. 'CONFIGURATION CHECK IN PROGRESS' will be displayed.

e) Wait until 'CONFIGURATION CHECK COMPLETE' is displayed. Estimated load times will be displayed.

f) Click ‘START LOAD’ in the menu.

g) Wait until ‘LOADING SEQUENCE COMPLETE’ is displayed. The loading process can take several minutes depending on the file to be loaded. Refer to applicable chapter for file loading times.

Note If ‘Ping/FTP Communication Fail” error message is displayed during load, cycle aircraft power, close/reopen the CMC application on the maintenance laptop, and repeat load procedure. If message persists, the file may be corrupt. Obtain a new copy of the file, if able.

h) On the Remote Terminal, verify installation completed successfully on each listed card. For files affecting a large number of cards, it may be necessary to scroll down, using the arrow keys, to view the entire list.

Note If the installation failed on any card, the process must be repeated. The installation will detect the status of each card, and only repeat the process on required cards.

i) Click on ‘MAIN MENU’ button, if more data files are to be loaded. Insert applicable media CD or USB stick. Repeat Para 1.H.1).

I. Closing CMC Remote Terminal

1) Proceed by the following:

a) Close CMC Remote Terminal Program.

b) Disconnect adaptor LAN cable from the maintenance panel LAN BUS A.

c) Reconnect LAN BUS A terminator on the maintenance panel.

d) Remove aircraft power from APEX for a minimum of one minute.

e) Reapply power to APEX.

f) Verify no APEX errors are displayed following initialization.

g) Via the aircraft cockpit displays and controls, the APEX SCMS/DL page may be used to verify APEX software, configuration settings and databus status.

Note Refer to the chapters applicable to the data that was loaded for verification procedures.

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Apex Options and Settings File – Load Procedures

1. General

This procedure requires the APEX Settings File and Options File.

2. Description

The Settings File part number is VSP25–xxxxx. The 'xxxxx' is replaced by the applicable file number.

The Options File part number is TT60000053–yyyyy (or Viking cross reference part number VSP24–yyyyy) (Pre Mod 6/2303); or TT600002657–yyyyy (or Viking cross reference part number VSP30–yyyyy) (Post Mod 6/2303). The 'yyyyy' is replaced by the applicable file number.

Note The correct Settings File and correct Options File must be obtained for the applicable aircraft S/N. The SETTINGS FILE aircraft registration and transponder code is aircraft serial number specific. The OPTIONS FILE is aircraft serial number specific.

Note The OPTIONS FILE requires a security code to be entered. The security code is noted on the Options File Media. When prompted in the CMC remote terminal program, a Honeywell agreement has to be accepted and the security code has to be entered.

Note Typical load time: 5 minutes each.

3. Apex Settings and Options File Load Procedure

Verify VSP25 document provided with the settings file matches the desired aircraft configuration.

Insert the Settings File Media or Options File Media and follow the loading procedure as described in 45-00-00, Central Maintenance Computer – General data.

When loading is complete, verify as follows:

A. Settings File

1) Use the arrow keys on the MF Controller to navigate to the SYSTEMS PAGE. (Refer to Figure 1).

2) Push the PAGE button the MF Controller. (Refer to Figure 2).

3) Use the Joystick to select SCMS/DL and push ENT on the MF Controller. (Refer to Figure 3).

Note There are multiple pages available under the Sys Config tab. (Refer to Figure 4). To Change pages, highlight Page Down or Page Up and press ENT on the MF Controller.

4) Scroll through the pages to view settings which are part of the Settings File. Verify the values displayed on Apex match what is listed in the VSP25–xxxxx document provided with the settings file. (Refer to Figure 4 and Figure 5).

The following is a sample only of items included in the VSP25–xxxxx document:

SETTINGS FILE ITEM	TYPICAL VALUE	NOTES
n/a	2049	Standard (STD)
n/a	SETT	STD
Aircraft Registration Number	C–GXXX	Aircraft Registration Specific
Aircraft Serial Number	879	Aircraft Serial Number Specific
Aural Warning Voice Type	1	0=Male, 1=Female (Default)

SETTINGS FILE ITEM	TYPICAL VALUE	NOTES
Aircraft S/N Digit Thousands	0x00	Aircraft Serial Number Specific
Aircraft S/N Digit Hundreds	0x08	Aircraft Serial Number Specific
Aircraft S/N Tens	0x07	Aircraft Serial Number Specific
Aircraft S/N Ones	0x09	Aircraft Serial Number Specific
Aircraft Transponder Code	C12ABC	Aircraft Registration Specific
AC Symbol In Air Charts Display	1	0=Disabled, 1=Enabled (Default)
Engine 1 Serial Number	PCERB0123	Example of Engine Serial Number
Engine 2 Serial Number	PCERB0124	Example of Engine Serial Number
Env. Temperature Units	0	0=DEG C (Default), 1=DEG F
Lightning Antenna Location	0	STD
ADF1 Quad Error	6	STD, the Quadrantal Error Correction number displayed shall be rounded to the nearest whole number
ADF2 Quad Error	6	'0' if S.O.O. 6217 is not installed, '6' if S.O.O. 6217 is installed, the Quadrantal Error Correction displayed is exact to 6 digits behind the decimal
COM1 Carrier Squelch	95	STD
COM1 Emerg Freq-100 MHz Digit	1	STD
COM1 Emerg Freq-10 MHz Digit	2	STD
COM1 Emerg Freq-1 MHz Digit	1	STD
COM1 Emerg Freq-0.1 MHz Digit	5	STD
COM1 Emerg Freq-0.01 MHz Digit	0	STD
COM1 Emerg Freq-0.001 MHz Digit	0	STD
COM1 Emergency Volume	20	STD
COM1 Sidetone Level	52	STD
COM1 Squelch	8	STD
COM1 Microphone Gain	21	STD
COM2 Squelch	95	STD
COM2 Emerg Freq-100 MHz Digit	1	STD
COM2 Emerg Freq-10 MHz Digit	2	STD
COM2 Emerg Freq-1 MHz Digit	1	STD
COM2 Emerg Freq-0.1 MHz Digit	5	STD
COM2 Emerg Freq-0.01 MHz Digit	0	STD
COM2 Emerg Freq-0.001 MHz Digit	0	STD
COM2 Emergency Volume	20	STD
COM2 Sidetone Level	52	STD

SETTINGS FILE ITEM	TYPICAL VALUE	NOTES
COM2 Squelch	8	STD
COM2 Microphone Gain	21	STD
Video Installed	0	STD
Weight Unit For Display	0	0=Pounds (Default), 1=Kilograms
n/a	0	not displayed
Binary File Embedded CRC	0xe14fe435	not displayed, example only
Full Bin CRC For HR	0x7bce34a2	not displayed, example only

5) When the Registration of the Aircraft has been changed, refer to manufacturer's procedure in Section 25-60-11, Emergency Locator Transmitter (ELT) – General Data).

B. Options File

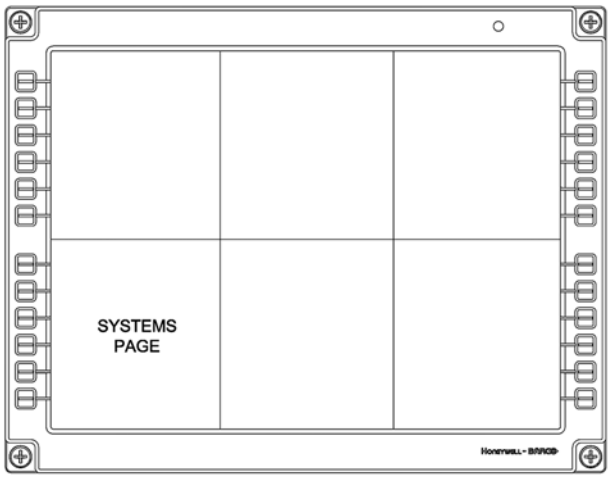
1) Use the arrow keys on the MF Controller to navigate to the SYSTEMS PAGE. (Refer to Figure 1).

2) Push the PAGE button on the MF Controller. (Refer to Figure 2).

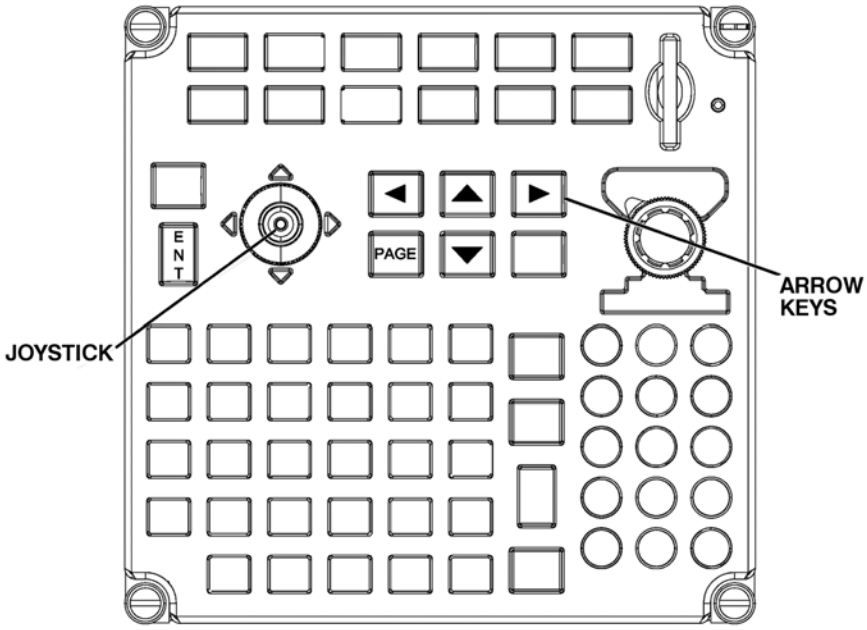
3) Use the Joystick to select SCMS/DL and push ENTER on the MF Controller. (Refer to Figure 3).

Note There are multiple pages available under the Sys Config tab. To change pages, highlight Page Down or Page Up and press ENT on the MF Controller.

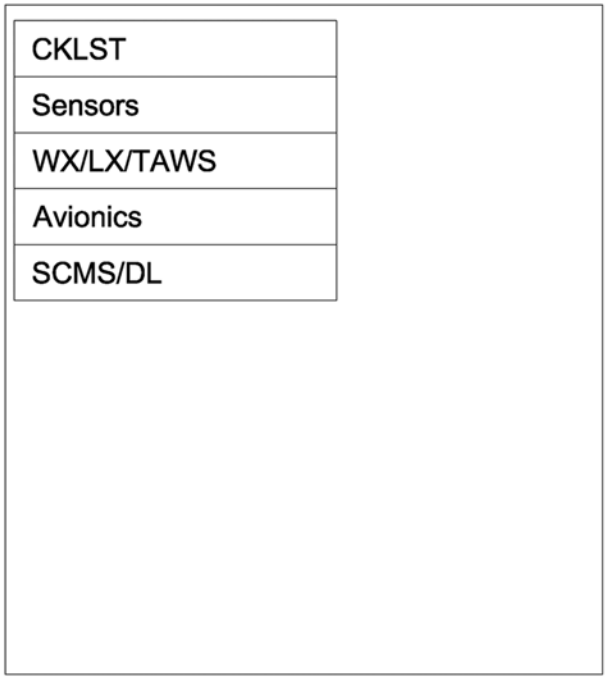
4) On Sys Config tab Page 1, view the part number listed under Configuration Part Number. Verify that the correct part number is displayed. (Refer to Figure 6).



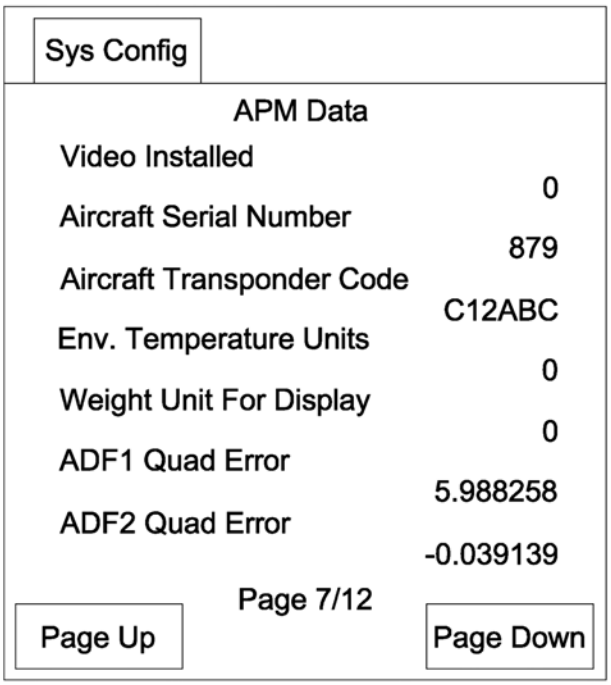
MFD (Lower Typical) – SYSTEMS PAGE Location
Figure 1



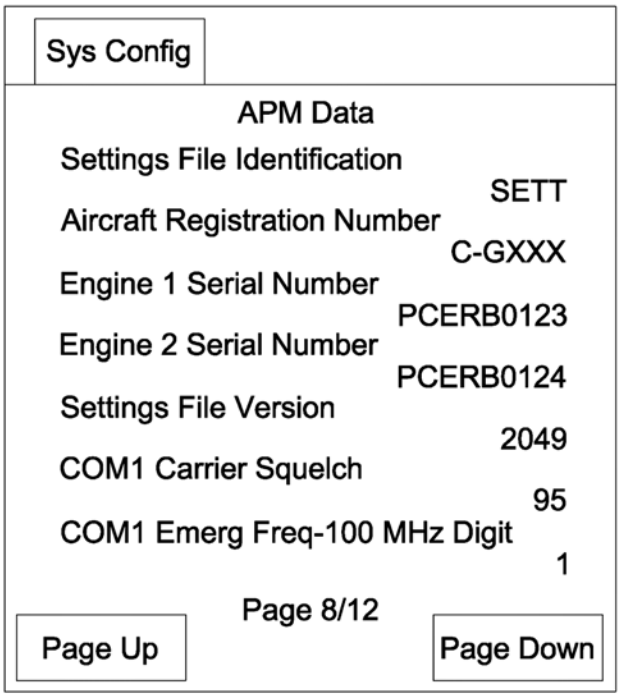
MF Controller
Figure 2



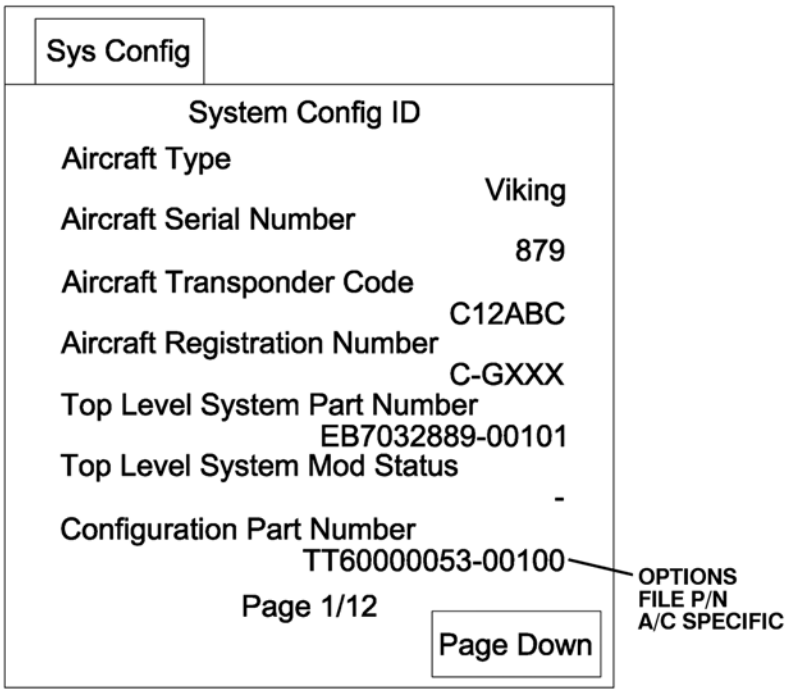
SYSTEMS PAGE Menu
Figure 3



Systems Configuration Tab – Various Settings File Values (Page 7 Shown as Example)
Figure 4



Systems Configuration Tab – Various Settings File Values (Page 8 Shown As Example)
Figure 5



Systems Configuration Tab (Page 1 Shown)
Figure 6

Apex Modular Avionics Unit Software – Load Procedures

1. General

This procedure requires the APEX MAU software. Media part number is MM60000469–xxx (Pre Mod 6/2303); or MM60002735–xxx (Post Mod 6/2303). The 'xxx' represent the latest approved version.

2. APEX Software Load Procedure

Insert the APEX MAU Software Media and follow the loading procedure as described in Chapter 45-00-00, Central Maintenance Computer – General data.

Note Typical load time: 5 minutes.

When loading is complete, verify as follows:

A. To verify the APEX MAU Software Part Number:

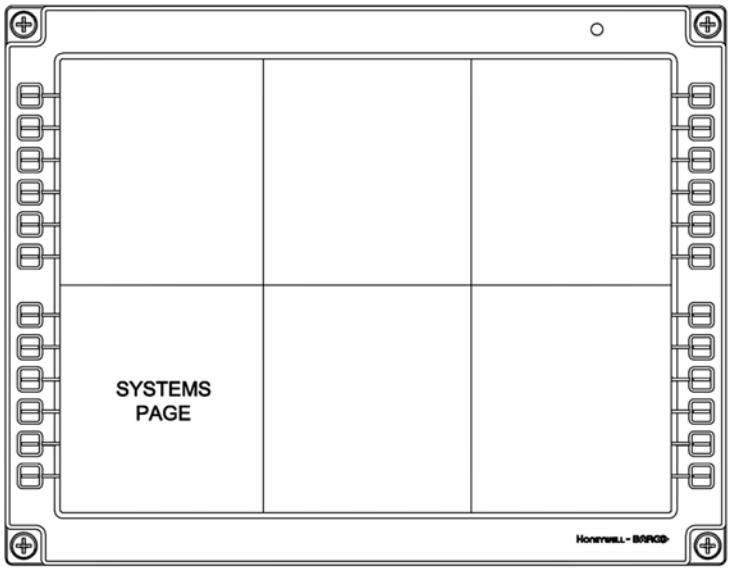
1) Use the arrow keys on the MF Controller to navigate to the SYSTEMS PAGE. (Refer to Figure 1).

2) Push the PAGE button on the MF Controller. (Refer to Figure 2).

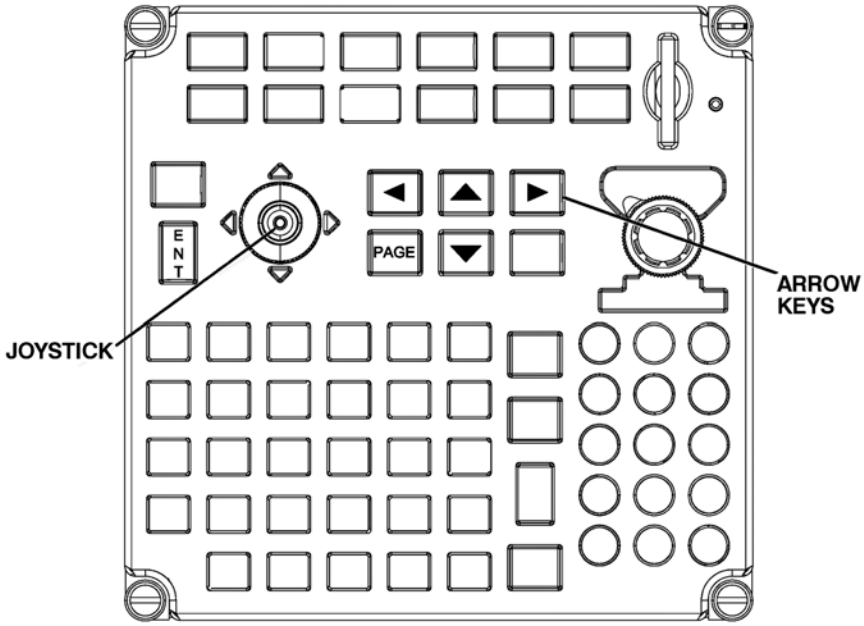
3) Use the Joystick to select SCMS/DL and push ENT on the MF Controller. (Refer to Figure 3).

Note There are multiple pages available under the Sys Config tab. To change pages, highlight Page Down or Page Up and press ENT on the MF Controller.

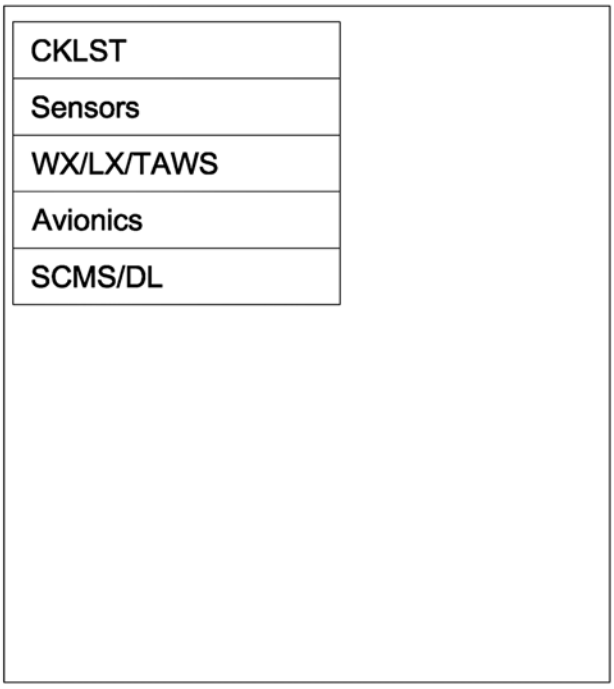
4) On Sys Config tab Page 1, view the part number listed under Top Level System Part Number. (Refer to Figure 4). Verify that the correct part number is displayed. The top level part number is noted on the APEX MAU Software Media.



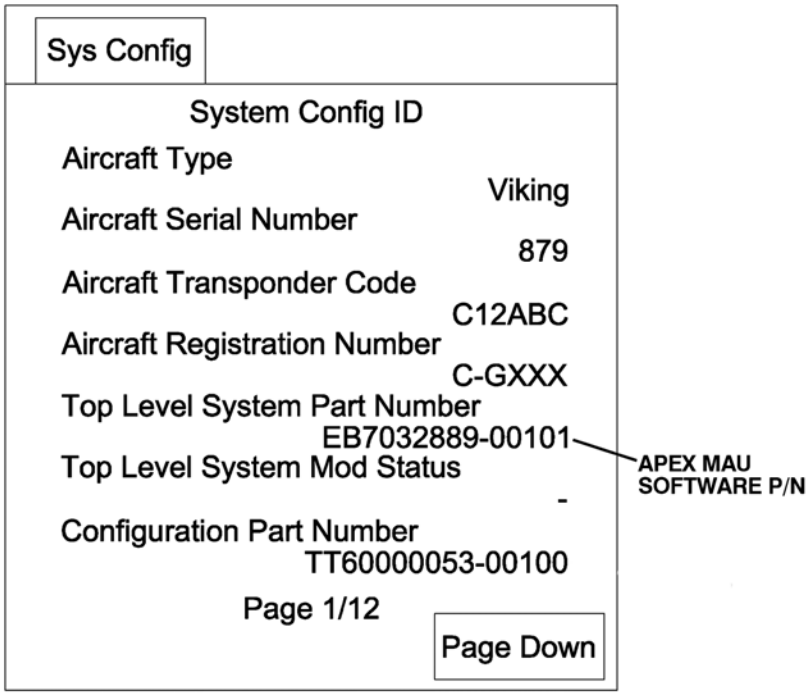
MFD (Lower Typical) – SYSTEMS PAGE Location
Figure 1



MF Controller
Figure 2



SYSTEMS PAGE Menu
Figure 3



Systems Configuration Tab (Page 1 Shown)
Figure 4

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Apex FMS Aircraft Performance Database Files – Load Procedures

1. General

This procedure requires the APEX FMS Aircraft Performance Database File. Media part number is MM60000461–xxx. The 'xxx' represent the latest approved version.

2. FMS Load Procedure

Insert the APEX FMS Aircraft Performance Database File Media and follow the loading procedure as described in 45-00-00, Central Maintenance Computer – General data.

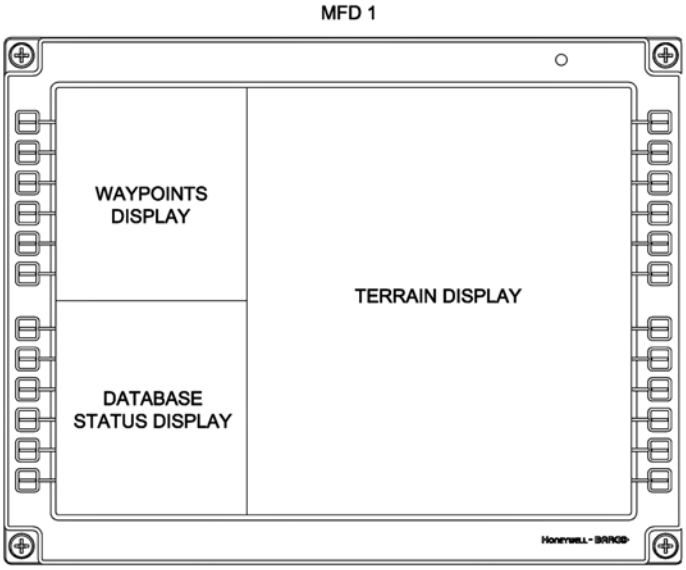
Note Typical load time: 5 minutes.

When loading is complete, verify as follows:

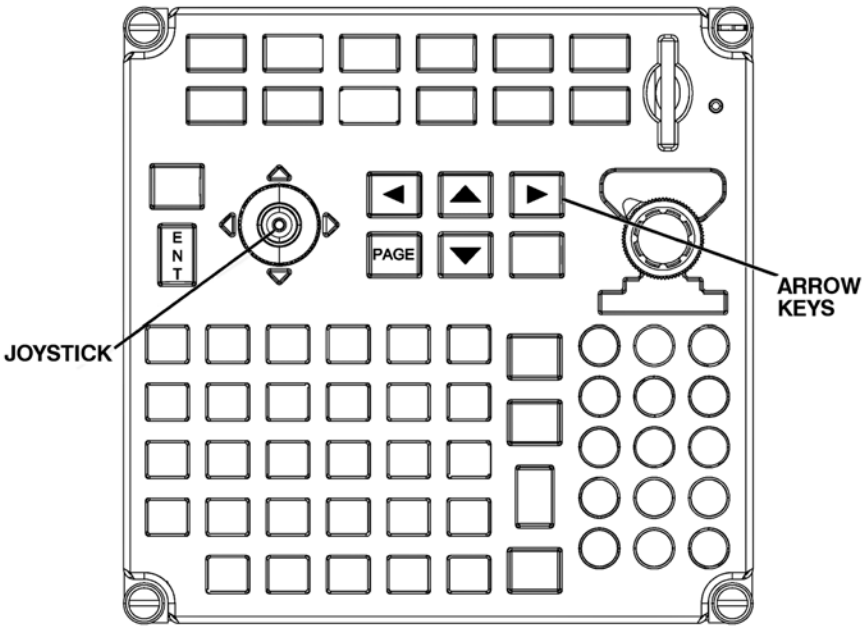
A. Use the arrow keys on the MF Controller to navigate to the DATABASE STATUS DISPLAY. (Refer to Figure 1).

B. Using the Joystick and ENT button on the MF Controller, select the Data Bases tab.

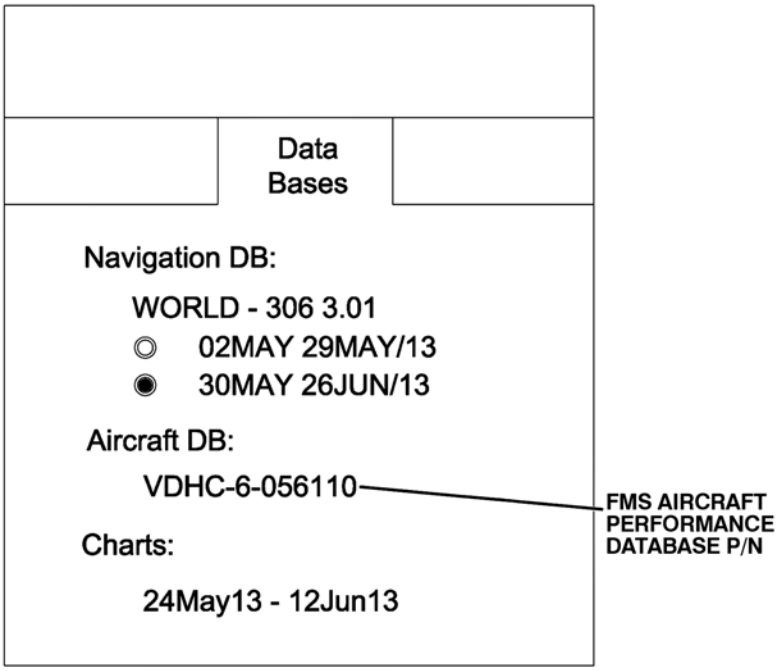
C. Under Aircraft DB, verify the correct part number VDHC–6–xxxxxx is displayed. (Refer to Figure 3).



MFD (Upper Typical)
Figure 1



MF Controller
Figure 2



Database Status Display – 'Databases Tab'
Figure 3

Apex Database Files – Load Procedures

1. General

This procedure requires the APEX Database Files from epicinds.com.

Note A subscription is required.

2. Description

A. The APEX System for the DHC6-400 Twin Otter has four databases that can be loaded:

1) TERRAIN Database (Green CD or downloaded (approximately 550MB))

2) NAV Database – Transportation and Cities (Purple CD or downloaded (approximately 30MB))

3) NAV Database – World Database (Purple CD or downloaded (file located with same media as step 2))

4) CHARTS Database (Blue CD or downloaded (approximately 500MB))

B. Typical loading times are as follows:

1) TERRAIN – 30 minutes

2) NAV – 10 minutes

3) CHARTS – 45 minutes (based on full world coverage)

3. APEX Database Load Procedure

Insert the TERRAIN, NAV, or CHARTS Database Media and follow the loading procedure as described in 45-00-00, Central Maintenance Computer – General data.

Note The NAV database (purple CD) has two separate files to be loaded. For each of these files there is an extra file of the same name which is not compatible with the DHC-6 Series 400 Twin Otter. Selecting this file will produce a 'SYSTEM SOFTWARE INCOMPATIBLE' message. If this message is witnessed, select the alternate file.

When loading is complete, verify as follows:

A. TERRAIN:

1) Use the arrow keys on the MF Controller to navigate to the SYSTEMS PAGE. (Refer to Figure 1).

2) Push the PAGE button on the MF Controller. (Refer to Figure 2).

3) Use the Joystick to select SCMS/DL and push ENT on the MF Controller. (Refer to Figure 3).

Note There are multiple pages available under the Sys Config tab. To change pages, highlight Page Down or Page Up and press ENT on the MF Controller.

4) Scroll to the page displaying the AGM 1 and AGM 2 Terrain Server on the Sys Config tab. (Refer to Figure 4).

5) Verify that AGM 1 and AGM 2 TERRAIN SERVER shows the correct part number (the last seven digits of the part number downloaded or received on the CD, omitting the dash, is the number which will be displayed here).

B. NAV:

1) Use the arrow keys on the MF Controller to navigate to the DATABASE STATUS DISPLAY. (Refer to Figure 5).

2) Using the Joystick and ENT button on the MF Controller, select the Data Bases tab. (Refer to Figure 2 and Figure 6).

3) Under Navigation DB the effective dates of the currently loaded databases are displayed.

Note

- It may take up to a minute to display valid data on this page. (Refer to Figure 7).
- The current database is displayed in white. An expired database is displayed in yellow.
- The APEX system automatically selects the current database. If both databases are expired, the APEX system selects the newest of the two.

4) Verify that the current database is loaded.

C. CHARTS:

Note The display of Jeppesen charts is a Standard Option Order obtained from OEM and requires a substantiation to be activated from Jeppesen.

1) Use the arrow keys on the MF Controller to navigate to the DATABASE STATUS DISPLAY. (Refer to Figure 5).

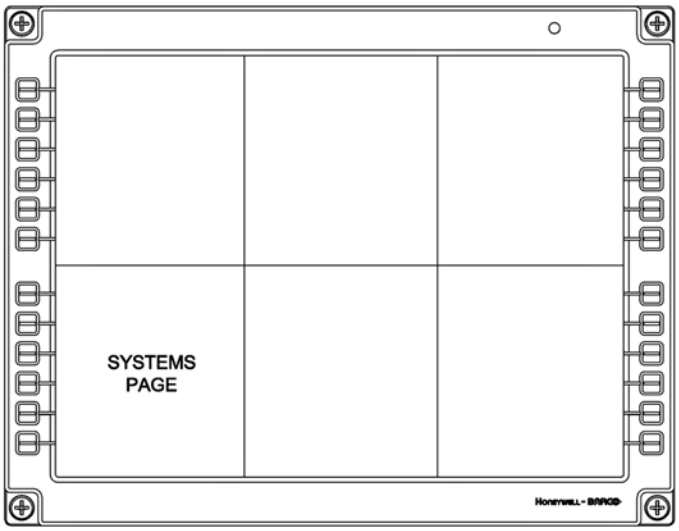
2) Using the cursor on the MF Controller, select the Data Bases tab. (Refer to Figure 2 and Figure 6).

3) Under Charts the effective date of the currently loaded database is displayed in green. (Refer to Figure 6).

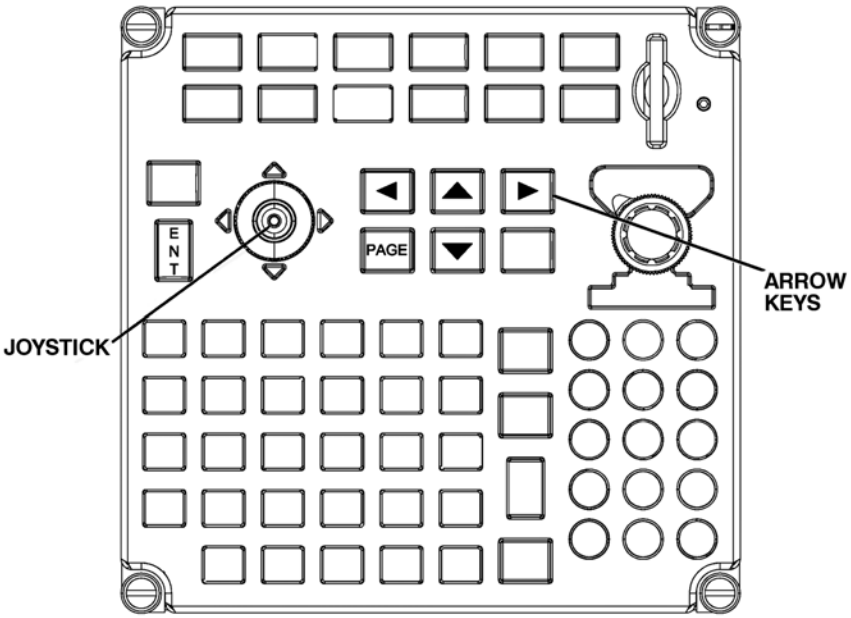
Note

- It may take up to a minute to display valid data on this page. (refer to Figure 7).
- If this option has not been activated, the display under Charts will display dashes.
- If the charts have not been loaded correctly but are activated, there will be no valid date displayed under Charts.

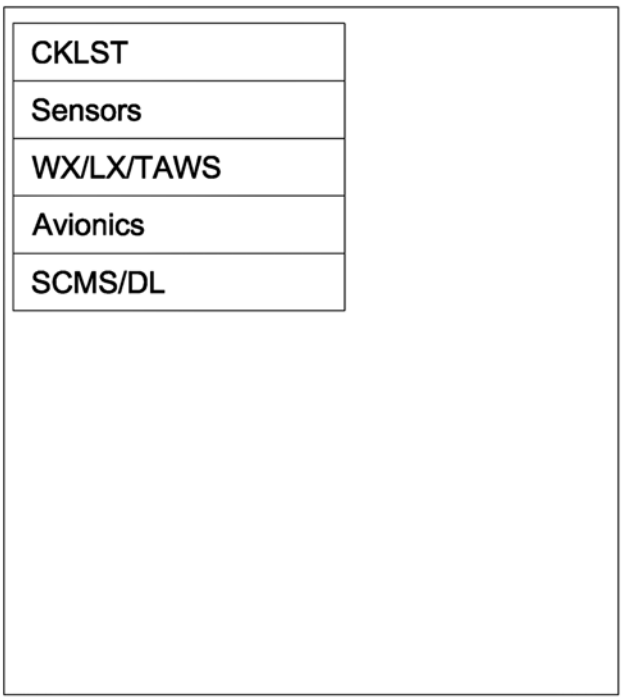
4) Verify that the date displayed is current.



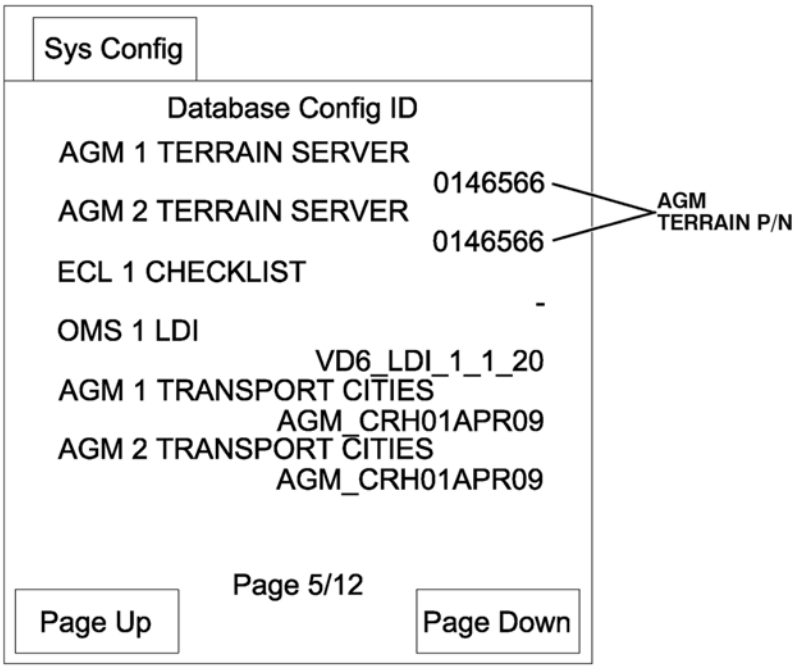
MFD (Lower Typical) – SYSTEMS PAGE Location
Figure 1



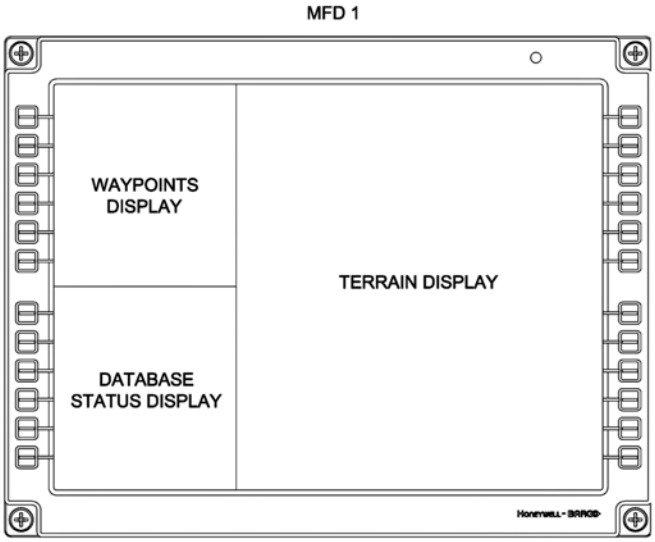
MF Controller
Figure 2



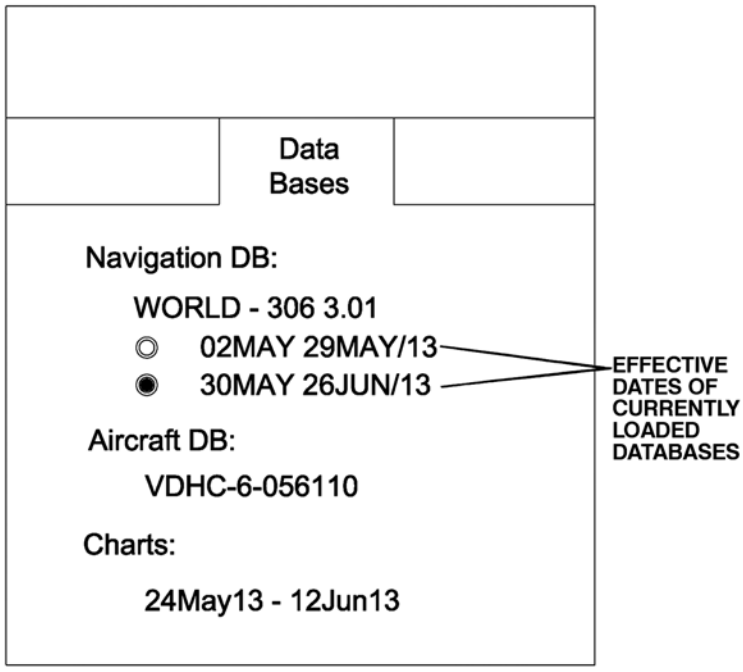
SYSTEMS PAGE Menu
Figure 3



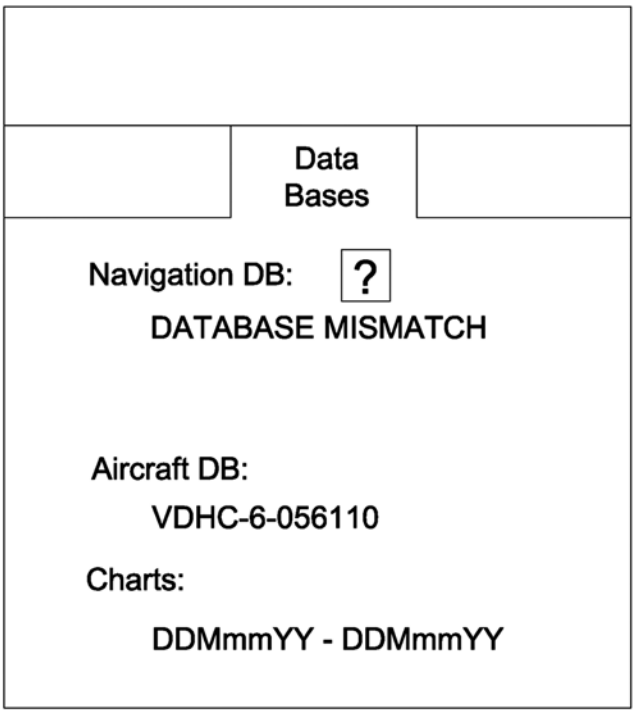
Systems Configuration Tab – (Page 5 Shown)
Figure 4



MFD (Upper Typical)
Figure 5



Databases Tab on Database Status Display
Figure 6



Databases Tab on Database Status Display
Figure 7

Loadable Diagnostic Interface File – Load Procedures

1. General

This procedure requires the Loadable Diagnostic Interface File (LDI). Media part number is VD6–LDI_1_1_xx (Pre Mod 6/2303); or VD6–LDI_4_1_4_xx (Post Mod 6/2303). The 'xx' represents the latest approved version.

2. LDI File Load Procedure

Insert the LDI File Media and follow the loading procedure as described in 45-00-00, Central Maintenance Computer – General data.

Note Typical load time: 5 minutes.

When loading is complete, verify as follows:

A. Use the arrow keys on the MF Controller to navigate to the SYSTEMS PAGE. (Refer to Figure 1).

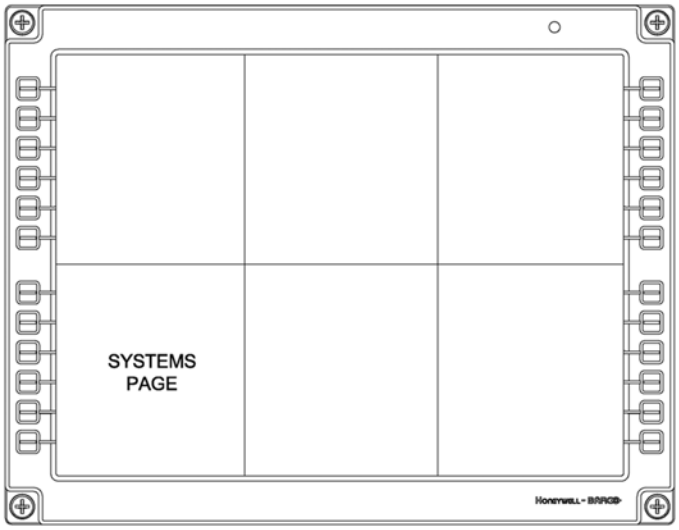
B. Push the PAGE button on the MF Controller. (Refer to Figure 2).

C. Use the Joystick to select SCMS/DL and push ENT on the MF Controller. (Refer to Figure 3).

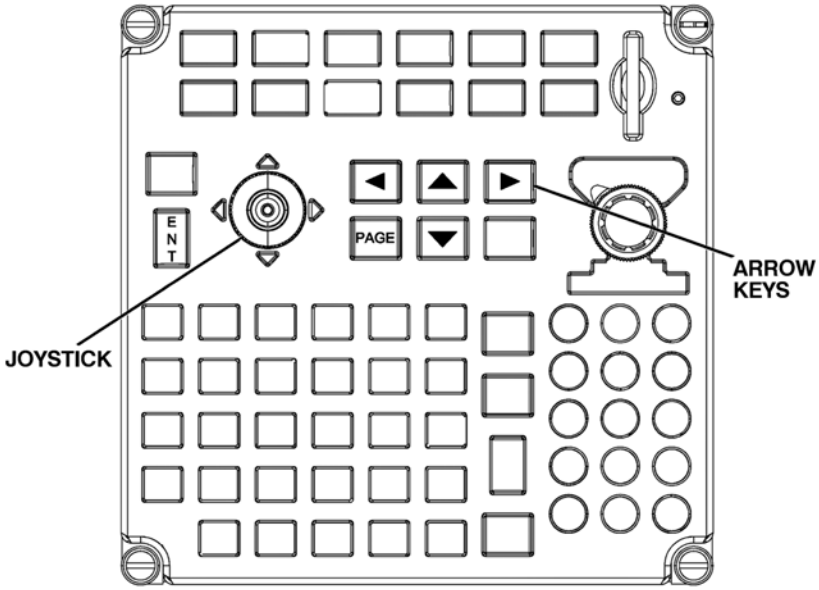
Note There are multiple pages available under the Sys Config tab. To change pages, highlight Page Down or Page Up and press ENT on the MF Controller.

D. Scroll to the page displaying the OMS 1 LDI on the Sys Config tab. (Refer to Figure 4).

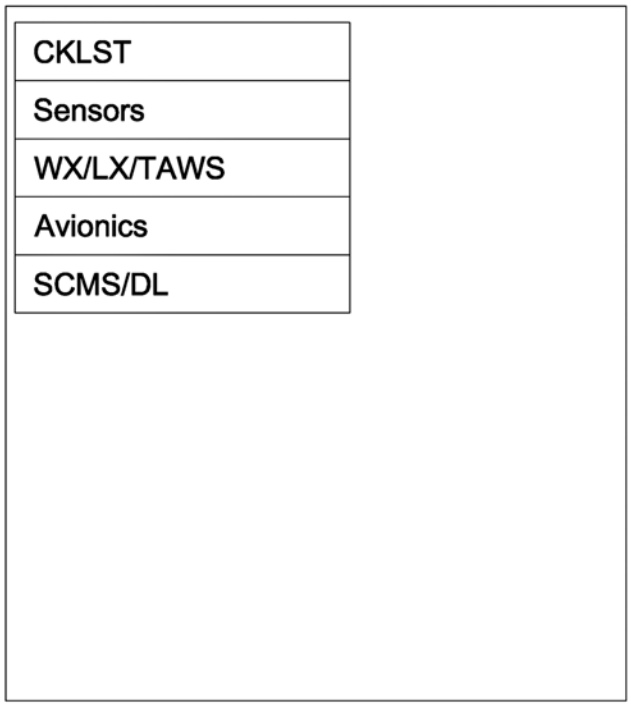
E. Verify that OMS 1 LDI shows the correct part number. (Refer to Figure 4).



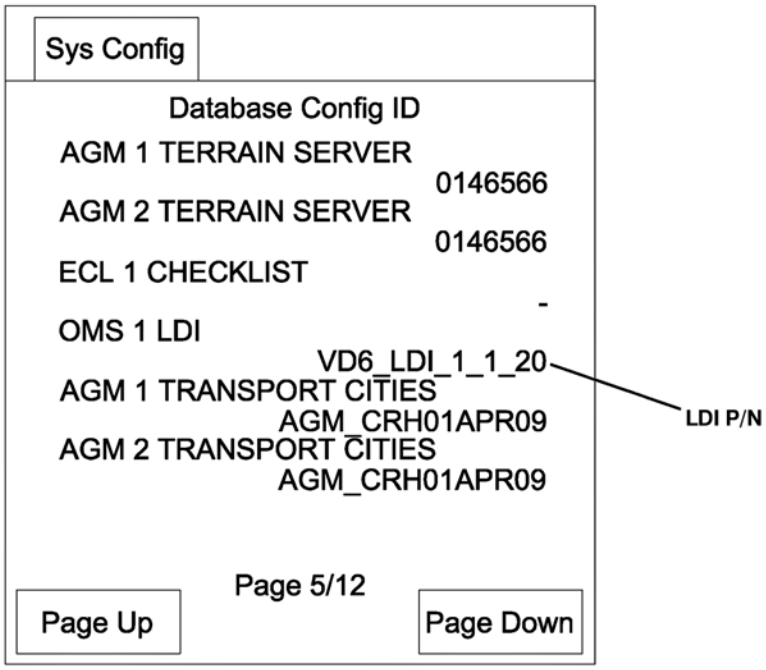
MFD (Lower Typical) – SYSTEMS PAGE Location
Figure 1



MF Controller
Figure 2



SYSTEMS PAGE Menu
Figure 3



Systems Configuration Tab – (Page 5 Shown)
Figure 4

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CHAPTER 46

INFORMATION SYSTEMS

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CHAPTER 46 – INFORMATION SYSTEMS

LIST OF EFFECTIVE PAGES

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	* 3	Sep. 28/18		* 5	Sep. 28/18
	* 4	Blank Page		* 6	Blank Page
	* 5	Sep. 28/18	46-50-00	* 1	Sep. 28/18
	* 6	Blank Page		* 2	Blank Page
46-00-00	* 1	Sep. 28/18	46-50-01	* 1	Sep. 28/18
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	* 2	Sep. 28/18		* 4	Sep. 28/18
46-20-01	* 1	Sep. 28/18		* 5	Sep. 28/18
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CHAPTER 46 – INFORMATION SYSTEMS

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Information Systems – General Data

1. General

Provisions have been made for three standard checklist procedures to be displayed through CAS. The standard checklists are for Normal procedures, Abnormal procedures and Emergency procedures. Refer to Pilot Operating Handbook and/or Aircraft Flight Manuals, PSM 1-64-1A/POH for details.

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Checklist – General Data

1. General

Provisions have been made for three standard checklist procedures to be displayed through CAS. The standard checklists are for Normal procedures, Abnormal procedures and Emergency procedures.

2. Description

The checklists (if installed) are displayed on the lower Multi-Function Display in the Electronic checklist window. (Refer to 31-10-00, Instrument and Control Panels – General Data).

3. Operation

The checklists are displayed in the Electronic checklist window and this window becomes interactive, with predefined soft keys down the port side of the window. There are corresponding bezel keys on the display unit for each of the soft keys. The multi-function controller keys allow navigation through the checklist menus, programming within the window and display format.

The predefined keys are:

RESET ALL – this key resets the ALL checklist procedure displayed by removing all checkmarks

RESET – this key resets the current checklist procedure displayed by removing all checkmarks

PREV ITEM – this key moves the cursor up the checklist screen one line at a time

NEXT ITEM – this key moves the cursor down the checklist screen one line at a time

PAGE UP – this key displays the previous page of the current checklist

PAGE DOWN – this displays the next page of the current checklist

ENT– this key selects the checklist category, and when selected in a checklist checks off the checklist line item

After all line items in a checklist are marked as complete the cursor is placed in the CHECKLIST COMPLETE selection. ENT is selected. Note that if all line items in the checklist are not complete the cursor will default to the first incomplete item.

4. Creating Checklist

To create a checklist, use the Electronic Checklist Designer software provided. This software is a ground base application which stored the checklist in a checklist database file.

This application is comprised of two components: Checklist Database development component and a checklist database verification component. The Electronic Checklist Designer allows the user to interface with the aircraft sub-database which provides the number of lines within the checklist, define the number of characters per line and assists the user in the development of the checklist procedures.

The Checklist database verification converts the checklist database file form a binary file to a text file. The text file is reviewed and approved by the appropriate authority. The generated checklist file once reviewed and approved is loaded on to the module in the Modular Avionics Unit.

5. Troubleshooting

In the event the checklist fails a checklist failure message is displayed on the Electronic checklist window.

Failure Message	Function
Checklist Unavailable	Checklist not installed, Checklist function has failed
ERROR (ECL): DB FILE OPEN	Copy of the database file is not loaded in the processor.
ERROR (ECL): DB FILE READ	Copy of the database file is loaded but processor is unable to read it.
ERROR (ECL): DB FILE INTEGRITY	The electronic checklist cyclic redundancy checks (CRC) and the database file CRC do not match.
ERROR (ECL): DB NOT APPROVED	The loaded database is not approved and the checklist is locked out until a cold start is done.

Apex Electronic Checklist File – Load Procedures

1. General

This document outlines the procedure for loading the APEX Electronic Checklist File.

2. Description

A. This procedure requires the following:

1) Laptop Computer with PCMCIA Card and 10BASE–T Network Adapter with LAN interface cable, or equivalent.

2) Proprietary Honeywell CMC Remote Terminal Software P/N PS7033676_001 V 5.0.2 or later applicable version.

3) APEX Electronic Checklist file as applicable. This file is prepared and edited using the Honeywell Electronic Checklist Designer software.

Refer to the Pilot Operating Handbook and/or Aircraft Flight Manual for details.

Note This procedure requires the COM 1 port of the laptop computer to be available. If problems are encountered disable other applications that may be using the COM 1 port.

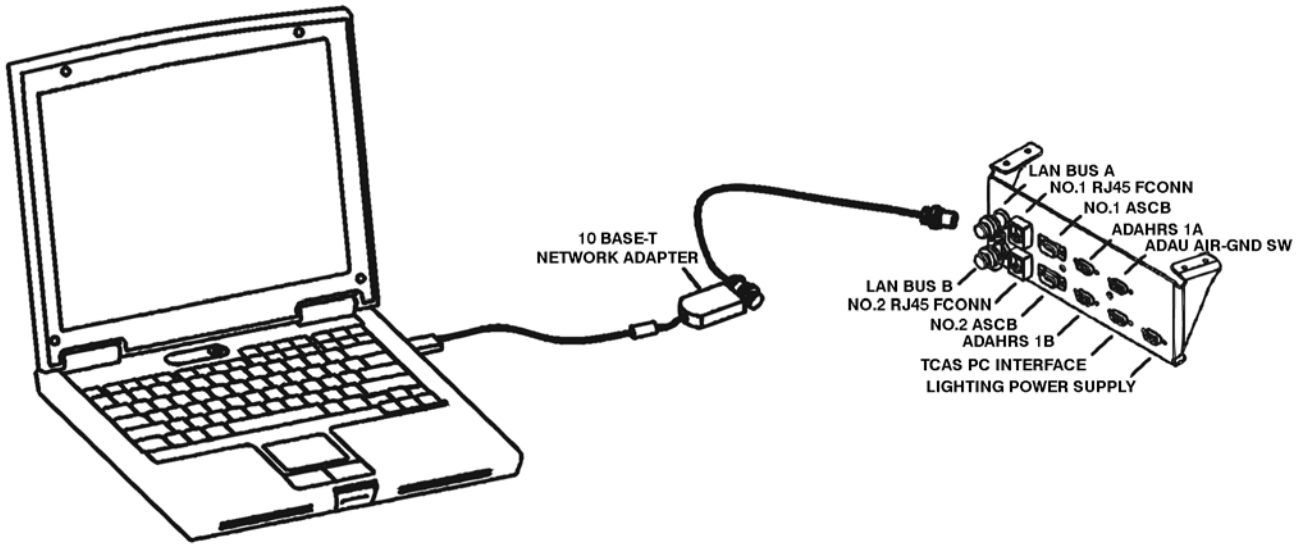
3. Apex Electronic Checklist Load Procedure

A. The procedure and photos detailed below are specific to loading the APEX Electronic Checklist.

1) Gain access to the Avionics Maintenance Panel in the forward avionics area.

2) Power up APEX IAW AFM procedures.

3) Plug 10BASE–T Network Adapter into laptop PCMCIA card.



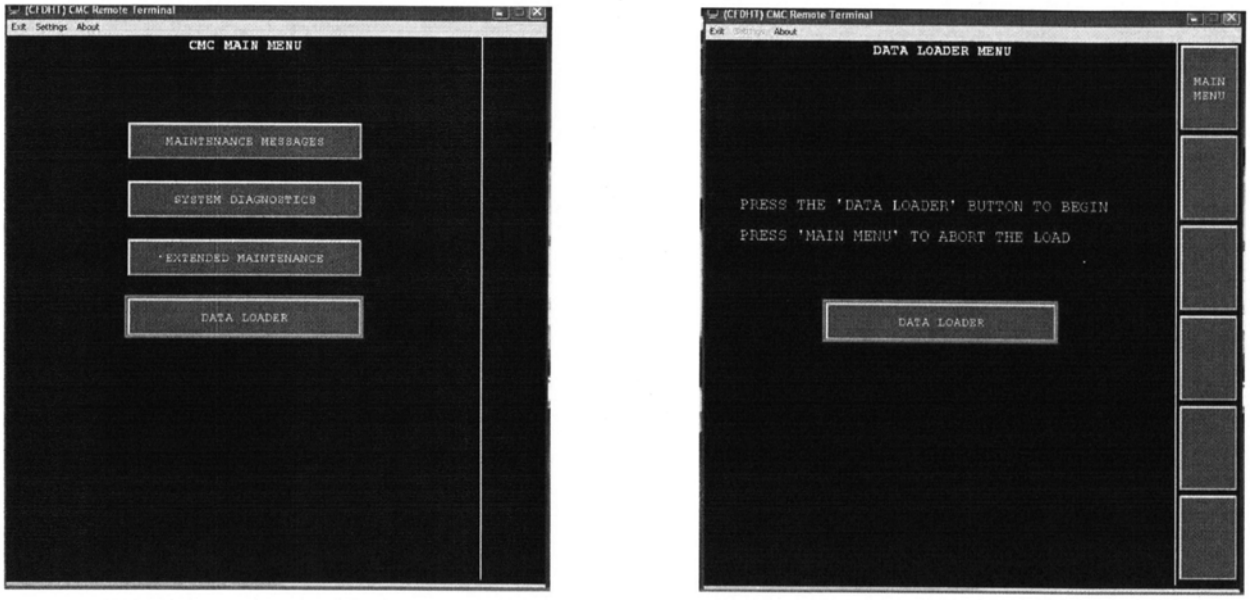
Forward Maintenance Panel
Figure 1

4) Plug the other end of Network Adapter into either LAN BUS A or LAN BUS B connector on the Forward Maintenance Panel.

   Note Either LAN connector will work.

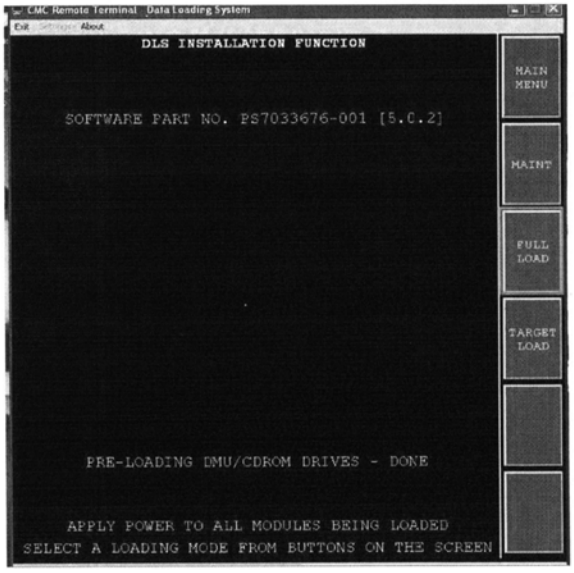
5) On laptop computer open the CMC Remote Terminal Program.

6) Select DATA LOADER twice. A “Connection Failed” message may be displayed after the first selection of DATA LOADER.



CMC Main Menu
Figure 2

7) The program will search the laptop computer for available drives. PRE-LOADING DRIVES will be displayed. This process may take up to 2 minutes. PRE-LOADING DMU/CDROM DRIVES – DONE will be displayed upon completion of this process.



DLS Installation Function
Figure 3

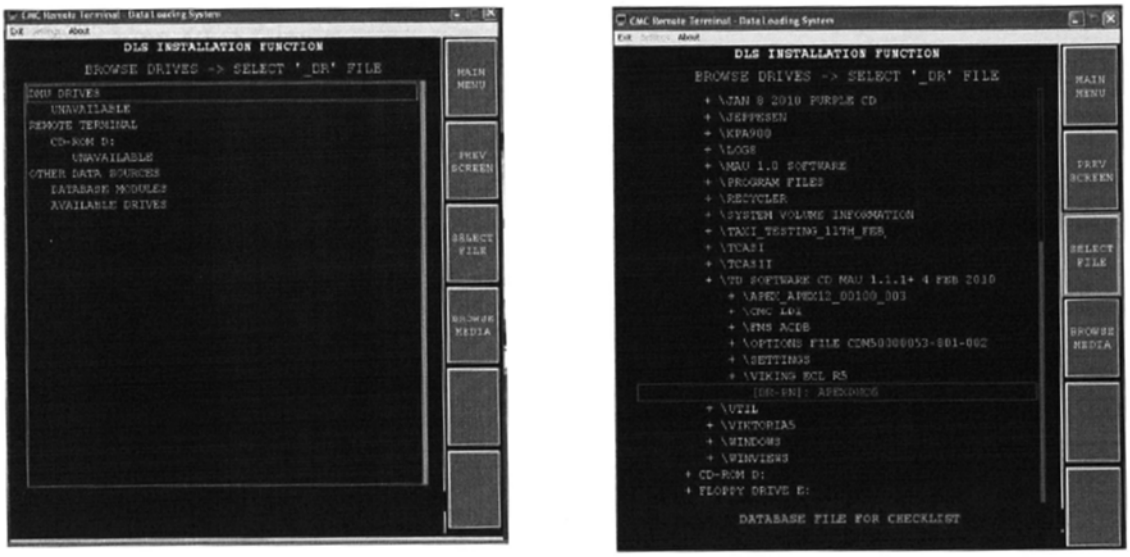
8) Select “FULL LOAD” from menu on RH side of program window.

9) Arrow down and select “AVAILABLE DRIVES” <ENTER>.

10) Arrow down and select desired Database File <ENTER>.

Note The file will be prefixed with [DR].

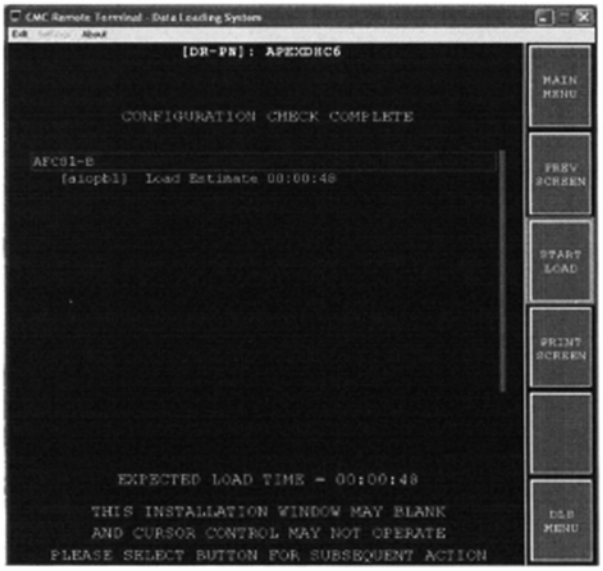
Note Figure 4 is a typical example only.



DLS Installation Function
Figure 4

11) Select SELECT FILE from menu on RH side of program window. “CONFIGURATION CHECK IN PROGRESS” will be displayed. Wait until complete. This may take up to two minutes. When complete “CONFIGURATION CHECK COMPLETE” will be displayed.

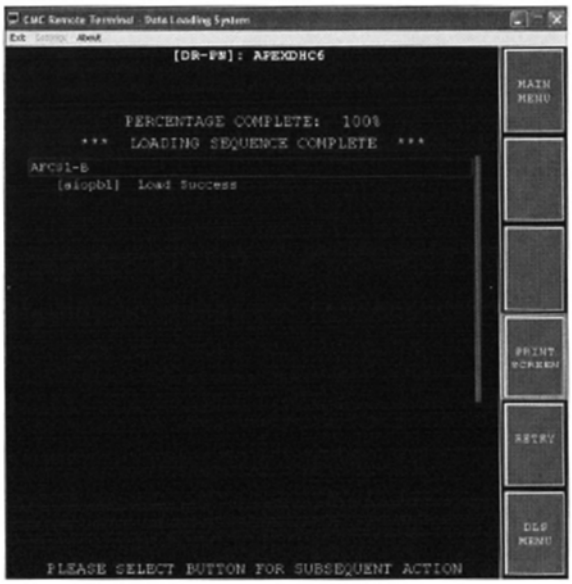
Note Figure 5 is a typical example only.



Configuration Complete
Figure 5

12) Select "START LOAD" from menu on RH side of program window. Wait until process is complete. The expected load time will be displayed. Upon process completion “LOADING SEQUENCE COMPLETE” will be displayed.

Note Figure 6 is a typical example only.



Loading Sequence Complete
Figure 6

13) Reserved.

14) Reserved.

15) Close CMC Remote Terminal Program. Disconnect laptop computer from the maintenance panel.

16) Remove aircraft power from APEX for minimum one minute.

17) Reapply power to APEX.

18) Verify no APEX errors are displayed following initialization. View the APEX Electronic Checklist and verify validity.

19) Via the aircraft cockpit displays and controls, the APEX SCMS/DL page may be used to verify APEX software, config settings and database status.

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Miscellaneous Information Systems – General data

1. General

Miscellaneous Information Systems are available by customer option order only, that provides on board information that support the pilots. The information systems are not related to flight deck, passenger cabin or maintenance functions.

Provisions are made for S.O.O. 6295 Float Mounted Depth Sounder. The depth sounder provides a water depth measurement when the aircraft is operating on Wipaire 13000 amphibious floats or Wipaire 13000 straight floats. The depth sounder is not to be used during flight.

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Float Mounted Depth Sounder (S.O.O. 6295) – General data

1. General

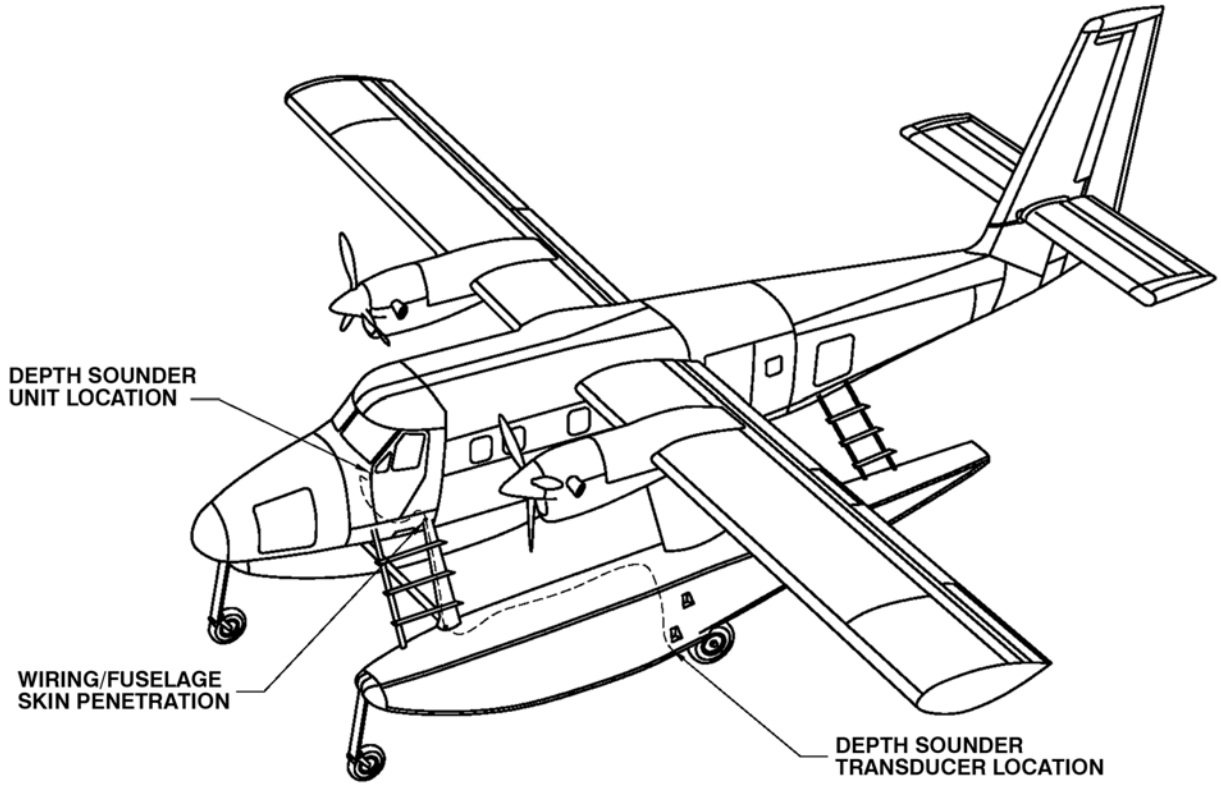
Provisions are made for S.O.O. 6295 Float Mounted Depth Sounder. The depth sounder provides a water depth measurement when the aircraft is operating on Wipaire 13000 amphibious floats or Wipaire 13000 straight floats. The depth sounder is not to be used during flight.

Refer to manufacturer's information regarding maintenance, servicing and repair of Depth Sounder.

2. Description

An illuminated Humminbird depth sounder is located on the pilot's sub panel to the right of the switch panel. The depth sounder provides digital depth reading in feet, meters or fathoms. The depth sounder may also be configured for a shallow water and deep water alarm. Power to the depth sounder is supplied by the 14V power converter through a 1 AMP circuit breaker, connected to the 'line side' of the 14V OUTPUT circuit breaker, located on the avionics circuit breaker panel.

Note Indicated depth measurement depends on bottom hardness, slope and water conditions. Typically, the depth sounder will indicate deeper depths in fresh water than salt water.



Depth Sounder Locations
Figure 1

3. Adjustment/Test

A. Set Measurement Units

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Select DEPTH SOUNDER switch ON.

3) Press the MENU key until the units icon is displayed and flashing.

4) Press the UP or DOWN key to scroll through available units:

- Ft: Feet
- M: Meters
- FA: Fathoms

5) After selection, wait 5 seconds for the unit to return to normal operation. Settings are saved even after depth sounder is powered off.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

B. Set Shallow Alarm

The Shallow Alarm will sound when the measured water depth is less than the saved setting. If the depth measurement of the water is less than the saved setting, the aural alarm will sound and the alarm icon will flash on the screen. Press any key on the depth sounder unit to mute the aural alarm.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Select DEPTH SOUNDER switch ON.

3) Press MENU key until the SHALLOW alarm icon is displayed and flashing.

4) Press the UP and DOWN arrow key to change to the desired alarm setting. Default setting is OFF.

5) After selection, wait 5 seconds for the unit to return to normal operation. SHALLOW will be displayed. Settings are saved even after depth sounder is powered off.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

C. Set Deep Alarm

The Deep Alarm will sound when the measured water depth is greater than the saved setting. If the depth measurement of the water is less than the saved setting, the aural alarm will sound and the alarm icon will flash on the screen. Press any key on the depth sounder unit to mute the aural alarm.

1) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Select DEPTH SOUNDER switch ON.

3) Press MENU key until the DEEP alarm icon is displayed and flashing.

4) Press the UP or DOWN arrow key to change to the desired alarm setting. Default setting is OFF.

Note The minimum Deep Alarm setting cannot equal or drop below the saved Shallow Alarm setting.

5) After selection, wait 5 seconds for the unit to return to normal operation. The DEEP will be displayed. Settings are saved even after depth sounder is powered off.

6) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

D. Set Offset

1) Keel

The Offset function adjusts the digital readout to display reading from the keel (in lieu of the transducer location). Measure the vertical distance between the face of the transducer and keel of the float.

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

b) Select DEPTH SOUNDER switch ON.

c) Press the MENU key until SET OFF SET is displayed and flashing on the screen.

d) Press the DOWN arrow key to enter the measurement as a negative (–) number.

e) After the offset is entered, wait 5 seconds for the unit to return to normal operation. An offset icon will be visible on the display. Settings are saved even after the depth sounder is powered off.

f) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

2) Waterline

The Offset function adjusts the digital readout to display reading from the waterline (in lieu of the transducer location). Measure the vertical distance between the face of the transducer and waterline.

a) Apply DC power (refer to 12-00-00, Servicing – General Data, Para 2.).

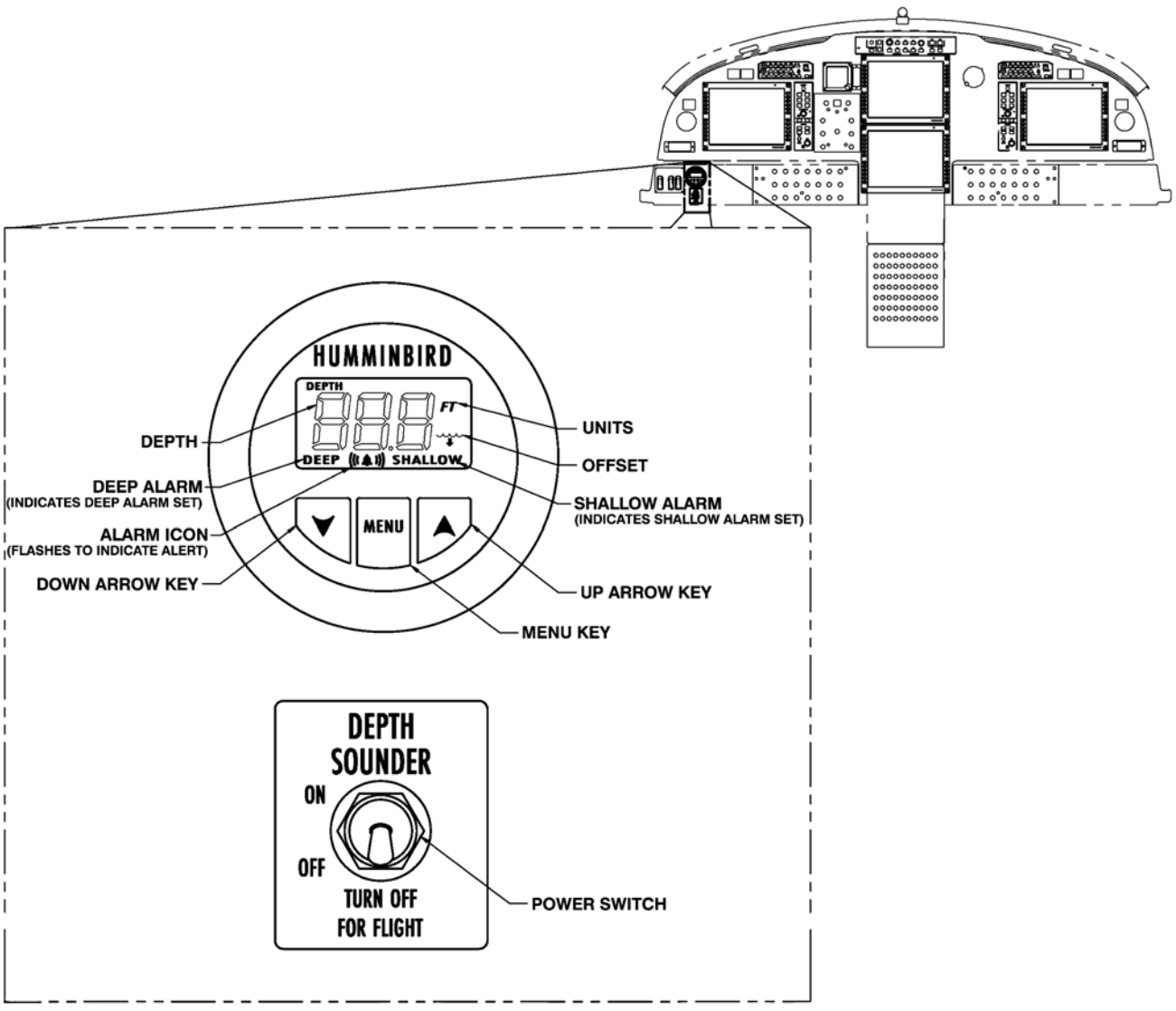
b) Select DEPTH SOUNDER switch ON.

c) Press the MENU key until SET OFF SET is displayed and flashing on the screen.

d) Press the UP arrow key to enter the measurement as a positive (+) number.

e) After the offset is entered, wait 5 seconds for the unit to return to normal operation. An offset icon will be visible on the display. Settings are saved even after the depth sounder is powered off.

f) Disconnect DC power (refer to 12-00-00, Servicing – General Data, Para 2.).



Depth Sounder Instrument Panel Location
Figure 2

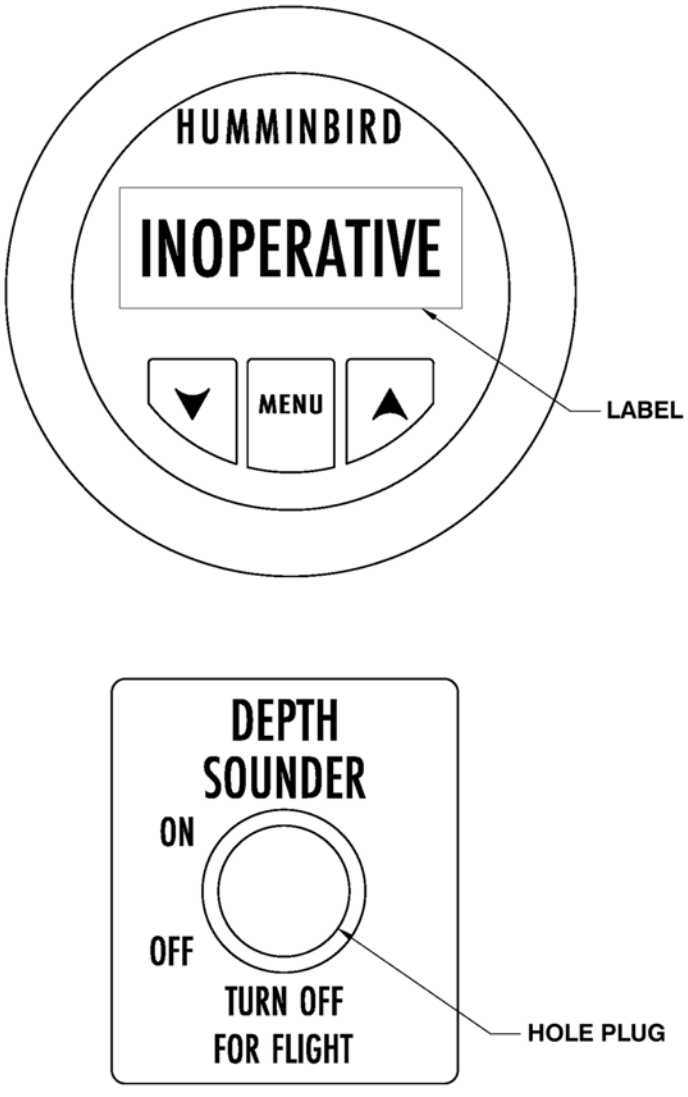
4. Removal

   A. Remove the DEPTH SOUNDER switch and plug hole with .50 inch diameter hole plug. Cap and stow wires.

   B. Remove DEPTH SOUNDER circuit breaker (located on avionics breaker panel) and plug hole with .50 inch diameter hole plug. Cap and stow wires.

   C. Disconnect depth sounder transducer wiring at inline splice located at transducer wire/fuselage penetration. Cap and stow wires.

   D. Fabricate 'INOPERATIVE' label with .25 inch lettering and apply to face of depth sounder display.



Depth Sounder Inoperative Configuration
Figure 3

5. Troubleshooting

Table 1 Troubleshooting

	SYMPTOM	POSSIBLE CAUSE	CHECK THAT	REMEDIAL ACTION
1. (a)	Blank HDR 650 display	Poor connection to power cable.	Condition of depth sounder wiring, splices and connections.	Replace damaged wires Replace damaged splices/connections.
(b)		Incorrectly wired power cable.	'Red' wire is connected to depth sounder switch. 'Black' wire connected to ground.	Correct wiring.
(c)		DEPTH SOUNDER RELAY is open.		Close depth sounder relay.
(d)		Supplied voltage below 10V.	Check output of 14V power converter Voltage should be 14V. Check condition or wiring and connectors.	Replace any damaged wiring. Replace damaged connectors. Replace damaged 14V power converter.
(e)		Damaged depth sounder unit.		Replace HDR 650 with new unit.
2. (a)	No bottom reading on display	Damaged transducer.	Check for visual damage to transducer.	Replace damaged transducer.
(b)		Damaged transducer wiring.	Check condition of transducer wiring in float. Check condition of transducer wiring in aircraft. Check condition of inline terminal module.	Replace damaged transducer wiring. Replace damaged transducer wiring. Replace damaged terminal module.
3.	No continuous depth display in very shallow water	Normal for water depths below one foot.		
4.	Screen fades, Images not sharp	Low voltage supplied to depth sounder.	Refer 1.(a), (b) and (c)	Refer 1.(a), (b) and (c)

CHAPTER 51

STANDARD PRACTICES AND STRUCTURES

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CHAPTER 51 – STANDARD PRACTICES AND STRUCTURES

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

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Structures – General Data

1. General

The DHC-6 Twin Otter is an all metal and composite, high-wing monoplane, powered by two Pratt and Whitney PT6A–34 (Mod 6/2012) engines driving Hartzell fully-feathering, reversing and constant-speed propellers. The aircraft is designed for passenger or general cargo carrying, and is provided with a fixed tricycle landing gear having a steerable nose wheel. Provision has been made for floats and amphibious floats; however, these are only available via incorporation of Supplemental Type Certificate.

The wing group consists of the metal wings, wing struts, wing fences and nacelles.

The fuselage is aluminum alloy, semimonocoque construction consisting of longerons, longitudinal stringers, intercostals, frames and stressed skin panels.

The tail group consists of horizontal and vertical stabilizers, elevators and a rudder. The horizontal and vertical stabilizers are both secured by front and rear fittings to corresponding fittings in the rear fuselage structure.

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CHAPTER 52

DOORS

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CHAPTER 52 – DOORS

LIST OF EFFECTIVE PAGES

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CHAPTER 52 – DOORS

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Doors (Mod 6/2024) – General Data

1. General

The aircraft entrances and exits, illustrated on Figure 1, consist of two flight compartment doors, two passenger cabin doors, front and rear baggage compartment doors, two emergency escape doors, one on each side of the aircraft, and two servicing access doors.

The forward baggage compartment door on the left side of the nose is hinged at its top edge and opens upward and outward and can be supported in the open position by a door stay. The door is secured in the closed position by two recessed latch assemblies. The forward latch can be key-locked. This baggage compartment door is included in the DOORS UNLOCKED annunciation circuit.

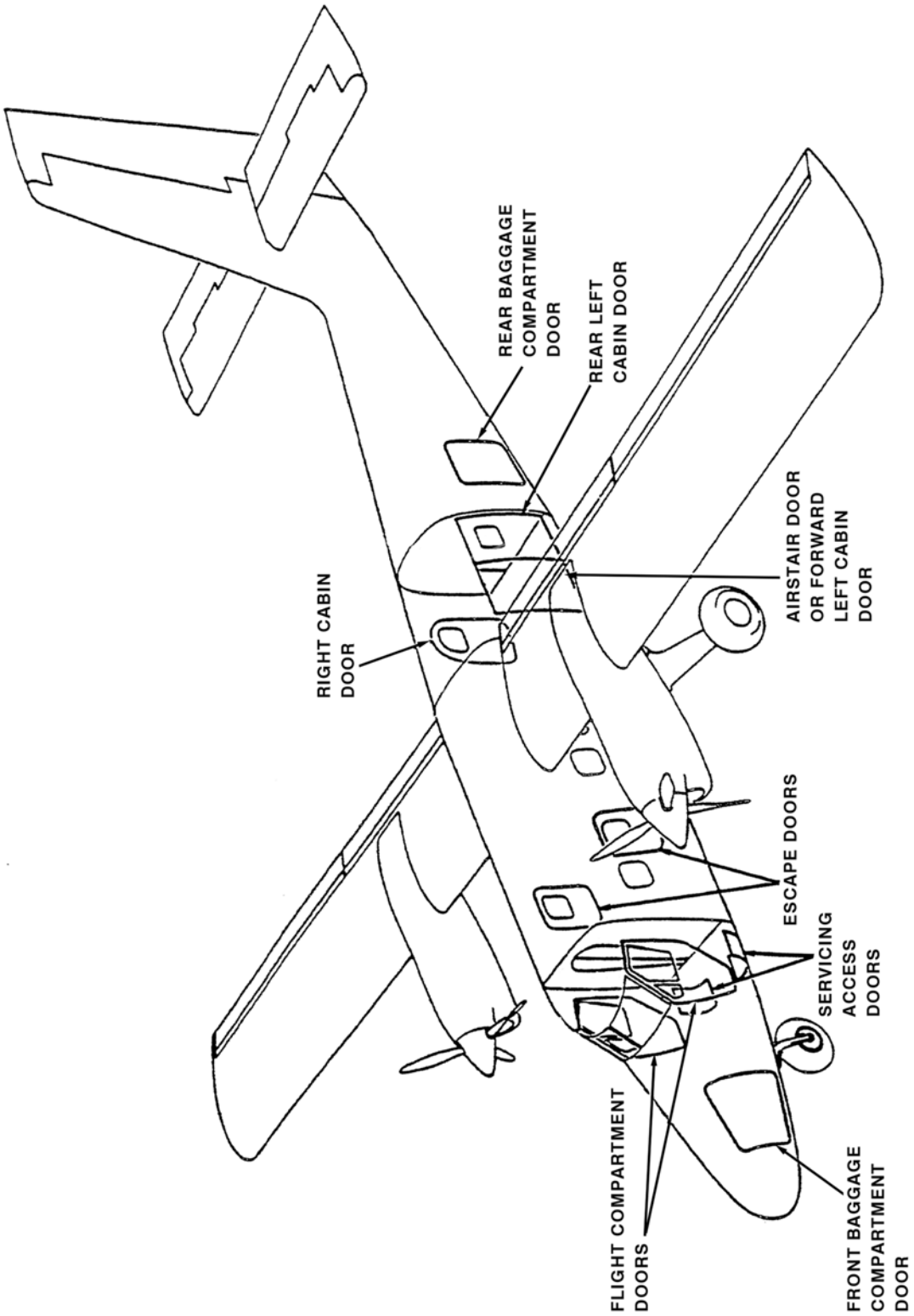
The flight compartment doors are hinged at their forward edges and latched by handles on the inside and outside. The outside handle incorporates a key-operated lock. A sliding window in each door can be adjusted to any position and be secured by a cam type latch. A map stowage pocket is provided on each door.

Annunciation of an open or improperly closed door is provided by the Crew Alerting System (CAS) portion of the avionics system. The nose baggage, rear baggage, right rear cabin, and forward portion of the left rear cabin door are monitored. Two pilot's doors, two plug type emergency exits, and the aft portion of the left cabin door are not monitored.

With a standard cabin interior fitted, no passageway is provided between the passenger cabin and rear baggage compartment at station 332.00. Two baggage restraint panels are permanently attached to the aperture in station 332.00 to prevent forward movement of baggage in the rear baggage compartment. The restraint panels are installed against the aft face of the bulkhead, one above the other.

Doors are considered to be secondary structure.

All doors are composite construction (Mod 6/2024), except Mod 6/2295 that introduces an airstair door of metal construction. It attaches to the airframe in the same manner as the composite door.



Doors – Location
Figure 1

Passenger/Crew – General Data

1. General

Two crew entry doors, one each side of the flight compartment, and two passenger doors, one each side of the cabin compartment are provided (refer to Figure 2).

A. Flight Compartment Doors

Each flight compartment door has a fixed window and a sliding window with a release handle. The doors are secured to the airframe structure by two hinges at the forward edge of each door, which allows either door to be opened outward and forward, to a position limited by a nylon restraining strap. An interior and exterior door handle is installed, which when turned clockwise, operates a latch mechanism which disengages a mid-door bolt, and an upper latch pin. Each exterior door handle can be locked by key from the outside. A map case is provided in the lower part of each door, and when a crew oxygen system is installed, clips are installed at the center of the door to support the flexible oxygen hoses.

B. Cabin Compartment Doors

Refer to Figure 1 for details.

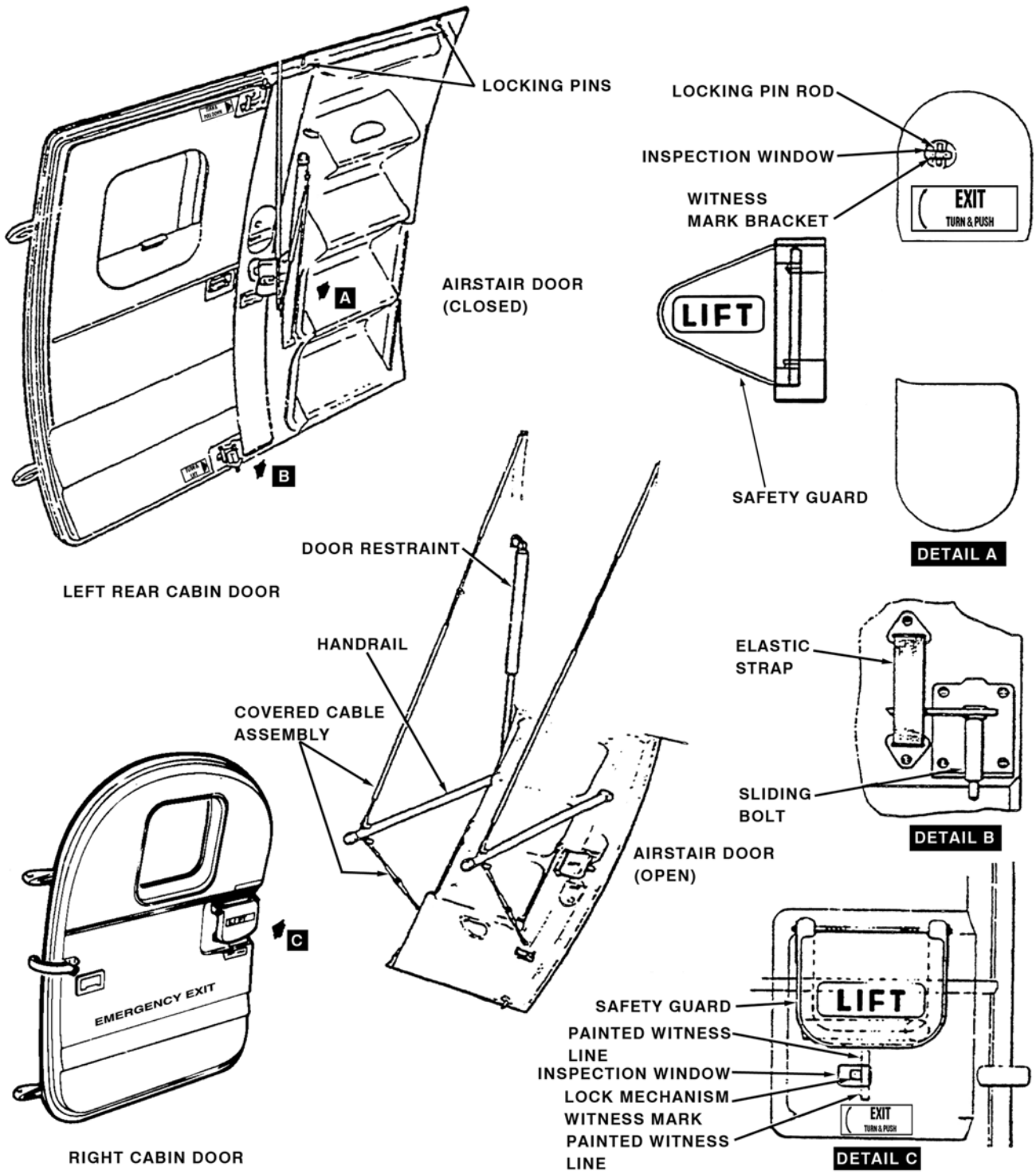
The cabin left compartment door is in two parts; the forward part is an airstair door for passenger entry, and the rear part to provide a larger entrance space for cargo loading.

The airstair door as installed to the basic aircraft, opens downward and outward, and is supported in the open position by two covered cable assemblies. One cable assembly is connected to the front edge of the door, and the other to the rear; both cables run parallel with each other and attach to the fuselage structure at the upper door frame. Aircraft have an adjustable eye fitting with a locknut on the rear cable assembly. The fitting is threaded onto the upper end of the rear cable assembly and connected to the rear door frame. The fitting can be adjusted to equalize cable tension between the front and rear cable assemblies. The airstair door is provided with a door restraint (door saver) which prevents the door falling free when being lowered and gives spring assist when being raised. The airstair door can be quickly removed to provide easier access for cargo loading. To assist in passenger entry and exit, the airstair door is provided with handrails. Each handrail is attached between its associated covered cable assembly and the appropriate side of the airstair door, and is so designed that as the door is closed, it will fold up against the inner side of the door. The airstair door has an internal and external handle which operates a latch mechanism to engage the two door locking pins in the fuselage upper door surround structure. An inspection window provides the means of checking that the two fluorescent orange dayglo paint witness marks (one on the locking pin rod and the other on a fixed bracket), align when the door is fully locked (refer to Figure 1, Detail A). Contacts, to complete the electrical circuit to the internal door unlocked microswitch (refer to 52-70-00, Doors Unlocked Caution System – General Data), are located on both the door and the fuselage door upper surround.

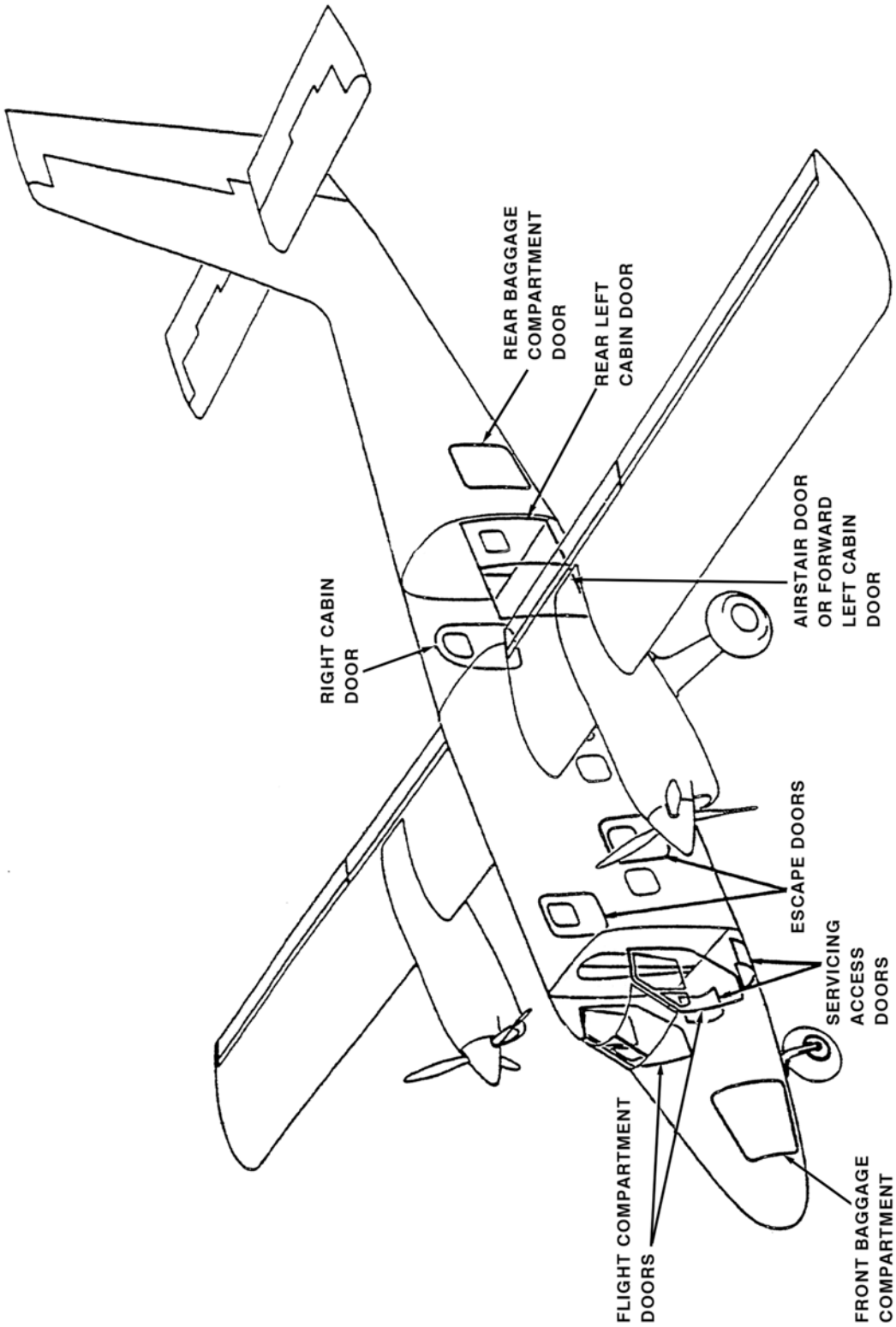
The cabin left rear door hinged to the rear part of the door frame, opens outward and rearward. When closed, the door is locked top and bottom by engaging sliding bolts which are then secured with elastic straps.

The cabin right door is hinged at the forward part of the door frame and incorporates a strap assembly which restricts door movement and prevents it from contacting and causing damage to the fuselage skin when the door is in the open position. The door latch mechanism, operated by an internal or external handle, has a latch bolt which engages in the door rear surround structure. An inspection window adjacent to the door lock handle shows, when the door is locked, a part of the door latch mechanism marked with fluorescent orange dayglo paint. With the door locked, this painted portion should be aligned with painted lines on each side of the inspection window.

The exterior cabin door handles can be locked by a key from the outside, and each interior handle is covered by a spring-loaded plastic safety guard which must be lifted in order to operate the handle.



Cabin Doors, Locks and Safety Guards
Figure 1



Doors – Location
Figure 2

2. Removal/Installation

A. Remove Airstair Door (Mod 6/2024)

1) Open and support airstair door.

2) Remove hardware attaching door saver to airstair and separate door saver from airstair.

3) Remove hardware attaching airstair handrail cables from aircraft fuselage.

4) Remove screws attaching piano hinge retainer and DZUS or camlock fasteners from aircraft and remove door.

5) Remove inner handle guard assembly (Mod 6/2326).

a) Remove screw attaching inner handle guard assembly and retain hardware.

b) Disassemble guard assembly by removing cotter pin at both ends of rod and retain.

c) Remove and discard damage springs.

d) Install new springs and assembly guard assembly by securing rod both ends with cotter pin.

B. Install Airstair Door (Mod 6/2024)

1) Position and support airstair door and align mounting holes of piano hinge to fuselage.

2) Secure piano hinge to fuselage with screws as required.

3) Secure airstair handrail cables to fuselage with attaching hardware.

4) Secure door saver to airstair door with attaching hardware.

5) Verify electrical conductivity between door and airframe. Refer to Para 3.C.

6) Install inner handle guard assembly (Mod 6/2326).

a) Install inner handle guard assembly, using retained hardware (refer to Para 2.A.5)).

b) Close airstair door and verify inner handle guard assembly is in contact with the handle.

C. Remove Airstair Door (Mod 6/2295)

1) Open and support airstair door.

2) Remove quick release pin attaching airstair door saver to fuselage.

3) Remove quick release pin attaching handrail cables to the door handrail.

4) Remove screws attaching piano hinge retainer and DZUS or camlock fasteners from aircraft and remove door.

D. Install Airstair Door (Mod 6/2295)

1) Position and support airstair door and align mounting holes of piano hinge to fuselage.

2) Secure piano hinge to fuselage with screws as required.

3) Secure airstair handrail cables to fuselage with quick release pin.

4) Secure airstair door saver to fuselage with quick release pin.

5) Verify electrical conductivity between door and airframe. Refer to Para 3.C.

E. Remove Cabin Left Rear Door

1) Open and support Cabin Left Rear Door.

2) Remove hinge pins attaching door to airframe and remove the door.

F. Install Cabin Left Rear Door

1) Install hinge pins attaching door to airframe.

2) Close door and check fit and alignment to airframe.

3) Verify electrical bonding of the door to airframe is maintained. Refer to Para 3.E.

G. Remove Cabin Right Rear Door

1) Open and support Cabin Right Rear Door.

2) Remove hinge pins attaching door to airframe and remove the door.

H. Install Cabin Right Rear Door

1) Install hinge pins attaching door to airframe.

2) Close door and check fit and alignment to airframe.

3) Verify electrical bonding of the door to airframe is maintained. Refer to Para 3.E.

I. Remove Flight Compartment Door

1) Open and support Flight Compartment Door.

2) Remove hinge pins attaching door to airframe and remove the door.

J. Install Cabin Flight Compartment Door

1) Install hinge pins attaching door to airframe.

2) Close door and check fit and alignment to airframe.

3) Verify electrical bonding of the door to airframe is maintained. Refer to Para 3.G.

3. Adjustment

Note Whenever airstair door is serviced or removed from aircraft, adjustment/check of the airstair door latch pin engagement is required.

Note Tapered section of latch pins is to be ignored when determining engagement.

Note If screws attaching the piano hinge or latch pin striker plates to the door structure are removed they must be reinstalled wet using PR–1764 CLASS B electrically conductive sealant with PR–148 adhesion promoter or equivalent.

A. Adjust the amount of latch pin engagement of the airstair door by adjusting the latch pin push rod assembly.

B. Ensure pin engagement on latch pins when door is in the closed position is minimum 0.25 of an inch.

C. Airstair Door Bonding Test

With door closed, measure the resistance between the airstair door and the airframe using a milliohm meter, resistance to be less than 10 milliohms. Probe measurements to be taken on one of the airstair door hinge fasteners and an airframe side rivet. Refer to Figure 3.

D. Airstair Door Bonding Adjustment/Rework

Remove the Airstair door hinge attachment fasteners and clean as required to achieve bonding requirement. For reference, refer to applicable standard electrical bonding procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

E. Cabin Left/Right Rear Doors Bonding Test

With door closed, measure the resistance between the rear cabin doors and the airframe using a milliohm meter, resistance to be less than 10 milliohms. Probe measurements to be taken on one of the rear door hinge fasteners and an airframe side rivet. Refer to Figure 4.

F. Cabin Left/Right Rear Doors Adjustment/Rework

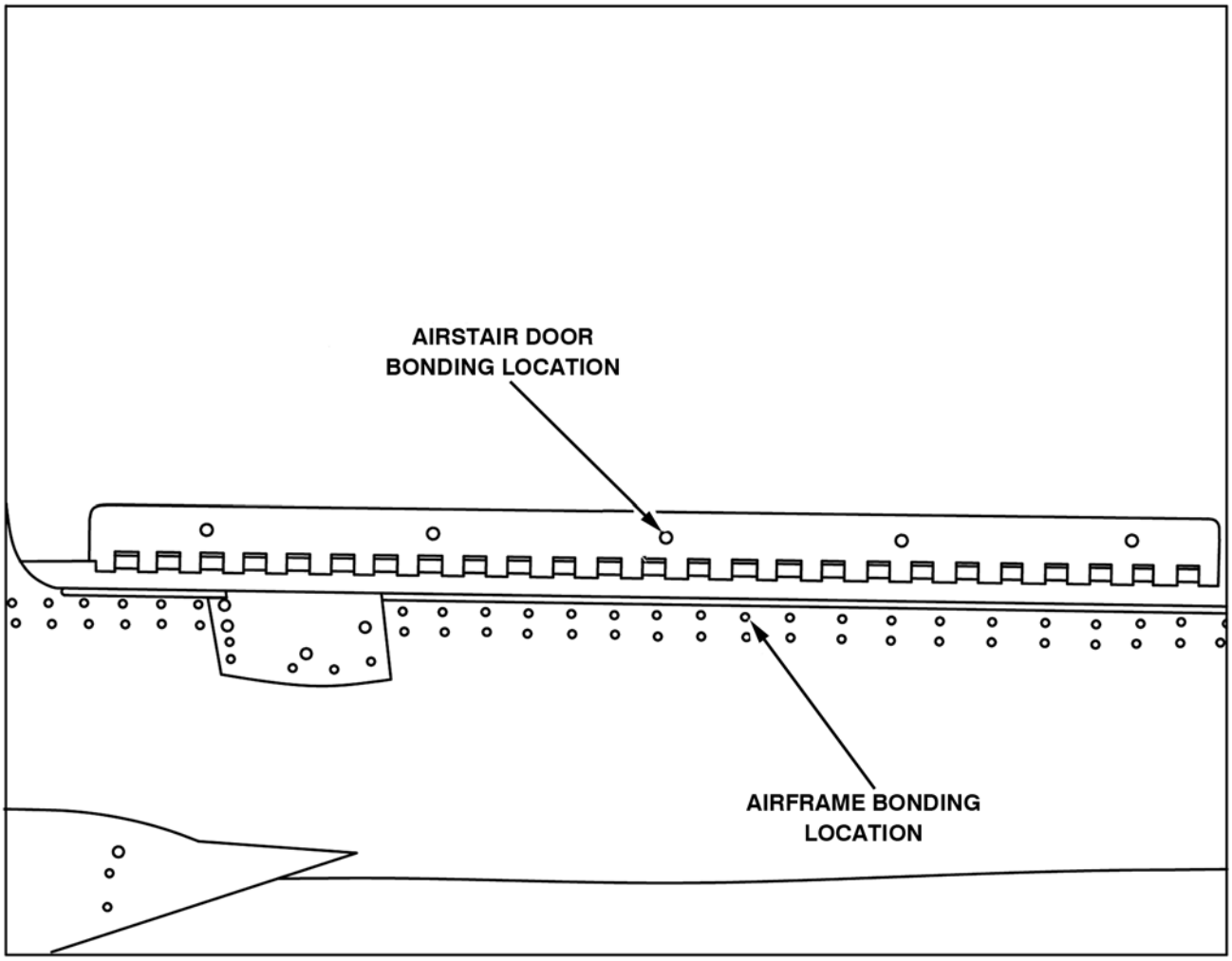
Remove the rear cabin door hinge attachment fasteners and clean as required to achieve bonding requirement. If required, remove rear cabin door to airframe hinge pin and clean to achieve bonding requirement. For reference, refer to applicable standard electrical bonding procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

G. Flight Compartment Doors Bonding Test

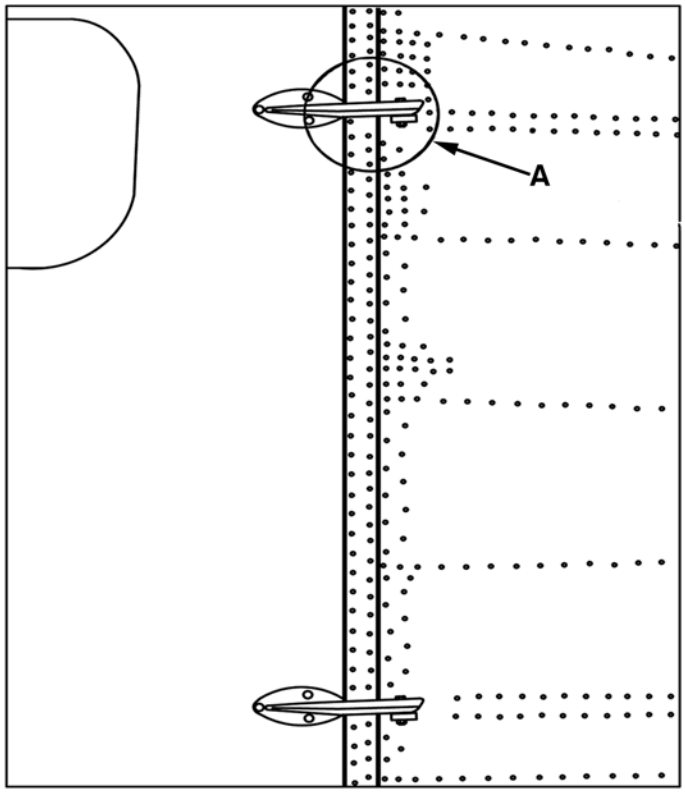
With door closed, measure the resistance between the flight compartment door and the airframe using a milliohm meter, resistance to be less than 10 milliohms. Probe measurements to be taken on one of the door hinge fasteners and an airframe side rivet. Refer to Figure 5.

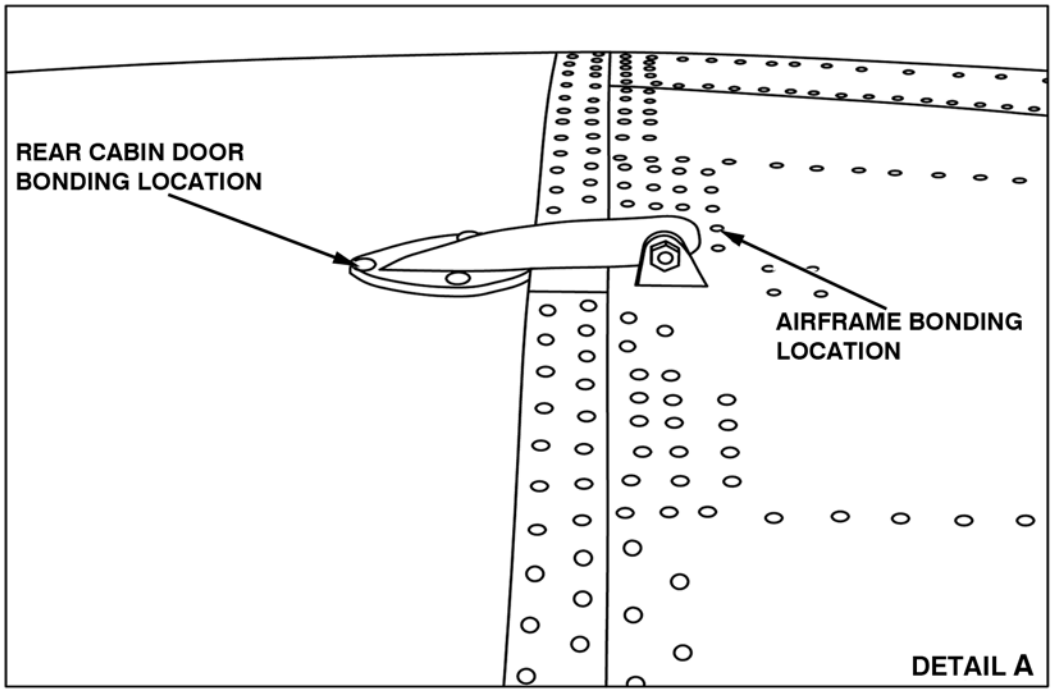
H. Flight Compartment Doors Bonding Adjustment/Rework

Remove the door hinge attachment fasteners and clean as required to achieve bonding requirement. If required, remove door to airframe hinge pin and clean to achieve bonding requirement. For reference, refer to applicable standard electrical bonding procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

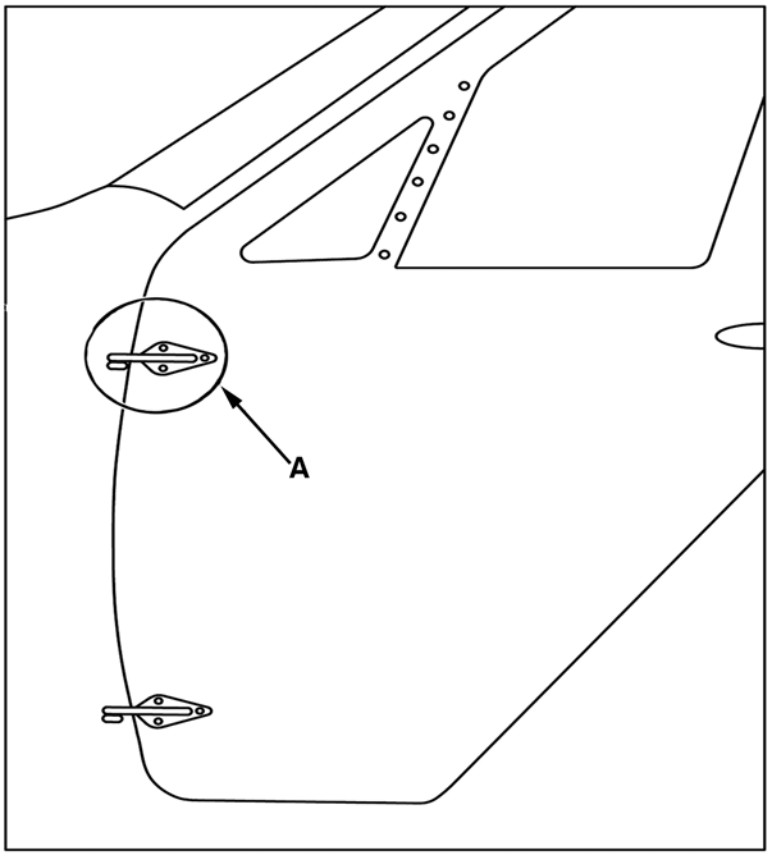


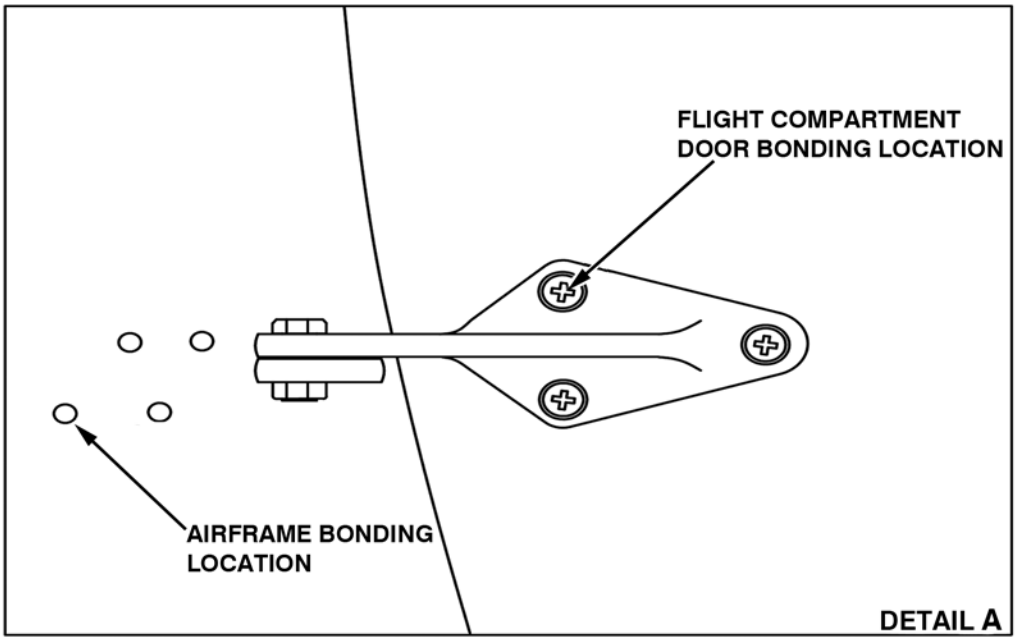
Airstair Door – Electrical Bonding Check Locations
Figure 3





Rear Cabin Doors – Electrical Bonding Check Locations (Left Door Shown)
Figure 4





Flight Compartment Doors – Electrical Bonding Check Locations (Left Door Shown)
Figure 5

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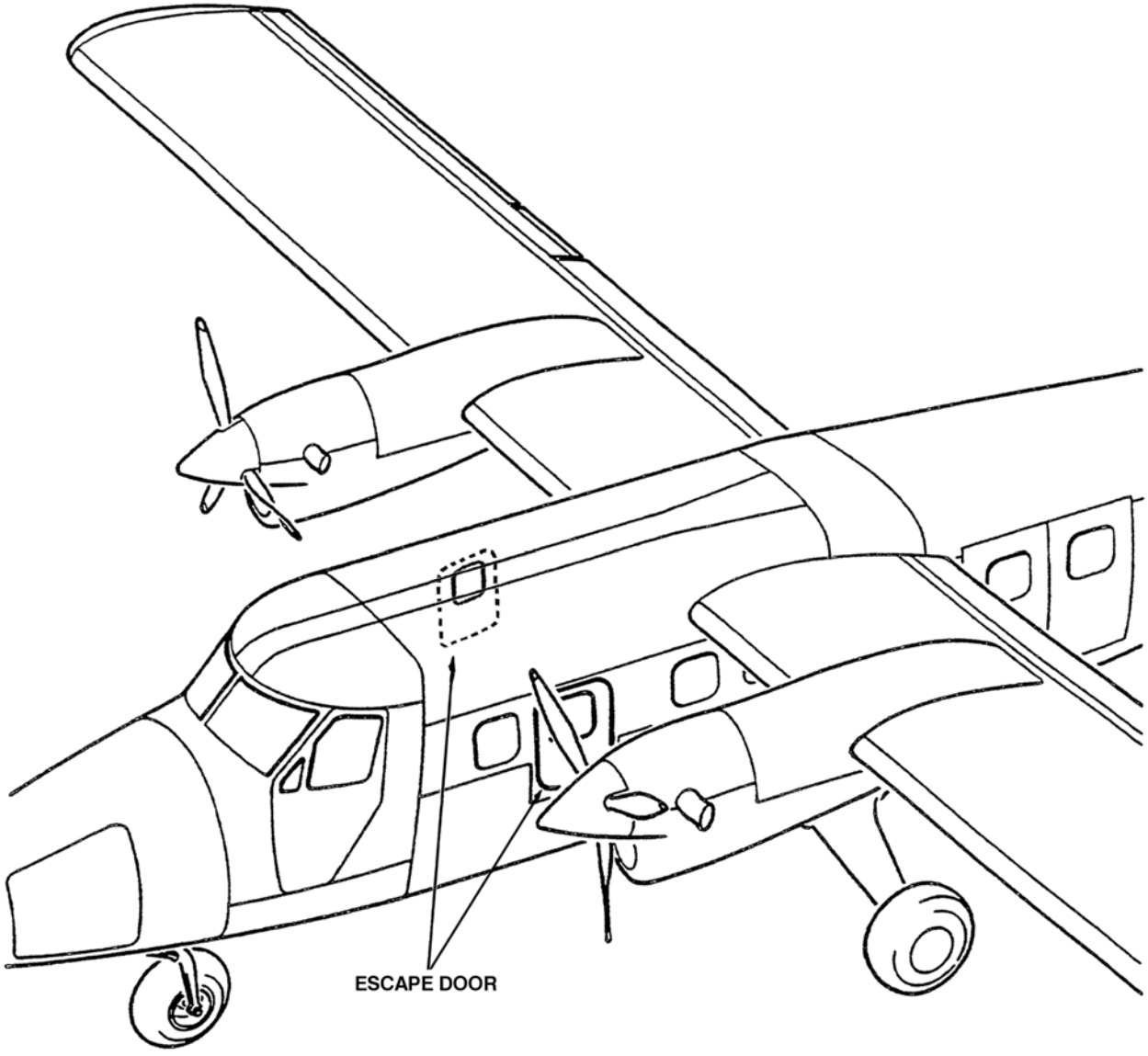
Emergency Exit (Mod 6/2024) – General data

1. General

   The emergency exits consist of two escape doors, one on each side of the passenger cabin compartment.

2. Escape Door

   Each escape door has a handle release mechanism, a window, door seal, and rubber pads bonded to the side and bottom structure. Pulling downward on the release handle disengages the release handle from fittings on the cabin structure and on the door. The top of the door is then pushed outwards until clear of the door frame.



Emergency Exits
Figure 1

3. Installation of Escape Door

   Note Two men are required, one to position and retain escape door from outside the fuselage, and one to install release handle.

   A. Position escape door so that two plates at bottom of door are located against pads on inside of fuselage. Push inwards at top of escape door so that door is flush with fuselage.

   B. Engage slots of two release fittings on release handle with two bolts on door fittings. Operate release handle downwards to withdraw two spring-loaded plungers, align the plungers with two holes in fittings on fuselage and raise release handle allowing the plungers to engage holes in fittings.

      Note If difficulty is experienced in aligning plungers with holes in fittings, it is recommended to loosen four screws on each plate at bottom of door to assist repositioning at top of escape door. Tighten screws on each plate when handle is installed.

   C. Lockwire forward arm of release handle to bolt attaching fitting to fuselage using 0.020–inch diameter copper locking wire. Use single method of lockwiring to ensure that it can easily be broken in an emergency.

   D. Install cover over release handle mechanism.

4. Adjustment/Test

   A. Operational Test – Escape Door

      Note Two men are required, one to operate release, and one to receive escape door outside fuselage.

      1) Remove cover at handle release mechanism.

      2) Pull down release handle sharply enough to break copper locking wire. Handle will separate from escape door and fuselage.

      3) With operator ready to receive escape door, push outwards at top of escape door sufficiently to free two plates locating bottom of door and fuselage. Escape door must push freely from door frame.

      4) Install escape door, refer to Para 3.

Baggage Compartment Doors (Mod 6/2020) (Mod 6/2024) – General Data

1. General

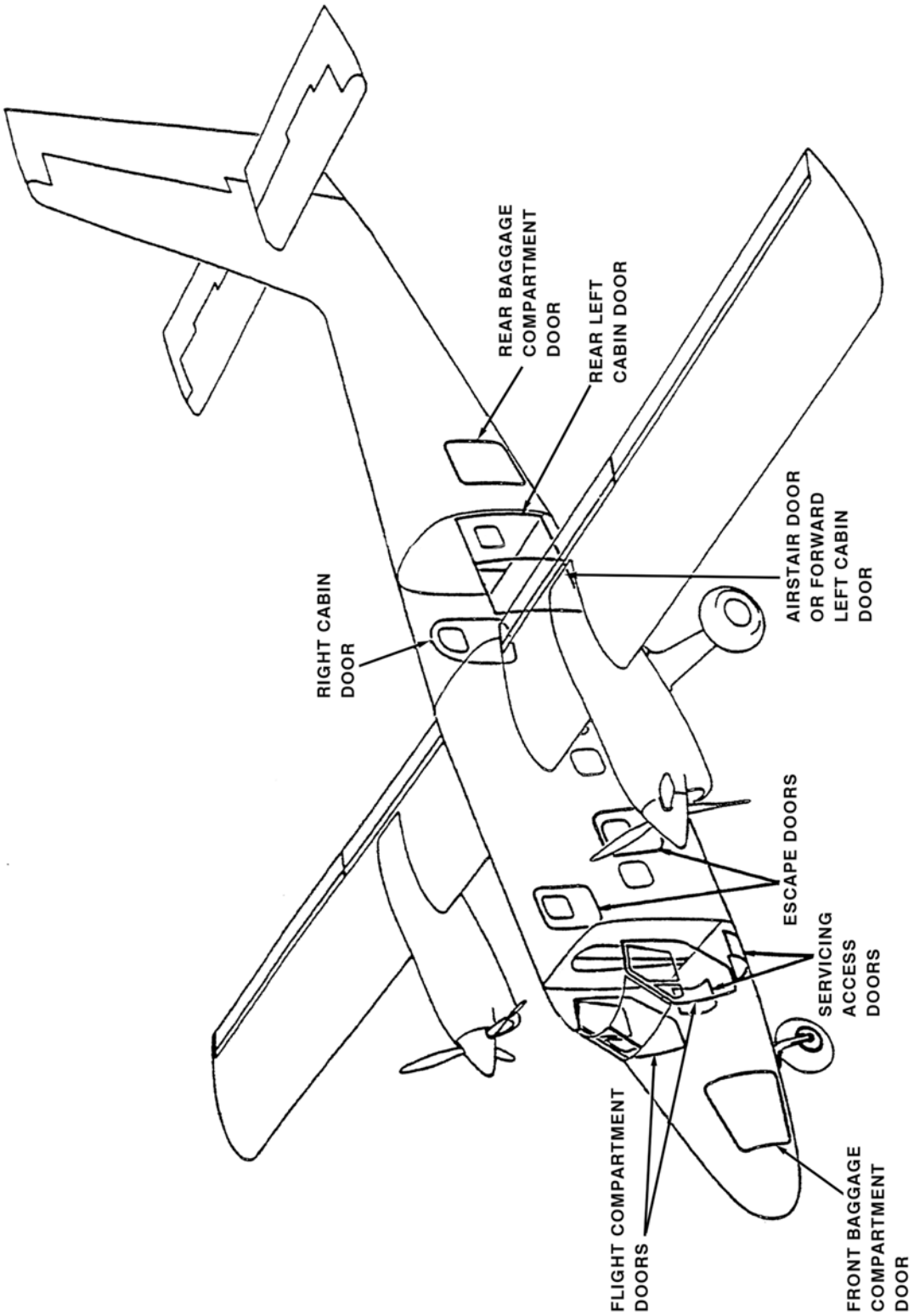
Front and rear baggage compartment doors are installed on the left side of the nose compartment and the aft fuselage respectively (refer to Figure 1).

A. Front
Baggage Compartment Door (Mod 6/2020 or Mod 6/2161)

The front baggage compartment door is comprised of an outer and inner skin made from pre-impregnated carbon cloth reinforced by a foam core framework, and is secured to the nose compartment at the top of the door by a piano type hinge assembly. The hinge is riveted to the door and is attached to the nose compartment by screws. Two spring-loaded latch assemblies at the bottom of the door, one at the fwd end and the other at the aft end secure the door in the closed position. A silicone foam seal is bonded to the inner skin to exclude drafts and moisture. A door stay assembly is provided to hold the door open and when not in use secured in a spring clip adjacent to the aft door latch. A door lock assembly is included in the forward latch assembly only and is operated by a key.

B. Rear Baggage Compartment Door (Mod 6/2024)

The rear baggage compartment door is fabricated from a composite sandwich construction of carbon fiber and low density foam core, and is secured to the rear fuselage at the top of the door by two hinge assemblies. Each hinge is secured to the door and the rear fuselage by screws. The door is secured to the fuselage at the lower end of the door frame by two spring-loaded latch pins, each connected to an arm assembly by a push rod, which in turn is rotated by operation of the door handle. A lock is incorporated in the door handle and is operated by a key. A silicone foam seal is bonded to the inner edge of the outer skin to exclude drafts and moisture. A door stay is provided to hold the door open and when not in use is secured in a spring clip at upper aft corner of the door.



Doors – Location
Figure 1

2. Removal/Installation

A. Remove Front Baggage Compartment Door

1) Open and support front baggage compartment door.

2) Remove screws attaching door hinge to nose compartment and remove door.

B. Install Front Baggage Compartment Door

1) Position and support door and align mounting holes in hinge at top of door and in nose compartment.

2) Secure hinge to nose compartment with screws as required.

3) Close and latch door.

4) Verify electrical conductivity between door and airframe. Refer to Para 3.A.

5) Ensure each fwd and aft latch pin engagement between the nose door and nose baggage compartment is minimum 0.25", in the closed position, measured from the outer latch pin surface.

C. Remove Front Baggage Compartment Door Latch Assembly

1) Open and support front baggage compartment door.

2) Remove hardware attaching latch assembly mounting plate to door outer skin and remove plate.

3) Remove rivets attaching latch to plate and withdraw latch assembly.

D. Install Front Baggage Compartment Door Latch Assembly

1) Position latch assembly on mounting plate and align attachment holes.

2) Secure latch assembly with six one-eighth diameter rivets on plate.

3) Secure plate on door with mounting hardware. Wet install hardware with PR–1764 CLASS B electrically conductive sealant with PR–148 adhesion promoter or equivalent.

4) Close and latch door.

5) Ensure each fwd and aft latch pin engagement between the nose door and nose baggage compartment is minimum 0.25", in the closed position, measured from the outer latch pin surface.

E. Remove Rear Baggage Compartment Door

Note Before removal of the Rear Baggage Compartment Door, confirm hinge configuration. If configuration comprises of a swaged pin, remove the Rear Baggage Compartment Door as per Para 2.E.1)2.5.1. If hinge configuration comprises of a cotter pin and washer configuration, remove Rear Baggage Compartment Door as per Para 2.E.2).

1) Baggage Door with Swaged Pin Configuration

a) Open and support rear baggage compartment door.

b) Remove two screws securing each hinge to airframe and remove door from aircraft.

2) Baggage Door with Cotter Pin Configuration

a) Open and support rear baggage compartment door.

b) Remove cotter pin and washer securing each hinge pin, remove hinge pins and remove door from aircraft.

F. Install Rear Baggage Compartment Door

Note Before install of the Rear Baggage Compartment Door, confirm hinge configuration. If configuration comprises of a swaged pin, install the Rear Baggage Compartment Door as per Para 2.F.1). If hinge configuration comprises of a cotter pin and washer configuration, install Rear Baggage Compartment Door as per Para 2.F.2).

1) Baggage Door with Swaged Pin Configuration

a) Position and support door and align mounting holes in hinges and aircraft.

b) Secure each hinge to aircraft with two screws.

c) Close and latch door.

d) Verify electrical conductivity between door and airframe. Refer to Para 3.C.

2) Baggage Door with Cotter Pin Configuration

a) Position and support door and align hinge halves on aircraft and door.

b) Insert hinge pins and lock in place with washer and cotter pin.

c) Close and latch door.

d) Verify electrical conductivity between door and airframe. Refer to Para 3.C.

G. Remove Rear Baggage Compartment Door Handle

1) Unlatch rear baggage compartment door.

2) Support door and remove two screws securing door handle to door.

3) Withdraw handle.

H. Install Rear Baggage Compartment Door Handle

1) Ensure door latches are in unlatched position.

2) Insert door handle in square hole in arm assembly and ensure tail of handle is pointing downwards.

3) Secure handle to door with two screws. Wet install screws with PR–1764 CLASS B electrically conductive sealant with PR–148 adhesion promoter or equivalent.

4) Close and latch door.

3. Adjustment/Test

A. Front Baggage Compartment Door Bonding Test

Note If hardware attaching the rear baggage compartment hinges to the composite door structure are removed they must be wet installed using PR–1764 CLASS B electrically conductive sealant with PR–148 adhesion promoter or equivalent.

With door closed, measure the resistance between the front baggage compartment door and the airframe using a milliohm meter, resistance to be less than 10 milliohms. Probe measurements to be taken on one of the front baggage compartment door hinge fasteners and an airframe side rivet. Refer to Figure 2.

B. Front Baggage Compartment Door Bonding Adjustment/Rework

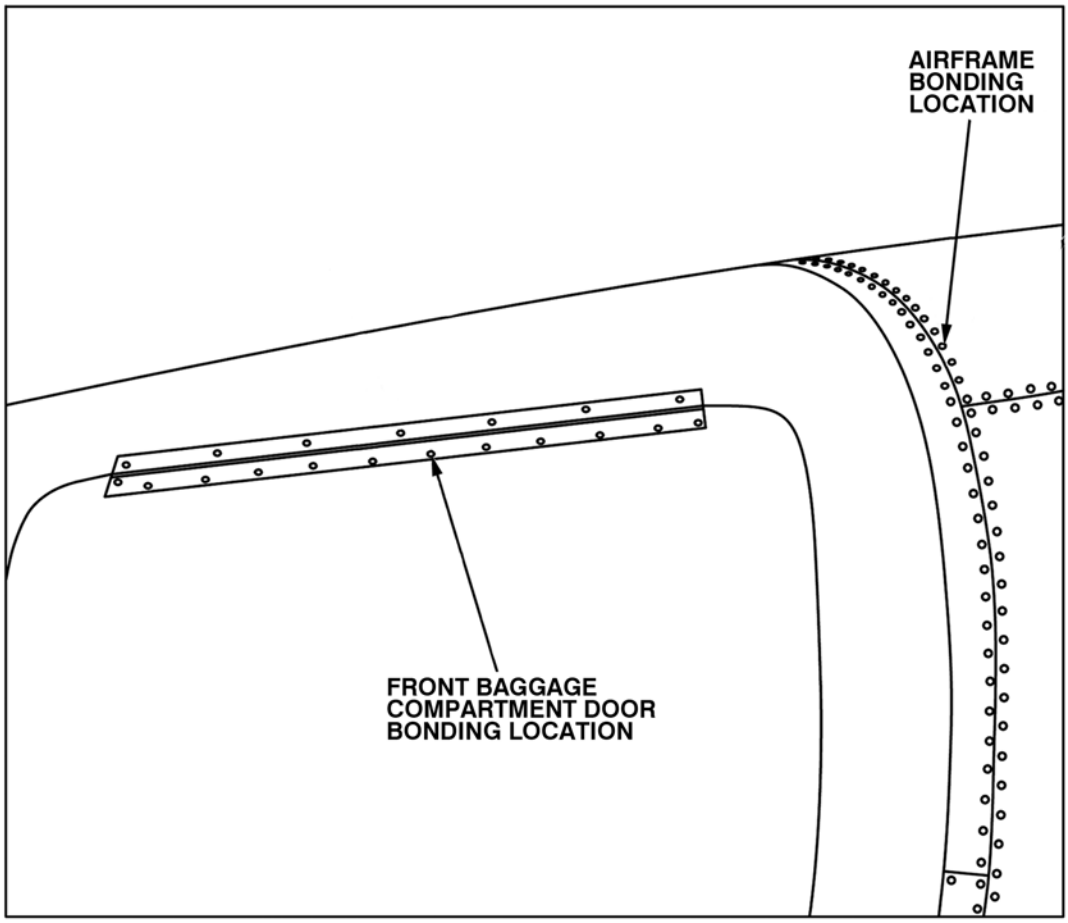
Remove the front baggage compartment hinge attachment fasteners and clean as required to achieve bonding requirement. If required, remove front baggage compartment to airframe bonding strap and clean mating surfaces to achieve bonding requirement. For reference, see applicable standard electrical bonding procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

C. Rear Baggage Compartment Door Bonding Test

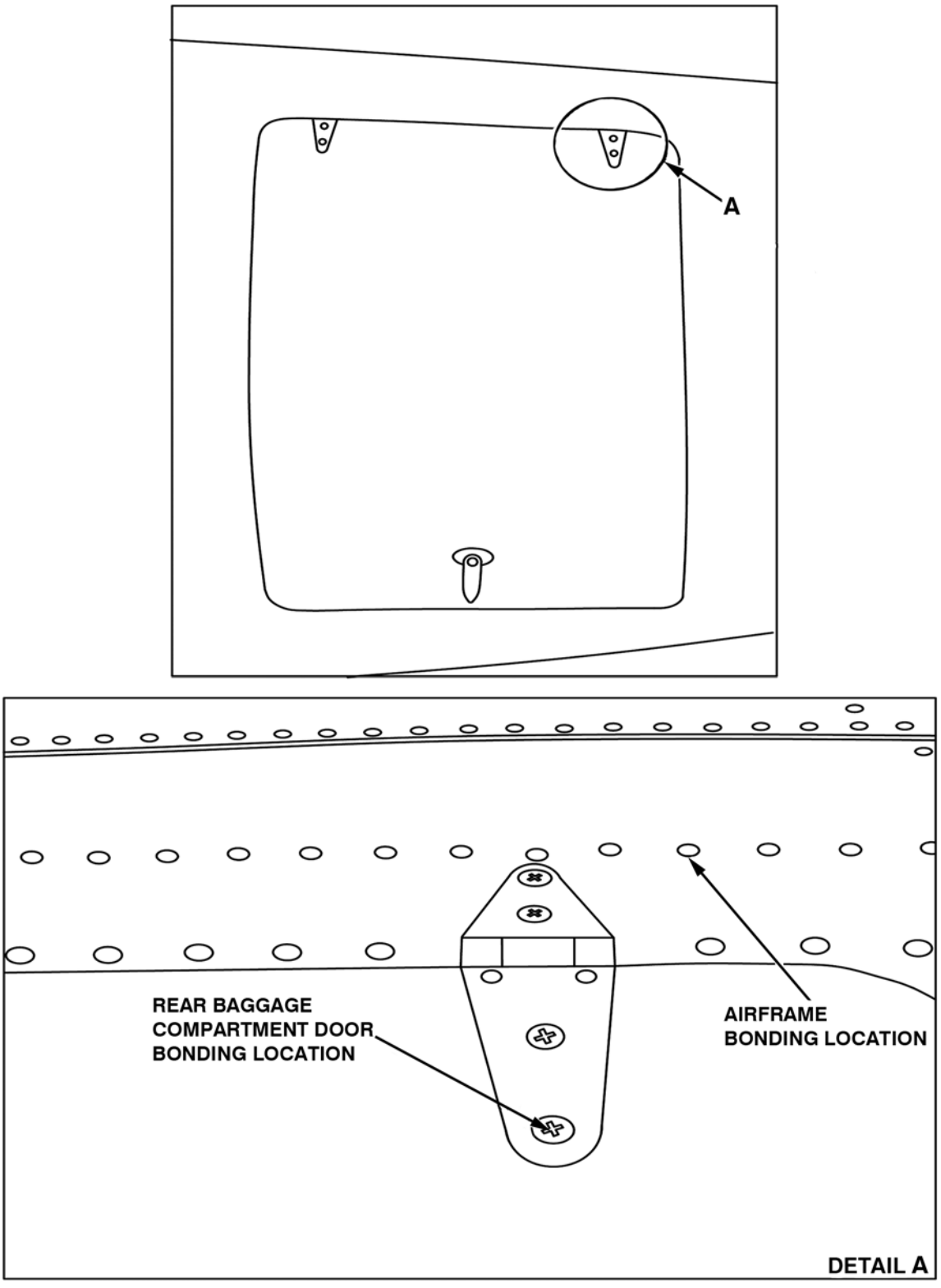
With door closed, measure the resistance between the rear baggage compartment door and the airframe using a milliohm meter, resistance to be less than 10 milliohms. Probe measurements to be taken on one of the rear baggage compartment door hinge fasteners and an airframe side rivet. Refer to Figure 3.

D. Rear Baggage Compartment Door Bonding Adjustment/Rework

Remove the rear baggage compartment hinge attachment fasteners and clean as required to achieve bonding requirements. If required, remove rear baggage compartment to airframe hinge pin and clean to achieve bonding requirement. For reference, see applicable standard electrical bonding procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.



Front Baggage Compartment Door – Electrical Bonding Check Locations
Figure 2



Rear Baggage Compartment Door – Electrical Bonding Check Locations
Figure 3

Doors Unlocked Caution System – General Data

1. General

Refer to Figure 1 for details.

The doors unlock caution system, provides the pilot with a visual and an aural indication if any one of the five doors are not locked (on aircraft with S.O.O. 6393, Rear Baggage Door Removal, there are only four doors). Microswitches installed on the right hand cabin door, the front and rear baggage compartment doors and the Airstair door, operate when the associated door (or doors) are opened and complete a ground circuit for a caution message to be displayed through the CAS and an aural warning generated.

Doors not monitored by the system are the left hand and right hand pilot doors, the two plug type emergency doors and the aft portion of the rear cabin door.

2. Description

The 28 Volt DC supply to the system is through a 5 ampere DOORS UNLOCKED circuit breaker. The caution light is connected to the MAU and displayed on the CAS.

The front baggage compartment door, right hand cabin door, and the rear baggage compartment door have a single microswitch installed at each door position, each being electrically connected in parallel.

Note Aircraft with S.O.O. 6393 incorporated do not have a rear baggage compartment door and associated microswitch.

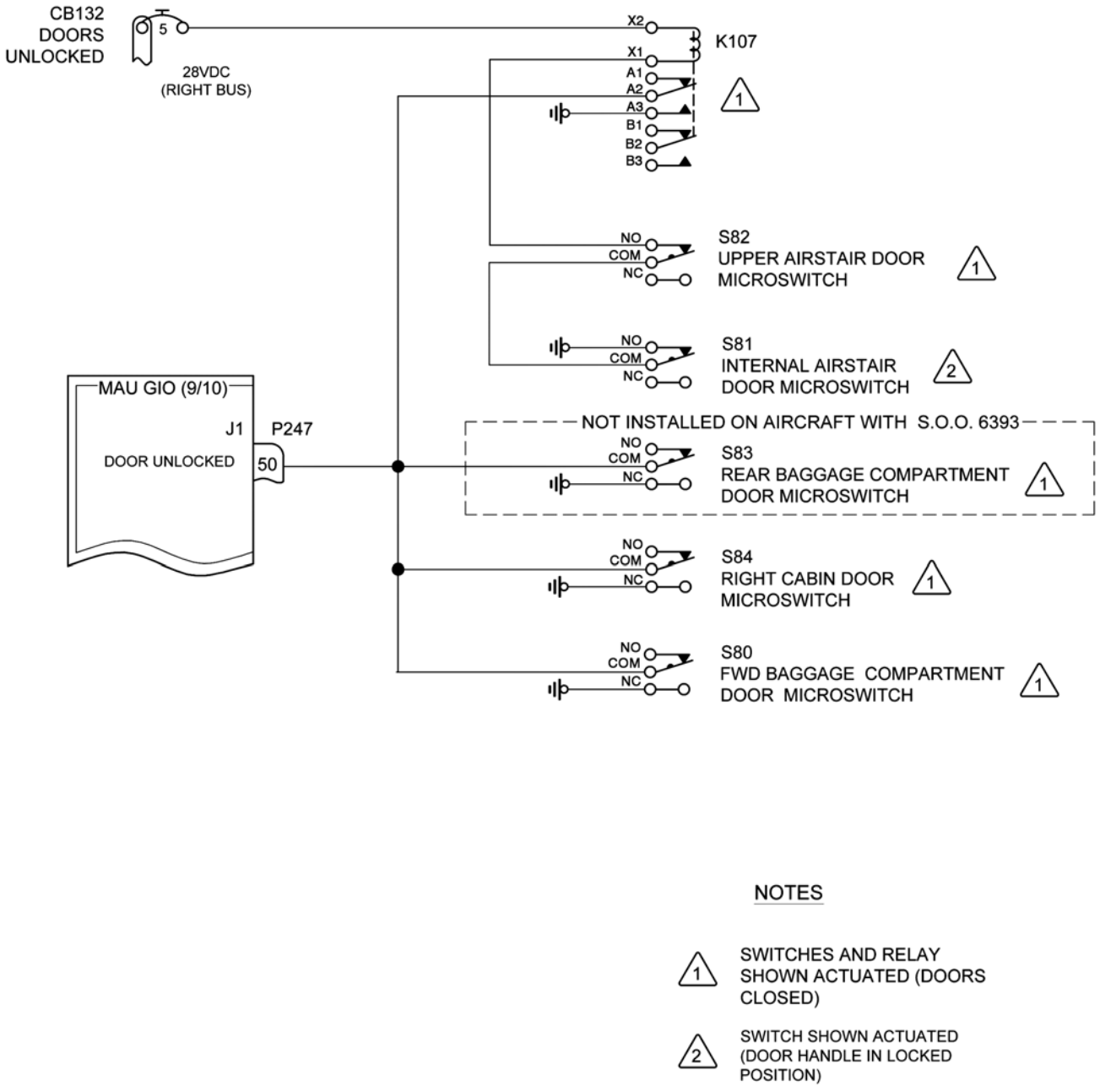
The airstair door has two microswitches connected in series with each other and in series with the control coil of an airstair door caution relay. The airstair door caution relay is located in the cabin roof at the rear of the fuselage station 262.00. One airstair microswitch is installed in the fuselage airstair door structure, and is actuated by the airstair door rear locking pin, the other microswitch is installed within the door and is operated by the door latch. With the aircraft electrical power on, and the airstair door closed and locked, the airstair door caution relay is energized (because of the ground provided through the microswitches for the relay coil), and normally closed contacts are open. Inversely, if the airstair door were unlocked or opened, the relay would deenergize, and its contacts would close, completing a ground circuit to the MAU (refer to Figure 1). Electrical connections to the airstair door lock microswitch are made by spring contacts on the door, and contact strips on the fuselage door surround structure, when the door is closed.

3. Operation

When the aircraft is on the ground or in the air and the Modular Avionics Unit detects a ground as a result of one or more doors being open, the following occurs:

1 MASTER CAUTION light on the instrument panel lights up. (Refer to 31-50-00, Monitor Warning System – General Data).

2 Doors Unlocked caution CAS message is displayed.



NOTES

1. SWITCHES AND RELAY SHOWN ACTUATED (DOORS CLOSED)
2. SWITCH SHOWN ACTUATED (DOOR HANDLE IN LOCKED POSITION)

Doors Unlocked Caution – Electrical Schematic
Figure 1

CHAPTER 53

FUSELAGE

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CHAPTER 53 – FUSELAGE

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Fuselage – General Data

1. General

The fuselage is aluminum alloy, semimonocoque construction consisting of longerons, longitudinal stringers, intercostals, frames, and stressed skin panels. It comprises five permanently attached sections: nose section (stations –21.00 to 60.00); flight compartment (stations 60.00 to 110.00); fuselage forward cabin (stations 110.00 to 262.00); fuselage aft cabin (stations 262.00 to 332.00); and rear fuselage (stations 332.00 to 535.24).

On aircraft with Mod S.O.O. 6168, removable ice shields are provided on both sides of the fuselage, in the plane of the propellers, to give protection from ice which may be shed by the propeller blades.

A. Nose Section (Mod 6/2020 or Mod 6/2160)

The cone-shaped nose section forms a baggage/equipment compartment with access through a door on the left side of the nose. Bonding straps from the door hinge and sill are connected to the fuselage at station 44.00. Removable screens at rear of the baggage compartment provide access to equipment forward of bulkhead at station 60.00. Lightning protection strips on the radome are bonded to the exterior of the nose section via metal tabs connected to the copper mesh. The nose is deemed to be a secondary structure.

B. Flight Compartment

The two-seat flight compartment has an entrance door on each side, with an L–type step and a hand-grip at the pilot's side; a similar step and hand-grip is installed on the co-pilot's side of the aircraft. The compartment rear bulkhead provides entrance from the cabin. Maintenance access to flight controls, hydraulic and heating system components below the flight compartment floor is provided by two hinged doors. The compartment front bulkhead carries the nose gear.

C. Cabin Compartment

The cabin has an entrance door on each side, and the cabin front and rear bulkheads provide entry into the flight compartment and baggage compartment respectively. Accommodation is provided for removable and folding seats and the installation of tie-down devices for cargo when carried in place of seats. Detachable upholstery panels provide access to flight and engine controls, bleed air, heating, a pneumatic package and deicing pipes, if fitted. Fuel tanks occupy the area below the cabin floor.

D. Rear Fuselage

The rear fuselage extends from the rear cabin semi-bulkhead to the tail in a gradual taper. A baggage compartment is provided with an external door on the left side. Access to flight controls cable runs in the rear fuselage is provided through the baggage compartment rear bulkhead.

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FWD and AFT Avionic Shelves – General data

1. General

Avionics components are installed on equipment racks located in the forward fuselage and in the aft empennage structure.

2. Description

The forward avionics equipment racks are entirely contained within the metallic section of the forward fuselage of the aircraft with the exception of the top shelf which has been extended into the nose baggage compartment to allow for mounting of the Apex MAU hardware. To provide protection from HIRF effects, the metallic structure has been extended into the baggage compartment to fully enclose the Apex avionics equipment within a metallic envelope. Metallic covers are provided to encase the entire avionics bay.

The aft avionics equipment racks are entirely contained within the metallic section of the aft empennage structure of the aircraft.

3. Adjustment/Test

A. Avionics Equipment Rack Test

Measure the resistance between the equipment rack and the airframe using a milliohm meter, resistance to be less than 2.5 milliohms. Probe measurements to be taken on the equipment rack mounting skin surface (as close as possible to the bonding stud bracket) and the corresponding airframe bonding stud.

B. Avionics Equipment Rack Adjustment

Rework bonding jumper attachments and/or brackets as required in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data to achieve bonding requirement.

4. Removal and Installation

A. Forward Avionics Equipment Racks

1) Removal of Equipment Racks

a) Gain access to the forward avionics racks through the forward baggage compartment.

b) Remove the top and middle covers attached to bulkhead at stations 37.00 & 44.00.

c) Disconnect and remove electrical components as required to allow removal of the equipment rack(s) (refer to associated individual component maintenance practices and 20-70-01, Handling of Static Sensitive Electronic Components and Devices – General Data).

d) Remove the bonding jumpers attached to the equipment rack(s).

e) Remove equipment rack fasteners and remove the equipment rack(s) from the avionics compartment.

2) Installation of Equipment Racks

a) Set equipment rack(s) in position and install fasteners.

b) Install electrical components and associated wiring (refer to individual component maintenance practices and 20-70-01, Handling of Static Sensitive Electronic Components and Devices – General Data).

c) Attach disconnected bonding jumpers to equipment rack(s).

d) Verify electrical bonding of equipment rack(s) to airframe is maintained. Refer to Para 3.A.

e) Install the top and middle covers attached to bulkhead at stations 37.00 and 44.00. Ensure the covers are mounted securely.

WARNING **ACCESS COVERS ARE REQUIRED TO MAINTAIN PROTECTION FEATURES THAT ARE ESSENTIAL TO MINIMIZING THE EFFECTS OF HIGH INTENSITY RADIATED FIELDS (HIRF).**

f) Close forward baggage compartment door.

B. Aft Avionics Equipment Racks

1) Removal of Equipment Racks

a) Gain access to the aft avionics racks through the aft baggage compartment.

b) Disconnect and remove electrical components as required to allow removal of the equipment rack(s) (refer to associated individual component maintenance practices and 20-70-01, Handling of Static Sensitive Electronic Components and Devices – General Data).

c) Remove the bonding jumpers attached to the equipment rack(s).

d) Remove equipment rack fasteners and remove the equipment rack(s) from the avionics compartment.

2) Installation of Equipment Racks

a) Set equipment rack(s) in position and install fasteners.

b) Install electrical components and associated wiring (refer to individual component maintenance practices and 20-70-01, Handling of Static Sensitive Electronic Components and Devices – General Data).

c) Attach disconnected bonding jumpers to equipment rack(s).

d) Verify electrical bonding of equipment rack(s) to airframe is maintained. Measure the resistance between the equipment rack and the airframe using a milliohm meter, resistance to be less than 2.5 milliohms. Probe measurements to be taken on the equipment rack mounting skin surface (as close as possible to the bonding stud bracket) and the corresponding airframe bonding stud. Rework bonding jumper attachments and/or brackets as required in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data to achieve bonding requirement.

e) Close aft baggage compartment door.

Radome – General data

1. General

The radome is installed on forward surface of the nose baggage compartment structure.

2. Description

The DHC-6 Series 400 retains the original DeHavilland DHC-6-300 radome design with the exception the Series 400 radome contains four lightning diverter strips as opposed to two.

The radome is attached to the nose using NAS 8403 (10 – 32) screws, four of which participate in connecting the lightning diverters to the bonding straps.

The nose assembly includes a continuation of the lightning diverting function of the radome diverters. The radome lightning diverters are attached the nose assembly's bonding straps and carbon fibre structure in a lap joint using NAS 8403 (10 – 32) screws. The bonding straps are also attached to the nose assembly using two blind rivets. The bonding straps channel the lightning current into the copper foil of the nose whereupon it is spread through the nose structure and into the aluminum fuselage.

At each of the bonding strap locations, an added local "reinforcement" ply of expanded copper foil is applied to provide additional foil cross section to better manage the current density that could occur in diverting the energy from a lightning strike. The bonding strap in both instances will provide a significant "footprint" for the conduction of lightning currents into the underlying expanded foil and carbon fibre structure.

3. Adjustment/Test

A. Radome Bonding Test

Measure the resistance between the radome diverter strips and the airframe using a milliohm meter, resistance to be less than 10 milliohms. Probe measurements to be taken on the radome diverter strip and a corresponding airframe side bonding strap attachment rivet. Repeat for each diverter strip (quantity 4 places).

B. Radome Bonding Adjustment/Rework

Remove the radome diverter strip to bonding strap attachment screw and clean as required to achieve bonding requirement. If required, remove the radome and clean the mating surface of the nose assembly bonding strap in accordance with standard procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data.

4. Removal and Installation

A. Removal of Radome

1) Remove radome fasteners and remove the radome from the forward nose assembly.

B. Installation of Radome

1) Set radome in position and install fasteners.

2) Ensure disconnected diverter strip to bonding strap screws are installed.

3) Verify electrical bonding of the radome to airframe is maintained. Refer to Para 3.A.

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Forward Baggage Compartment – General data

1. General

The forward baggage compartment is installed on forward surface of the fuselage nose structure.

2. Description

The forward baggage compartment is constructed using the outer piles of carbon prepreg, a 5/16 phenolic honeycomb core, and two inner piles of carbon prepreg. In order to provide electromagnetic shielding for electric and magnetic fields associated with lighting and high powered radio frequency transmitters, an expanded copper foil is applied as an outer layer to the carbon nose baggage compartment and door. The baggage compartment is attached to the fuselage FS 44.00 via a circumferential aluminum strap, riveted to the baggage compartment and fuselage structure. This joint is the principal path of lightning energy transfer between the baggage compartment and the fuselage.

3. Adjustment/Test

A. Forward Baggage Compartment Joint Bonding Test

1) Measure the resistance between one of the forward baggage compartment diverter strip bonding strap attachment rivets and the airframe using a milliohm meter, resistance to be less than 10 milliohms.

2) Probe measurements to be taken on one of the diverter strips bonding strap attachment rivets and an airframe side rivet. Refer to Figure 1.

B. Forward Baggage Compartment Joint Bonding Adjustment/Rework

1) Remove the forward baggage door hinge-to-airframe bonding strap and clean as required in accordance with 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General data to achieve bonding requirement.

4. Removal and Installation

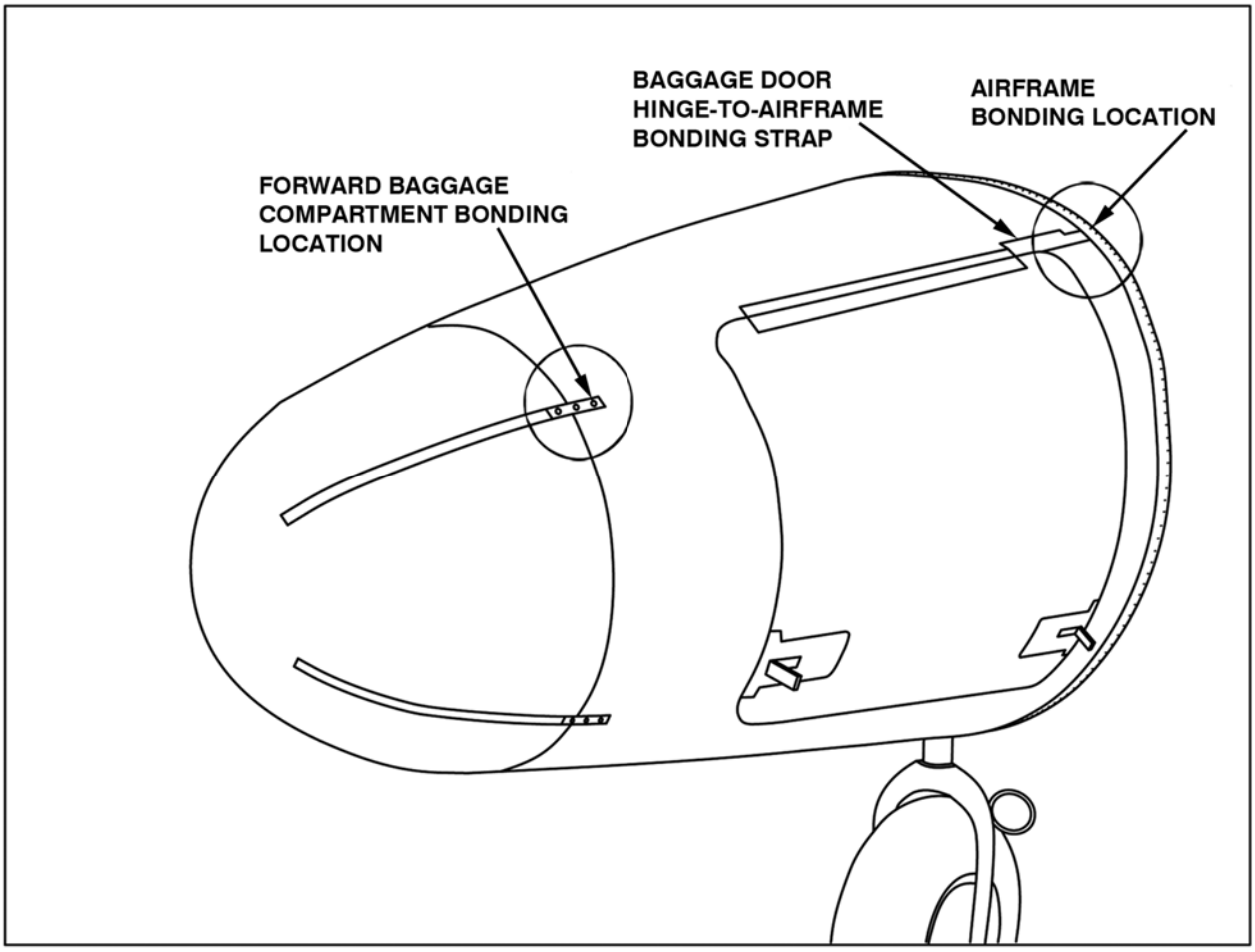
A. Removal of Forward Baggage Compartment

1) Remove circumferential joint fasteners and remove the forward baggage compartment from the forward fuselage.

B. Installation of Forward Baggage Compartment

1) Set forward baggage compartment in position and install fasteners.

2) Verify electrical bonding of the forward baggage compartment to airframe is maintained. Refer to Para 3.A.



Forward Baggage Compartment Joint – Electrical Bonding Check Locations
Figure 1

Floors (Mod 6/2027) (Mod S.O.O. 6244) (Mod S.O.O. 6398) – General Data

1. General

The floors consist of nine panels in the cabin compartment, skin panels and a plywood center panel in the flight compartment, and skin panels in the rear baggage/equipment compartment. The coin matting covers the flight compartment and cabin floor.

A. Cabin Floor

The cabin floor panels are each constructed from sandwich panels with upper and lower aluminum facing sheets with an aluminum honeycomb core. Panel edges are sealed with an epoxy-based potting compound. The panels are supported by transverse beams. The top surface of the panels is coated with light olive urethane resin to form a non-skid surface. Transverse joints of the floor panels are secured by stud rails and straps, and longitudinal joints by seat rail and track assemblies.

B. Flight Compartment Floor

The flight compartment floor consists of aluminum alloy skin panels riveted to the floor support structure, and a plywood center panel which is sealed and coated with Interpon epoxy paint.

C. Coin Mat

The coin mat provides a vinyl, nonslip, skid resistant floor covering for the flight compartment and cabin floor panels.

2. Removal/Installation

A. Remove Cabin Floor Panels

1) Remove passenger seats and coin mats as per Para 2.C. Remove panel assemblies around bulkhead at station 110.00 for access to bulkhead assembly. Remove screws and temporarily move bulkhead assembly to gain access for removal of screws attaching right and left-hand seat rail and track assemblies at station 110.00.

2) Remove screws attaching right and left-hand seat rails to floor support structure aft of station 262.00.

3) Lift right and left-hand seat rails complete with aft floor panels sufficiently to disengage center floor panel from seat rails.

Note When lifting seat rails and floor panels, wedge right and left-hand floor panels, if necessary.

4) Remove seat rails and right and left-hand floor panels.

5) Remove screws attaching right and left-hand seat rail and track assemblies to floor support structure forward of station 262.00.

6) Lift seat rails complete with floor panels, wedge right and left-hand floor panels, if necessary, and disengage and remove two center panels from seat rails.

7) Remove seat rails and two right and left-hand floor panels.

B. Install Cabin Floor Panels

1) Check cork packing and sponge rubber strips on floor support structure for condition and adhesion.

2) Position right-hand, center and left-hand stud rails at station 180.75. Assemble two right-hand panels and seat rail and track assembly, and position under angle on right side aircraft structure. Assemble two left-hand panels and seat and track assembly and position under angle on left side aircraft structure.

3) Lift seat rails and floor panels, wedge floor panels if necessary, and engage two center panels in seat rails. Push seat rails and panels downward into position.

4) Install screws securing seat rail and track assemblies to floor support structure.

5) Position right-hand, center and left-hand stud rails at station 262.00. Position right-hand aft floor panel under angle and step plate on right side aircraft structure and install seat rail. Position left-hand aft floor panel under angle and step plate on left side aircraft structure and install seat rail.

6) Lift seat rails and floor panels, wedge floor panels if necessary, and engage center panel in seat rail. Push seat rails and floor panels downward into position.

7) Install screws securing seat rails to floor support structure.

8) Position bulkhead assembly at station 110.00 and install screws. Install coin mats as per Para 2.D and passenger seats.

C. Coin Mat Removal

1) Remove screws and hot air duct fairing assemblies at station 110.0 and station 239.92 left and right side of aircraft.

2) Remove screws and left, center, right straps from stud rails and transverse joints at station 180.75 and 262.0. Remove screws and strap from transverse joint at station 110.0 (cabin to cockpit doorway).

3) Remove screws and step plates at airstair door and emergency exit door.

4) Remove sealant around seat tracks and coin mat track cut outs.

5) Roll cabin coin mat either aft to fwd or fwd to aft.

6) Remove cabin and cockpit coin mats from aircraft.

7) Remove tape from cabin floor and clean any residue.

8) Clean any tape residue from bottom of coin mat with mild methanol solution or Loncare cleaning solution (S.O.O. 6244) or with mild isopropyl solution (S.O.O. 6398).

D. Coin Mat Installation

1) Apply carpet tape to cabin floor, inset from perimeter. Ensure a non-permanent tape is used. Ensure taped area is clean and grease free.

2) Position respective coin mat in cockpit and cabin.

3) Do not remove release liner from tape. Unroll cabin mat ensuring positioning correct along bulkheads and seat rail cutouts, place cockpit mat in cockpit.

4) Once correct cabin coin mat positioning has been confirmed roll coin mat aft to fwd. Remove release liner from tape and unroll cabin mat.

5) Install screws and step plates at airstair door and emergency exit door.

6) Install left, center, right straps and screws to stud rails and transverse joints at station 110.0, 180.75, and 262.0.

7) Install screws and hot air duct fairing assemblies at station 110.0 and 239.92 on left and right sides of aircraft.

8) Apply PR1422B–1/2 or non-acidic RVT sealant between coin mat cut outs and seat tracks, clean any excess.

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CHAPTER 54

NACELLES /PYLONS

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CHAPTER 54 – NACELLES/PYLONS

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Composite Nacelle Fairings – General data

1. General

The composite nacelle fairings are installed on bottom surface of the wing aft of the main engine nacelle structure.

2. Description

The composite nacelle fairings are attached to the aluminum airframe using perimeter 10 - 32 screws spaced approximately every 3 inches. The fairings are attached to the metal firewall on the forward edge and to the lower wing surface on the sides and aft edge. Expanded foil is applied from the forward to the aft end of the fairings so that it is captured by the mounting screws. The foil stops just short of the side flanges for manufacturing reasons. No conductive path to the airframe is provided along the side attachment fasteners. The forward and aft fastener locations are considered the principal lightning energy transfer paths. A solid aluminum strip is bonded into the aft edge of the fairing to assist in energy conduction.

3. Removal/Installation

Refer to Figure 1 for details.

A. Removal of Composite Nacelle Fairings

1) Remove the screws from the strut-to-nacelle.

2) Pull the strut-to-nacelle fairing away from the nacelle fairing.

3) Remove the screws from the fairing that attach the fairing to the wing.

4) Remove the screws from the nacelle fairing.

5) Remove the nacelle fairing from the main engine nacelle structure.

B. Installation of Composite Nacelle Fairings

1) Position the nacelle fairing with the main engine nacelle structure.

2) Install the screws on the nacelle fairing and the fairing.

Note For aircraft incorporating Mod 6/2340, the fairings have rubber washers installed, torque the screws to compress the rubber washers (which are underneath the metal washers) to approximately one-half of their original thickness. For fairings without rubber washers (Pre Mod 6/2340) torque the #10 screws to 10 – 16 in–lbs.

3) Position the strut-to-nacelle fairing on the nacelle fairing.

4) Install the screws on the strut-to-nacelle fairing.

Note For aircraft incorporating Mod 6/2340, the fairings have rubber washers installed, torque the screws to compress the rubber washers (which are underneath the metal washers) to approximately one-half of their original thickness. For fairings without rubber washers (Pre Mod 6/2340) torque the #10 screws to 10 – 16 in–lbs.

5) Do a bonding test (refer to Para 4.).

4. Adjustment/Test

Refer to Figure 2 for details.

A. Composite Nacelle Fairing Bonding Test

1) With a milliohm meter, measure the resistance between the nacelle fairing and the airframe. Resistance must be less than 10 milliohms.

Note Probe measurements to be taken on one of the strut-to-nacelle fairing screws or nut-plate fasteners and an airframe side blind rivet.

2) If the measurement is more than the bonding requirement of less than 10 milliohms, do the Bonding Adjustment/Rework (refer to Para 4.B).

B. Composite Nacelle Bonding Adjustment/Rework

1) Remove the nacelle fairing forward and aft attachment screws and clean the screws as required to achieve bonding requirement.

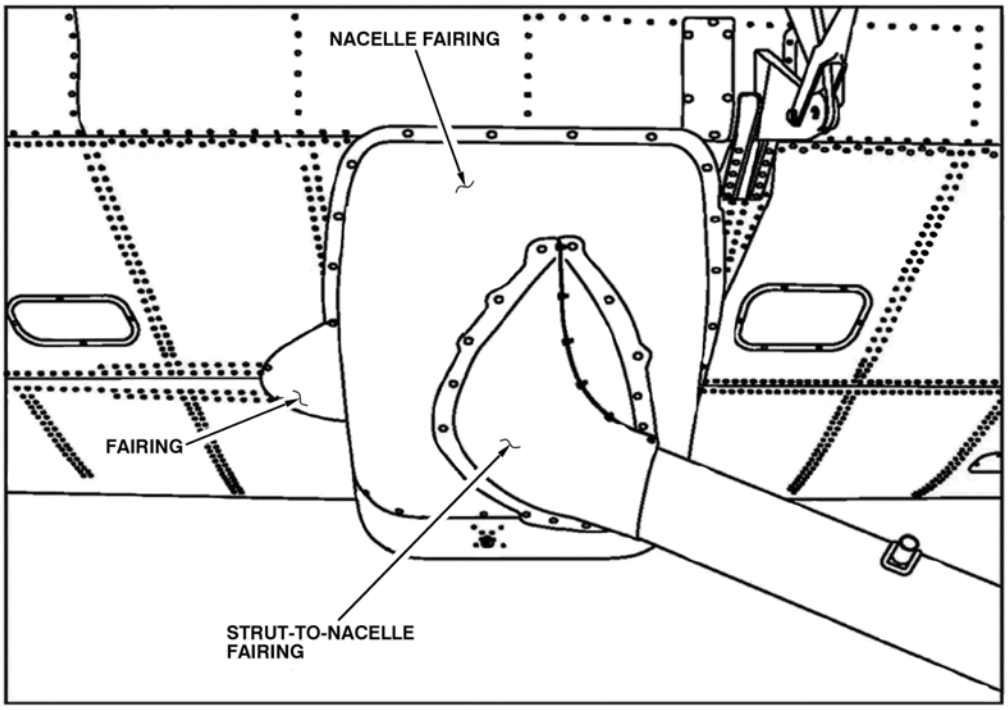
2) Remove the strut-to-nacelle fairing screws and clean the screws as required to achieve bonding requirement.

3) If required, remove the composite nacelle fairing (refer to Para 3.A).

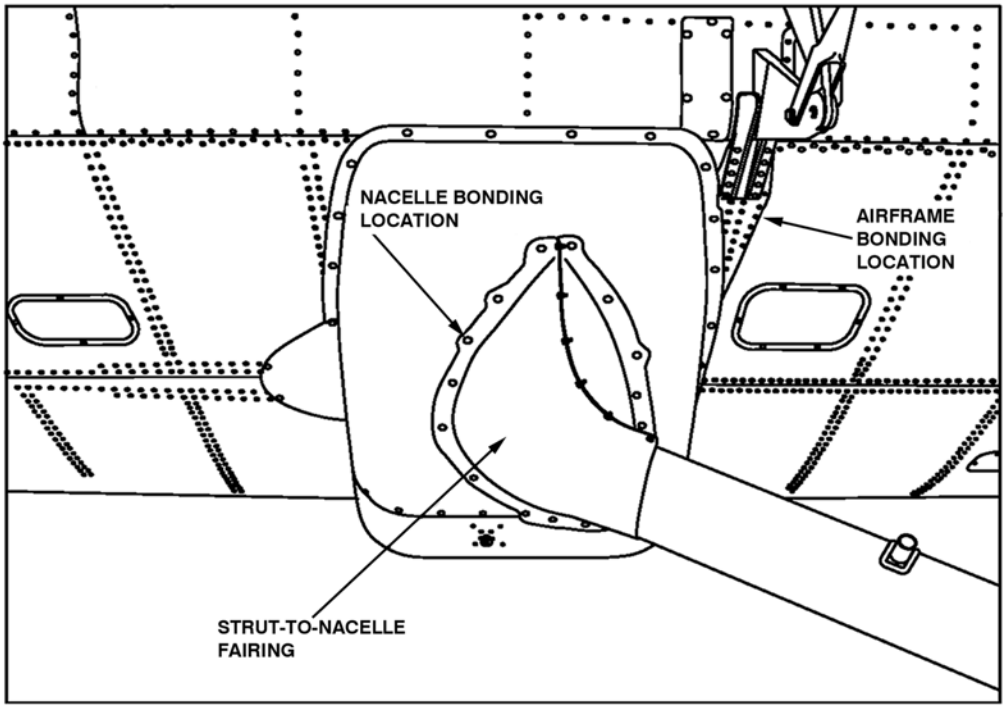
4) Clean the aft mating surface of the nacelle aluminum strip.

5) If removed, install the composite nacelle fairing (refer to Para 3.B).

Note For reference, see applicable standard electrical bonding procedures in 20-75-00, Electrical Bonding and Grounding of Aircraft Structures – General Data.



Composite Nacelle Fairing – Removal/Installation
Figure 1



Composite Nacelle Fairing – Bonding Locations
Figure 2

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CHAPTER 55

STABILIZERS

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CHAPTER 55 – STABILIZERS

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Stabilizers – General Data

1. General

The stabilizers consist of horizontal and vertical stabilizers, elevators and a rudder. The horizontal and vertical stabilizers are both secured by front and rear fittings to corresponding fittings in the rear fuselage structure.

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Horizontal Stabilizer – General Data

1. General

The horizontal stabilizer is housed in a recess in the vertical stabilizer structure, and is secured by four bolts to the rear fuselage structure. A spacing block is inserted between the forward attachments to provide the correct degree of incidence.

The horizontal stabilizer comprises front and rear spars, and ribs reinforced by spanwise stringers. Hinge plates are provided at two places on each side for attachment of elevator hinge brackets, and a central mounting on the rear of the stabilizer supports the elevator torque tubes. As a customer option, de-icer boots may be fitted to the leading edge of the stabilizer.

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Horizontal Stabilizer

1) Remove vertical stabilizer. (Refer to 55-30-00, Vertical Stabilizer – General Data).

2) Remove elevators. (Refer to 27-30-11, Elevator – Maintenance Practices). Coil and stow loose cables.

3) If de-icer boots are fitted, disconnect pipe lines.

4) Attach sling SD12508 to horizontal stabilizer and take weight. (Refer to 07-00-00, Lifting and Shoring – General Data).

5) Remove four bolts, saddle washers and washers at front and rear attachment fittings of horizontal stabilizer to rear fuselage.

6) Lift horizontal stabilizer clear of aircraft. Remove spacing block, four barrel nuts and sleeves from rear fuselage attachment fittings.

B. Install Horizontal Stabilizer

1) Position spacing block at front attachment fittings on rear fuselage.

2) Attach sling and hoist horizontal stabilizer into position at front and rear attachment fittings on rear fuselage. Ensure spacing block lines up with front attachment fittings.

3) Position four sleeves in fuselage attachment fittings, insert four attachment bolts with saddle washers and washers, and check that plain shank of each bolt is approximately flush with inside surface of each sleeve. If necessary, add washer under existing washer on bolt to achieve this condition.

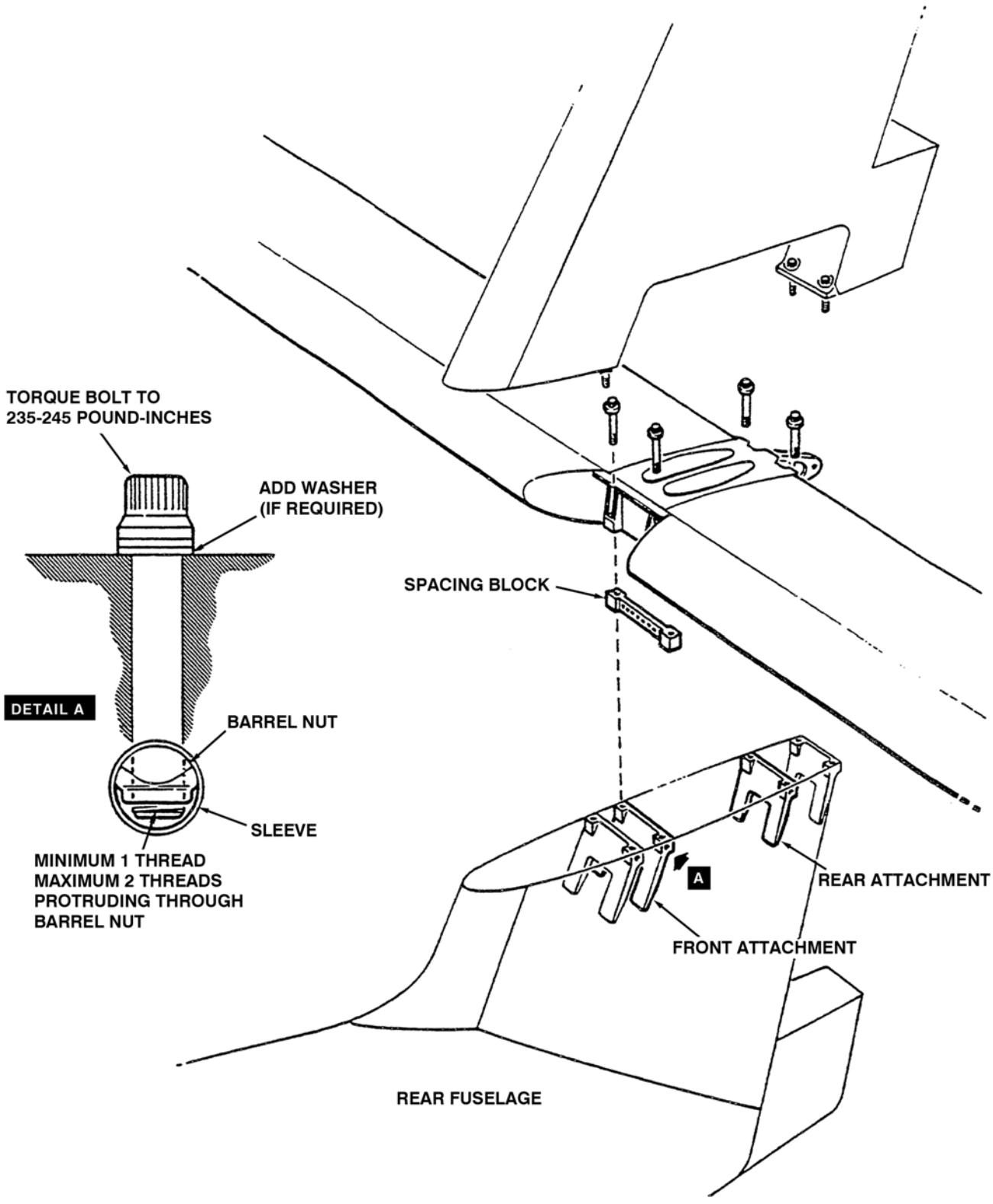
4) Withdraw bolts sufficiently to install barrel nuts in sleeves. Tighten bolts to 235 to 245 pound–inches. Check bolt thread protruding through barrel nut is between one thread minimum and two threads maximum.

5) Remove sling from horizontal stabilizer.

6) If de-icer boots are fitted connect pipe lines.

7) Install elevators. (Refer to 27-30-11, Elevator – Maintenance Practices).

8) Install vertical stabilizer. (Refer to 55-30-00, Vertical Stabilizer – General Data).



Horizontal Stabilizer Installation
Figure 1

Elevators – General Data

1. General

The elevators each comprise a front and rear spar supported by ribs, nose spar and nose ribs, and skin panels. Two hinge attachments are mounted on the front spar, and a third is located on the flange on the end of the torque tube. When elevators are installed, the torque tube flanges, when aligned and bolted together, also provide maximum up and down travel stops. The left elevator carries the elevator trim tab and the right elevator carries the flap/elevator interconnect tab. Refer to 27-30-00, Elevator Control System – General Data and 27-30-11, Elevator – Maintenance Practices, for details on removal, installation, and rigging of the elevator.

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Vertical Stabilizer – General Data

1. General

The vertical stabilizer is secured by four bolts at front and rear attachment fittings to corresponding fittings on the rear fuselage.

The vertical stabilizer comprises spars, ribs, and skins reinforced by stringers. The front and rear attachment fittings are riveted to the bottom of the front and rear spars. The top and center hinge plates for rudder attachment are riveted to the ribs. A detachable cap on top of the stabilizer houses the VOR antenna and the anti-collision light. On aircraft with lightning protection (Mod 6/1272), a horn assembly is attached to the cap, and protection straps and a conduit extension assembly are fitted between the cap and stabilizer structure. Structural provision has been made on each side of the vertical stabilizer for the installation of a half loop type VOR antenna (Mod 6/2003). On aircraft with Mod 6/2171 incorporated, two KMG 7010 magnetometers are mounted in the vertical stabilizer, one for each channel of the ADAHRS. (Refer to 34-20-00, Attitude and Direction – General Data).

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Vertical Stabilizer

1) Remove fairings enclosing vertical stabilizer and horizontal stabilizer to rear fuselage.

2) Remove rudder. (Refer to 27-20-11, Rudder – Maintenance Practices).

3) Disconnect electrical wiring at disconnect in forward end of vertical stabilizer.

4) Disconnect VOR antenna wiring in rear fuselage.

5) Disconnect ADAHRS magnetometers (Post Mod 6/2171).

6) Attach sling SD12507 to vertical stabilizer and take weight. (Refer to 07-00-00, Lifting and Shoring – General Data).

7) Remove four bolts, saddle washers and washers at front and rear attachment fittings of vertical stabilizer to rear fuselage.

8) Lift vertical stabilizer clear of aircraft. Remove four barrel nuts and barrel nut retainers from rear fuselage attachment fittings.

B. Install Vertical Stabilizer

1) Attach sling and position vertical stabilizer at front and rear attachment fittings on rear fuselage.

2) Install four bolts with saddle washers and washers at attachment fittings. Check that plain shank of each bolt is approximately flush with inside surface of fitting.

Note Maximum length of thread in fitting 0.10 inch. Maximum length of plain shank of bolt extending through fitting 0.10 inch. If necessary, add washers under existing washer on bolt.

3) Withdraw bolts sufficiently to install barrel nuts and barrel nut retainers. Tighten front and rear attachment bolts to 340 to 350 pound–inches.

4) Remove sling from stabilizer.

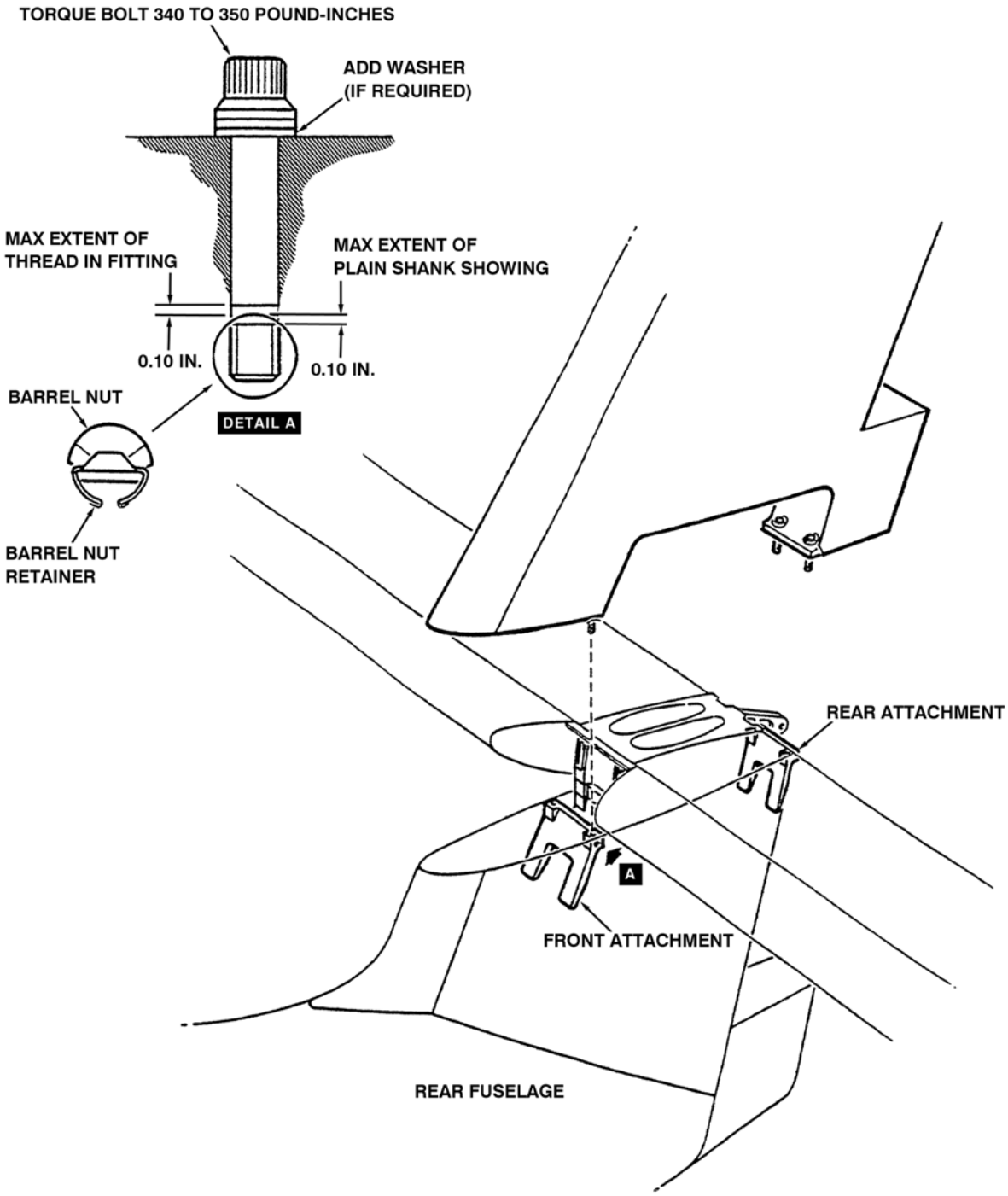
5) Connect VOR antenna wiring in rear fuselage.

6) Connect ADAHRS magnetometers (Post Mod 6/2171).

7) Connect electrical wiring at disconnect in forward end of vertical stabilizer.

8) Install rudder. (Refer to 27-20-11, Rudder – Maintenance Practices).

9) Install fairings enclosing vertical stabilizer and horizontal stabilizer to rear fuselage.



Vertical Stabilizer Installation
Figure 1

Rudder – General Data

1. General

The rudder is hinged to the vertical stabilizer and rear fuselage at three points. A lever attached to the rudder main spar is connected to the rudder control cables. Two tabs are hinged to the trailing spar, the upper being the trim tab and the lower tab being the geared tab.

The rudder comprises a main spar, rear spar, nose spar, ribs, and skin panels. Fluted skin panels are fitted aft of the rudder main spar.

Refer to 27-20-00, Rudder Control System – General Data and 27-20-11, Rudder – Maintenance Practices for removal, installation, and rigging of the rudder details.

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CHAPTER 56

WINDOWS

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CHAPTER 56 – WINDOWS

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

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Windows – General Data

1. General

The windows comprise pilot's and co-pilot's windshields, a fixed window and sliding window on each flight compartment door, passenger windows in the cabin compartment, and cabin door windows. As a customer option, window shade tracks and window shades are fitted to the cabin window reveal (surround).

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Flight Compartment Windshields – General Data

1. General

Basic pilot's and co-pilot's windshields Mod 6/2032 embodies 3/8 inch thick acrylic plastic, bird impact resistant windshields.

Mod S.O.O. 6187 Mod 6/2045 embodies electrically heated, bird impact resistant windshields manufactured from laminates of glass and vinyl. Sensing and heating elements are incorporated within the laminations during manufacture.

Windshield heating control is achieved through a windshield heat control switch, circuit breakers, two temperature controllers, a heat control relay and associated wiring.

Both windshields are secured in the flight compartment windshield frames, against retaining angles, by retaining plates attached by screws. Sealing strips are located each side of the windshield between retaining angles and plates. During installation, the space between the edge of the windshield and the frame structure is filled with sealant (refer to 20-50-01, Weather Sealing – General Data) to prevent entry of moisture.

2. Removal/Installation

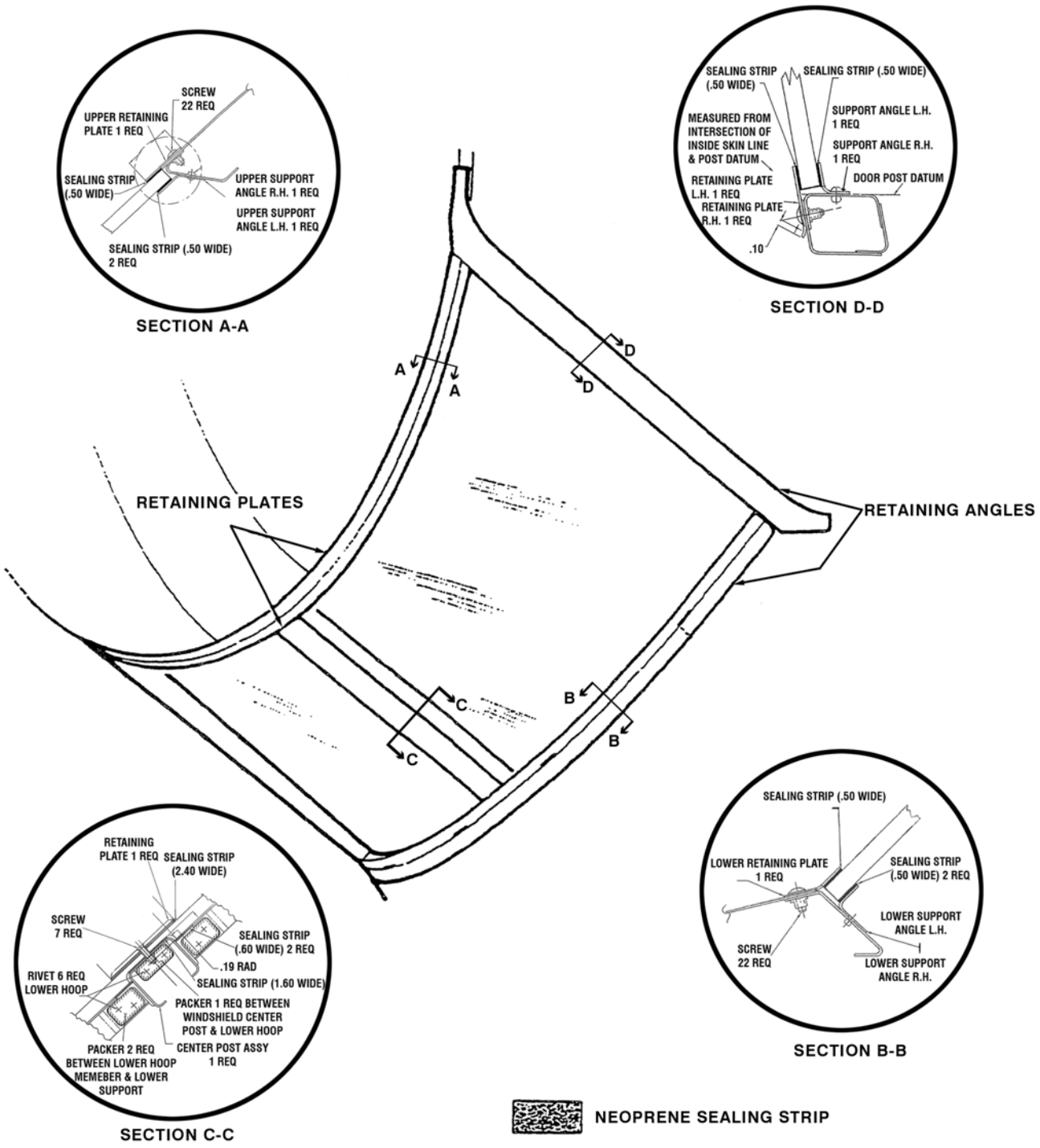
Refer to Figure 1 for details.

A. Remove Windshield

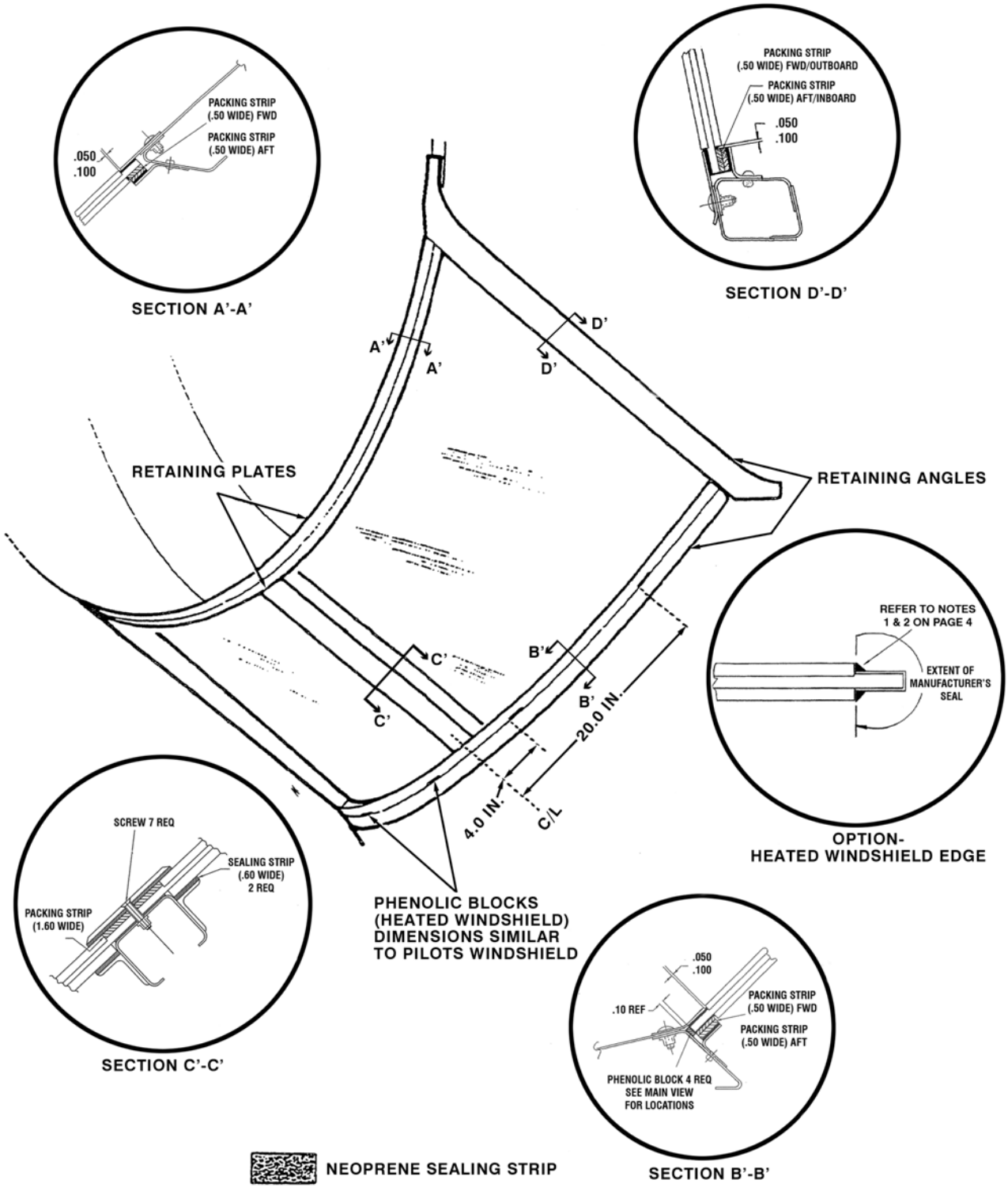
1) Disconnect windshield heat wiring at terminals (electrically heated windshield only, refer to 30-40-00, Windshield Heating System (Mod S.O.O. 6187 and Mod 6/2045) – General Data).

2) Remove screws securing retaining plates and angles.

3) Remove windshield. On electrically heated windshields only, mark position of the two phenolic blocks to facilitate reinstallation.



Windshield Installation – Base Installation (Mod 6/2032)
Figure 1



Windshield Installation – Option (S.O.O. 6187 Mod 6/2045)
Figure 2

B. Install Windshield

Note Electrically heated windshields are supplied with sealing strips and packing strips; standard windshields are supplied only with sealing strips.

1) On electrically heated windshield, check condition of sealant fillets on windshield.

Note 1. Prior to installation check that edge sealing as supplied by manufacturer is intact (refer to Figure 1 and Figure 2). If seal is broken, inspect to ensure no delamination has occurred and repair seal using PR1776M–B2 sealant.

Note 2. Prior to installation apply a fillet of PR1776M–B2 sealant to joint between glass and vinyl laminations (refer to Figure 1 and Figure 2), seal inside and outside joints around full perimeter of windshield and allow to cure for a minimum of 2 hours.

WARNING ADEQUATE VENTILATION MUST BE PROVIDED AND THE USE OF PROTECTIVE CLOTHING STRICTLY ADHERED TO AT ALL TIMES DUE TO THE TOXIC NATURE OF THE SOLVENTS.

2) Clean off any old sealant on windshield structure, retaining plates and retaining angles using a non-metallic scraper and naphtha.

3) On electrically heated windshields, position phenolic blocks at base of windshield structure.

4) Prepare adhesive and sealants as follows:

a) In a clean, wax free container, mix 10 parts by weight of Part A base PR1428–B2 with 100 parts by weight of Part B. The resulting sealant will have an application time of 2 hours and a full cure time of 36 hours at 75°F and 50% R.H.

b) In a clean, wax free container, mix 10 parts by weight of Part A base PR1776M–B2 with 100 parts by weight of Part B. The resulting sealant will have an application time of 2 hours and a full cure of 72 hours at 75°F and 50% R.H.

c) Thoroughly agitate the EC1357N adhesive in its own container, ensuring that all solids are dispersed by scraping the bottom of the container with a suitable tool.

5) For electrically heated windshields, bond pairs of packing strips, -55 and -57, -59 and -61, -63 and -65 together as follows using EC1357N:

a) Apply a thin uniform coat of adhesive to both bonding surfaces.

b) Allow to air dry until it is tacky, but does not transfer to the finger when touched lightly.

c) Press the parts together and roll down using a rubber or stitch roller to ensure intimate contact over the full bonding surface.

6) Apply two thin coats of release agent (parting Agent 13, Alpha/Owens Coming) to the faying surfaces of the windshield frame structure. Allow each coat to touch dry between coats and before assembly or applying sealant.

7) Apply PR1428–B2 sealant to windshield frame structure. Ensure that sufficient sealant is applied to fill gaps. Seal all gaps in excess of 0.10 inch with 2 plies of aluminum foil tape (3M 425 or Mystic 7453). Immediately install the windshield together with the sealing or packing strips.

8) Apply PR1428–B2 sealant to the retaining strips and angles and immediately install them ensuring that sufficient sealant is applied to seal all screw holes. Smooth extruded material into a fillet. Install screws.

9) Retighten screws after one hour and again after two hours. Sealant will be cured sufficiently after a further two hours to permit flight.

10) Peel or scrape off excess sealant. Do not use solvent near windshield seal.

11) On electrically heated windshields, connect electrical wiring. Function test windshield heating system (refer to 30-40-00, Windshield Heating System (Mod S.O.O. 6187 and Mod 6/2045) – General Data, Para 2.A).

12) Install windshield wiper arm and blade (if removed, refer to 30-45-12, Windshield Wiper Arm and Blade (Mod 6/2088) – Maintenance Practices, Para 2.B).

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Flight Compartment Heated Windshield (Mod S.O.O. 6187 and Mod 6/2045) – Maintenance Practices

1. Inspection/Check

   A. Delamination Check

      There are no technical limitations on delamination of the heated windshield. If delamination is observed, the serviceability of the windshield is dependent on the optical distortion caused by the delamination. The acceptance of optical distortion must remain with the pilot.

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Cabin Windows – General Data

1. General

The cabin windows are made of acrylic plastic, and are each mounted in a window frame between two seals and retained by a ring riveted to the window frames. The thickness of the window varies and the edges are shouldered or joggled, according to the position of the window in the cabin. As a customer option, inner liners are provided for each cabin window, and are a press fit into a ridge in the window reveal (surround).

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Cabin Window

1) Remove window trim panels, inner liners (if fitted) and window reveal (surround) with window shade and window shade tracks, if fitted.

2) Remove rivets securing retaining ring to window frame. Withdraw retaining ring and window.

B. Install Cabin Window

1) Check seals on retaining ring and window frame for condition and adhesion. Replacement seals to be bonded to retaining ring and/or window frame using Bostik 1142 (B.B. Chemical Co.).

2) Prepare PR1776–B2 sealant. (Refer to 20-50-01, Weather Sealing – General Data).

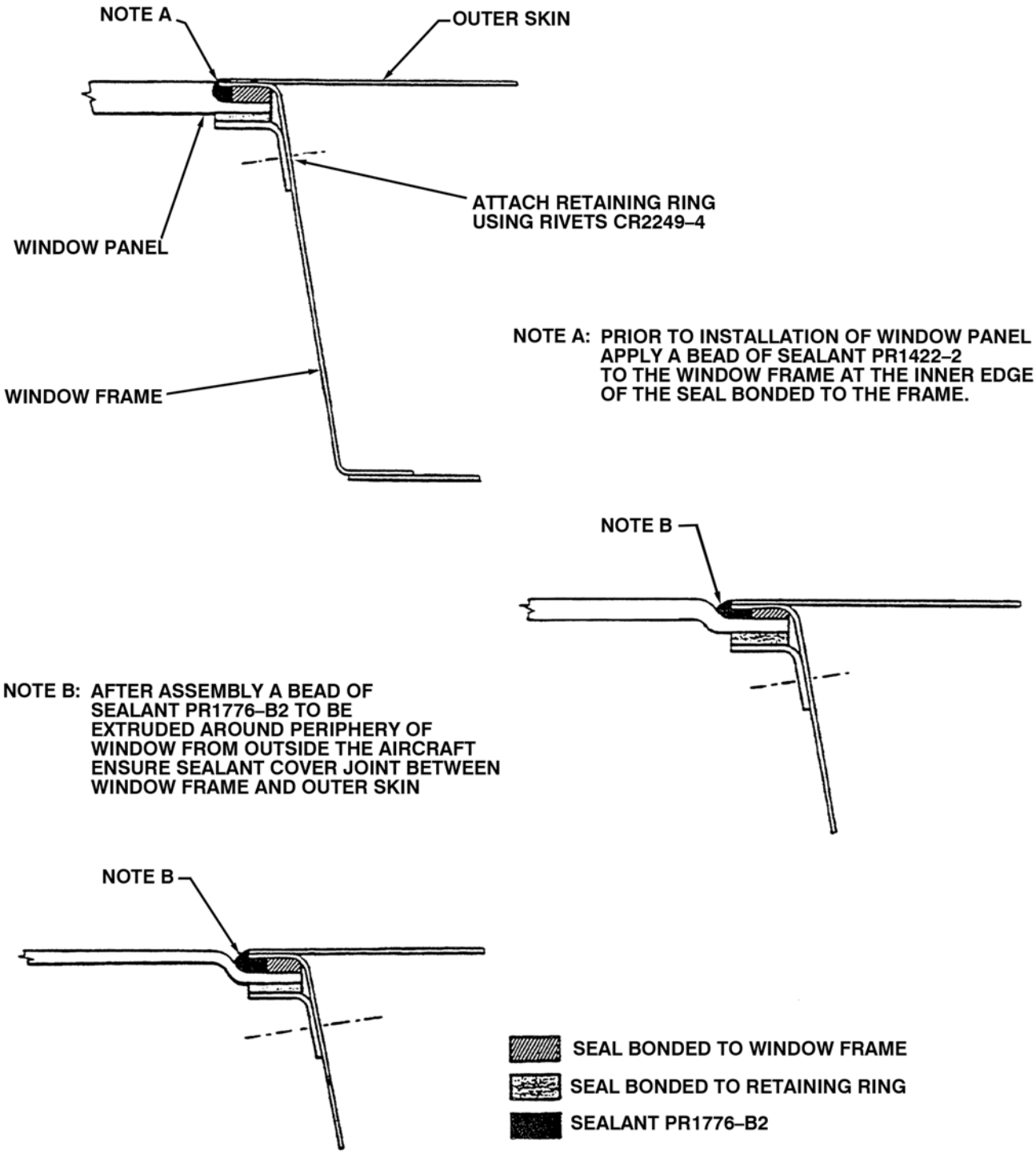
3) When installing a window with shouldered edge, apply a bead of sealant to the window frame at the inner edge of the seal bonded to the frame.

4) Install window and retaining ring using rivets CR2249–4.

5) On windows with shouldered edge, trim off any excess sealant flush with outer skin. On windows with joggled edge, apply a bead of PR1776–B2 sealant around periphery of window from outside the aircraft. Ensure that sealant covers joint between window frame and fuselage skin.

6) Install window reveal (surround) with window shade tracks and window shades, if removed, and trim panels.

7) Install inner liners (if removed) to window reveal (surround).



Cabin Window Installation
Figure 1

Door Windows – General Data

1. General

The door windows consist of flight compartment, cabin compartment and cabin escape door windows.

2. Flight Compartment

Each flight compartment door has a fixed window, and a sliding window panel. The sliding window is locked in any position by a cam type window latch.

3. Cabin Compartment

Windows are installed to the right-hand cabin door aft, left-hand cabin door aft, and to each escape door.

A. Left-hand and Right-hand Aft Cabin Doors

The windows in the left-hand and right-hand aft cabin doors are made of acrylic plastic, and are each mounted in a window frame between two seals and retained by a ring riveted to the window frame.

B. Cabin Escape Door

The window in each cabin escape door is the same as the windows installed in the left-hand and right-hand aft cabin doors.

4. Removal/Installation

Refer to Figure 1 for details.

A. Remove Right-Hand and Left-Hand Cabin Escape Door Windows

1) Remove window trim panels, inner liners (if fitted) and window reveal (surround) with window shade and window shade tracks, if fitted.

2) Remove rivets securing retaining ring to window frame. Withdraw retaining ring and window.

B. Install Right-Hand and Left-Hand Cabin Escape Door Windows

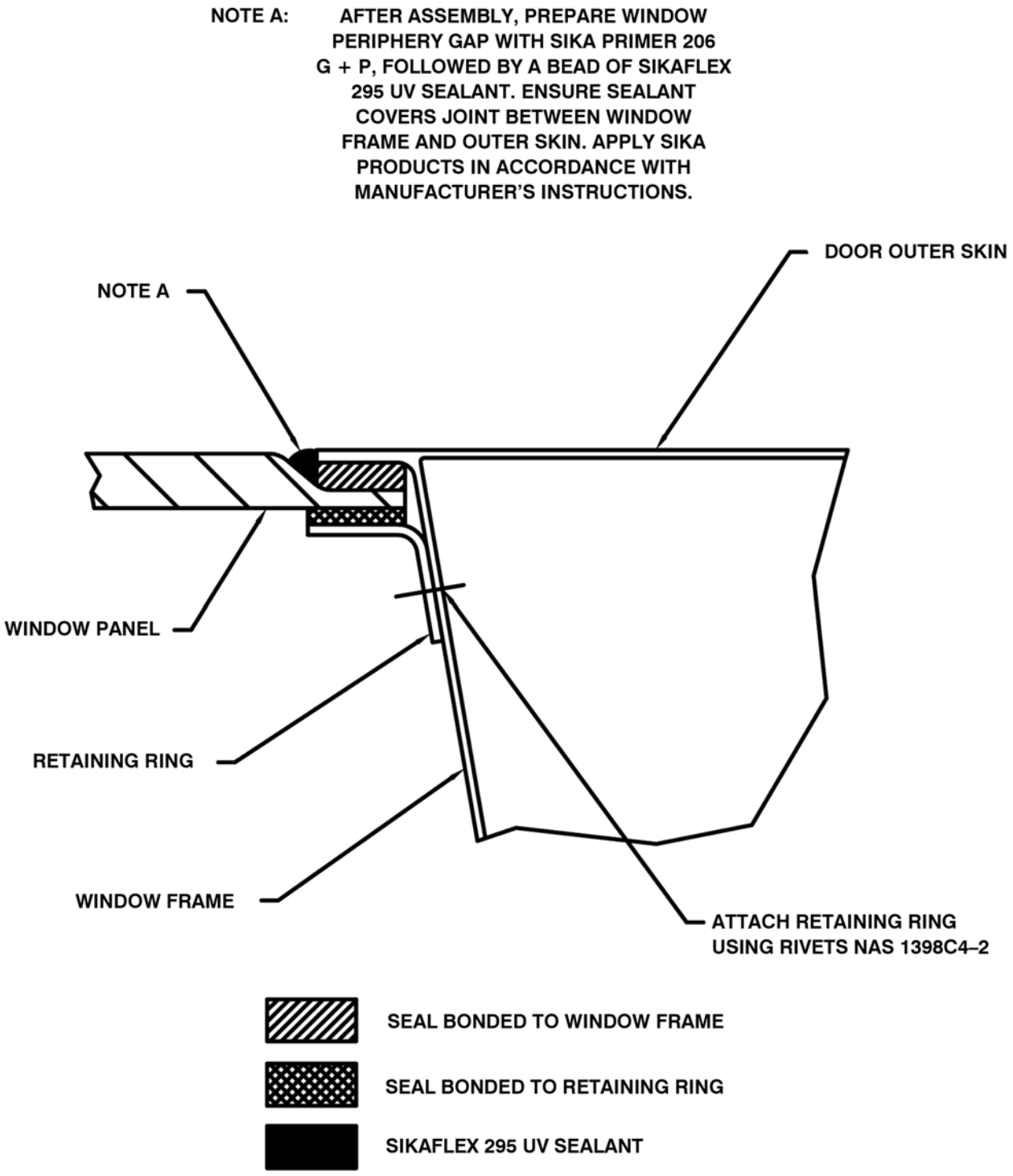
1) Check seals on retaining ring and window frame for condition and adhesion. Replacement seals to be bonded to retaining ring and/or window frame using Bostik 1142 (B.B. Chemical Co.).

2) Install window and retaining ring using rivets NAS1398C4–2. Wet install rivets using MIL–PRF–81733 TYPE II corrosion inhibitive sealant.

3) After assembly, prepare window periphery gap with SIKA PRIMER 206 G + P, followed by a bead of SIKAFLEX 295 UV Sealant. Ensure sealant covers joint between window frame and outer skin.

4) Install window reveal (surround) with window shade tracks and window shades, if removed, and trim panels.

5) Install inner liners (if removed) to window reveal (surround).



Cabin Door Window Installation
Figure 1

CHAPTER 57

WINGS

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CHAPTER 57 – WINGS

LIST OF EFFECTIVE PAGES

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Wings – General Data

1. General

The wing group consists of the wings, nacelles, wing struts and wing fences. A detachable wing tip is installed on each outer wing. The wings are provided with attachment fittings for the flap/aileron system at the trailing edge, and bleed air piping.

The wing structure consists of front, main, and rear spars, ribs, and skins reinforced with spanwise corrugations bonded to the skins. The nacelle structure is riveted to the top and bottom surfaces of the wing. A front and rear attachment fitting is provided on the root end of the wing, and a wing strut attachment fitting is located on the main spar. Hinge fittings for the wing flaps and aileron systems are carried at intervals along the rear spar, and a flap shroud is riveted to the spar. Nose ribs mounted forward from the front spar support the fixed leading edge skin which extends outboard from the nacelle structure. Inboard of the nacelle structure, the leading edge is hinged to provide access to engine control runs. Access panels in the bottom skin give access to the wing flap and aileron mechanism.

Aircraft incorporating Mod S.O.O. 6247 have integral fuel tanks installed in the leading edge of each wing. Each tank occupies three bays between the four outer leading edge ribs, forward of the main spar.

On aircraft with lightning protection (Mod 6/1272), a horn assembly is attached to each wing tip, and protection straps and a conduit assembly are installed between the wing tip and wing structure.

For aircraft incorporating Mod 6/2351, the wing tips have rubber washers installed. Torque the screws to compress the rubber washers (which are underneath the metal washers) to approximately one-half of their original thickness. For Pre Mod 6/2351 wing tips, without rubber washers, torque the #10 screws to 10 – 16 in–lbs.

The engine compressor wash (Mod S.O.O. 6050 or Mod 6/1316), a tube assembly with spray holes is installed in each nacelle structure.

2. Removal/Installation

Refer to Figure 1 for details.

A. Remove Wing (Mod 6/1752)

1) Remove wing root, wing strut and nacelle fairings. Raise wing inboard leading edge. Remove upholstery in cabin roof.

2) Trestle tail and opposite wing.

3) Remove engine. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

4) Disconnect five wing-to-fuselage electrical connectors. Disconnect generator cable assembly at power distribution box and remove clamps in cabin roof. Disconnect and tag remaining electrical wiring at wing-to-fuselage joint. Withdraw cable and wiring clear of wing joint.

5) Disconnect aileron cables at turnbuckles in cabin roof.

6) Disconnect fuel shut-off, throttle, and propeller pitch engine control cables at turnbuckles adjacent to wing nose spar.

7) If wing de-icer boots are fitted, loosen clamps and remove hose from two de-icing lines at wing root. Cap open connections.

8) Unbutton insulation blanket at bleed air line and check connector for freedom of movement. Loosen clamps and remove hose from fuel vent line. Cap all open connections.

9) Remove flap control rod seal at wing root. Remove cotter pin, nut, washer and bolt at flap control rod. Remove cotter pin, nut and washer attaching flap hinge arm to fitting on fuselage.

10) Attach sling SD12505 to wing and take weight on crane. (Refer to 07-00-00, Lifting and Shoring – General Data).

11) Disconnect flexible fuel line from elbow at top of wing strut. Remove oil breather pipe adjacent to wing strut, disconnecting pipe at wing strut on aircraft embodying Mod 6/1326. Cap open lines.

12) Remove cotter pin and slacken nut at wing strut to fuselage attachment. If Mod S.O.O. 6247 is embodied, withdraw wing tank system electrical wiring from wing strut. Support wing strut, remove cotter pin, nut, washer, and bolt at wing strut to wing attachment and lower wing strut to rest on ground.

    Note If required, wing strut may be removed completely by disconnecting flexible fuel pipe at elbow and removing nut, washer and bolt at wing strut/fuselage attachment.

13) Remove cotter pins, washers, pins and the rear spar wing-to-fuselage attachment pin at the rear wing/fuselage attachment and nut, washer, bolt and shims at front wing/fuselage attachment. Note position and thickness of shims used between front attachments fittings for reinstallation. Carefully move wing outboard and withdraw cables at joint. Ensure bleed air pipe is withdrawn from connector and flap hinge arm is released from fitting on fuselage.

14) Lower wing onto transportation cradle.

B. Install Wing (Mod 6/1752)

1) Attach sling SD12505 to wing and hoist wing into position (refer to 07-00-00, Lifting and Shoring – General Data). Ensure flap hinge arm engages fuselage fitting, and bleed air pipe slides into connector.

2) Apply light coating of corrosion preventive compound (refer to 20-30-01, Protective Coatings Hard and Soft Film Corrosion Preventive Compound – General Data), to plain portion of wing/fuselage attachment bolts. Ensure threads of bolt are dry.

3) Line up wing/fuselage attachment fittings and insert bolt using the front acorn SD12514. Insert shims at front attachment only, between fuselage beam and rear lug of wing fitting. Shims to be sufficient to take up end float with wing pushed back as far as possible, and a minimum gap of 0.010 inch at rear attachment between fuselage beam and front and rear lugs of wing fitting.

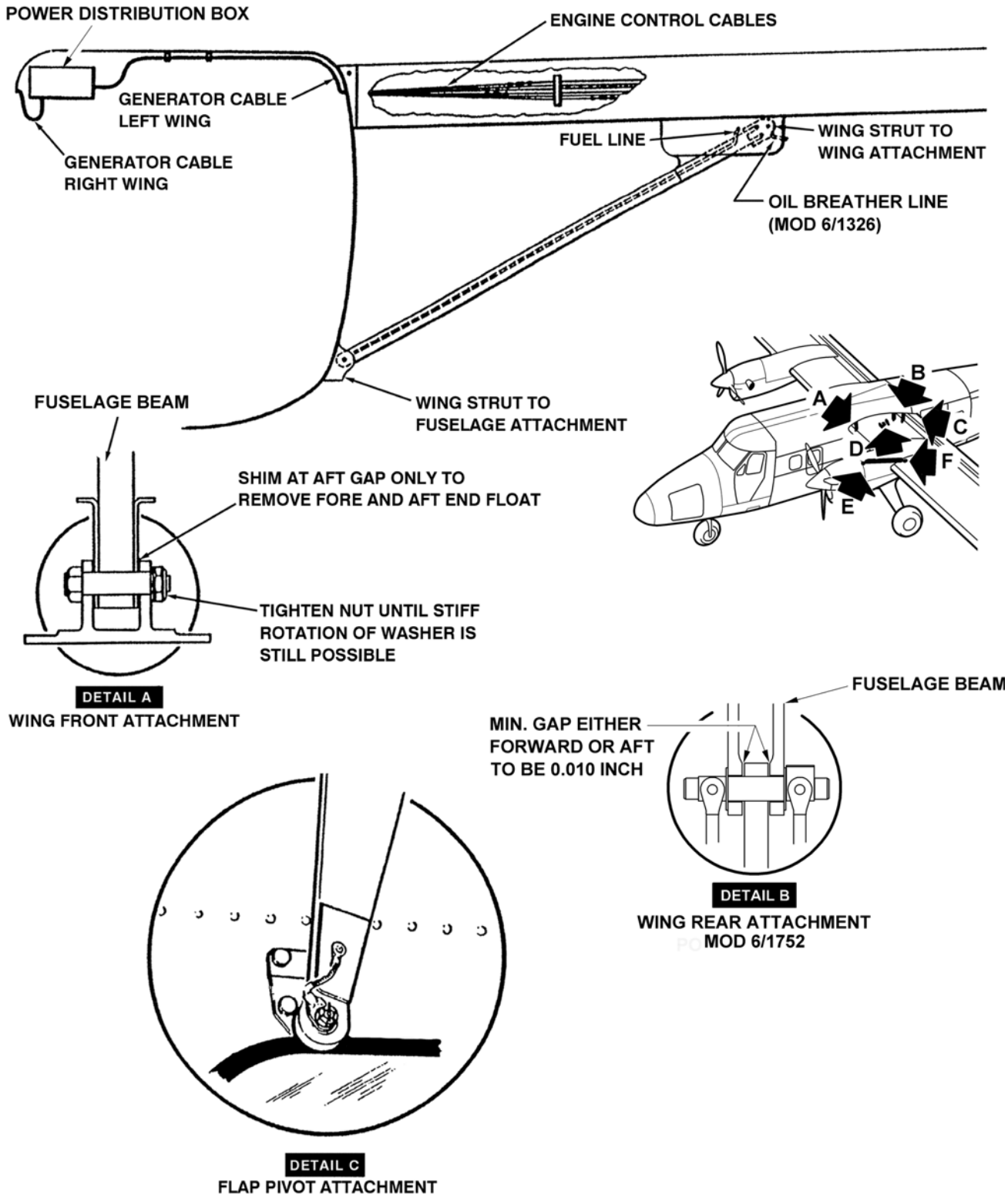
4) At front attachment bolt, ensure that washer installed below nut and painted face of adapter lug are both clean; tighten nut until stiff rotation of washer by fingers is still possible. Install the rear spar wing-to-fuselage attachment pins with washers on outside if fuselage fitting. Install pin, washers and cotter pins in both ends of rear spar wing-to-fuselage attachment pin.

5) Swing wing strut into position at wing attachment and install bolt, washer, nut and cotter pin. Tighten nut at wing strut/fuselage attachment and install cotter pin. If Mod S.O.O. 6247 is embodied, feed wing tank system electrical wiring through wing strut, and reconnect at bottom of strut.

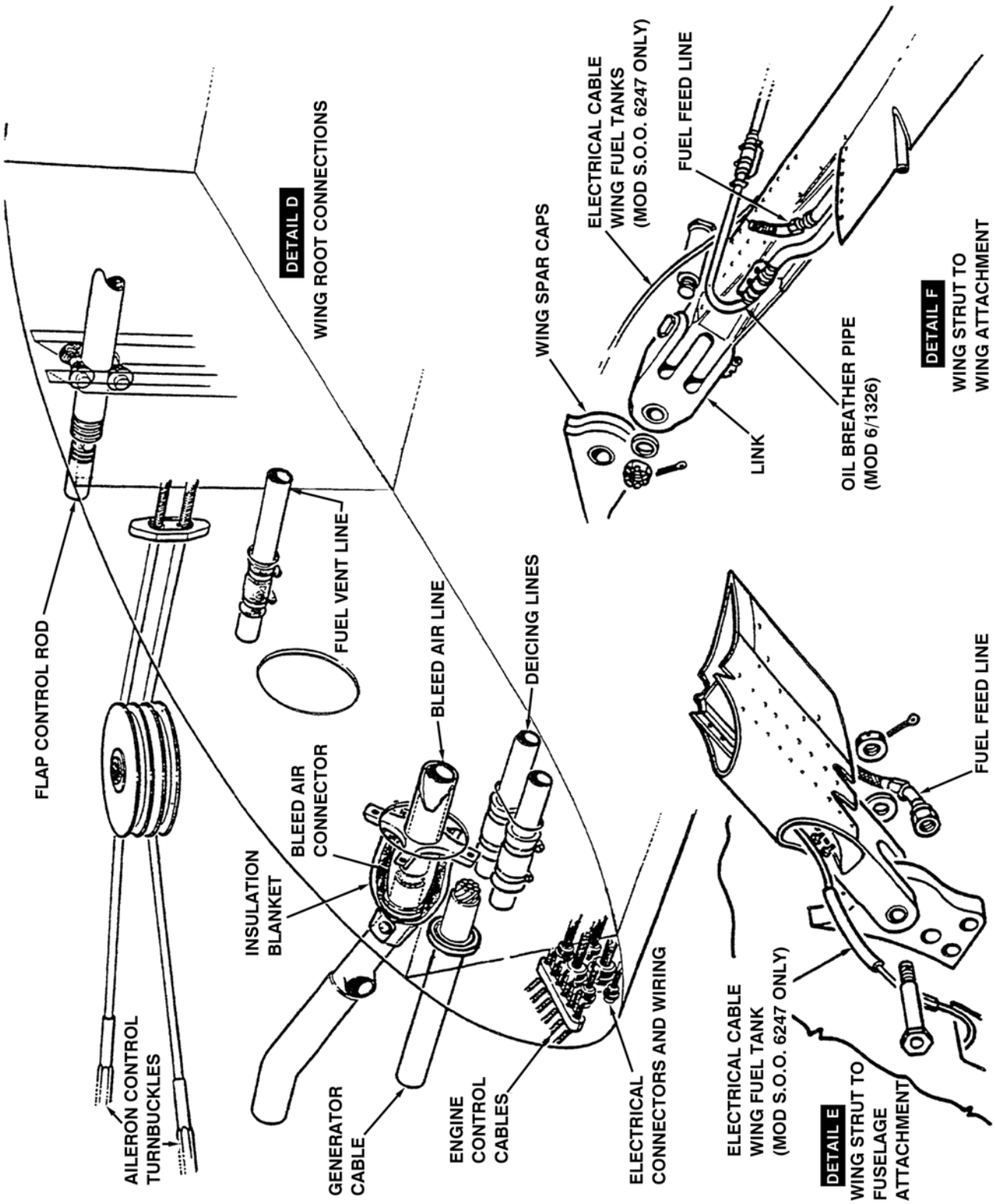
    Note If wing strut was removed completely, install bolt, washer, nut and cotter pin at wing strut/fuselage attachment. Connect flexible fuel line to elbow at bottom of strut.

6) Remove sling SD12505 from wing.

7) Connect flexible fuel line to elbow at top of wing strut. Install oil breather pipe adjacent to wing strut. Tighten clamps and secure bonding.



Wing Installation
Figure 1 (Sheet 1 of 2)



Wing Installation
Figure 1 (Sheet 2 of 2)

8) Engage flap push-pull rod connection and install bolt, washer, nut and cotter pin. Install washer, nut and cotter pin attaching flap hinge arm to fitting on fuselage. Install flap control rod seal at wing root.

9) Install hose connecting fuel vent line at wing root and tighten clamps. Check bleed air connector at wing joint, and button insulation blanket over connector. Examine fume seals and replace if unserviceable.

10) If wing de-icer boots are fitted, install hose and tighten clamps on two de-icing lines at wing root.

11) Connect turnbuckles of fuel shut-off, power and propeller control cables at wing nose spar.

12) Connect turnbuckles of aileron cables in cabin roof.

CAUTION **IF BOTH LEFT AND RIGHT AILERON CONTROLS HAVE BEEN DISCONNECTED, ENSURE THAT AILERON CABLES AT AILERON CONTROL QUADRANT IN CABIN ROOF ARE CORRECTLY INSTALLED. (REFER TO Figure 2, DETAIL B).**

13) Connect electrical wiring, wing-to-fuselage. Route and connect generator cable assembly to power distribution box and install clamps in cabin roof. Connect five wing-to-fuselage electrical connectors.

14) Install engine (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data). Rig fuel shut-off, power and propeller controls (refer to 76-10-00, Power Control – Maintenance Practices).

15) Remove tail and wing trestles.

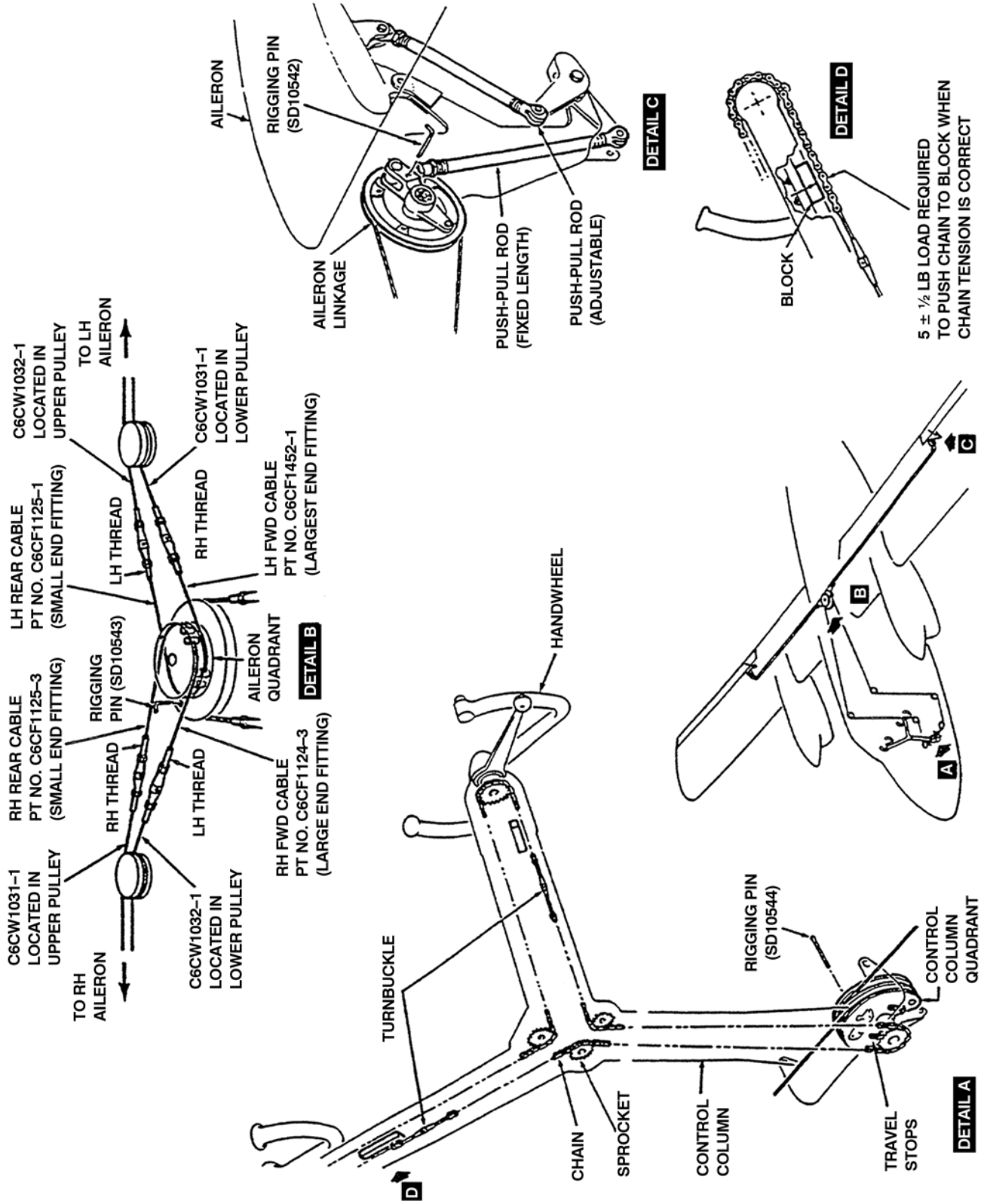
16) Check rigging of wing flaps, tension aileron control cables and rig ailerons and aileron tabs. (Refer to 27-10-00, Aileron Control System – General Data).

17) Function electrical systems to engine and wing affected by removal and installation. With booster pumps operating, check fuel lines for leaks.

18) Install upholstery in cabin roof. Lower and secure wing inboard leading edge. Install nacelle, wing strut and wing root fairings.

Note For aircraft incorporating Mod 6/2340, the fairings have rubber washers installed, torque the screws to compress the rubber washers (which are underneath the metal washers) to approximately one-half of their original thickness. For fairings without rubber washers (Pre Mod 6/2340) torque the #10 screws to 10 – 16 in–lbs.

19) During engine run, check bleed air system. (Refer to 21-40-11, Bleed Air Shut-off Valve – Maintenance Practices).



Aileron System – Rigging
Figure 2

CHAPTER 61

PROPELLERS

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CHAPTER 61 – PROPELLERS

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Propellers – General Data

1. General

Both the engines are equipped with a Hartzell constant speed, reverse pitch, fully feathering propeller, the operation of which is controlled in the constant speed and feather range by the propeller lever, and in the beta range by the engine power lever. The propeller (refer to 61-10-00, Propeller Assembly – General Data), is operated in one direction by engine oil, pressurized and controlled by a constant speed governor, which is delivered through the hollow propeller shaft. Counterweights and feather return springs operate the propeller in the opposite direction.

The propeller control system (refer to 61-20-00, Controlling – General Data), includes the constant speed and overspeed governors, and overspeed test switch.

Other controlling factors are the autofeather system and blade zero thrust stops if fitted.

Beta range indication is displayed through EFIS. Reset Props caution light will be displayed in CAS.

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Propeller Assembly – General Data

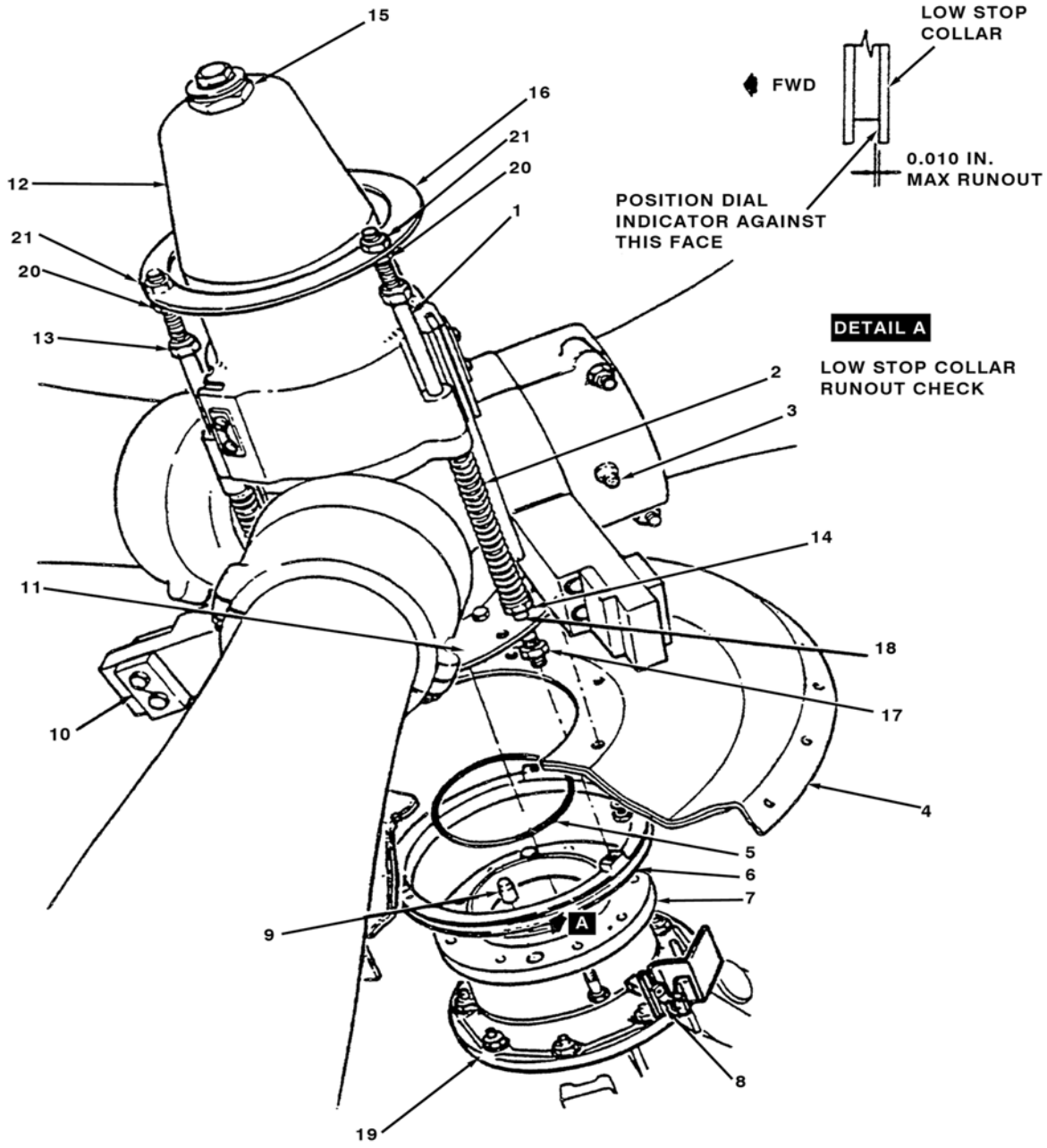
1. General

Refer to Figure 1 for details.

The propeller consists of a hollow spider hub which supports three propeller blades and houses the feather return springs. The propeller has a diameter of 8 feet 6 inches and a pitch range of 102 degrees, from –15 degrees reverse to +87 degrees feathered. When blade zero thrust stops are fitted, the zero stop setting is at +1 degree. The propeller is dowelled and bolted to the front face of the engine propeller shaft flange. Blade movement is controlled by a servo piston, mounted on the front of the propeller spider hub, which is connected by links to each blade root. During propeller operation, centrifugal counterweights attached to each blade, and the feather return springs in the spider hub, tend to move the blades into the high pitch or feather position, but this movement is opposed by oil pressure (controlled by the constant speed governor) acting on the piston. An increase in oil flow moves the blades towards the low pitch position (increased rpm), and relieving oil pressure allows the blades to move to the high pitch position, under the influence of the feather return springs and the blade counterweights. The piston is also connected to a low stop collar, mounted behind the propeller, by three spring loaded rods. Movement of the low stop collar in the beta range is relayed to a beta control valve (incorporated in the constant speed governor) by a carbon block and a propeller reversing lever, to control blade angles from the normal forward low pitch stop to full reverse.

Propeller blade zero thrust stop assemblies may be fitted by the propeller manufacturer for operation on floats. (Refer to 61-26-00, Blade Zero Thrust Stop System (S.O.O. 6203) – General Data).

On special order, a propeller de-icing system may be installed. (Refer to 61-10-11, Propeller De-icing without Pitch Latches (S.O.O. 6202) – General Data or 61-10-12, Propeller De-icing with Pitch Latches (S.O.O. 6237) – General Data).



1. LOW PITCH STOP ROD
2. REVERSE RETURN SPRING
3. LUBRICATION FITTING
4. SPINNER BULKHEAD
5. O-RING
6. LOW STOP COLLAR
7. ENGINE SHAFT FLANGE
8. MICROSWITCH SLIDE BLOCK
9. DOWEL PIN
10. COUNTERWEIGHT
11. SPINNER SUPPORT RING
12. PISTON UNIT
13. PITCH SETTING NUT
14. STOP ROD WRENCH FLATS
15. FLEXLOCK NUT
16. ROD END RING
17. LOCKNUT
18. ROD GUIDE
19. ENGINE REDUCTION GEAR CASE
20. ROD END RING REAR LOCKNUTS
21. ROD END RING FORWARD LOCKNUTS

Propeller Installation
Figure 1

2. Removal/Installation

Refer to Figure 1 for details.

Equipment Required:

- Sling SD5534
- Wrench SD12540
- Retraction Clamp SD12559
- Wrench SD12549
- Container to receive oil spillage

A. Remove Propeller

1) Remove spinner, nut and bolt securing teleflex control clevis rod end to propeller reversing lever, and pin from pivot at beta control valve clevis slot, retain spacer and bushing. Disengage and remove reversing lever complete with carbon block from low stop collar.

2) Attach sling (SD5534) to propeller (refer to 07-00-00, Lifting and Shoring – General Data) and support weight.

3) Position container to receive oil spillage.

4) If retraction clamp (SD12559) is to be used during removal, proceed as follows:

CAUTION DO NOT DISTURB PITCH SETTING NUTS (13) OR THE REAR ROD END RING LOCKNUTS, FOR THE ADJUSTMENT OF THE PITCH SETTING NUTS. REFER TO 71-00-00, POWER PLANT (MOD 6/2012) – GENERAL DATA ADJUSTMENT/TEST.

a) Remove the three forward locknuts securing rod end ring (16). Do not allow stop rods to rotate.

b) Remove rod end ring (16).

c) Using retraction clamp, screw clamp bolts onto ends of low pitch stop rods. Do not allow stop rods to rotate. Insert clamp socket over hexagon on piston unit ensuring that clamp bolts are evenly adjusted. Rotate center tee-bolt to pull low pitch stop rods and low stop collar forward to allow access to propeller retaining bolts.

d) Using wrench (SD12540) remove propeller retaining bolts and remove propeller. Retain the beta microswitch carbon block and slide which becomes detached from switch mounting as propeller is moved forward.

e) Remove propeller sling and retraction clamp.

f) Screw two of the three rear rod end ring locknuts further onto low stop rods; do not disturb third rear locknut. Install rod end ring (16) and secure with one locknut at the undisturbed locknut position. Turn the two rear rod end ring locknuts (previously screwed further onto rods) until they just contact the rear face of the rod end ring; install the two forward locknuts at these positions. Tighten all three forward locknuts. Do not allow stop rods to rotate.

5) If retraction clamp is not to be used during a propeller removal, proceed as follows:

a) Remove flexlock nut (15) using wrench (SD12549).

b) Rotate all blades simultaneously slowly to full reverse. Ensure locknuts do not foul guide rod lugs. Secure blades in this position.

c) Install sling SD5534 to support propeller.

d) Using wrench (SD12540) remove propeller retaining bolts and remove propeller. Retain the beta microswitch carbon block and slide which becomes detached from switch mounting as propeller is moved forward.

e) Remove propeller sling and install flexlock nut (15), ensuring O-ring seal is in position. Using wrench (SD12549) torque flexlock nut to 120 pound–feet.

B. Install Propeller

1) Attach sling (SD5534) to propeller and raise into position in front of engine shaft. (Refer to 07-00-00, Lifting and Shoring – General Data).

2) Ensure propeller and engine flanges are clean and free from scoring or burring which could prevent mating.

3) If retraction clamp (SD12559) is to be used, position low stop collar as detailed in Para 2.A.4).a), and as applicable Para 2.A.4).c) or Para 2.A.4).d).

4) If retraction clamp is not to be used, position low stop collar as detailed in Para 2.A.5).a) and Para 2.A.5).b).

5) Install propeller according to Hartzell Owner's Manual 139 (61–00–39) (latest revision) using wrench (SD12540), or (SD12513) if retraction clamp is not being used.

3. Inspection/Check

A. Check Propeller Low Stop Collar Runout

1) Ensure propeller is feathered.

2) Suitably mount a dial indicator on engine reduction gear case.

3) Rotate propeller slowly and check runout of working face of low stop collar.

Runout must not exceed 0.010 inch.

Note If runout exceeds 0.010 inch, the collar may be adjusted by slackening the low stop locknuts adjacent to collar, rotating the low stop rods to adjust insertion of rod in collar and retightening locknuts. Torque load locknuts to 12 pound–feet.

CAUTION IF THE LOW STOP RODS ARE ADJUSTED, THE PITCH SETTING NUTS MUST BE READJUSTED (REFER TO 71-00-00, POWER PLANT (MOD 6/2012) – GENERAL DATA, ADJUSTMENT/TEST).

Propeller De-icing without Pitch Latches (S.O.O. 6202) – General Data

1. General

Propeller de-icing system components installed on the propeller are; an electrically heated de-icing boot fitted to each blade, and a slip ring assembly. Each blade boot contains an inner and outer heater element which is connected electrically to the slip ring assembly mounted behind the spinner back plate on the propeller hub. An engine mounted brush block assembly makes contact with the slip ring to provide electrical power to the blade elements. The propeller de-icing system is fully described in Hartzell Propeller Ice Protection System Manual, manual # 180 (30–61–80).

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Propeller De-icing with Pitch Latches (S.O.O. 6237) – General data

1. General

Propeller de-icing system components installed on the propeller are; an electrically heated de-icing boot fitted to each blade, and a slip ring assembly. Each blade boot contains an inner and outer heater element which is connected electrically to the slip ring assembly mounted behind the spinner back plate on the propeller hub. An engine mounted brush block assembly makes contact with the slip ring to provide electrical power to the blade elements. The propeller de-icing system is fully described in Hartzell Propeller Ice Protection System Manual, manual # 180 (30–61–80).

Note This option is applicable for propeller with zero thrust stop.

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Controlling – General Data

1. General

The propeller is controlled by the propeller and engine power levers in conjunction with a constant speed governor, overspeed governor, and autofeather system. An overspeed governor test facility is provided for testing the overspeed governor on the ground. When aircraft operated on floats, the blade zero thrust stop system (S.O.O. 6203) needs to be fitted. A propeller tachometer indicating system and a Reset Props caution message is provided through the CAS.

2. Description

The constant speed governor combines the functions of a normal constant speed unit (CSU), beta range control valve, and fuel governor. As a CSU, it regulates power turbine speed by varying the pitch of the propeller to match the load torque to engine torque in response to varying conditions of flight. The beta range control valve controls the propeller blade angles in the beta range as selected by the engine power lever. The beta range is that segment in the overall pitch range of the propeller, which is directly controllable by the power lever; it extends from +17 degrees in the forward range to –15 degrees in the reverse range. During beta range operation, the propeller is kept in the under speeding condition as a function of airspeed. The fuel governor reset lever, which is linked to the power lever control operating the reversing lever, moves progressively to reset a lower maximum power turbine speed, relevant to the selected speed, to limit the power output of the gas generator, by restricting fuel flow to a value which prevents propeller speed exceeding a value 5% below selected propeller speed. The fuel governor section of the constant speed governor also protects the engine against propeller overspeeding, in the event of an overspeed governor failure during normal forward thrust operation, by reducing fuel flow.

The overspeed governor provides automatic control of a propeller overspeed condition by increasing the pitch of the propeller blades to absorb engine power, thereby decreasing propeller speed. The overspeed test switch, when operated on the ground, simulates an overspeed condition of the propeller by resetting the governor to a lower value.

Propeller feathering is accomplished by selecting the propeller lever to FEATHER.

A propeller lever/power lever interlock mechanism is fitted, which prevents movement of the power levers below IDLE, if both propeller levers are positioned at less than 91% propeller rpm. Individual operation of either propeller lever above 91% rpm disengages the interlock, to permit the power levers to be moved below IDLE.

3. Operation

A. Propeller On-speed

In the on-speed condition, forces acting on the engine, propeller and propeller constant speed governor are in balance. The pilot valve within the CSU section of the governor has closed off the oil ports to meter only sufficient oil to the propeller servo piston, to maintain blade pitch.

B. Propeller Overspeed

In an overspeed condition, the governor mechanism causes the pilot valve to lift and uncover oil ports so that oil acting on the propeller servo piston is allowed to return to the reduction gear sump. This results in the propeller piston moving rearwards under the combined influence of the blade counterweights and feather return springs, to turn the blades into a higher pitch. Increasing blade pitch decreases propeller speed and consequently governor speed, so that the pilot valve moves to close off the oil ports when an on-speed condition is reached. In the event of a propeller maximum overspeed condition occurring, the propeller overspeed governor mechanism lifts the overspeed governor pilot valve, to allow propeller servo piston pressure oil to be damped to the reduction gearbox, so that the blades can move to increase their pitch and so reduce propeller speed.

C. Propeller Underspeed

In an underspeed condition, the constant speed governor mechanism lowers the pilot valve and uncovers ports to allow pressure oil to be directed to the propeller servo piston, which moves forward and turns the blades into a lower pitch, so allowing propeller speed to increase till the on-speed condition is reached.

D. Reverse Thrust Operation

During reverse thrust operation when the power lever is moved into the REVERSE range, the reversing lever and consequently the beta control valve is moved rearwards. In this position the beta control valve allows pressure oil to act on the propeller servo piston to turn the blades into reverse pitch. As the propeller piston moves forward, the low stop collar also moves forward and operates the beta control valve to close off the oil supply, so that when the propeller blades reach the pitch determined by power lever selection, the oil supply to the servo piston is just sufficient to maintain a balance.

E. Propeller Feathering

When the propeller lever is selected to feather, the CSU pilot valve is lifted and oil pressure acting on the propeller servo piston is relieved. This permits the combined effect of the blade counterweights and the feather return springs, to turn the blades into the feathered position (a pitch of +87 degrees).

F. Propeller Unfeathering

To unfeather a propeller, the propeller lever must be moved to full INCREASE to reset the constant speed governor, which in effect moves the pilot valve to open the governor oil ports, so that when the engine is started, pressure oil from the governor pump is directed to the propeller servo piston to turn the blades into low pitch. As the propeller piston moves forward, the low stop collar also moves forward and operates the beta control valve to close off the oil supply, so that when the blades reach the pitch determined by power lever selection, the oil supply to the servo piston is sufficient to maintain a balance.

Constant Speed Governor – Maintenance Practices

1. Removal/Installation

   A. Remove Constant Speed Governor

      1) Gain access to constant speed governor mounted on top of the engine reduction gearbox.

      2) Disconnect control from governor pitch lever.

      3) Disconnect sense tube at governor.

      4) Remove bolt, washer and nut connecting fuel governor lever to interconnecting rod.

      5) Remove bolt, spacer and nut connecting power lever-operated flexible control fork end to propeller reversing lever.

      6) Remove clevis pin from fork end of beta control valve.

      7) Remove nuts and washers and withdraw governor from mounting studs. Remove gasket.

   B. Install Constant Speed Governor

      1) Examine replacement constant speed governor for signs of external damage.

      2) Install gasket over studs on governor mounting pad. Ensure that raised side of screen faces upward.

      3) Install governor and secure with four washers and self-locking nuts. Torque load nuts to between 170 to 190 pound–inches.

      4) Connect fork end of beta control valve to propeller reversing lever with clevis pin. Secure with washer and cotter pin.

      5) Secure fork end of power lever operated flexible control to reversing lever with bolt, washer and nut. Secure nut with cotter pin.

      6) Connect fuel governor lever to interconnecting rod (inner hole) with bolt, washer and nut. Secure nut with cotter pin.

      7) Connect sense tube to governor. Tighten and lockwire coupling nut.

      8) Adjust propeller pitch and engine power lever-operated controls as necessary (refer to 76-10-00, Power Control – Maintenance Practices).

      9) Carry out engine ground run and complete propeller function checks (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

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Overspeed Governor – Maintenance Practices

1. Removal/Installation

   A. Remove Overspeed Governor

      1) Gain access to overspeed governor mounted on left side of engine reduction gearbox.

      2) Disconnect overspeed test and autofeather solenoid electrical connections.

      3) Remove nuts and washers securing overspeed governor to reduction gearbox, and withdraw overspeed governor from mounting studs. Discard mounting gasket.

      4) Remove overspeed governor drive, examine for damage and wear, if satisfactory retain for installation.

   B. Install Overspeed Governor

      1) Examine replacement governor for signs of external damage.

      2) Install new mounting gasket over studs on overspeed governor mounting pad.

      3) Install a serviceable overspeed governor drive shaft, and ensure engagement with engine drive.

      4) Install overspeed governor and secure with washers and self-locking nuts. Torque nuts to between 170 to 190 pound–inches.

      5) Connect overspeed test and autofeather solenoid electrical connections.

      6) Carry out engine run and complete propeller functional checks. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

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Propeller Autofeather System (Mod 6/2022) – General Data

1. General

Automatic propeller system is provided which automatically feathers the propeller of an underpowered engine when a decrease in torque to below approximately 20 psi is detected. Subsequent autofeathering of the other propeller is prevented by a blocking relay which disarms the autofeather system after the first propeller has feathered. The system is armed for operation when the following three conditions have been met:

1 AUTO FEATHER switch is selected ON

2 Torque has risen above a specific threshold on both engines, and;

3 The power levers are advanced beyond 86 – 88% N_G

Pulling either or both back below 86 – 88% N_G (for example, in the event of a rejected take-off) disarms the autofeather system. Two autofeather indicator lights illuminate to signify when the system is selected on and when it is armed. An autofeather test switch is provided to permit a ground check of the autofeather system. The autofeather system is powered from the left DC bus and is protected by a PROPELLER AUTOFEATHER circuit breaker on the main circuit breaker panel.

A two-second delay between the detection of low torque and automatic feathering of the propeller is provided. This allows propeller discing to be used in the event of a rejected take-off due to a loss of torque. Dual circuitry is used on all autofeather sensors and relays to reduce the probability of a single system failure causing an unwanted autofeathering.

2. Description

The system comprises of two propeller autofeather push button switch, autofeather test switch, indication lights, two power lever-operated microswitches, two arming light relays, two bleed resistors, two time delay relays, and a torque pressure switch and feathering solenoid valve installed on each engine. The system obtains power through a 5–ampere circuit breaker from the left main DC bus.

A. Propeller Autofeather Switch

Two identical propeller autofeather switches are provided, located outboard of each pilot's primary flight display. When activated (accomplished by pressing either switch, selected autofeather armed) electrical power is supplied to the autofeather system as indicated by the illumination of SELECT on two switches. The dual switch design eliminates the need for the right seat pilot to stretch across the instrument panel to operate the switch.

The two switches operate in the same manner as a household three-way switch system that is found at the top and bottom of a flight of stairs. Pressing either switch once changes the state of the system, from ON to OFF or from OFF to ON as appropriate.

B. Autofeather Test Switch

The propeller autofeather test switch is located on the left sub panel and is a guarded two-position momentary-on switch labelled AUTOFEATHER. The switch is spring loaded to the center off position and has an unmarked test (up) position. The test switch permits a ground test of the autofeather system with engines operating at a low speed, by bypassing the power lever-operated microswitches.

C. Power Lever-Operated Microswitches

Two microswitches are installed on a slide assembly located in the cabin roof. A slide, attached to each engine power lever cable, operates the appropriate microswitch when the power levers are advanced. Electrically both microswitches are connected in series.

D. Arming Light Relay

Two arming light relays are installed in an autofeather relay box in the cabin roof. When both relays are energized, by operation of the left and right torque pressure switches, they connect electrical power to the SELECTED AUTOFEATHER ARMED lights on the pilot's flight instrument panel.

E. Time Delay Relay

Two time delay relays, one for each engine, are installed in the autofeather relay box in the cabin roof. When energized, the time delay relay, after a time delay of two seconds, connects electrical power to its respective feather solenoid valve.

F. Feathering Solenoid Valve

A solenoid-operated valve is installed on each engine's overspeed governor. When operated electrically, the valve opens to allow oil pressure, acting on the pitch change mechanism of the propeller installed on that engine, to be relieved, so causing the propeller to feather.

G. Torque Pressure Switch

A double pole, low pressure torque switch is connected to the engine torque transmitter on each engine. The switch completes an electrical circuit to its respective time delay relay, if the torque pressure of its engine falls below 11 to 13 psi. The switch also controls its respective arming light relay when it operates on increasing torque pressure at 17 psi.

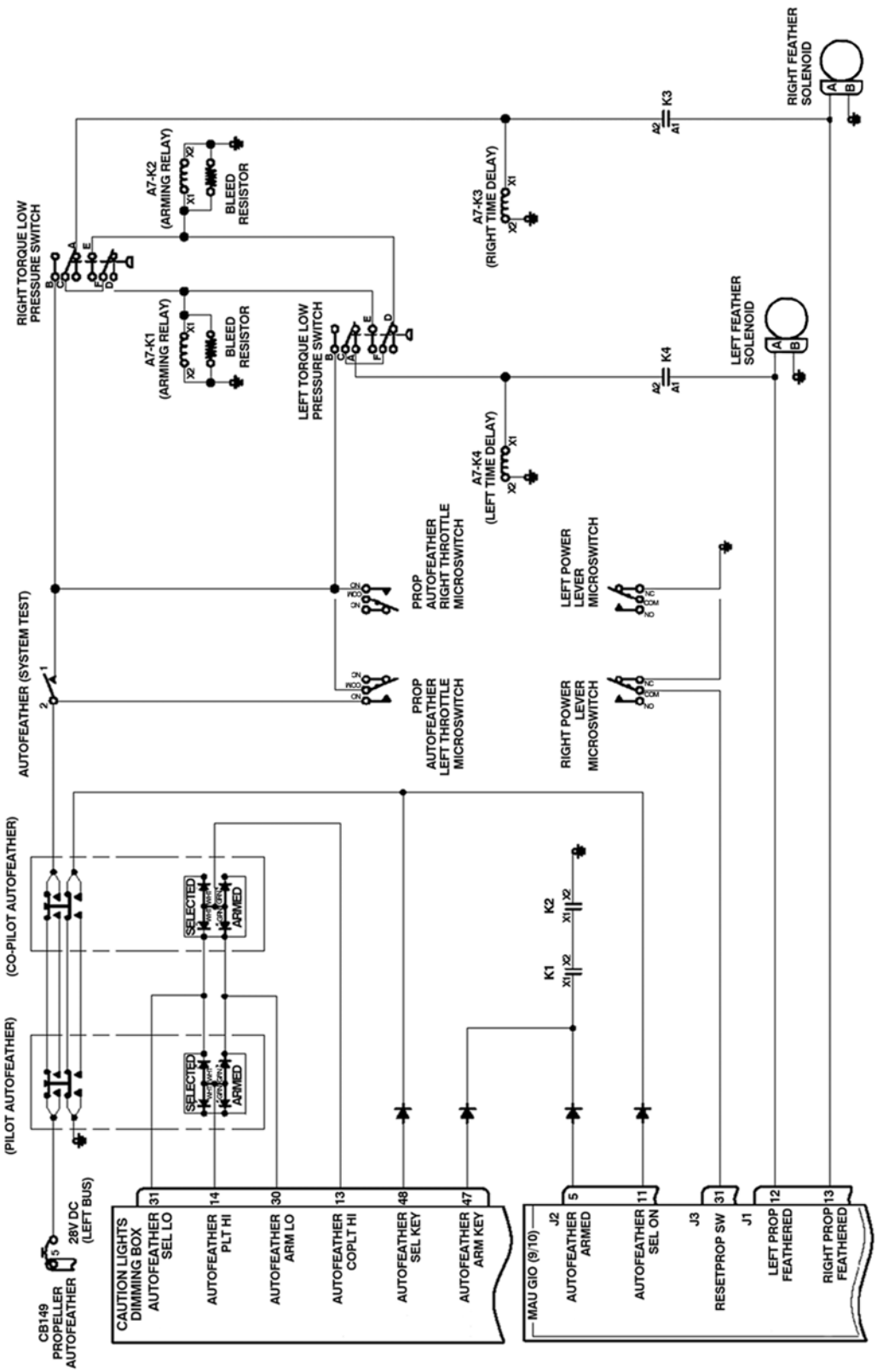
3. Operation

Refer to Figure 1 for details.

The propeller autofeather indicator lights are integral with the autofeather switches. The indicators consist of two rectangular lights, one above the other, in each switch; the upper light is labelled SELECTED and when illuminated indicates that the system has been selected on. The lower light is labelled ARMED and when illuminated indicates that the autofeather system is armed for operation. The ARMED indicator light illuminates when the autofeather is on and both power levers are advanced to 86 – 88% N_G or beyond; it goes out if either or both power levers are retarded and/or when automatic feathering of a propeller has occurred. The autofeather indicator lights are included in the caution lights test and intensity switch circuit. To comply with contemporary guidelines for the use of colour in switches and indicators, the SELECTED light is white and the ARMED light is green.

With the PROPELLER AUTOFEATHER circuit breaker engaged and the SELECTED AUTOFEATHER ARMED switch selected to ON, the selected white SELECTED indicator light illuminates. If both engine power levers are advanced to obtain 86 to 88% gas generator rpm (N_G), the two power lever-operated microswitches operate and electrical supply is provided to both torque pressure switches. Increasing torque pressure (by 17 psi) in each engine operates the torque pressure switches to energize the arming light relays K1 (by right torque pressure switch) and K2 (by left torque pressure switch) which cause the ARMED indicator green light to illuminate. In the event of the torque pressure of one engine falling to below 11 to 13 psi, when both power levers are set above 86% N_G, the torque pressure switch of that engine closes to complete a circuit to its respective time delay relay (K4 for the left engine or K3 for the right engine), which after a delay of approximately two seconds, energizes the feather solenoid valve of that engine resulting in the propeller being feathered. The closing of the torque switch also de-energizes one of the arming light relays and the ARMED indicator light goes out.

If with one propeller feathered, the torque pressure of the opposite engine should fall below 11 to 13 psi, the torque pressure switch of that engine will close and disarm the autofeather system. The feather solenoid valve of the feathered propeller is de-energized and the propeller will unfeather.



Propeller Autofeather – Electrical Schematic
Figure 1

4. Adjustment/Test

Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data, Para 10.B.12) for Adjustment/Test of the Autofeather System.

Autofeather Relay Box Assembly – General Data

1. Description

The Autofeather Relay Box Assembly forms part of the autofeather system, and consists of an aluminum alloy box housing miniature relays. Electrical connection to the box is via a multipin connector. The control box is located in the cabin roof, forward of station 219.525.

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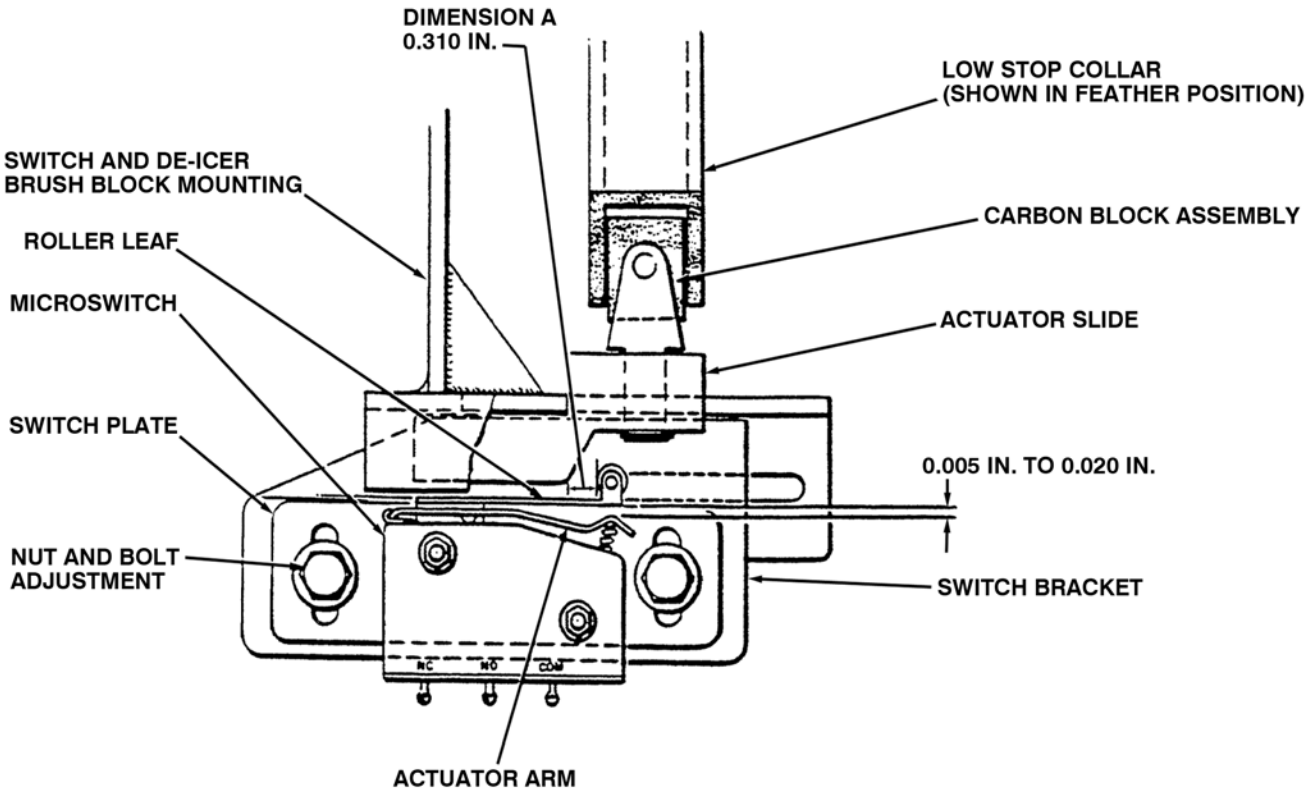
Beta Microswitch Assembly (Mod 6/2022) – General Data

1. General

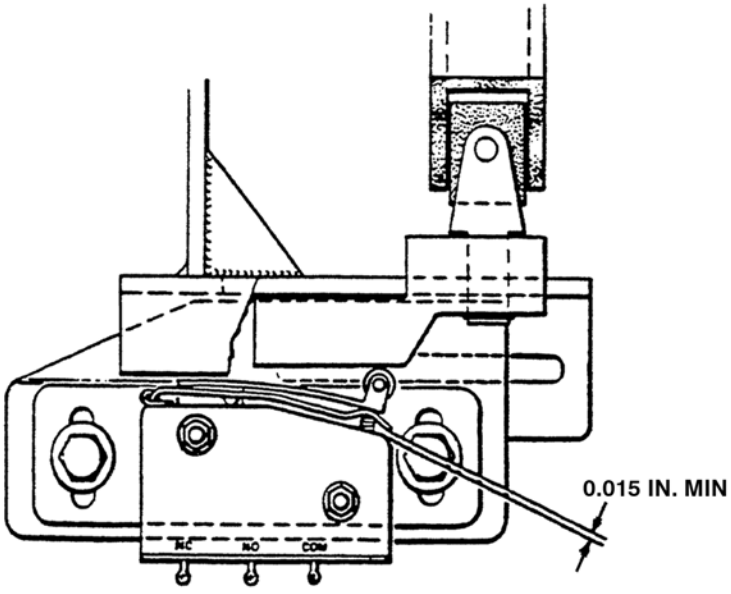
Refer to Figure 1 for details.

The beta microswitch assembly comprises a switch and de-icer brush block mounting, switch bracket, switch plate, microswitch, actuator slide, and carbon block assembly. The switch and de-icer brush block mounting, secured to the engine reduction gear case by two nuts, incorporates a slot in which operates the actuator slide, and two mounting faces. During manufacture, the mounting and actuator slide are treated with a dry film lubrication process (Electrofilm), making it unnecessary to lubricate these parts in service. One of the mounting faces supports the propeller de-icing brush block assembly (when fitted refer to 30-60-11, Propeller De-icing Brush Module Assembly (Mod 6/1674) – Maintenance Practices), and the other the switch bracket. The switch bracket, secured to the mounting with two bolts, washers and anchor nuts, has a lengthwise slot, and the switch plate, bolted to the switch bracket, has two slots across its width. The slot in the switch bracket allows the microswitch operating position to be adjusted, and the slots in the switch plate permit correct microswitch engagement with actuator slide.

During operation the carbon block assembly, actuated by the propeller low stop collar, moves the actuator slide which operates the microswitch when the propeller blades reach an angle of +9 degrees. The microswitch contacts are normally open above, or closed below +9 degrees. A propeller blade movement of 1 degree moves the actuator slide approximately 0.030 inch, during the range in which the low stop collar operates.



WITH PROPELLER BLADES AT ZERO PITCH AND SWITCH OPERATED, CHECK THAT AN OVERTRAVEL, CHECK THAT AN OVERTRAVEL CLEARANCE OF 0.015 IN. MIN EXISTS BETWEEN ROLLER AND SLIDE



Beta Microswitch Assembly
Figure 1

2. Removal/Installation

A. Remove Beta Microswitch

1) Gain access to front of engine reduction gear case.

2) Remove and retain the two bolts and washers securing switch plate to switch bracket.

3) Disconnect microswitch wires at disconnect splices.

4) Remove the two nuts, screws and washers securing microswitch to switch plate. Retain nuts, screws, washers and switch plate.

B. Install Beta Microswitch

1) Secure microswitch to switch plate, using plate, screws, washers and nuts retained during microswitch removal.

2) Connect microswitch wires at disconnect splices.

3) Locate switch plate, complete with microswitch, on switch bracket and attach using bolts and washers retained during microswitch removal. A washer should be located under the bolt heads and under each nut. Do not fully tighten bolts at this stage.

4) Adjust microswitch; refer to Para 3.A.

5) Secure access panels as necessary.

C. Remove Beta Microswitch and Propeller De-icing Brush Block Assembly

1) Remove propeller de-icing brush block (refer to 30-60-11, Propeller De-icing Brush Module Assembly (Mod 6/1674) – Maintenance Practices), if fitted.

2) Disconnect microswitch wires at disconnect splices.

3) This may be achieved as in Para 2.C.3).a) or Para 2.C.3).b) as follows:

a) Using retraction clamp (SD12559) as in propeller installation (refer to 61-00-00, Propellers – General Data).

Note When using the retraction clamp, strictly observe the procedures regarding installation and use.

b) Turn the propeller blades into reverse pitch and secure in this position.

Note To facilitate turning of blades, remove propeller piston flexlock nut, using wrench (SD12549).

4) Remove the two nuts securing switch and de-icer brush block mounting to reduction gearbox. Remove mounting, complete with actuator slide, carbon block, switch bracket, switch plate and microswitch. Retain nuts.

D. Install Beta Microswitch and Propeller De-icing Brush Block Assembly

1) Before installation, check that the actuator slide moves freely in slot in switch bracket. If a new carbon block is being fitted, the following points should be noted:

a) No tight spots exist when sliding block around groove in low stop collar. If necessary, lightly sand block to give a clearance of 0.001 to 0.002 inch at tightest point.

b) Lacquer has been removed from stem of block.

c) Cotter pins retaining carbon block in axle are tightly fitted and trimmed to prevent fouling on low stop collar.

d) Ensure carbon block swivels freely in actuator slide; lightly lubricate carbon block stem and each end of axle with one drop of engine oil.

2) Turn propeller blades into reverse pitch.

This may be achieved as in Para 2.D.2).a) or Para 2.D.2).b) as follows:

a) Using retraction clamp (SD12559) as in propeller installation (refer to 61-10-00, Propeller Assembly – General Data).

Note When using the retraction clamp, strictly observe the procedures regarding installation and use.

b) Turn the propeller blades into reverse pitch and secure in this position.

Note To facilitate turning of blades, remove propeller piston flexlock nut, using wrench (SD12549).

3) Using the two nuts retained during mounting removal, install mounting complete with actuator slide, carbon block, switch plate and microswitch. Ensure carbon block engages low stop collar. Torque nuts securing mounting to engine reduction gearbox to between 145 to 165 pound–inches.

4) Connect microswitch wires at disconnect splices.

5) Adjust microswitch; refer to Para 3.A.

6) Install propeller de-icing brush block assembly (refer to 30-60-11, Propeller De-icing Brush Module Assembly (Mod 6/1674) – Maintenance Practices).

3. Adjustment/Test

Refer to Figure 1 for details.

A. Adjust Beta Microswitch

1) Slacken bolts securing switch plate to switch bracket.

2) Check low stop collar runout (refer to 61-10-00, Propeller Assembly – General Data).

3) Check that a clearance of 0.005 to 0.020 inch exists between roller leaf and actuator arm when roller is not contacting slide.

4) Position propeller low stop collar so that microswitch roller is positioned on operated face of slide. This may be achieved as in Para 3.A.4).a) or Para 3.A.4).b) as follows:

Note When using the retraction clamp, strictly observe the procedures regarding installation and use.

a) Using retraction clamp (SD12559) as in propeller installation (refer to 61-10-00, Propeller Assembly – General Data.

Note To facilitate turning of blades, remove propeller piston flexlock nut, using wrench (SD12549).

b) Turn the propeller blades into zero pitch and secure in this position. If zero thrust stop latches are fitted, turn blades until latches engage.

5) Move switch plate assembly upwards against slide face until switch actuates (audible click), then tighten retaining bolts.

6) Check gap between actuator arm and microswitch body is not less than 0.015 inch.

7) Position low stop collar to feathered position using Para 3.A.7).a) or Para 3.A.7).b) as follows:

   Note When removing retraction clamp, observe correct procedures for replacement of rod end ring.

   a) Remove retraction clamp and replace rod end ring as in propeller installation (refer to 61-10-00, Propeller Assembly – General Data).

   b) Turn propeller blades to the feathered position. If zero thrust stops are fitted, release latches before attempting to turn blades. Install flexlock nut, ensuring O-ring seal is in position, and torque to 120 pounds feet using wrench (SD12549).

8) Slacken bolts securing switch bracket to mounting.

9) Adjust clearance between microswitch roller and actuator slide (dimension A) to 0.310 inch, utilizing the switch bracket slot.

10) Tighten bolts securing switch bracket to mounting.

11) Recheck clearance.

12) Complete ground run and check for beta microswitch operation (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

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Blade Zero Thrust Stop System (S.O.O. 6203) – General Data

1. Description

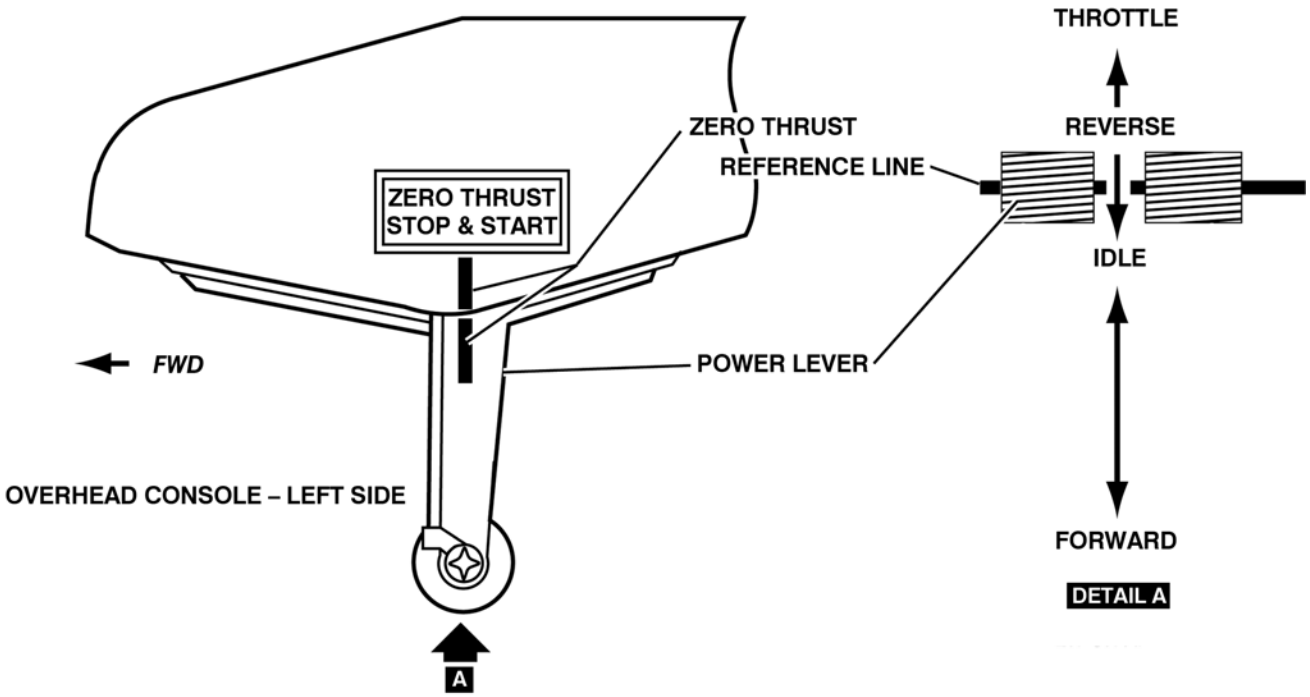
Refer to Figure 1 for details.

Each propeller is fitted with a blade zero thrust stop mechanism, which engages at a blade pitch angle of +1 degree when the engine is shutdown. This permits engine starts with zero thrust at idle power. The mechanism consists of a spring-loaded plunger and stop plate assembly installed at the root end of each blade. Reference lines on the power lever and on the overhead console, marked ZERO THRUST STOP AND START, are used to position the power lever correctly when engaging and releasing the zero thrust stops. This installation is only fitted on aircraft with floats.

2. Operation

If the zero thrust pitch stops are to be engaged on engine shutdown, the propeller lever must be set to full INCREASE, and the power lever set with the zero thrust reference lines aligned prior to stopping the engine. When the engine ceases to rotate, the oil pressure acting on the pitch change mechanism decays, due to a slow internal leakage, and the plungers engage the stop plate.

When starting an engine with blade zero thrust stops engaged, the power lever will have remained set with the zero thrust reference lines aligned and the propeller lever will be set according to OAT (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data). As the engine commences to turn, constant speed governor oil pressure rises and is directed to the propeller pitch change mechanism, which moves the blades slightly into reverse pitch, allowing the plungers to move away from the zero thrust stop plates. When the propeller speed increases to above 30% rpm, the centrifugal effect acting on the plungers moves the plungers outwards, against spring tension, to clear the stop plates. In this condition, the engine power lever can be advanced to idle and the blades move into a low forward pitch.



Location of Zero Thrust Reference Lines
Figure 1

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Propeller Indicating – General Data

1. General

The propeller control systems, incorporate and provide the following indications:

- Propeller speed in percent rpm
- Propeller reset caution
- Beta range indicting
- Propeller autofeather select and arm

A. Propeller Tachometer Indication

A tachometer mounted on the reduction gearbox provides a signal that is translated by the DAU into a % RPM indication (N_P) which is displayed as part of the engine pane on each PFD.

B. Propeller Reset Caution Light

An amber CAS message with the text Reset Props will appear if the power levers are retarded below the 75% N_G position with the lever set at 91% N_P or less. This caution message indicates the need to move the propellers forward to full INCREASE in order to put the propeller governor into an underspeed condition and provide the pilot with direct control of propeller blade angle via the power levers. The Reset Props message electrical circuit is energized through two microswitches connected in series, one operated by the left propeller lever and the other by the right power lever. The circuit is powered from the right DC bus and is protected by the PROPELLER OVERSPEED GOVERNOR circuit breaker on the main circuit breaker panel.

C. Beta Range Indicating Lights

The beta backup system has been deleted from the Series 400 Twin Otter. There is Indication that a propeller is in ground fine range (blade angles of +9° or less) during normal operations (power lever grips twisted). If a propeller enters ground fine range and the power lever grip switch has not been twisted, a warning level message Prop Low Pitch will be displayed, and an aural alert PROPELLER LOW PITCH will be voiced, in addition to the indications shown above.

Correct operation of the propeller position sensing microswitches can be confirmed by observing that the cyan BETA annunciation appears when the power lever grips are twisted and moved aft. Correct functioning of the propeller lever grip switch can be confirmed by observing the appearance of the status CAS message Interlock Open whenever the power lever grips are twisted, and confirming that the Interlock Open message extinguishes when the power lever grips are no longer twisted.

D. Propeller Autofeather Indicator Lights

Refer to 61-22-02, Propeller Autofeather System (Mod 6/2022) – General Data.

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Propeller Tachometer Generator – Maintenance Practices

1. Adjustment/Test

A. Bench Test Tachometer Generator

Equipment Required:

- Variable speed motor
- Drive speed indicator
- Tachometer generator output indicator (percent rpm)
- Mounting fixture and wiring
- Motor control switch
- HI–RPM–LOAD (Three 40 ohm, Y–connected resistances)
- LO–RPM–LOAD (Three 20 ohm, Y–connected resistances)
- Load ON–OFF switches (2)
- Multimeter

1) Test Continuity

a) Using multimeter (set to OHMS R x 1) measure continuity resistance of tachometer generator between each pair of stator windings A–B, A–C and B–C (C–case). Readings must be within 2.0 ohms of each other and each reading must be less than 50 ohms.

2) Test Rotation

a) Connect tachometer generator to output indicator (refer to Figure 1).

Note Tachometer generator is rotated in a CCW direction, viewed from the shaft end.

b) Operate motor and check output indicator pointer moves in a clockwise direction as motor speed is increased.

c) Shutdown motor.

3) Test Voltage Output

a) Disconnect wiring from output indicator (refer to Para 1.A.2).a)) and connect to test point terminals (refer to Figure 1).

b) Ensure LO–RPM–LOAD switch is OFF, and select HI–RPM–LOAD switch ON.

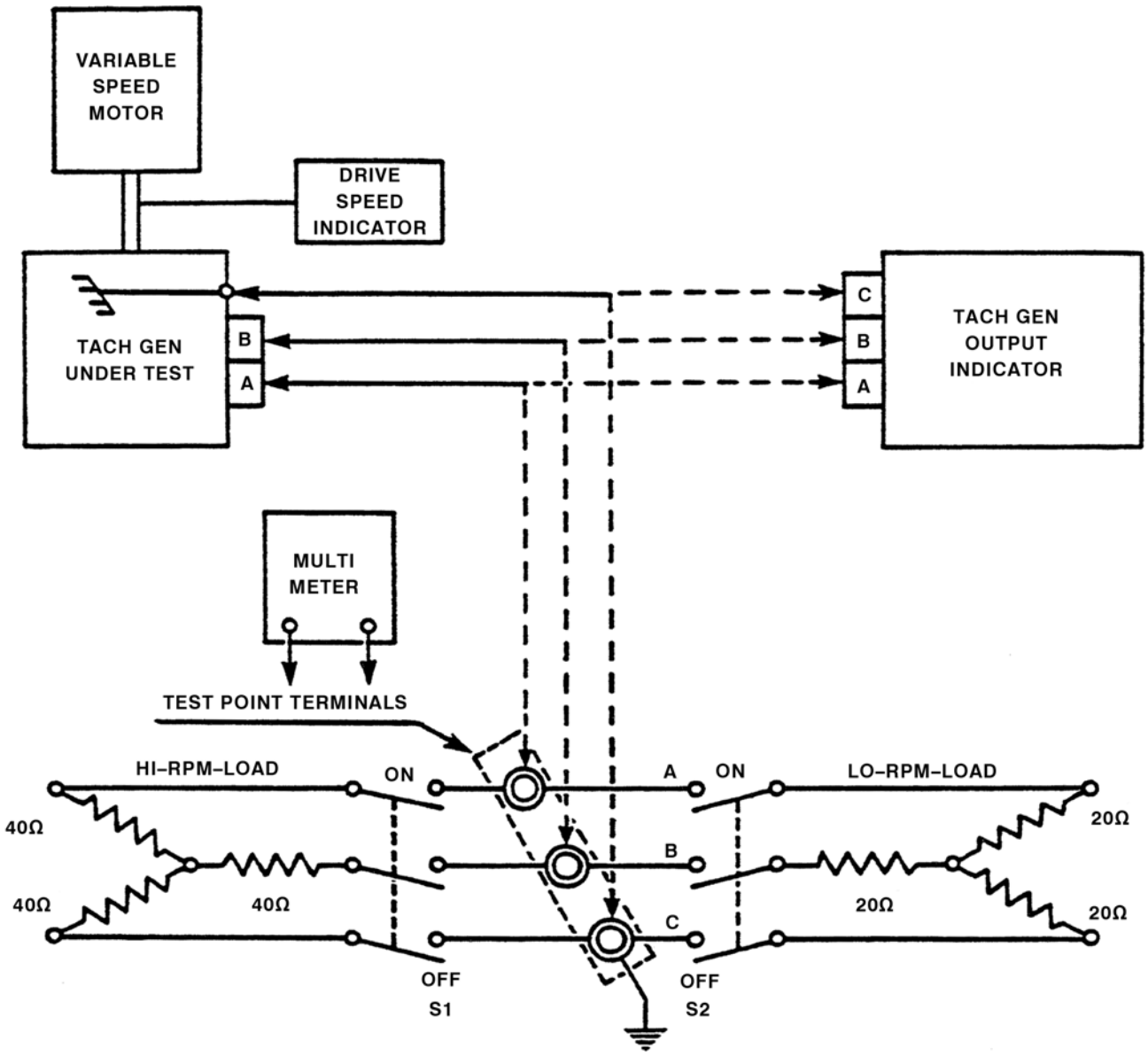
c) Operate motor and increase speed to 4200 rpm (drive speed indicator).

d) Set multimeter to 30 volts AC range and measure voltage outputs at the test point terminals across each of the phases A–B, A–C and B–C. The voltage across each phase shall be 21.0 ± 0.5 volts AC.

e) Decrease motor speed to 1000 rpm, select HI–RPM–LOAD switch to OFF and LO–RPM–LOAD switch to ON.

f) Repeat Para 1.A.3).d) and check phase voltages are not less than 3.5 volts AC.

g) Remove tachometer generator from test bench.



Test Rig Layout Schematic
Figure 1

CHAPTER 71

POWER PLANT

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CHAPTER 71 – POWER PLANT

LIST OF EFFECTIVE PAGES

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Power Plant (Mod 6/2012) – General Data

1. General

Each power plant consists of a United Aircraft PT6A–34 engine, engine mount assembly, exhaust stubs, and the equipment and accessories required for the supply of fuel and air for the correct functioning of the engine under all operating conditions. The engine is supported at three points by means of vibration isolators attached to the front face of the engine nacelle structure. Enclosing the engine are two removable engine cowlings, the lower cowling housing a snow and ice deflector system. The power plant is protected by fire detection and fire extinguishing systems, and two fireseals fitted to the front and rear of the engine compressor intake.

2. Troubleshooting

TROUBLE	POSSIBLE CAUSE	CORRECTION
Failure to light up.	Ignition system defective.	Refer to PT6A Engine Maintenance Manual, Ignition Section. 74-00-00, Ignition – General Data
Acceleration rates varying between engines.	Incorrect engine rigging.	Recheck engine rigging and adjust as required.
Propeller rpm and torque pressure fluctuation.	Interplay between propeller constant speed and overspeed governors.	Check CSU governor for correct operation.

Note For further information on power plant troubleshooting, refer to PT6A Engine Maintenance Manual or 72-00-00, Engine – General Data.

3. Removal

Refer to Figure 1 for details.

A. Remove Power Plant

1) With electrical power ON, close fuel emergency shut-off valve using firewall shut-off valve switch on emergency panel. Close fuel shut-off valve by setting fuel lever to OFF position.

2) Disconnect electrical power.

3) Support rear fuselage using steady jacks and bar.

4) Disconnect leads from upper nacelle fire detector.

5) Remove cowlings (refer to 71-10-00, Cowlings – General Data), and access panel from top of nacelle.

6) Remove propeller (refer to 61-10-00, Propeller Assembly – General Data).

7) Disconnect power, propeller, and fuel shut-off controls at engine.

8) Disconnect main fuel feed, and fuel purge lines at main firewall.

9) Disconnect electrical connectors and leads at lower left firewall.

10) Remove electrical leads from starter/generator.

11) Disconnect oil breather line at main firewall.

12) Disconnect bleed air line at main firewall.

13) Attach engine sling SD12541 and take up power plant weight (refer to 07-00-00, Lifting and Shoring – General Data).

14) Remove single attachment bolt, using wrench SD12524, from each vibration isolator and lower power plant (refer to 71-20-11, Vibration Isolator – Maintenance Practices).

15) Remove vibration isolators from each side of engine easing (refer to 71-20-11, Vibration Isolator – Maintenance Practices).

16) Lower power plant onto maintenance stand. Remove engine sling and upper vibration isolator (refer to 71-20-11, Vibration Isolator – Maintenance Practices).

B. Remove Power Plant Accessories

CAUTION BLANK OFF COMPRESSOR INTAKE PRIOR TO REMOVAL OF ACCESSORIES.

1) Remove fuel and oil drain lines, and drain line brackets.

2) Remove oil cooler (refer to 79-20-12, Oil Cooler (Mod 6/1438 and Mod 6/1481) – Maintenance Practices).

3) Remove from engine accessory case the oil pressure switch, pressure line, switch mounting bracket, oil temperature bulb and adapter and oil pressure transmitter from mounting face (refer to 79-00-00, Oil – General Data).

4) Remove external oil lines from engine accessory case.

5) Remove fire detectors.

6) Remove the following controls from the engine (refer to 76-00-00, Engine Controls – General Data):

   a) Propeller control complete with brackets and hardware for propeller governor.

   b) Beta control cam, cam follower lever, and FCU actuating lever complete with hardware from accessory gear case.

   c) Fuel shut-off link lever bracket on top of FCU.

7) Remove starter/generator cooling duct (refer to 80-10-11, Starter/Generator – General Data).

8) Remove bleed air line, and intake deflector line.

9) Remove starter/generator and drive mount pad and clamp (refer to 80-10-11, Starter/Generator – General Data).

10) Remove exhaust nozzles (refer to 78-00-00, Exhaust – General Data).

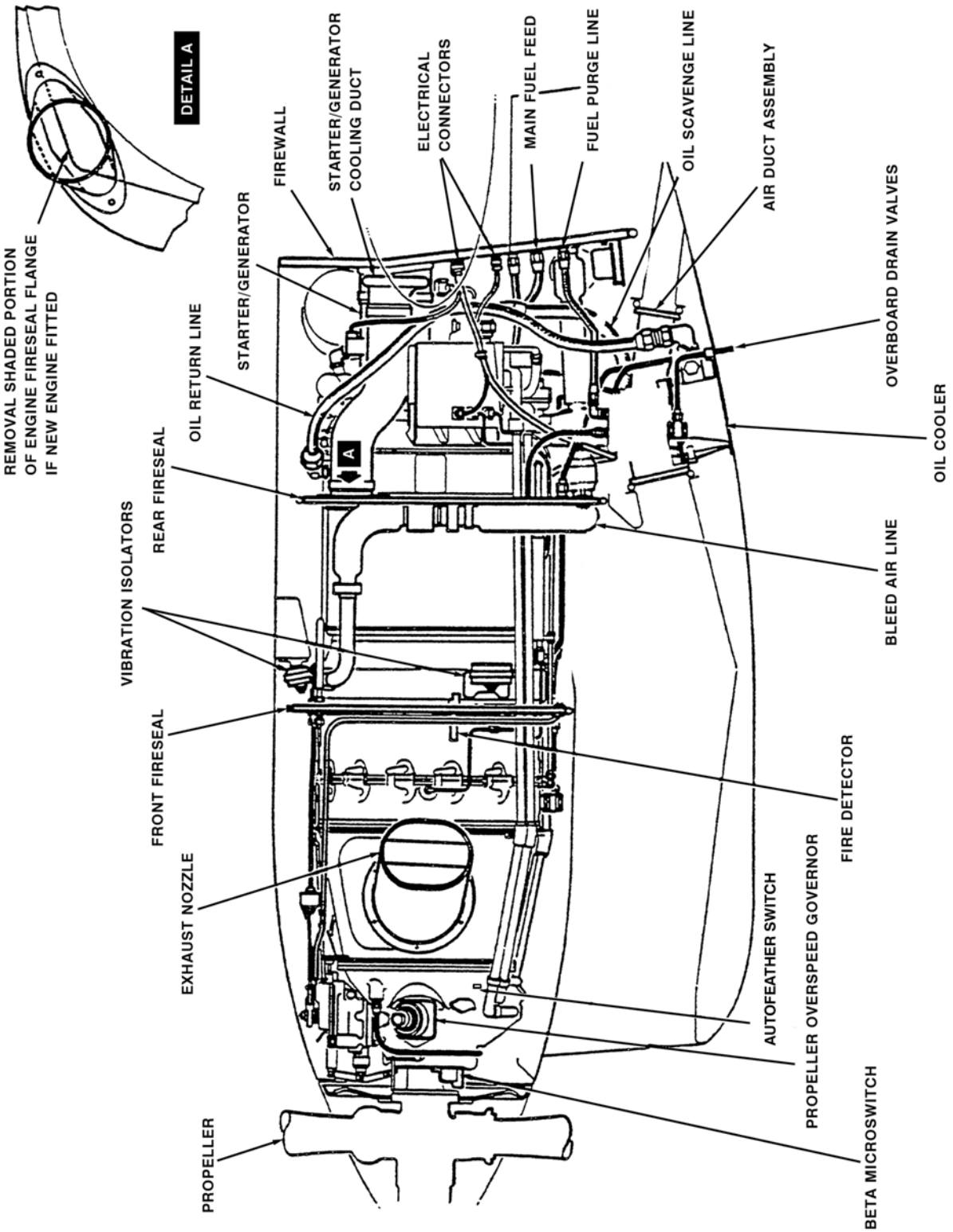
11) Remove electrical harness and leads (refer to 71-50-00, Electrical Harness – General Data).

12) From reduction gear case, remove torque pressure transmitter and vent line, autofeather switch and hose, beta microswitch and bracket (refer to 61-24-13, Beta Microswitch Assembly (Mod 6/2022) – General Data), propeller overspeed governor, (refer to 61-20-21, Overspeed Governor – Maintenance Practices), and propeller tachometer generator.

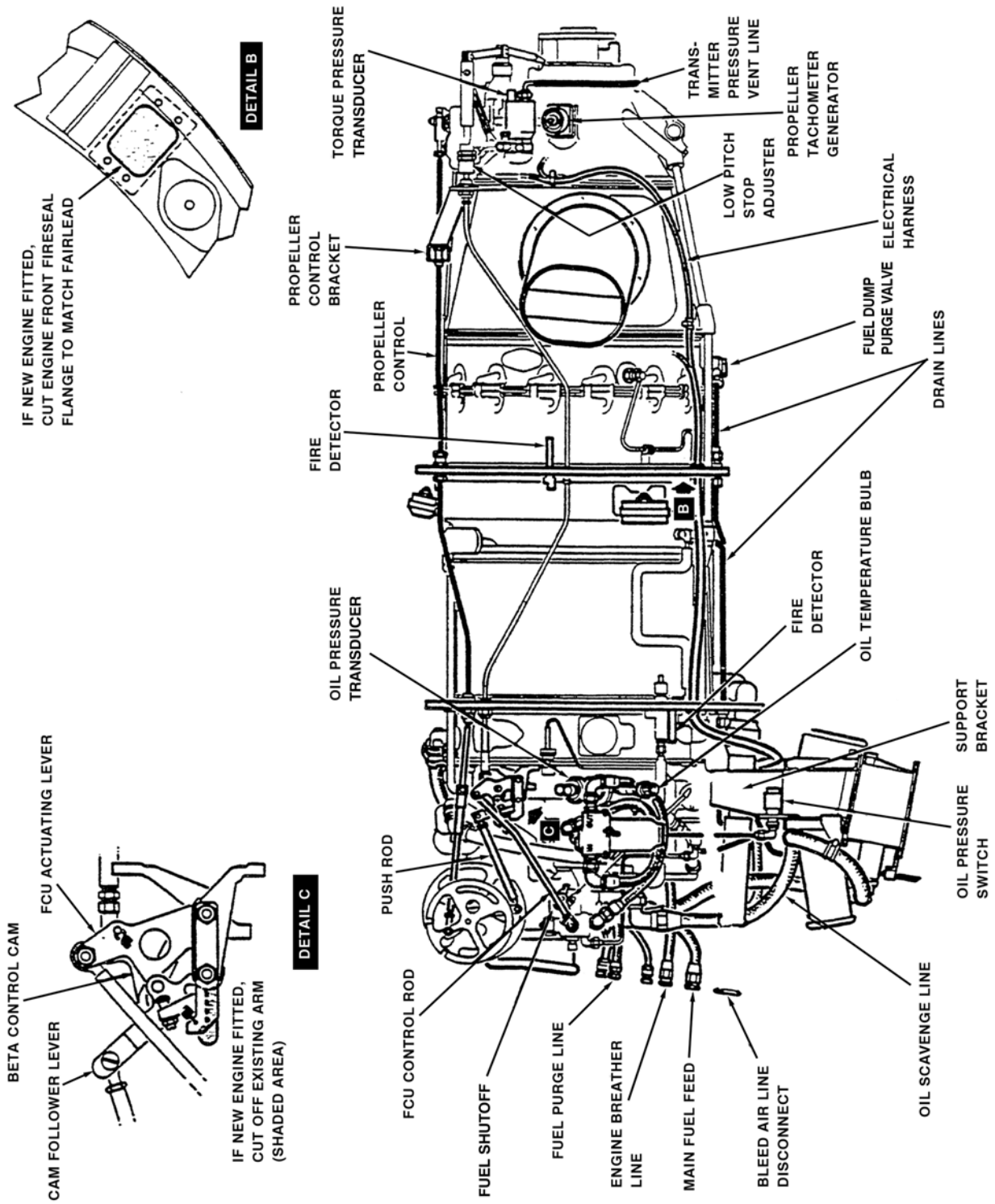
13) Remove gas generator tachometer generator from engine accessory case.

14) Remove front and rear fireseals.

15) Remove air duct assembly and support bracket from underside of accessory gearbox.



Power Plant Installation
Figure 1 (Sheet 1 of 2)



Power Plant Installation
Figure 1 (Sheet 2 of 2)

4. Installation

CAUTION **IF PREVIOUSLY INSTALLED ENGINE HAS BEEN REMOVED BECAUSE OF METAL CONTAMINATION OF OIL SYSTEM, PROPELLER, PROPELLER OVERSPEED GOVERNOR, OIL COOLER, AND ASSOCIATED PIPE LINES AND HOSES MUST BE THOROUGHLY CLEANED BY FLUSHING. IF OIL COOLER CANNOT BE ADEQUATELY CLEANED BY FLUSHING, IT MUST BE REPLACED.**

Note Unless otherwise specified, install engine nuts and/or bolts to torque values stated in PT6A Engine Maintenance Manual.

Refer to Figure 1 for details.

A. Install Power Plant Accessories

CAUTION **BLANK OFF COMPRESSOR INTAKE PRIOR TO INSTALLATION OF ACCESSORIES.**

Note All splines of accessory drives to be lubricated with Molykote M77 or Plastilube No. 3 (Warren Refining Co., Cleveland, Ohio).

1) Install support bracket to underside of accessory gearbox.

2) Install air duct assembly to bracket.

3) Install front and rear fireseals.

Note If new engine fitted, cut engine front fireseal flange to match front fireseal electrical harness fairlead and rear fireseal flange to match starter/generator cooling duct intake (refer to Figure 1 Detail A and B).

4) Install gas generator tachometer generator to engine accessory case.

5) Install to reduction gear case, the torque pressure transmitter and pressure line, autofeather switch and hose, beta microswitch and bracket (refer to 61-24-13, Beta Microswitch Assembly (Mod 6/2022) – General Data), propeller overspeed governor and drive (refer to 61-20-21, Overspeed Governor – Maintenance Practices), and propeller tachometer generator.

6) Install electrical harness and leads (refer to 71-50-00, Electrical Harness – General Data).

7) Install exhaust nozzles (refer to 78-00-00, Exhaust – General Data).

8) Install gasket, drive mount pad (with dowel at 12 o'clock position), and starter/generator (refer to 80-10-11, Starter/Generator – General Data).

9) Install bleed air line, and intake deflector line. Torque each Janitrol clamp to value quoted on clamp body.

10) Install starter/generator cooling duct (refer to 80-10-11, Starter/Generator – General Data).

11) Install the following controls to the engine (refer to 76-00-00, Engine Controls – General Data):

a) Beta control cam, cam follower lever, and FCU actuating lever complete with hardware.

b) Propeller control complete with brackets and hardware for propeller governor.

c) Fuel shut-off link lever bracket to the top of the FCU.

12) Install fire detectors.

13) Install external fuel and oil lines to engine accessory case (refer to 79-00-00, Oil – General Data).

14) Install to engine accessory case, the oil pressure switch mounting bracket, oil pressure switch and line, oil temperature bulb and adapter, and oil pressure transmitter.

15) Install oil cooler (refer to 79-20-12, Oil Cooler (Mod 6/1438 and Mod 6/1481) – Maintenance Practices).

16) Install fuel and oil drain lines, and drain lines bracket to oil cooler.

17) Rig engine controls (refer to 76-00-00, Engine Controls – General Data and 76-10-00, Power Control – Maintenance Practices).

B. Install Power Plant

1) Install engine bracket assembly and upper vibration isolator on engine casing using the four attachment bolts (refer to 71-20-11, Vibration Isolator – Maintenance Practices).

2) Install engine sling SD12541 and take up power plant weight (refer to 07-00-00, Lifting and Shoring – General Data).

3) Remove power plant from maintenance stand.

4) Install vibration isolators on each side of engine casing using the four attachment bolts per isolator on each side of engine casing (refer to 71-20-11, Vibration Isolator – Maintenance Practices).

5) Raise power plant into position in engine nacelle.

6) Secure power plant to nacelle with vibration isolators attachment bolts (refer to 71-20-11, Vibration Isolator – Maintenance Practices). Remove engine sling.

7) Connect bleed air line to firewall connection.

8) Connect oil breather line to firewall connection.

9) Connect electrical leads to starter/generator.

10) Connect electrical connectors and leads at lower left main firewall.

11) Connect power, propeller, and fuel shut-off controls to engine.

12) Install propeller (refer to 61-10-00, Propeller Assembly – General Data).

13) Install cowlings (refer to 71-10-00, Cowlings – General Data).

14) Connect leads to upper nacelle fire detector.

15) Connect electrical power.

16) Rig power plant controls (refer to 76-00-00, Engine Controls – General Data to 76-10-11, Overhead Console – General Data).

Note Ensure low pitch stop adjuster spring has been removed.

17) Install access panel in top of nacelle.

18) Remove rear fuselage steady jacks and bar.

19) Check for fuel and oil leaks on subsequent engine run.

20) Carry out post installation procedures, refer to Para 10.

5. Ground Running – General Precautions

   A. Action Prior to Ground Running

      1) Position aircraft facing into wind.

      2) Chock wheels.

      3) Fire extinguisher in position.

      4) Ensure area around aircraft is free of loose objects.

      5) Check engine oil level (refer to 12-10-25, Oil System – General Data).

      6) Check engine intakes and exhausts are free from foreign objects.

      7) Ensure danger areas around aircraft are clear of personnel.

6. Engine Depreservation

   A. Depreserve Engine

      1) Fill engine oil tank (refer to 12-10-25, Oil System – General Data).

         Note Oil system does not require any depreservation unless different oil is to be used in system. If different oil used, flush system after Engine Leak Test Run (refer to 12-10-25, Oil System – General Data).

      2) Carry out preliminary action prior to ground running (refer to Para 5.).

      3) Pull IGNITION circuit breakers and fit safety clip.

      4) POWER SOURCE switch – EXTERNAL (Pre Mod 6/2314); or EXTERNAL switch – ON (Post Mod 6/2314).

      5) Booster pumps – FWD BOOST or AFT BOOST for right or left engine.

      6) Disconnect inlet fuel line at bottom fuel purge/dump valve adapter (refer to PT6A Engine Maintenance Manual); to permit drainage into container.

      7) Ensure external electrical power is connected and MASTER switch is ON (Pre Mod 6/2314); or EXTERNAL switch is ON (Post Mod 6/2314)

      8) Power lever – IDLE.

      9) Fuel lever – ON.

      10) START switch – hold at LEFT or RIGHT (as applicable) for approximately 10 seconds, then release switch.

      11) Check fuel flow from line into containers is free of preservative fluid.

      12) Fuel lever – OFF.

      13) Check fuel flow stops flowing release start switch lever.

CAUTION OBSERVE THE STARTER OPERATING TIME LIMITS AS FOLLOWS:

- **25 SECONDS ON, 1 MINUTE OFF**
- **25 SECONDS ON, 1 MINUTE OFF**
- **25 SECONDS ON FOLLOWED BY A 30 MINUTE COOLING-OFF PERIOD**

14) Fuel booster pump – OFF.

15) Reconnect fuel line to bottom of fuel dump and purge valve.

Note An eight hour soaking period is recommended after fuel system deprivations to allow bypass valve diaphragm to return to the required degree of pliability, and also to permit synthetic packing to swell and provide satisfactory seals.

16) Check for leaks in engine fuel system.

17) Check safety and security of all connections.

7. Fuel Drains and Dump Test (Wet Run)

A. Test FCU Purge Valve

1) Remove the fuel manifold purge line aft of the firewall at the bulkhead fitting. Position container to receive spillage.

2) Pull applicable IGNITION L or IGNITION R circuit breaker.

3) Power lever – IDLE.

4) Booster pump on – FWD BOOST or AFT BOOST (as applicable).

5) Engine fuel lever – OFF.

6) START switch – hold to LEFT or RIGHT (as applicable) to motor engine.

CAUTION OBSERVE THE STARTER OPERATING TIME LIMITS AS FOLLOWS:

- **25 SECONDS ON, 1 MINUTE OFF**
- **25 SECONDS ON, 1 MINUTE OFF**
- **25 SECONDS ON FOLLOWED BY A 30 MINUTE COOLING-OFF PERIOD**

7) Observe fuel flows from FCU purge valve fitting.

8) Move engine fuel lever to OFF.

9) Observe fuel flows from fuel manifold purge line at the bulkhead fitting.

10) Release starter switch and observe fuel flows from combustion chamber drain lines.

11) Check fuel pump and generator seal overboard drains for leakage.

12) Switch off booster pumps.

13) Reinstall the fuel lines C6PF1026–61 LH and C6PF1026–59 RH.

WARNING ENSURE RESIDUAL FUEL IN THE ENGINE COMBUSTION CHAMBER HAS STOPPED DRAINING BEFORE ATTEMPTING TO CARRY OUT ENGINE START.

Table 1 Operating Limits

OPERATING CONDITION	PT6A–34 OPERATING LIMITS					
POWER SETTING	TORQUE PSI (Ti)	TURBINE TEMP °C (T_5)	GAS GEN SPEED % (N_G)	PROP SPEED % (N_P)	OIL PRESS PSI (L_P)	OIL TEMP °C (LT)
Starting		1090 max (limit 2 secs)				–40 min
Idle		685 max	52		40 min	–40 to 99
Acceleration	68.7 max (limit 2 secs)	825 max (limit 2 secs)	102.6 max	110 max	85 to 105	0 to 99
Take-off and max continuous	50 max	790 max	101.6	96	85 to 105	10 to 99

CAUTION IF OIL PRESSURE DROPS BELOW 85 PSI WHEN GAS GENERATOR SPEED IS GREATER THAN 72% N_G, ENGINE RUN SHOULD BE TERMINATED.

8. Engine Starting and Shutdown

A. Prestart Checks

Note All switches to be off, unless otherwise stated.

1) Parking brake – ON (check pressure).

2) Circuit breakers – IN.

3) MASTER switch – ON (Pre Mod 6/2314).

4) POWER SOURCE switch – As required (Pre Mod 6/2314); or EXTERNAL and/or BATTERY switch – ON (Post Mod 6/2314).

5) IGNITION switch – NORMAL.

6) GENERATORS switches – OFF. Check caution message displayed.

7) BUS TIE switch – NORMAL.

8) CAUTION LIGHTS test switch – TEST.

9) Check position of blades on each propeller and set propeller levers and power levers accordingly as follows:

a) Propeller latches (if fitted) engaged.

CAUTION WITH ENGINES STOPPED AND PROPELLER BLADE LATCHES ENGAGED, NO ATTEMPT MUST BE MADE TO MOVE POWER LEVERS FORWARD TO IDLE OR MORE THAN 1.25 INCHES AFT OF IDLE.

1 Propeller levers – FEATHER (full INCREASE for pressure altitudes below 3000 feet when temperature is below 50°F (10°C) OAT).

2 Power levers – Zero thrust reference lines aligned.

CAUTION **OBSERVE OPERATING LIMITATIONS (REFER TO Table 1 — Operating Limits) DURING ENGINE RUNNING.**

b) Propeller feathered.

1 Propeller levers – FEATHER (full INCREASE for pressure altitudes below 3000 feet when temperature is below 50°F (10°C) OAT).

2 Power levers – IDLE.

CAUTION **WITH ENGINES STOPPED DO NOT ATTEMPT TO MOVE POWER LEVERS AFT OF IDLE TOWARD THE ZERO THRUST STOP/START LINE OR PROPELLER CONTROL LINKAGE MAY BE DAMAGED.**

10) Fuel levers – OFF (fully aft).

11) FIRE WARNING test switch – test.

12) FUEL EMERGENCY shut-off switches – NORMAL.

13) FUEL SELECTOR – NORMAL.

14) BOOST PUMP switches – FWD BOOST and AFT BOOST. (Check all four boost pump indication are not displayed.)

B. Engine Starting, Using External Electrical Power

CAUTION **IN AMBIENT CONDITIONS OF 90°F (32°C) AND OVER, GROUND OPERATION OF ENGINES IN FEATHERED, DISCING (ZERO THRUST), OR REVERSE, MUST BE KEPT TO AN ABSOLUTE MINIMUM (LESS THAN ONE MINUTE), OR INTERNAL NACELLE TEMPERATURES WILL EXCEED SAFE LIMITS.**

1) Carry out prestart checks as in Para 8.A.

2) AVIONICS START BATTERY switch (Post Mod 6/2315) – AUTO.

WARNING **ON AIRCRAFT WITH THE GPU RECEPTACLE LOCATED FORWARD OF THE PROPELLER (S.O.O. 6371), THE GPU PLUG SHOULD BE DISCONNECTED BEFORE STARTING THE LEFT ENGINE. DISCONNECTING THE GPU PLUG AFTER STARTING THE LEFT ENGINE PUTS THE GROUND CREW AT RISK OF BEING STRUCK BY THE PROPELLER.**

3) START switch – Hold at LEFT or RIGHT, as required. Check for oil pressure rise. Check appropriate L or R GENERATOR caution light goes out.

4) Allow gas generator speed to stabilize above 12% N_G, then move applicable engine fuel lever ON.

CAUTION **DO NOT SELECT ENGINE FUEL LEVER ON (ALLOWING FUEL TO FLOW) BEFORE STABILIZED RPM ARE REACHED. THE MINIMUM SPEED TO OBTAIN SATISFACTORY LIGHT-UP IS 12% N_G. ANY START ATTEMPT BELOW THIS SPEED IS CONSIDERED HAZARDOUS AND SHOULD BE MONITORED CLOSELY TO PREVENT AN OVER TEMPERATURE CONDITION.**

5) Light-up – Check that engine accelerates to normal idle rpm for prevailing ambients and turbine temperature does not exceed 1090°C. (Time limit at 1090°C is two seconds; operators are advised to keep hand on engine fuel lever during this period).

CAUTION **WHEN A GAS GENERATOR FAILS TO LIGHT UP WITHIN 10 SECONDS OF ADVANCING ENGINE FUEL LEVER, OR ENGINE OVER TEMPERATURE IS IMMINENT, SET ENGINE FUEL LEVER OFF TO STOP FUEL FLOW, AND CONTINUE MOTORING THE ENGINE WITH STARTER FOR APPROXIMATELY 10 SECONDS. OBSERVE STARTER LIMITATIONS (REFER TO Para 8.E.5)) AND ALLOW THE ENGINE TO COME TO A COMPLETE STOP BEFORE ATTEMPTING A NEW START.**

6) START switch – Release. Check appropriate L or R GENERATOR light comes on.

CAUTION **IF EITHER L OR R GENERATOR CAUTION LIGHT DOES NOT COME ON, IMMEDIATELY SELECT BUS TIE SWITCH TO OPEN AND EXTERNAL/BATTERY SWITCH TO OFF. SHUTDOWN ENGINE(S) AND RECTIFY UNSERVICEABILITY.**

7) Oil pressure – At least 40 psi, and ENGINE OIL PRESSURE caution light out.

8) Repeat Para 8.B.3) through Para 8.B.7) to start second engine.

9) Propeller levers – Full INCREASE.

10) Power levers – Advance to idle N_G + 15%.

CAUTION **IF THERE IS RESISTANCE TO FORWARD MOVEMENT OF THE POWER LEVERS, INDICATING FAILURE OF LATCHES TO WITHDRAW, THE POWER LEVERS MUST NOT BE FORCED AHEAD OR DAMAGE TO CONTROL MECHANISM MAY BE CAUSED.**

Note If blade latches were engaged when engines were started, a momentary movement of the power levers toward reverse should be made to ensure latch disengagement, before advancing power levers.

11) POWER SOURCE switch – BATTERY (Pre Mod 6/2314); or BATTERY switch – ON (Post Mod 6/2314).

12) GENERATORS switches – Individually RESET and release to ON. Check caution message displayed.

13) Check generator load by selection.

CAUTION **DO NOT SWITCH EITHER GENERATOR TO ON WITHOUT FIRST ADVANCING APPLICABLE POWER LEVER TO IDLE N_G + 15%. SUBSEQUENTLY DO NOT RETARD POWER LEVER(S) FROM THIS SETTING UNTIL GENERATOR LOAD IS 100 AMPS OR LESS. THE GENERATOR(S) MUST BE SWITCHED OFF IF POWER IS REDUCED AND THIS CONDITION IS NOT MET.**

14) Disconnect external electrical power.

C. Engine Starting, Using Battery

1) Carry out prestart checks as in Para 8.A.

2) When a battery start is necessary, start engines as in Para 8.B.3) through Para 8.B.12). If, however, during the start cycle of the first engine there is any reason to doubt the capability of the battery to start the second engine, proceed as follows:

a) Propeller lever of first engine – Full INCREASE.

b) Power lever of first engine – Advance to idle N_G + 15%.

c) GENERATORS switch of first engine – RESET and release to ON. Check that appropriate L or R GENERATOR caution light goes out.

d) Observe battery charge current until load is 40 amps or less.

e) GENERATORS switch of first engine – OFF.

CAUTION **CROSS GENERATOR STARTING IS DETRIMENTAL TO GENERATOR BRUSHES AND TO REVERSE CURRENT CUT-OUT OPERATION (CONTACT WELDING MAY EVENTUALLY RESULT). CROSS GENERATOR STARTING IS PROHIBITED, HOWEVER, GENERATOR ASSISTED STARTING IS ALLOWED.**

f) Start second engine as in Para 8.B.3) through Para 8.B.7).

g) As second engine speed moves up through 12% N_G, select GENERATORS switch of first engine to RESET and release to ON. Check that appropriate L or R GENERATOR caution light goes out.

h) Complete engine starting procedure as in Para 8.B.9), Para 8.B.10), Para 8.B.12), and Para 8.B.13).

D. Engine Starting of Cold Soaked Engines (Below –20°F (–29°C) OAT), Using Battery

1) Carry out prestart checks as in Para 8.A.

2) Dry motor of first engine to be started (for 5 seconds) by holding START switch at LEFT or RIGHT as required.

3) After approximately one minute, start first engine in accordance with Para 8.B.3) through Para 8.B.7).

4) From stabilized idle with 0°C oil temperature achieved, advance power lever to idle N_G + 15%.

5) GENERATORS switch of normal (operating) engine – RESET and release to ON. Check that appropriate L or R GENERATOR caution light goes out.

6) Observe battery charge current until load is 0.4 or less.

7) GENERATORS switch of operating engine – OFF.

8) Check L or R GENERATOR caution light illuminates.

CAUTION **CROSS GENERATOR STARTING IS DETRIMENTAL TO GENERATOR BRUSHES AND TO REVERSE CURRENT CUT-OUT OPERATION (CONTACT WELDING MAY EVENTUALLY RESULT). CROSS GENERATOR STARTING IS PROHIBITED, HOWEVER, GENERATOR ASSISTED STARTING IS ALLOWED.**

9) Start second engine as in Para 8.B.3) through Para 8.B.7).

10) As second engine speed moves up through 12° N_G, select GENERATORS switch of first engine to RESET and release to ON. Check that appropriate L or R GENERATOR caution light goes out.

11) Complete engine starting procedure as in Para 8.B.9), Para 8.B.10), Para 8.B.12), and Para 8.B.13).

E. Clearing an Engine – Motor Run

CAUTION **OBSERVE THE STARTER OPERATING TIME LIMITS AS FOLLOWS:**

- **25 SECONDS ON, 1 MINUTE OFF**
- **25 SECONDS ON, 1 MINUTE OFF**
- **25 SECONDS ON FOLLOWED BY A 30 MINUTE COOLING-OFF PERIOD**

Note The following procedure is used to clear an engine when considered necessary to remove internally trapped fuel and vapor, or if there is evidence of a fire within the engine. Air passing through the engine serves to purge fuel, vapor, or fire from the combustion section, gas generator turbine, power turbine, and exhaust system.

1) Engine fuel lever – OFF.

2) Pull applicable IGNITION L or IGNITION R circuit breaker.

3) Power lever – IDLE.

4) Booster pump on – FWD BOOST (RH engine) or AFT BOOST (LH engine) (as applicable).

5) START switch – Hold to LEFT or RIGHT as applicable. Maintain for desired duration, then release switch.

Note If an engine fire is not extinguished during clearing of engine, switch off booster pumps and FUEL EMERGENCY switches. POWER SOURCE switches to OFF. Then vacate aircraft and apply fire extinguisher into exhaust stubs.

F. Engine Shutdown

WARNING IF THERE IS ANY EVIDENCE OF FIRE IN EITHER ENGINE AFTER SHUTDOWN, PROCEED IMMEDIATELY AS FOR CLEARING AN ENGINE IN Para 8.E.

1) Power levers – IDLE.

Note Allow engines to idle for a minimum of one minute at a turbine temperature of 585°C or below. Increase gas generator rpm (N_G) as required to maintain temperature limit.

2) Parking brake – ON.

3) Turbine temperature (T_5) stable.

4) BLEED AIR – OFF.

5) Radios – OFF.

6) GENERATORS – OFF. Check caution message displayed.

CAUTION IF EITHER L OR R GENERATOR CAUTION LIGHT DOES NOT COME ON, IMMEDIATELY SELECT BUS TIE SWITCH TO OPEN AND POWER SOURCE SWITCH TO OFF.

7) Propeller levers – Full INCREASE (latches to be engaged).
or
– FEATHER (latches not to be engaged).

8) Fuel levers – OFF.

Note If propeller blade latches are to be engaged, align zero thrust reference lines prior to, or immediately following, the selection of fuel levers to OFF. The latches will not engage until the propeller rpm has decreased below 30%.

9) BOOST PUMP switches – OFF.

10) AVIONICS START BATTERY switch (Post Mod 6/2315) – OFF.

11) POWER SOURCE switch – As desired.

12) MASTER switch – OFF (Pre Mod 6/2314); or EXTERNAL and BATTERY switches – OFF.

Note During engine shutdown, listen for any unusual scraping or rubbing noises.

9. Engine Leak Test Run

A. Engine Leak Test Run (Cowls Off)

1) Start engine (refer to Para 8.).

2) Check acceleration time from light-up to idle (approximately 12 seconds).

3) Check for:

a) Turbine temperature indication (T_5).

b) Oil pressure indication (LP).

c) Gas generator rpm indication (N_G).

d) Torque pressure indication (Ti).

e) Propeller rpm indication (N_P).

f) Oil temperature indication (LT).

4) With power lever at IDLE, move propeller lever through full range four times, feathering and unfeathering the propeller each time to purge system.

5) Check feathering and unfeathering times at idle setting are approximately 10 and 30 seconds respectively.

CAUTION IF FUEL IS SHUT OFF INADVERTENTLY BY RETARDING FUEL LEVER, DO NOT OPEN FUEL LEVER AGAIN IN ANY ATTEMPT TO REGAIN LIGHT. THE STARTING SEQUENCE MUST BE REPEATED FROM BEGINNING AFTER ALLOWING THE ENGINE TO COME TO COMPLETE STOP.

6) Run at idle speeds until conditions stabilize.

7) Shutdown engine (refer to Para 8.F).

8) Check compressor decelerates freely (minimum time approximately 35 seconds).

9) Check for oil and fuel leaks and security of equipment.

10) Check and top up oil tank (refer to 12-10-25, Oil System – General Data).

11) Install cowlings.

10. Post Installation Procedures

Note The following adjustments and tests are to be carried out after installation of an engine or power plant, but each adjustment or test can be carried out independently if a specific component or setting is to be checked.

A. Adjustments Prior to Ground Running

1) Calibrate turbine temperature (T_5) indicating system (refer to 77-20-00, Turbine Temperature Indicating System – General Data).

2) Calibrate engine torque pressure (Ti) system.

B. Ground Running Adjustments and Test

Note Unless otherwise stated, engines are to be run with no load applied (generators OFF, bleed air switch OFF, and intake deflector retracted).

Ensure engine to be run is depreserved (refer to Para 6.), prior to carrying out the following adjustments or tests.

1) Carry out engine leak test run (refer to Para 9.A).

2) Adjust gas generator idle speed (N_G) setting.

CAUTION DO NOT DO A CHECK AT BAROMETRIC PRESSURES BELOW 25.2 IN. HG OR ALTITUDES IN EXCESS OF 3500 FEET.

a) Set aircraft altimeter window scale to 29.92. Use the BARO knobs on the pilot's and co-pilot's primary flight display controller to set the altimeter. Record pressure altitude.

b) Start engine (refer to Para 8.).

c) Allow engine to warm up to operating temperature and idle rpm to stabilize with no load on engine.

d) Ensure fuel control unit (FCU) linkage is in the idle null position by checking that 1/8 to 1/4 inch forward movement of the power lever knob is required to increase rpm (N_G).

Note If unable to achieve an idle null position, recheck power control rigging (refer to 76-10-00, Power Control – Maintenance Practices).

e) Adjust FCU minimum speed adjustment (if necessary) on top of FCU (refer to Figure 2) to give 52 ± 1% N_G. Adjust clockwise to increase rpm (1/16 turn 1% N_G).

CAUTION TURN SCREW IN MINUTE INCREMENTS AS ADJUSTMENT IS EXTREMELY SENSITIVE.

Note If tolerances in Para 10.B.2).e) cannot be achieved refer to Pratt and Whitney Engine Manuals 3021242 Chapter 71-00-00: Fuel Control Unit Checks.

f) Record N_G at idle speed.

3) Adjust prime blade angle.

a) Record outside air temperature (OAT) and pressure altitude (refer to Para 10.B.2).a)).

b) Start engine, refer to Para 8.

c) Allow engine to stabilize.

d) With propeller lever at full INCREASE, advance power lever to obtain 77% propeller speed (N_P).

Note Para 10.B.3).d) should be carried out under zero wind conditions, but an average reading of direct up and down winds can be used. If wind speed is greater than 15 mph, data obtained is unreliable.

e) Read resultant engine torque pressure and record.

f) From curve in Figure 5, read desired torque pressure for ambient conditions.

g) Shutdown engine (refer to Para 8.F).

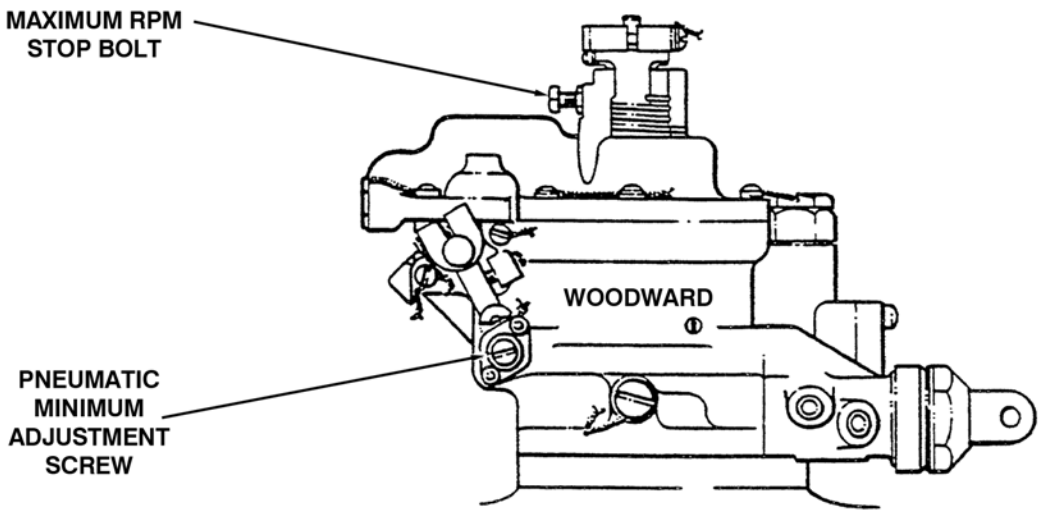
h) Remove propeller spinner.

i) Adjust pitch setting nuts (refer to Figure 3), if necessary, to give desired torque (refer to Para 10.B.3).f)). One flat rotation of nut gives 1 psi torque change.

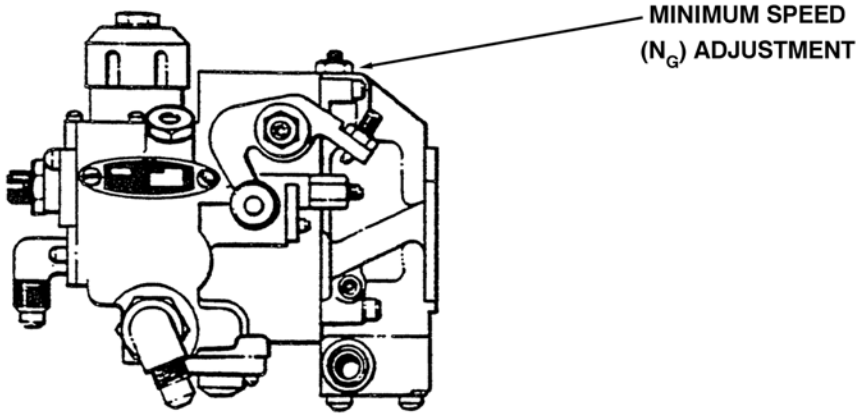
Note Mark position of each nut relative to low pitch stop rod, and turn each nut one flat at a time to ensure nuts are not moved out of position to each other. Adjust nuts out to decrease torque. Minor adjustment only should be required. If any larger change is necessary, the basic setting of beta valve (bottom of clevis flush with valve body) should first be checked (refer to 76-10-00, Power Control – Maintenance Practices, Para 3., Rig Beta Controls).

CAUTION SUBSEQUENT RUNS MAY BE DONE WITH SPINNER REMOVED, BUT IF PROPELLER DE-ICING IS FITTED, SPINNER MUST BE REINSTALLED.

j) Repeat Para 10.B.3).b) to Para 10.B.3).e) and adjust if necessary.

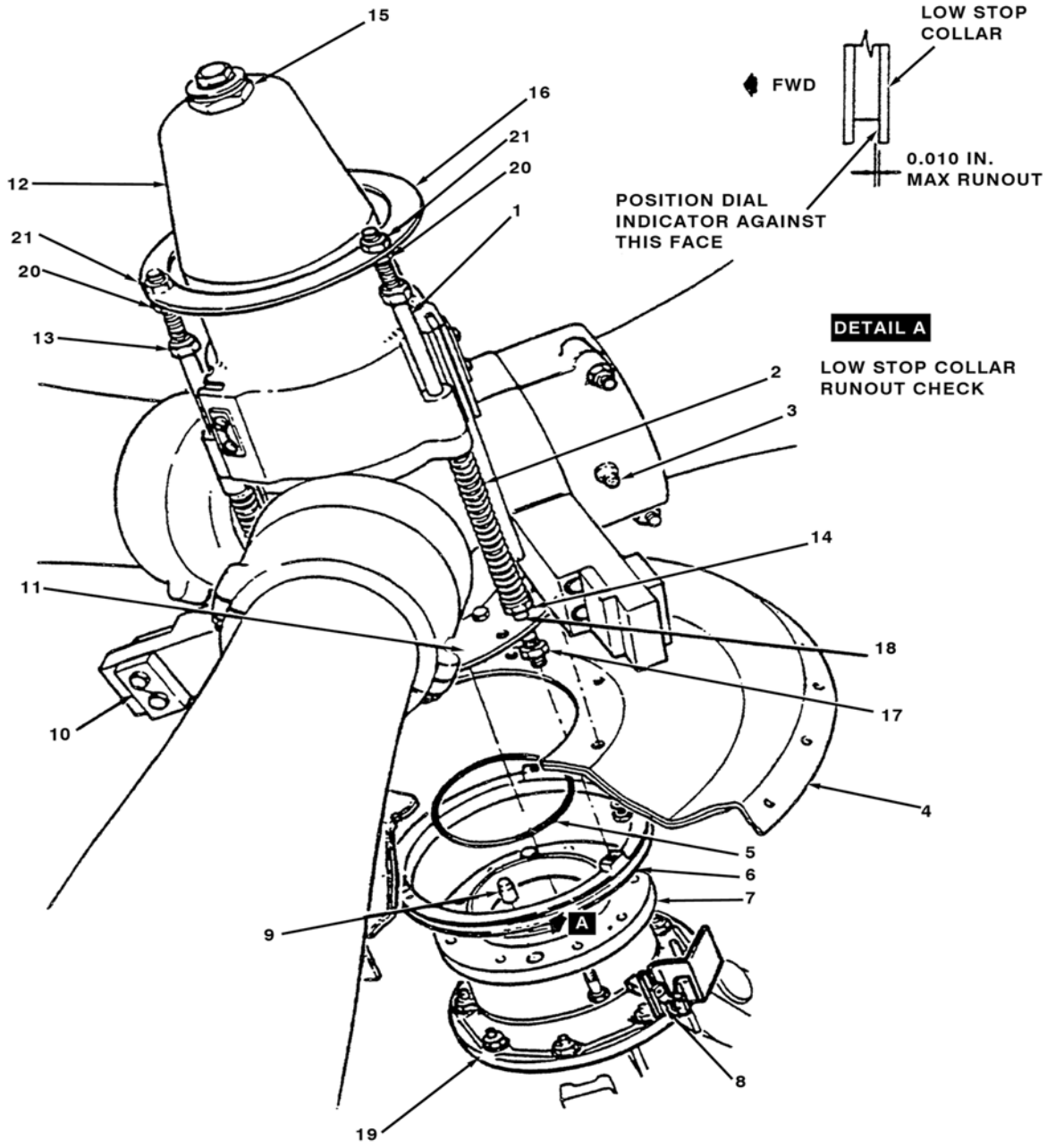


PROPELLER GOVERNOR



FUEL CONTROL UNIT

Running Adjustment Settings
Figure 2



1. LOW PITCH STOP ROD
2. REVERSE RETURN SPRING
3. LUBRICATION FITTING
4. SPINNER BULKHEAD
5. O-RING
6. LOW STOP COLLAR
7. ENGINE SHAFT FLANGE
8. MICROSWITCH SLIDE BLOCK
9. DOWEL PIN
10. COUNTERWEIGHT
11. SPINNER SUPPORT RING
12. PISTON UNIT
13. PITCH SETTING NUT
14. STOP ROD WRENCH FLATS
15. FLEXLOCK NUT
16. ROD END RING
17. LOCKNUT
18. ROD GUIDE
19. ENGINE REDUCTION GEAR CASE
20. ROD END RING REAR LOCKNUTS
21. ROD END RING FORWARD LOCKNUTS

Propeller Installation
Figure 3

4) Adjust idle blade angle (11°).

a) Start engine (refer to Para 8.), and stabilize at idle with no load on engine.

b) With power lever at IDLE and propeller lever at full INCREASE, read resultant propeller (N_P) and gas generator (N_G) speeds, and record. Wind effect applies as Note in Para 10.B.3).d).

c) Shutdown engine (refer to Para 8.F).

d) From curve in Figure 6, read desired N_P for N_G recorded in Para 10.B.4).b).

Note If N_P recorded in Para 10.B.4).b) is within tolerance limit (± 1%) of desired N_P shown on curve in Figure 6, disregard Para 10.B.4).e) to Para 10.B.9).

e) With power lever at IDLE, move propeller lever to minimum rpm position and install rigging pin through propeller and power control pulleys.

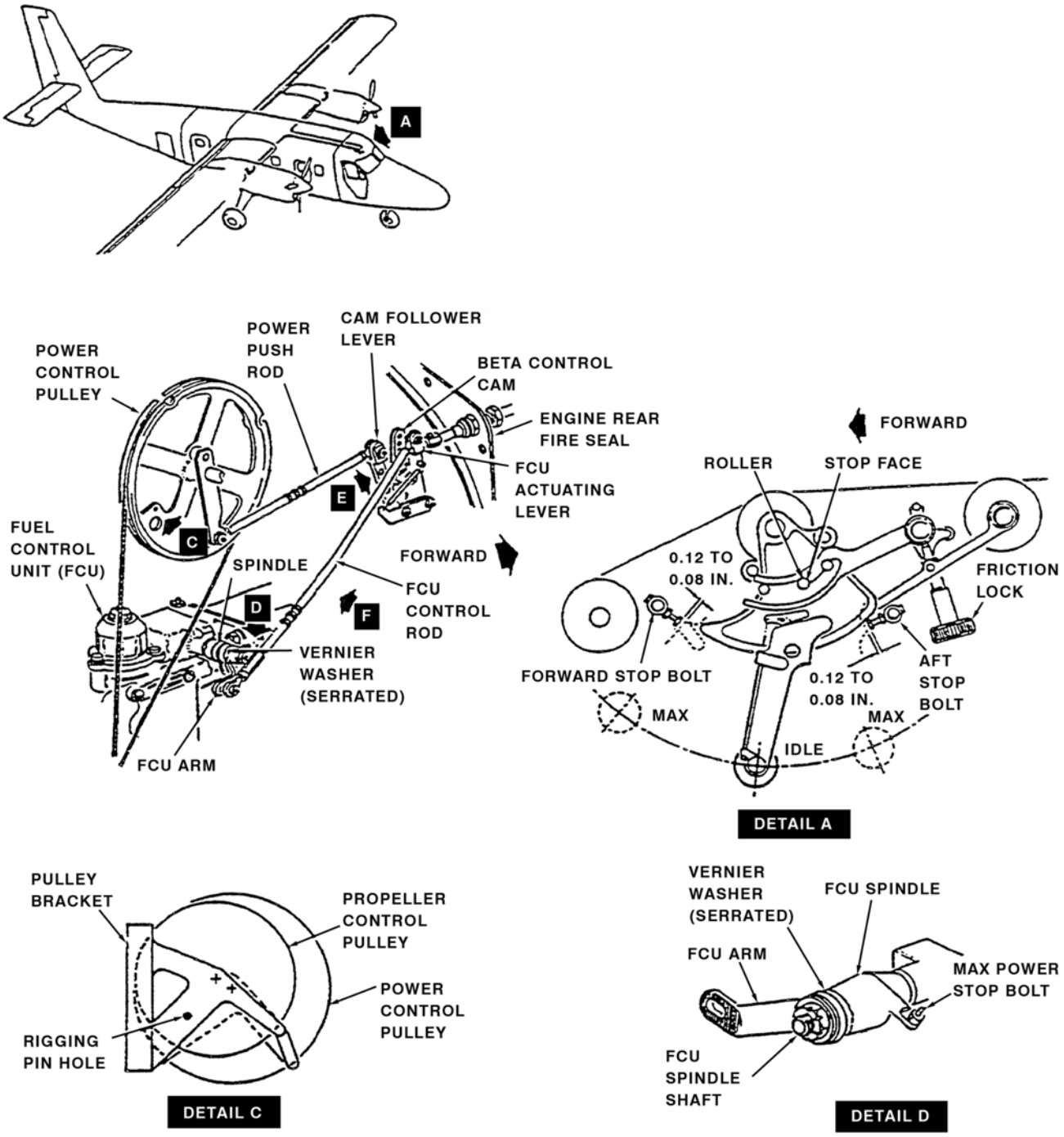
f) Adjust length of push rod (refer to Figure 4) to bring N_P within limits, shown in Figure 6. Adjust in small increments 3/4 turn of adjuster, is equivalent to 1% N_P. Increasing rod length decreases N_P.

Note Small adjustment only is permitted. If N_P requires other than minor change, the teleflex control movement in low pitch adjuster should first be checked (refer to 76-10-00, Power Control – Maintenance Practices, Para 3. Rig Beta Controls).

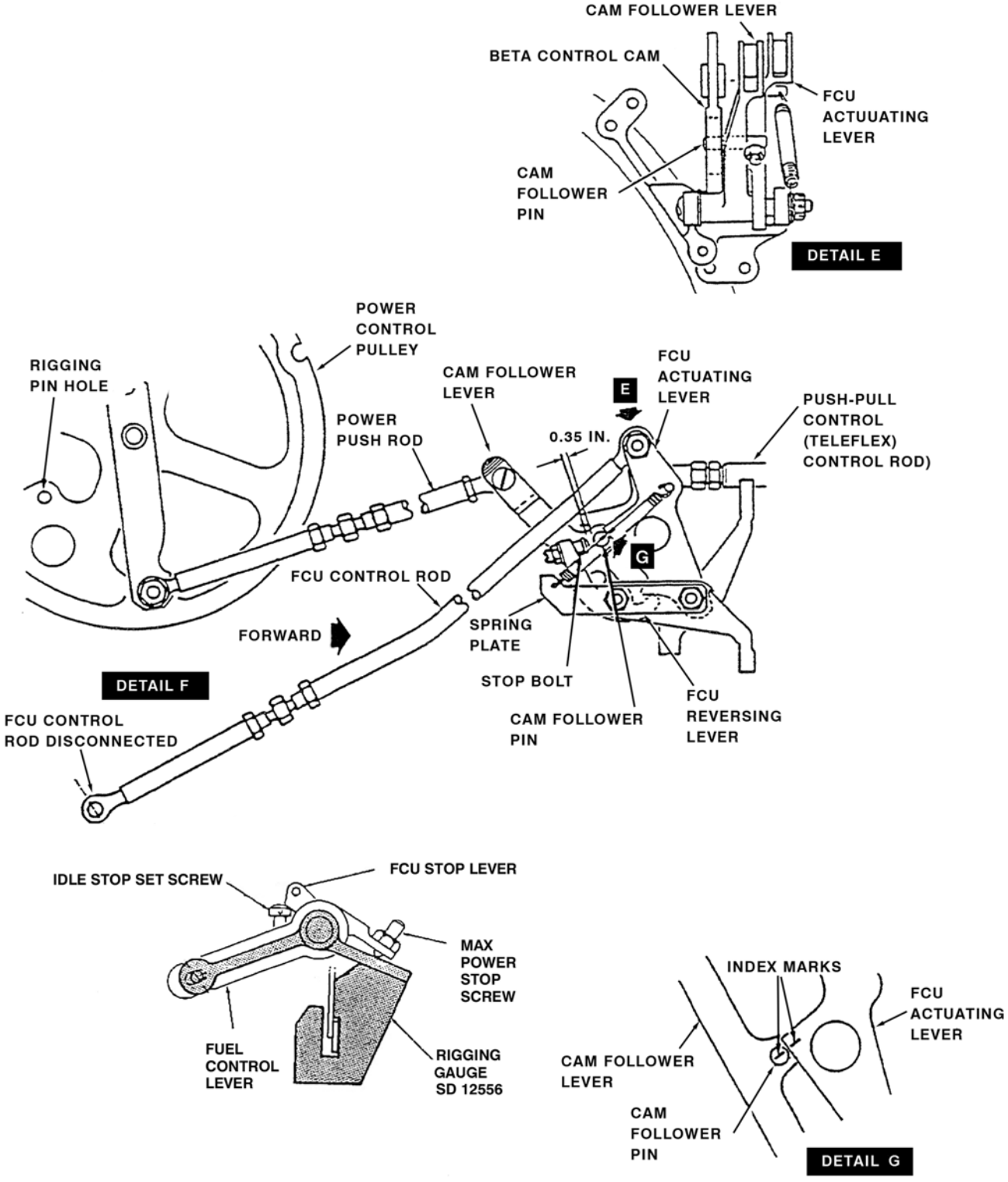
g) Adjust length of FCU control rod (refer to Figure 4) to compensate for any change in FCU linkage rigging (refer to Rig Power Lever Controls or 76-10-00, Power Control – Maintenance Practices, Para 2. Rig Power Lever Control. Ensure alignment of index marks on cam follower pin and FCU actuating lever.

h) Remove pulley rigging pin.

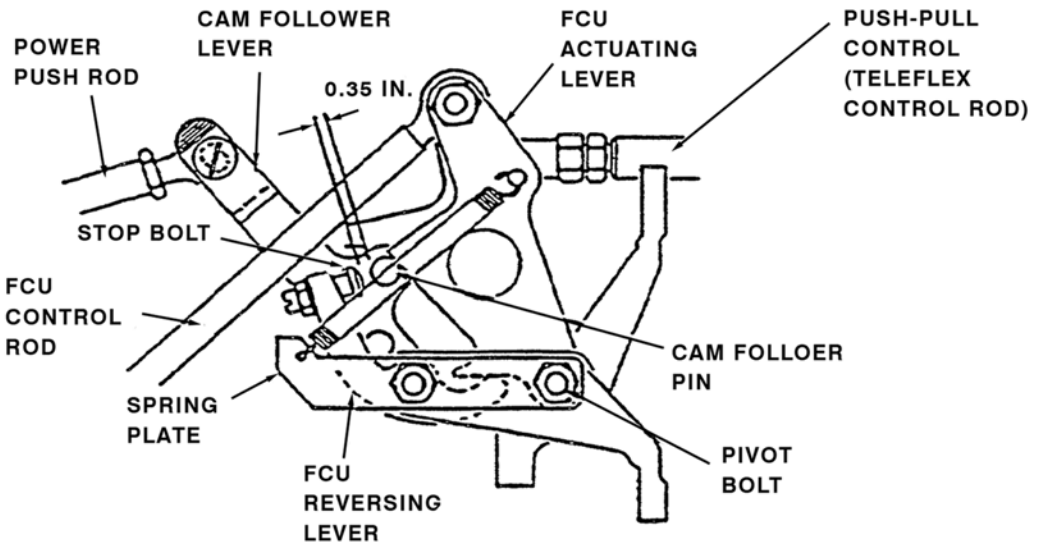
i) Repeat Para 10.B.4).a) to Para 10.B.4).d).



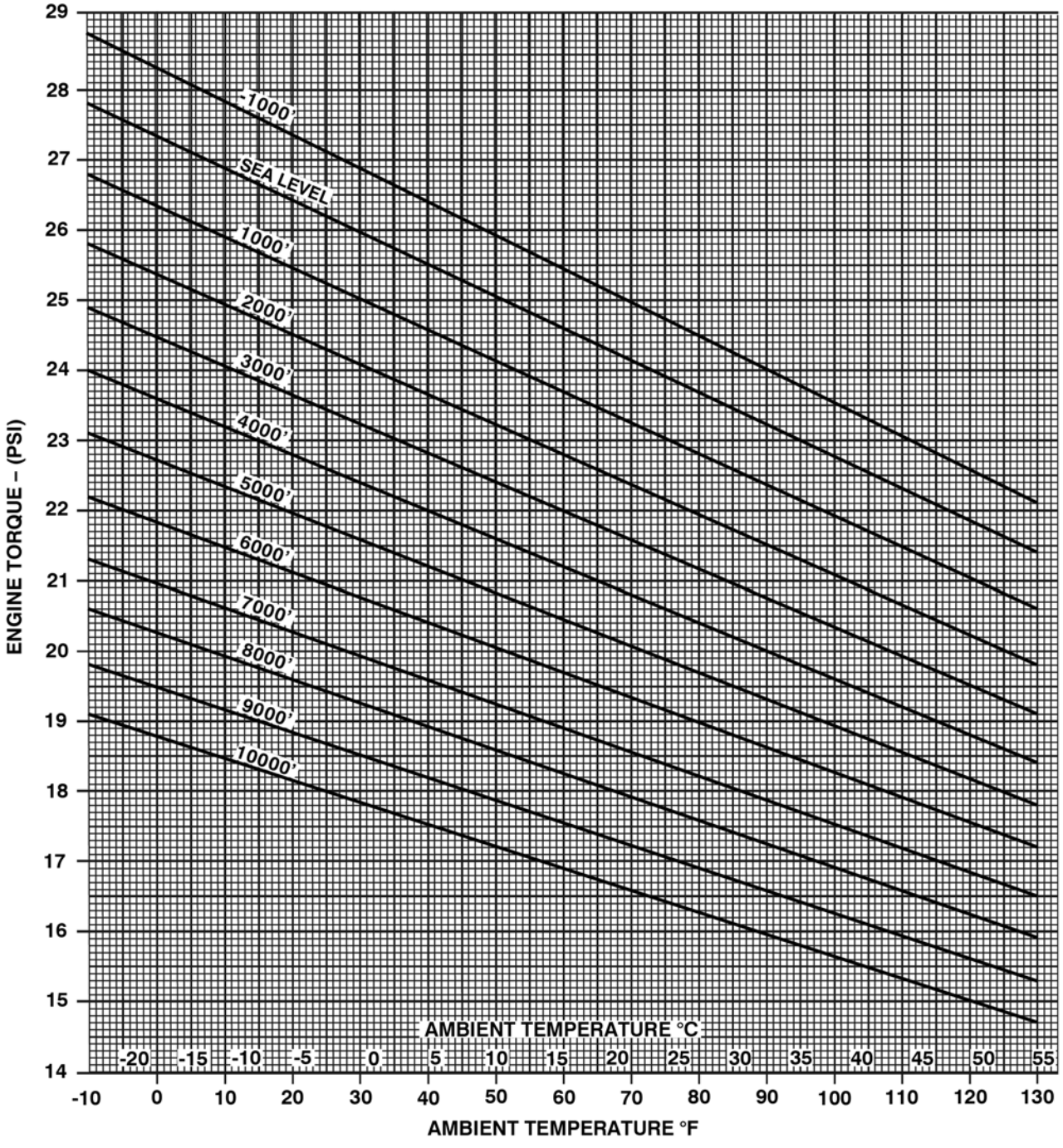
Power Lever Control System Rigging
Figure 4 (Sheet 1 of 3)



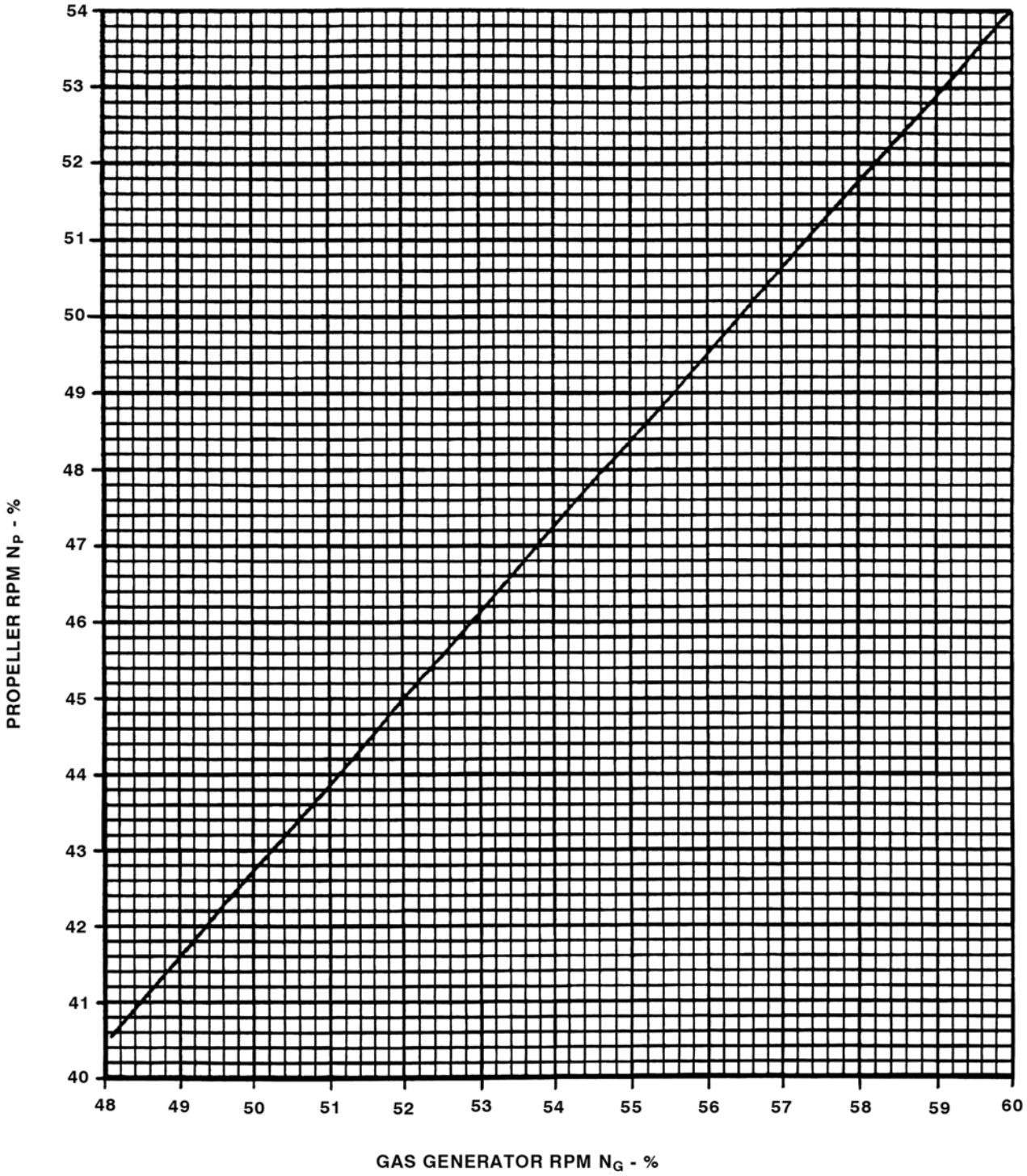
Power Lever Control System Rigging
Figure 4 (Sheet 2 of 3)



Power Lever Control System Rigging
Figure 4 (Sheet 3 of 3)



Prime Blade Angle Curve
Figure 5



Idle Blade Angle Curve
Figure 6

5) Test beta light system.

Note For test and adjustment of power lever-operated microswitch refer to 61-24-13, Beta Microswitch Assembly (Mod 6/2022) – General Data, Para 3. Adjustment/Test.

Reverse power operation is limited to 50 psi torque pressure with propeller speeds varying between 82% to 87% for ambient temperatures of +125° to –40°F (+51° to –40°C).

a) Run engine at stabilized idle.

b) With propeller lever at full INCREASE, rotate power lever knob and pull power lever approximately 0.5 inch (measured at fairing) into REVERSE. BETA RANGE light should come on after 0.2 inch travel of power lever.

c) Return power lever to IDLE. BETA RANGE light should go out.

d) Repeat steps for other engine.

Note If BETA RANGE lights remain on at idle, this is caused by a slip ring microswitch being incorrectly set, the circuit being closed at idle blade angle instead of approximately 2° below it. To correct this, increase gap (0.030–inch = 1° blade angle) between microswitch roller and slide block (refer to 61-24-13, Beta Microswitch Assembly (Mod 6/2022) – General Data), by adjusting microswitch position until correct operation is obtained.

6) Adjust power lever idle flat.

a) Run engine at stabilized idle.

b) Rotate power lever knob and pull power lever slowly into REVERSE until gas generator speed starts to increase. Measure movement of power lever on console fairing from IDLE detent to point of N_G speed increase.

c) If power lever movement (idle flat) measured in Para 10.B.6).b), is less than 1.2 to 1.3 inches, increase 0.350 inch gap between stop bolt and cam follower pin (refer to Figure 4). If power lever movement in Para 10.B.6).b) is greater than 1.2 to 1.3 inches, decrease gap.

d) Repeat Para 10.B.6).a) and Para 10.B.6).b) and, if necessary Para 10.B.6).c), until an idle flat dimension of 1.2 to 1.3 inches is achieved, to ensure satisfactory zero thrust stop (if fitted) engagement.

7) Adjust propeller control settings.

a) Run engine at stabilized idle.

b) Advance power lever until propeller speed (N_P) is governed at 96% to achieve 620 hp.

c) Move power lever to IDLE.

Note If propeller speed is governed outside limits in Para 10.B.7).b), carry out Para 10.B.7).d) to Para 10.B.7).f), otherwise omit these steps.

d) Shutdown engine (refer to Para 8.F).

e) Adjust maximum rpm stop bolt on propeller governor (refer to Figure 2) to achieve 620 hp propeller speed.

f) Start engine (refer to Para 8.) and stabilize at idle.

g) Retard propeller lever to minimum rpm detent and apply friction lock.

h) Advance power lever to achieve 75% ± 1% N_P. Adjust sliding rod end of teleflex control (refer to Figure 9) if necessary, to achieve 75% ± 1% N_P.

Note Compensating adjustments to match propeller levers can be achieved by moving position of sliding rod end in the slot of speed select lever (refer to Figure 9). Power lever matching can be achieved by moving position of push rod in the slot of cam follower lever (refer to Figure 4).

i) Advance propeller lever to full INCREASE, apply power as required and check propeller speed is 96% + 0% or –1%.

j) Retard power lever to IDLE, and propeller lever to minimum rpm detent.

k) Advance power lever and check 75% ± 1% N_P is achieved.

8) Test maximum setting of fuel governing section of propeller governor (CSU).

a) Position aircraft so that nose wheel is approximately 10° away from engine to be run.

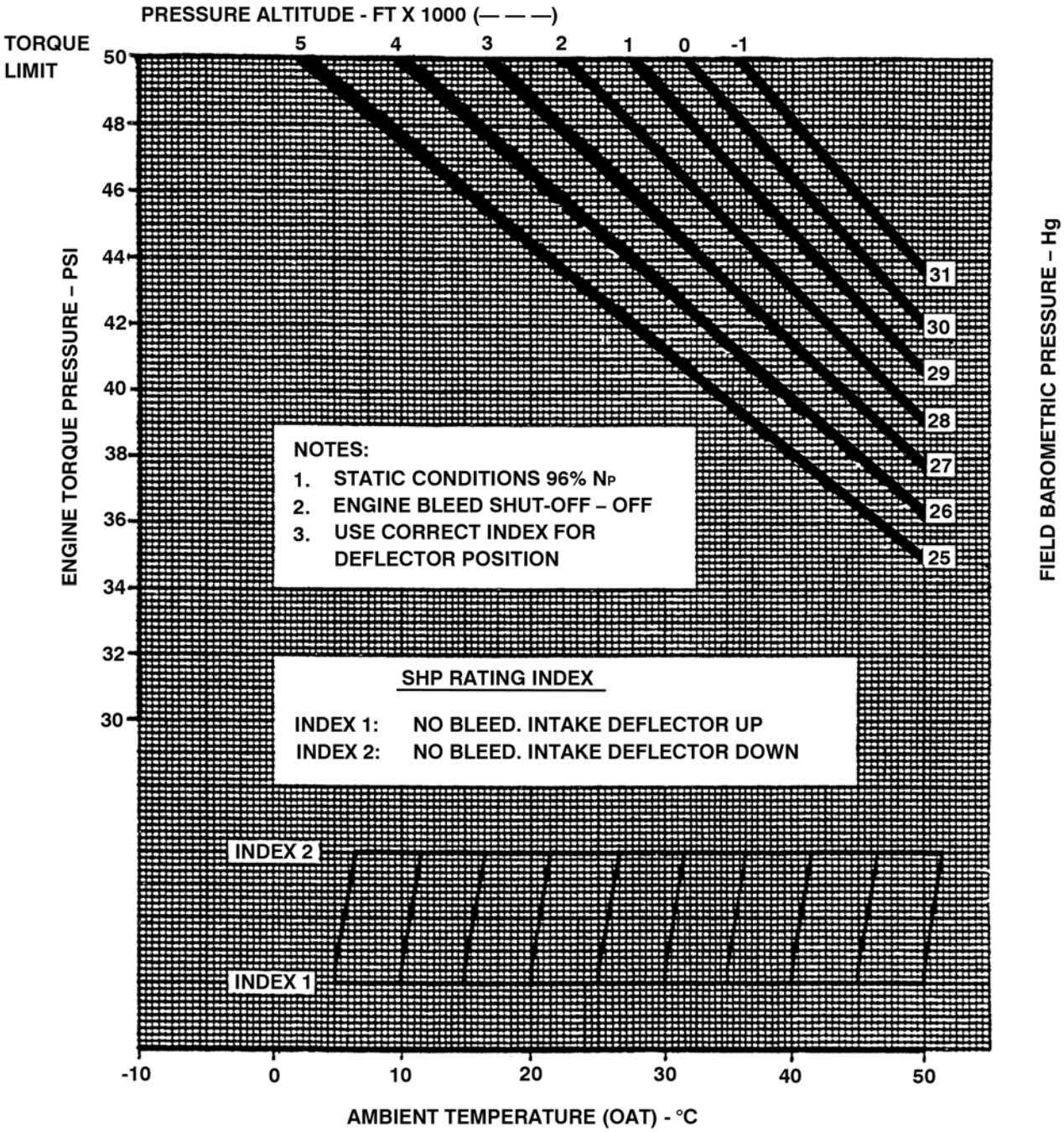
b) Obtain Outside Air Temperature (OAT) and pressure altitude. (Refer to Para 10.B.2).a).

c) Enter take-off power setting curve (refer to Figure 7) with OAT and field barometric pressure or pressure altitude and read off desired torque pressure.

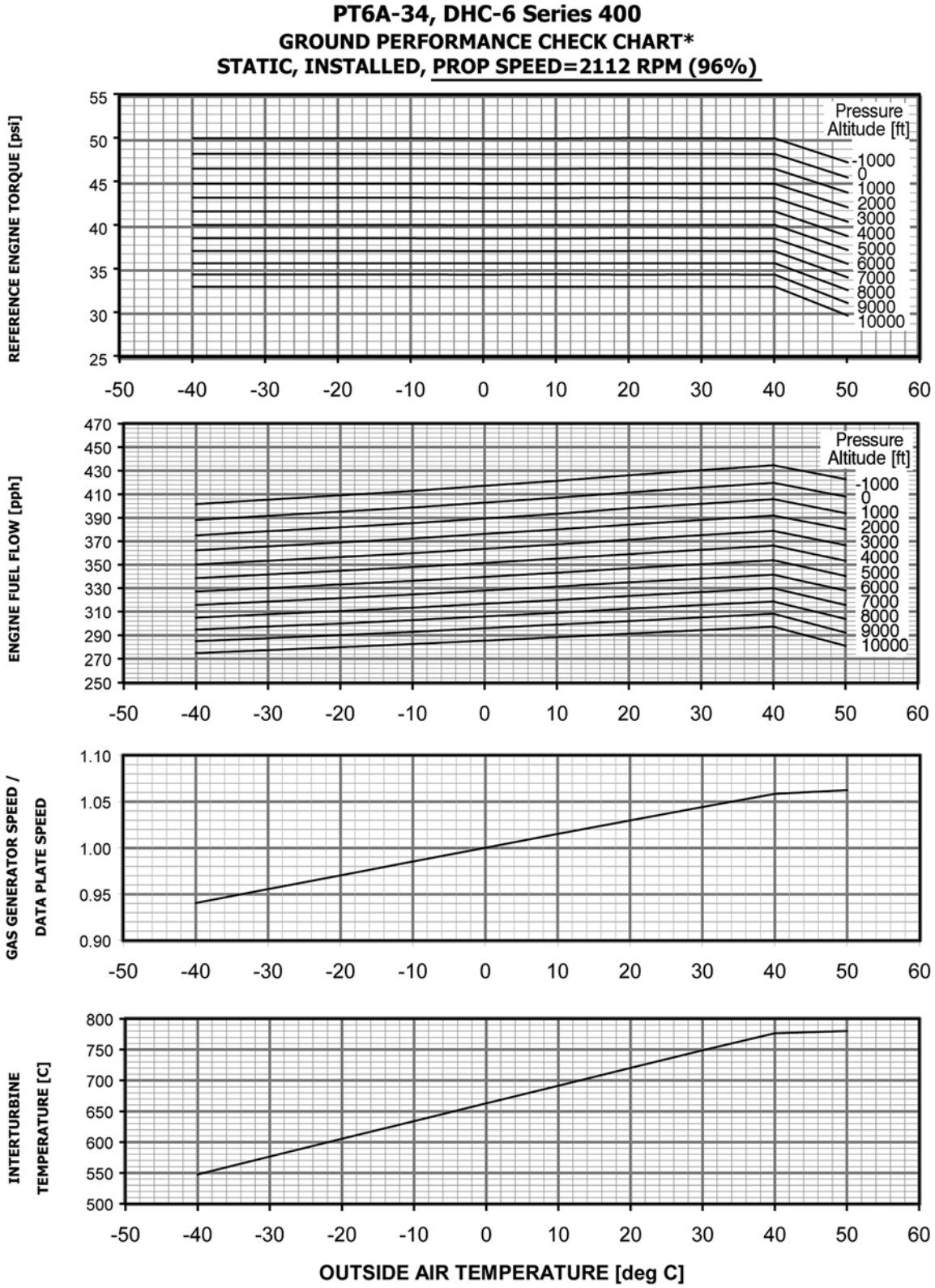
d) Ensure no leaks exist in pneumatic line from CSU to fuel control unit.

e) With engine running at stabilized idle, increase power to desired torque pressure obtained in Para 10.B.8).c). Observe operating limits (refer to Table 1 — Operating Limits).

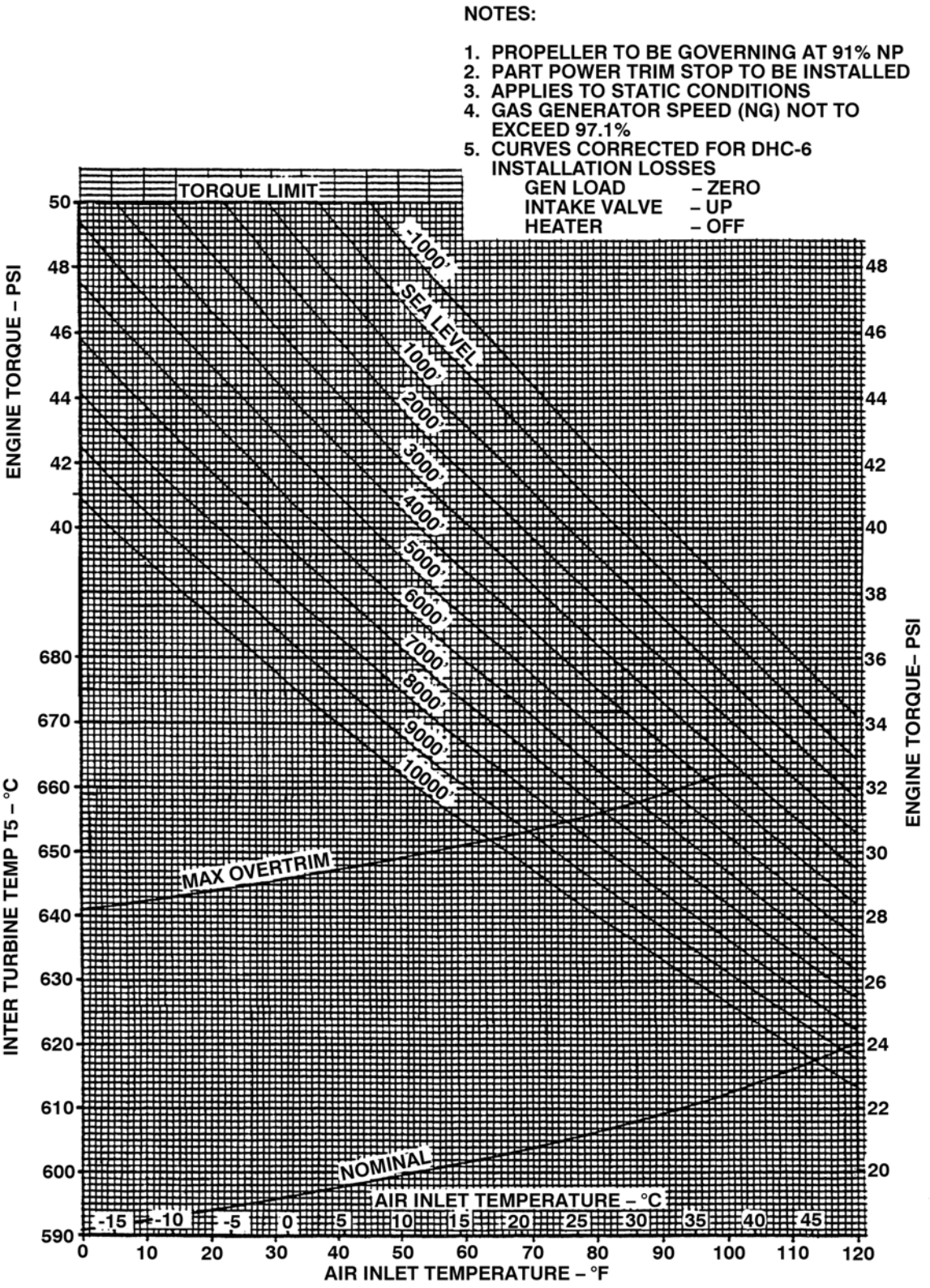
f) Ensure no hesitation in gas generator speed increase is experienced with advancing power lever and speed remains steady when desired torque pressure is reached. If unable to obtain these conditions, reject propeller governor.



Take-off Power Setting Curve
Figure 7



Engine Performance & Data Plate Speed Check
Figure 8



Part Power Trim Check Curve
Figure 9

9) Test minimum setting of fuel governing section of propeller governor (CSU).

   a) Disconnect fuel governor reset arm from fuel governor link.

   b) Move fuel governor reset arm rearwards until internal minimum stop is contacted; secure arm in this position.

   c) Run engine at stabilized idle.

   d) Advance power lever and check that propeller speed (N_P) of 91% + 1% is reached and remains steady.

   e) If N_P is out of tolerance, adjust at pneumatic minimum adjustment screw in 1/16th turn increments (refer to Figure 2).

      Note The adjustment screw connects to an internal eccentric cam, so direction of adjustment rotation cannot be defined for the initial adjustment.

   f) Run engine and recheck N_P is 91% ± 1%. Readjust if necessary.

   g) Shutdown engine (refer to Para 8.F).

   h) Connect fuel governor reset arm to fuel governor link.

10) Test propeller overspeed governor.

   a) Start engine (refer to Para 8.) and stabilize at idle.

   b) Select PROPELLER GOVERNOR test switch to up position.

   c) Advance power lever and observe that propeller governs at 70% ± 2% propeller speed as power is applied.

      Note This should occur at approximately 20 psi, torque pressure varying slightly with changing ambients and altitude. Limit torque pressure is 25 psi during test.

      CAUTION DO NOT RELEASE TEST SWITCH WHILE POWER LEVEL IS ADVANCED.

   d) Reduce power to idle.

   e) Release test switch.

11) Test propeller reset caution light and power/propeller lever interlock.

   a) Run both engines at stabilized idle.

   b) Advance power levers to 75% N_G. Mark position of levers.

   c) Retard propeller levers to just below 90% N_P. Mark position of levers, and check Reset Props caution message is illuminated.

   d) If the caution fails to illuminate, check that the PROPELLER OVERSPEED GOVERNOR circuit breaker is set.

      Note If the caution remains out proceed with Para 10.B.11).e) to Para 10.B.11).l), otherwise omit.

   e) Shutdown engines (refer to Para 8.F) – MASTER switch ON (Pre Mod 6/2314); or EXTERNAL and BATTERY switches ON (Post Mod 6/2314).

   f) Position propeller and power levers as in Para 10.B.11).b) and Para 10.B.11).c).

g) Remove upholstery from cabin roof in vicinity of stations 140.05 to 147.85 (refer to Figure 10).

h) Check microswitches are not actuated.

i) Adjust if necessary by loosening bolt securing slide to cable, and moving slide so that it contacts but does not operate the microswitch.

j) Secure slide to cable by tightening bolt.

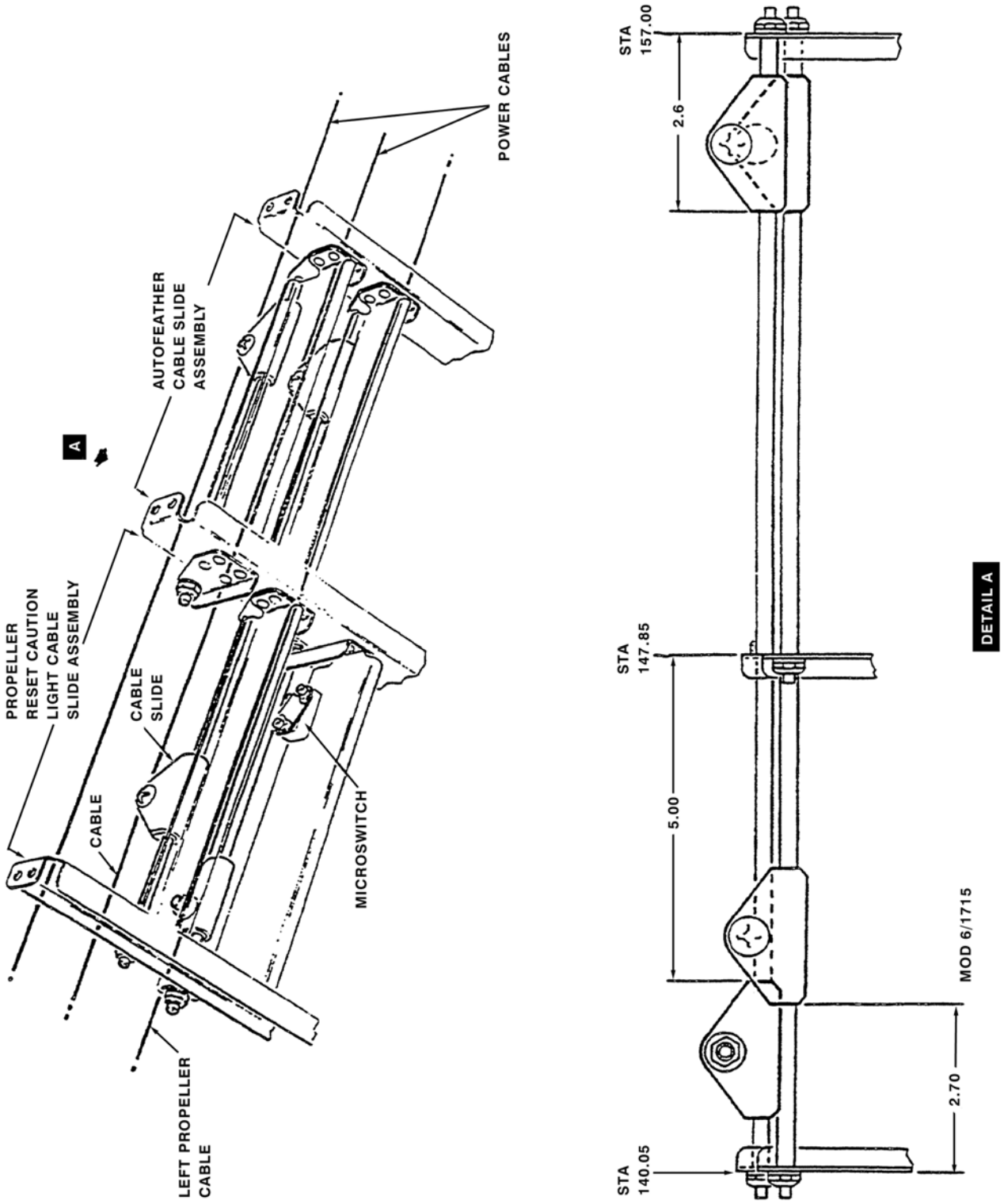
k) Check Reset Props caution message is ON.

l) Replace upholstery in cabin roof.

m) Check the caution goes out when the power levers or propeller levers are advanced past positions marked in Para 10.B.11).b) and Para 10.B.11).c).

n) Position propeller levers to 90% N_P and retard power levers to IDLE.

o) Check that power levers cannot be retarded through the idle gate at the propeller lever setting in Para 10.B.11).n) or at any lower propeller lever setting.



Rigging of Cable Slides
Figure 10

12) Test autofeather system.

a) Run both engines at stabilized idle.

b) Advance both power levers to obtain torque greater than 26 psi.

c) Select AUTOFEATHER switch to ON and check that SELECTED light illuminates.

d) Hold AUTOFEATHER test switch to test and check that ARMED light illuminates.

e) Retard left power lever to give less than 12 psi torque. Check that ARMED light goes out and two seconds later left propeller feathers. Retard right power lever to give less than 12 psi torque and check that left propeller unfeathers.

f) Release AUTOFEATHER test switch.

g) Repeat Para 10.B.12).b) to Para 10.B.12).f) for right engine, substituting right for left and left for right in Para 10.B.12).e).

h) Advance both power levers to obtain torque greater than 26 psi.

i) Advance left power lever to above 88% N_G. Check that ARM light does not illuminate.

j) Advance right power lever to above 88% N_G and check that ARM light illuminates.

k) Retard left power lever to below 88% N_G and check that ARM light goes out.

l) Retard right power lever to below 88% N_G and check that ARM light remains out.

m) Return both power levers to IDLE. Select AUTOFEATHER switch to OFF and check that SELECTED light goes out.

13) Test engine performance and data plate speed.

Note This test must be carried out with generator OFF, intake deflector retracted and bleed air shut-off valve closed. The Engine Performance and Data Plate Speed Check Curve (refer to Figure 8) is used to establish desired torque pressure, fuel flow, gas generator speed/data plate speed ratio, and turbine temperature. By periodic checking of these parameters, any changes in engine performance can be noted.

a) Obtain ambient air temperature (OAT); maximum limit is 43°C (109.4°F).

b) Obtain pressure altitude (refer to Para 10.B.2).a)).

c) Obtain reference rpm from engine data plate.

d) Enter engine performance check curve with pressure altitude and proceed horizontally left to read torque pressure and record.

e) Enter curve with ambient temperature and proceed vertically upward to intersect appropriate pressure altitude line. From this intersection, proceed horizontally left to read fuel flow in lb/hr and record.

f) Enter curve with ambient temperature and proceed vertically upward to intersect gas generator/data plate speed line. From this intersection, proceed horizontally left to read desired ratio (gas generator speed/data plate speed) and record.

g) Enter curve with ambient temperature and proceed vertically upward to intersect turbine temperature line. From this intersection, proceed horizontally right to read turbine temperature and record.

h) Start engine (refer to Para 8.).

i) Run engine and stabilize at idle.

j) Set propeller control lever to give 96% N_P, with power lever set to give torque setting previously determined in Para 10.B.13).d).

k) Compare observed fuel flow, gas generator speed and turbine temperature with values previously recorded.

l) Ensure that values observed during engine performance and data plate speed test are within the following limits:

	Fuel flow	+15 or –25 lb/hr
Ratio:	Gas generator speed / Data plate ratio	± 0.02
	Turbine temperature T_5	within maximum limit as determined from Figure 8

Note If performance deviates from preceding limits, an instrument or engine fault is indicated (refer to Para 2. and Pratt and Whitney PT6A–34 Maintenance Manual).

m) Shutdown engine (refer to Para 8.F).

14) Check part power trim.

Note Fuel control units are calibrated during bench test to provide maximum power and temperature rating on an 18°C (65°F) day. If engine inlet temperature is other than 18°C (65°F), it is not possible to set maximum gas generator speed stop, since either a turbine temperature (T_5) or a torque limit would be reached first. To overcome this, a part power trim stop is provided. The stop is a movable spacer placed between gas generator maximum power stop anvil and max power stop bolt (refer to Figure 4). The part power trim stop is attached by a screw in its stowed position to the side of the maximum power stop anvil. This check cannot be carried out at ambient temperatures below –1°C (+30°F) as the upper torque pressure limit will be reached before contacting part power trim stop.

a) Loosen screw securing part power trim stop and rotate into position against face of maximum power stop anvil. Tighten screw.

b) Obtain outside air temperature (OAT).

c) Obtain pressure altitude (refer to Para 10.B.2).a)).

d) Enter part power trim check curve (refer to Figure 9) with ambient temperature, and proceed vertically upward to intersect pressure altitude line. From this intersection, proceed horizontally left and read engine output torque. Record value obtained.

e) Enter curve with ambient temperature and proceed vertically upward to intersect turbine temperature line. From this intersection, proceed horizontally left to read turbine temperature T_5. Record value obtained.

f) Start engine (refer to Para 8.) and stabilize at idle with bleed air valves off, generator off and intake deflector retracted.

g) Advance power lever to part power trim stop and allow engine to stabilize at this setting for approximately five minutes at a propeller speed of 91%.

h) Observe torque pressure, turbine temperature and maximum gas generator speed.

CAUTION **GAS GENERATOR SPEED MUST NOT EXCEED 97.1% AT ANY TIME DURING CHECK AND TURBINE TEMPERATURE MUST NOT EXCEED THE VALUE SHOWN ON MAXIMUM OVERTRIM CURVE (REFER TO Figure 9).**

i) Check that the maximum N_G speed is set at 97.1%, the T_5–ITT limit is not exceeded, and the torque meter pressure is equal to or above the referenced pressure (refer to Figure 9). Adjust the N_G maximum power stop bolt on FCU spindle arm, as necessary (refer to Figure 4, Detail D). If the T_5–ITT limit is exceeded, or the torque meter pressure is below the referenced pressure at 97.1% N_G, rectification is necessary prior to further flight.

Note If after adjustment, flight snags indicate power limitations at altitude (operating limit not reached), refer to engine manufacturer for analysis and recommendations.

j) Shut down engine (refer to Para 8.).

k) Stow and lockwire part power trim stop.

CAUTION **PART POWER TRIM STOP MUST BE STOWED AFTER COMPLETION OF PART POWER TRIM CHECK. ENSURE THAT ATTACHING SCREW IS TIGHTENED AND LOCKWIRED.**

l) Lockwire maximum power stop bolt.

15) Match power levers.

a) Attach a strip of masking tape along both edges of face plate between power levers.

b) From Figure 5 obtain required torque pressure for prime blade angle at prevailing ambient temperature and pressure altitude.

c) Start both engines and stabilize at idle.

d) With propeller levers at full INCREASE, advance both power levers to obtain torque pressure determined in Para 10.B.15).b).

e) Mark positions of power levers on masking tape. Shutdown engines and set power levers to IDLE.

f) From marked positions, determine if one power lever is leading.

g) Adjust leading power lever by repositioning power push rod in cam follower lever slot and adjusting its length accordingly (refer to Figure 4). Move power push rod towards inner end of slot to reduce lead.

Note Ensure that cam follower lever position is not moved during adjustment and that power lever is at IDLE.

h) Repeat Para 10.B.15).c) to Para 10.B.15).g) until satisfactory matching is obtained.

Note If unable to match after adjusting as in Para 10.B.15).g), carry out Para 10.B.15).i) as follows:

i) Adjust length of FCU control rod and reposition in slot to FCU arm (refer to Figure 4). Move FCU control rod towards inner end of slot to reduce lead.

Note Ensure that FCU arm and FCU actuating lever are not moved during adjustment.

j) Start both engines and stabilize at idle.

k) Advance both power levers, in increments of 5 psi torque pressure, through full power range, and check matching at each increment.

l) Shudown engines and remove masking tape.

16) Match propeller levers.

a) Attach a strip of masking tape along the outer edges of each propeller slot.

b) Start both engines and stabilize at idle.

c) With propeller levers at full INCREASE, advance both power levers to obtain 95% N_P and tighten friction damper knobs.

d) Retard both propeller levers from full INCREASE to obtain 90% N_P.

e) Mark position of propeller levers on masking tape. Shutdown both engines.

f) From marked positions, determine if one propeller lever is lagging.

g) Adjust lagging propeller lever as follows:

1 Position lagging propeller lever on its 90% mark and secure speed select lever on governor (refer to Figure 11).

2 Disconnect sliding rod end of teleflex control from speed select lever (refer to Figure 11).

3 Position lagging propeller lever in line with 90% mark of leading propeller lever.

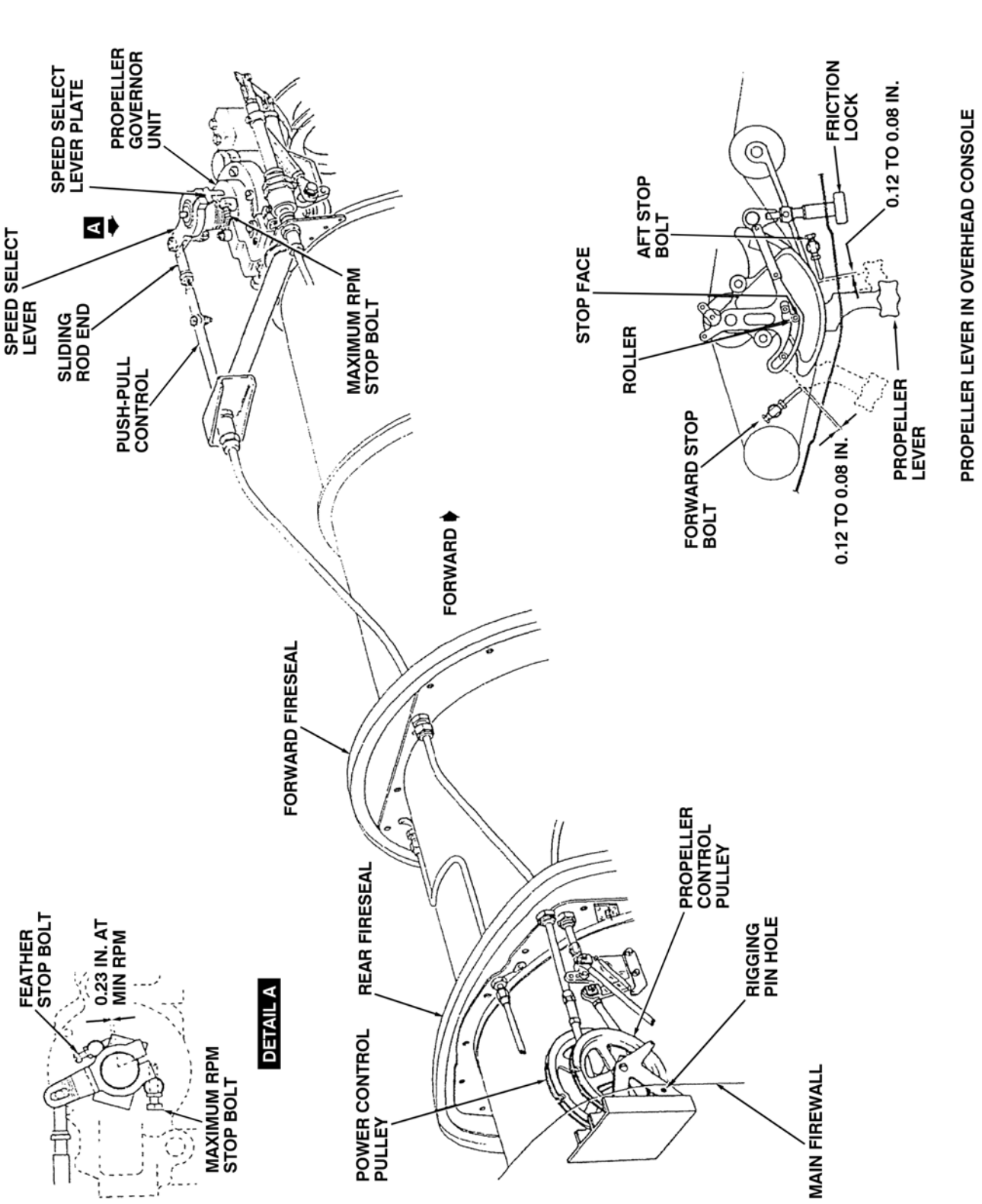
4 Connect teleflex control to speed select lever by moving sliding rod end along slot in speed select lever (refer to Figure 11). Sliding rod end should move towards inner end of slot to reduce lag.

Note If sufficient adjustment is not available in slot, split the difference by also adjusting the leading propeller lever in the opposite direction.

5 Release speed select lever and check that speed select lever plate contacts the maximum rpm stop bolt and feather stop bolt (refer to Figure 11) when the propeller lever is operated through the full range of travel.

h) Repeat Para 10.B.16).b) to Para 10.B.16).g) until satisfactory matching is obtained.

i) Remove masking tape.



Rigging Propeller Control System
Figure 11

11. Check Ground Running Data

The ground run check sheet (refer to Figure 12) can be used to record information obtained during engine running as detailed in Para 10.B.

GROUND RUN CHECK

DHC-6 Twin Otter (Series 400)

TYPE CHECK		LOCATION	
AIRCRAFT NUMBER		SHUT DOWN	L. R.
ENGINE NUMBERS	L. R.	START UP	L. R.
DATE		RUNNING TIME	L. R.

✲ AMBIENT CONDITION	OAT	FBP	PRESS. ALT

ENGINE RUN	LEFT	RIGHT
MAX. TURB TEMP AT START (LIMIT 1090°C FOR 2 SECS)		

✲ NOTES:
1. FOR FUNCTION OF SYSTEM OR CHECK. USE √ FOR SATISFACTORY OR X FOR UNASTISFACTORY IN SPACE PROVIDED.
2. OBTAIN (a) FROM APPLICABLE CURVE (FIGS 5 TO 9) AND (b) FROM APPLICABLE INDICATOR.
3. ABBREVIATIONS:
   OAT - OUTSIDE AIR TEMPERATURE.
   FBP - FIELD BAROMETRIC PRESSURE.
4. MAKE LOG BOOK ENTRIES AS REQUIRED.
5. TO ACHIEVE 620 HP TAKE-OFF.

✲ GAS GENERATOR IDLE	GAS GEN	RPM (N_G)
52% N_G $^{+1\%}_{-0\%}$	%	%
	%	%

✲ PROPELLER PRIME	TORQUE PRESS.	
BLADE ANGLE CHECK	psi	psi
(77% Np)	psi	psi

✲ PROPELLER IDLE	GAS GEN	RPM (N_G)
BLADE ANGLE (b)	%	%
CHECK	PROP. RPM (N_P)	
	%	%
	%	%

	✲ FUNCTION	
PROP. CONTROL CHECK	PROP. RPM (N_P)	
✲ GOVERNED (96% N_P) (5)	%	%
MIN. RPM (74% - 76% N_P)	%	%
PROPELLER GOVERNOR FUEL SECTION MIN. SETTING CHECK (90% - 92% N_P)	PROP. RPM (N_P)	
	%	%

PROPELLER O/SPEED GOV CHECK (68% - 72% N_P)	PROP. RPM (N_P)	
	%	%
POWER/PROP. LEVER INTERLOCK CHECK	✲ FUNCTION	
RESET PROP. CHECK		
AUTOFEATHER CHECK		

✲ DATA PLATE SPEED CHECK (MAX. OAT LIMIT 110°F)	(a)	L.	R.
TORQUE PRESS. psi			
FUEL FLOW pph			
Ng/DATA PLATE SPEED			
TURBINE TEMP °C			

✲ PART POWER TRIM CHECK (MIN. OAT LIMIT 30°F)		L.	R.
TORQUE PRESS. psi			
TURBINE TEMP °C			
MAX. N_G (LIMIT 97.1%) %	X		

CONTROLS MATCH PROP LEVERS

AIRCRAFT SYSTEMS CHECK	
SYSTEM	✲ FUNCTION

REMARKS/FAULTS:

✲ OPERATOR'S SIGNATURE _____

Ground Run Check Sheet
Figure 12

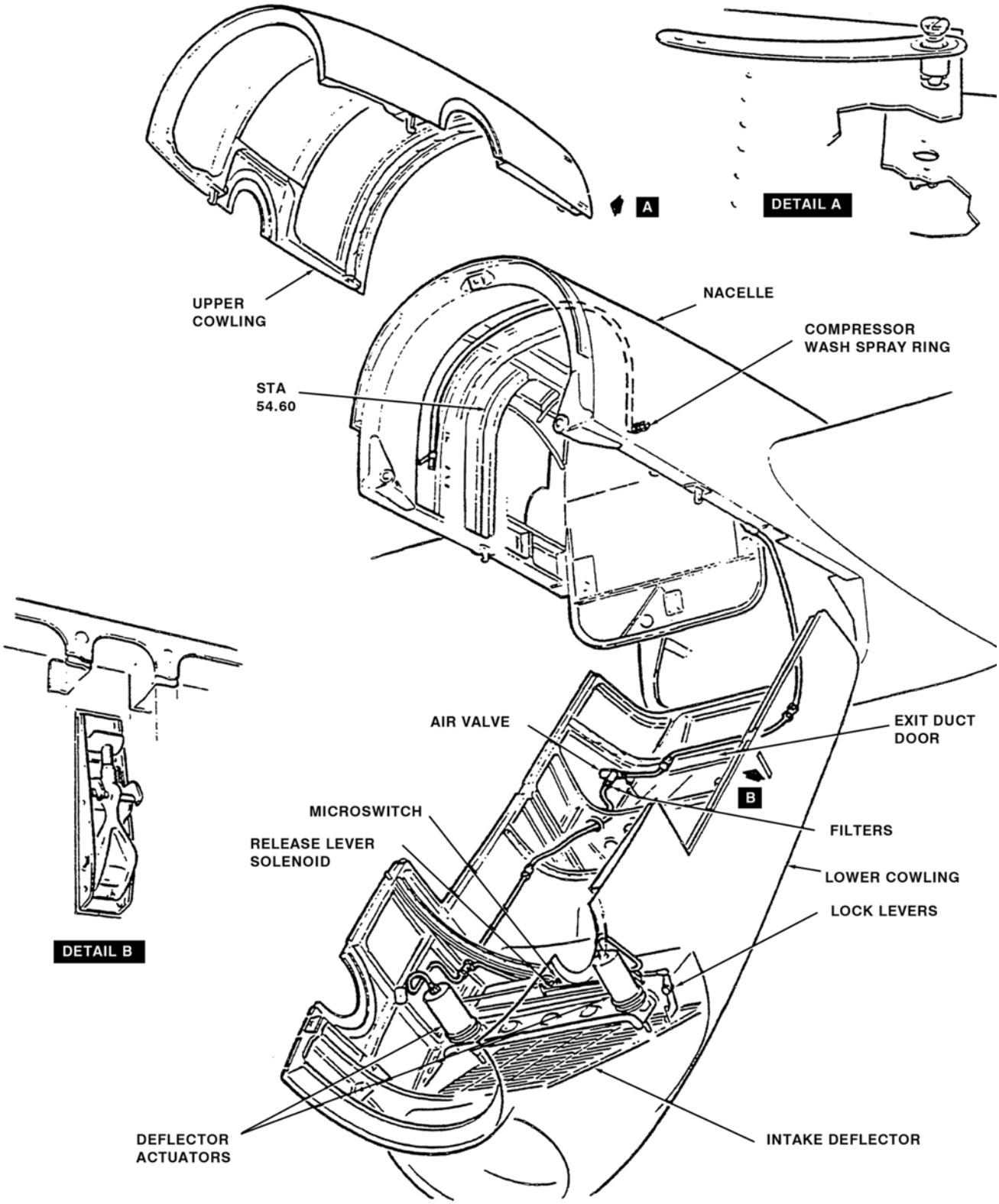
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Cowlings – General Data

1. General

Refer to Figure 1 for details.

Two engine cowlings (upper and lower) are provided to enclose each engine and accessories. Both cowlings fasten to each other and to the nacelle structure. The upper cowling is equipped with an aft facing air duct which provides an exhaust outlet for the engine compartment ventilation. Cooling air enters through an air inlet duct in the lower cowling. The lower cowling also contains an air intake duct for the engine and an air duct to direct a flow of air through the oil cooler. As protection against snow and ice, a retractable intake deflector and exit duct door, both electro-pneumatically operated, are installed in the intake duct. Further protection of the intake is achieved by a permanent screen fitted to the intake throat.



Engine Cowlings
Figure 1

2. Removal/Installation

A. Remove Cowlings

Note Two people are required for removal of lower cowling.

1) Release stud fasteners along rear end and latches on lower sides of upper cowling.

2) Lift off upper cowling.

3) Support lower cowling, release side latches and lower cowling.

4) Disconnect flexible air line and electrical connector to intake deflector.

5) Remove bolts securing rear hinge to cowling. Remove cowling.

6) Blank open air line.

B. Install Cowlings

1) Position lower cowling to rear hinge on nacelle and bolt cowling to hinge.

2) Connect flexible air line and electrical connector to intake deflector system.

3) Raise lower cowling and fasten side latches.

4) Place upper cowling in position and fasten rear stud fasteners and lower side latches.

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Mounts – General Data

1. General

Refer to Figure 1 for details.

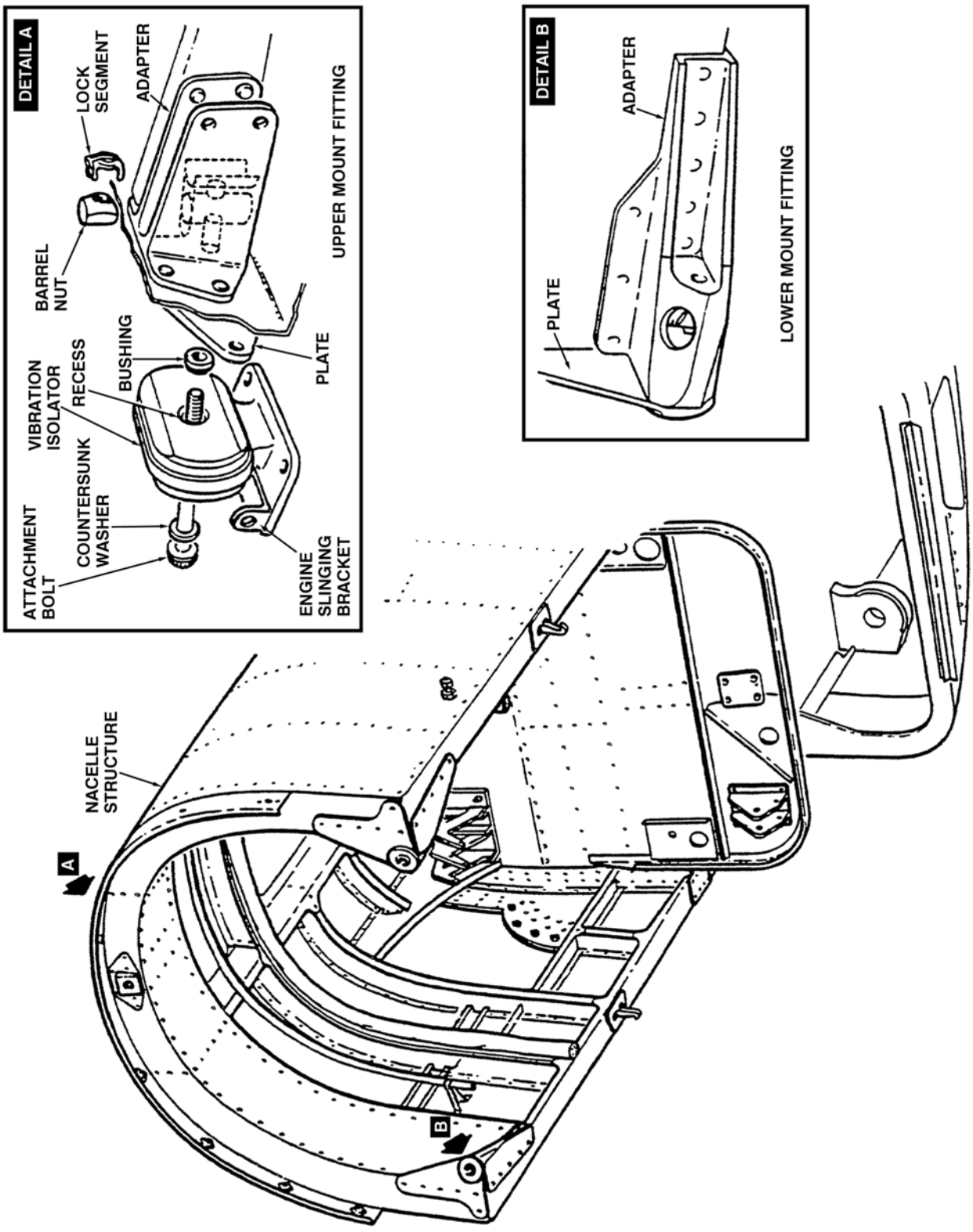
The engine is supported in the nacelle at three points and attached to each point by a single engine attachment bolt. Engine vibration is dampened by means of vibration isolators, mounted on the engine, through which pass the engine attachment bolts.

A. Engine Mount

The engine mount consists of the front frame of the nacelle structure and three fittings which are riveted to the frame. Each fitting is in three parts; adapter, plate and bushing. The adapter is riveted to the nacelle structure on the rear side of the front frame, and the plate is positioned on the front face of the front frame and riveted to the adapter flange. A bushing is inserted in the plate of the fitting and can be replaced if damaged. Each adapter is drilled to provide the housing for a barrel type nut. The fittings are not interchangeable, but the bushings are.

B. Vibration Isolators

Three vibration isolators are required to support each engine in the nacelle. The vibration isolators are each secured to the engine combustion casing by four bolts, and by a single bolt to the nacelle structure attachment point. The two front bolts securing the vibration isolator on the top of the engine also secure the rear engine slinging bracket. A vibration isolator can be replaced without removing the engine provided the engine weight is supported.



Engine Mounting
Figure 1

Vibration Isolator – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Vibration Isolator

1) Remove engine cowlings.

2) Support engine weight.

3) Remove engine to mount attachment bolt and washer, using wrench SD12524. Remove barrel nut and lock segment.

4) Cut lockwire and remove four bolts and washers securing vibration isolator to engine. On top isolator only, remove engine slinging bracket with the two front securing bolts.

5) Remove vibration isolator and gasket.

6) Check vibration isolator for deformation, overhaul, or replacement. (Refer to Para 3.A.1)).

Note Change complete vibration isolator engine mount or partial, as required in accordance with Para 3., Inspection/Check.

B. Install Vibration Isolator

1) Install countersunk washer under engine to mount attachment bolt head and insert bolt in vibration isolator. Ensure recess in vibration isolator is towards nacelle structure attachment point.

2) Check that mounting faces are clean and undamaged.

3) Position gasket (supplied with vibration isolator) and vibration isolator on engine mounting face.

4) Install four vibration isolator retaining bolts and washers. Torque to 200 to 250 pound–inches and lockwire.

Note Install four new vibration isolator retaining bolts at each engine change (refer to Table 1 — Hardware List to Attach Engine Vibration Mount to Nacelle and Engine).

On top isolator only, the two front securing bolts are longer and also secure the rear engine slinging bracket. One washer is installed between the slinging bracket and the isolator and one washer under the head of each bolt.

5) Secure engine attachment bolt to barrel nut and lock segment, using wrench SD12524. Torque bolt to 325 to 375 pound–inches.

6) Remove engine sling.

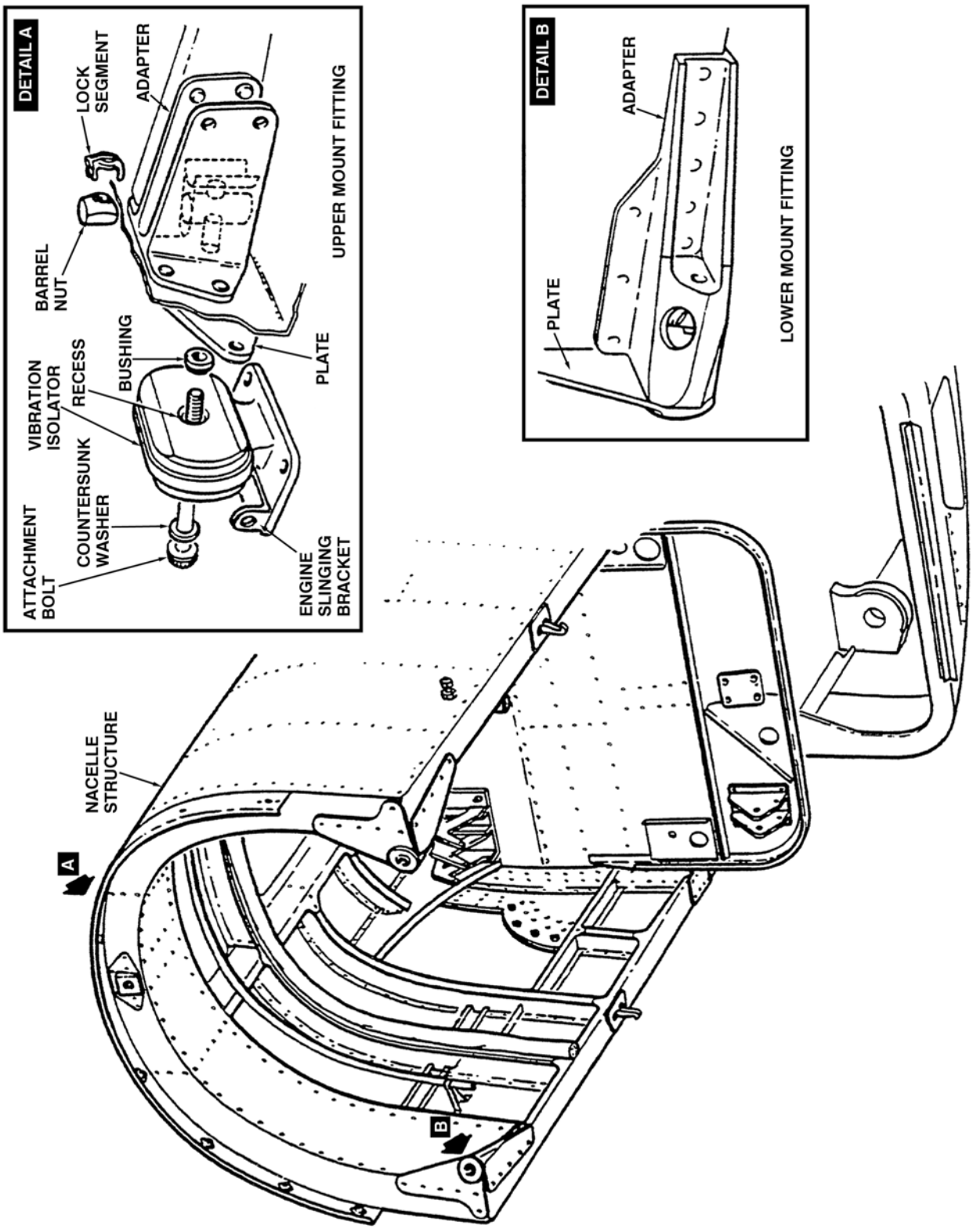
7) Install cowlings.

Table 1 Hardware List to Attach Engine Vibration Mount to Nacelle and Engine

QTY PER ENGINE	DESCRIPTION	400 P/N
3	BOLT – VIBRATION ISOLATOR ASSY TO NACELLE	NASM21250–06042
3	WASHER – VIBRATION ISOLATOR ASSY TO NACELLE	NASM20002C–6

Table 1 Hardware List to Attach Engine Vibration Mount to Nacelle and Engine (continued)

QTY PER ENGINE	DESCRIPTION	400 P/N
3	NUT – BARREL, VIBRATION ISOLATOR ASSY TO NACELLE	B42–624
2	FOR TOP MOUNT ONLY REAR BOLTS – SPECIAL, VIBRATION ISOLATOR ASSY TO ENGINE CASE	CSP282–29
	AND	
2	FRONT BOLTS – VIBRATION ISOLATOR ASSY TO ENGINE CASE	AN6H6A
8	FOR SIDE MOUNTS BOLT – SPECIAL, VIBRATION ISOLATOR ASSY TO ENGINE CASE	CP282–29
14	WITH MOD 6/1706 WASHER – FLAT, VIBRATION ISOLATOR ASSY TO ENGINE CASE	NAS1149C0663R
3	GASKET – VIBRATION ISOLATOR ASSY TO ENGINE CASE	R20802–2



Engine Mounting
Figure 1

2. Disassembly/Assembly

Refer to Figure 2 for details.

A. Disassemble Engine Vibration Isolator Assembly

1) The core (2) is press fit on stud (3). Pull core off stud by hand or use soft mallet to tap housing away from core.

2) Remove core (8) and stud (3) from housing (4).

3) Remove bumper tube (6) from stud (3).

CAUTION **WHEN DRILLING, BE CAREFUL NOT TO DAMAGE CORE (8). CORES DAMAGED DURING DRILLING MUST BE REPLACED.**

4) Hold protruding end of stud (3) and drill out flared portion of stud (3) in core (8) and discard stud.

5) Remove and discard gasket (5).

6) Visually inspect cores for damage.

Note Core assemblies removed from an engine vibration isolator are to be retained as a matched pair, visually inspected and re-installed as a matched pair.

7) If one core has visible damage, replace both cores as a pair.

Note If engine vibration isolator housing does not pass the inspection requirement, replace the entire engine vibration isolator assembly and associated hardware as shown in Table 2 — Component List for Figure 2.

B. Assemble the Engine Vibration Isolator

1) Insert small end of stud (3) into core (8).

a) Flare the stud in the countersink of core (8) with 100° flare tool.

Note Flare stud only enough to hold core so it will not come off the stud. Too much flare could crack the stud. The deep countersink in the core is to ensure the flared end of the stud is below the surface of the core.

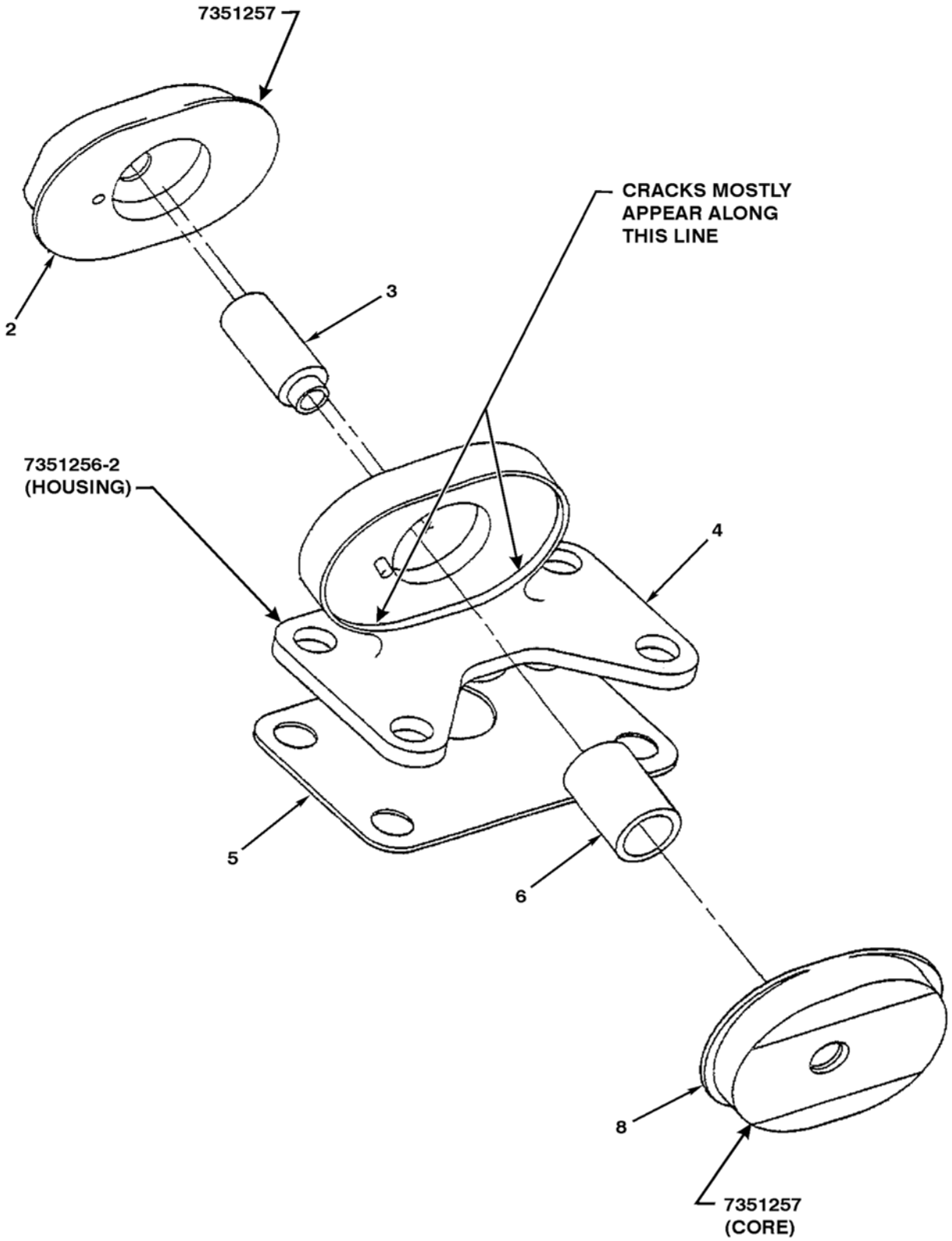
2) Install bumper tube (6) over stud and place sub-assembly on housing (4) aligning the pin on the housing with the hole in the core.

Note The spring pin used on housing P/N 7351256–2 (refer to Figure 2) and the cast pin used on the current production housing assure correct sub-assembly orientation and do not carry load. The blind hole in the current production core assemblies has been enlarged to simplify assembly. If core assemblies with a smaller diameter blind hole are to be installed on the cast pin housing, enlarge blind hole with a 5/32 inch drill to facilitate installation.

3) Press fit the core (2) onto stud (3).

4) To keep the press fit core (2) on the stud (3) for handling and storage purposes, place a plastic bonding tie strap or equivalent through the stud hole and secure it around the two cores. In some instances the press fit is not sufficient to hold the isolator together. This happens occasionally and is considered normal. The isolator can be used as is if this occurs.

5) Install new gasket (5) on housing.



Engine Vibration Isolator
Figure 2

Table 2 Component List for Figure 2

ITEM	QTY	DESCRIPTION	PART NUMBER	SERIES
-1	1	ISOLATOR ASSEMBLY	7350721–7	
-2	1	CORE	7351257–9	
-3	1	STUD	7351261–1	
-4	1	HOUSING	7351256–2	
-5	1	GASKET	R20702–2	
-6	1	TUBE, BUMPER	7351260–1	
-8	1	CORE	7351257–7	

3. Inspection/Check

A. Condition, Separation, Corrosion Cracking, Deformation and Security Check

Refer to Figure 3 for details.

1) Remove vibration isolator and check for permanent set or sag. Maximum allowable 0.125 inch.

2) Check for delamination of elastomer. Maximum allowable 1.0 inch length or 0.0625 inch depth.

3) Check for corrosion and cracking. Pay particular attention to corrosion and cracks at base of housing (P/N 7351256–2) right next to the edge of the cores (pucks).

4) Replace vibration isolator if limits in Para 3.A.1) or Para 3.A.2) are exceeded.

5) Install vibration isolator.

B. Overhaul Check

1) Inspect the engine vibration isolator in accordance with Para 3.A.

2) Inspect the engine vibration isolator housing P/N 7351256–2, in accordance with Para 3.C.

3) If vibration isolator engine mount does not pass the Fluorescent Inspection, then the engine mount must be replaced, including the associated hardware (refer to Table 1 — Hardware List to Attach Engine Vibration Mount to Nacelle and Engine).

Note Rework limits as per PSM 1-6-6A, Component Maintenance Manual (Barry Controls Type 7350721), DO NOT APPLY.

C. Fluorescent Penetrant Inspection

Refer to Figure 2 for details.

1) Function

a) This procedure documents the Fluorescent Penetrant process used to detect discontinuities on the surface of the DHC-6 Engine Vibration Isolator Housings, P/N 7351256–2.

b) Materials:

- 17–4PH Stainless Steel Casting (130–150KSI)

c) Type/Sensitivity:

- Type I, Level 3 or 4 in accordance with ASTM–E–1417

2) Applicability

This process is applicable for maintenance (in-service) inspections of the vibration isolator housing only.

3) Tools, Materials, Equipment and Personnel

a) Fluorescent Penetrant

1 Black Lights:

The black light must have a minimum power rating of 100 Watts and a minimum acceptable intensity of 1000 µWatts/cm2 measured at a distance of at least 38 cm (15 inches) from the front of the bulb or filter. A suitable model is Magnaflux 2Z–24A P/N: 513721 or equivalent. Black light reflectors and filters shall be checked for cleanliness and integrity prior to carrying out the FPI process.

2 Materials:

- Penetrant System shall be Type I, Level 3 or 4, in accordance with ASTM–E–1417.
- The materials used for the penetrant system above shall be compatible, come from the same manufacturer and be listed on QPL–AMS–2644.
- Materials shall be tested in accordance with Table 1 of ASTM–E–1417.

3 Inspection Area:

The inspection area for stationary systems for Type I penetrants shall be checked as specified in Table 1 — Hardware List to Attach Engine Vibration Mount to Nacelle and Engine and shall be clean and free from excessive fluorescent contamination and excessive visible light background in accordance with ASTM–E–1417.

4 Personnel Qualification:

Personnel doing examinations shall be qualified and certified in accordance with ASTM–E–1417, sensitivity Level 3, or to a standard acceptable to the operators cognizant airworthiness authority.

4) Procedure

a) Surface Preparation:

Thoroughly clean the inspection surface by solvent, chemical, or mechanical cleaning method in accordance with ASTM–E–1417.

b) Penetrant System:

Penetrant system shall be Type I, Level 3 or 4 in accordance with ASTM–E–1417.

1 Penetrant Application:

Penetrant shall be applied in accordance with ASTM–E–1417.

2 Penetrant Dwell Time:

Dwell time shall be 60 minutes. The drain off time shall be considered as part of the penetrant dwell time. If the method used is immersion, parts shall be immersed in penetrant for no more than one half the dwell time. The long dwell time may result in drying of the penetrant, fresh penetrant shall be reapplied periodically to avoid drying.

3 Penetrant Removal:

Penetrant shall be removed by applicable method in accordance with ASTM–E–1417.

4 Drying:

Parts shall be dried prior to the application of developer in accordance with ASTM–E–1417.

5 Developing:

Developer application shall be done with a developer listed in QPL–AMS–2644 for Type I, Level 3 or 4 in accordance with ASTM–E–1417.

6 Inspection:

Inspection of parts shall be conducted in accordance with ASTM–E–1417, in a darkened booth using black light with a minimum intensity of 1000 µWatts/cm2 on the surface of the part being inspected. The inspector shall be in the darkened booth for at least one minute prior to doing inspection so that his/her eyesight will adapt to dark viewing. The inspector shall not wear light sensitive or permanently darkened lenses while doing the inspection.

5) Inspection Criteria

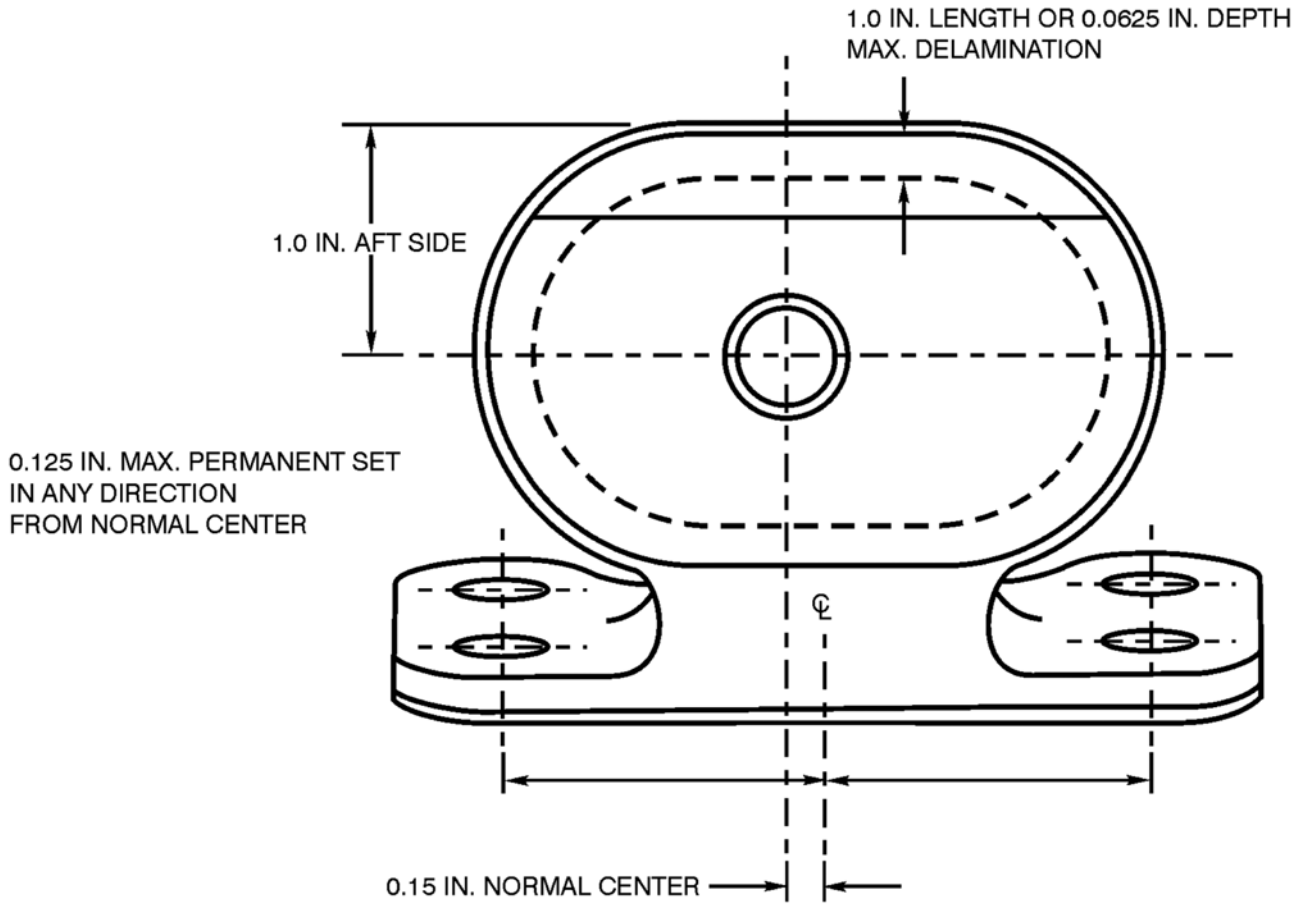
Linear indications and non-linear indications shall be interpreted. All indications interpreted to be defects shall be cause for isolation housing REJECTION.

6) Post Inspection Processes

Post-inspection cleaning and marking shall be carried out in accordance with ASTM–E–1417.

7) Inspection Records

Inspection records, at the inspection facility, shall show the part number, date of inspection, number of parts inspected, number of parts accepted and rejected, reference to any rejection documents and the identity of the inspector. Additionally, the log should record the nature, size, number and location of the rejectable indications and shall be noted on the rejection document.



Deformation Check
Figure 3

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Fireseals – General Data

1. General

Two fireseals are fitted to the engine, one fitted forward and the other fitted to the rear of the engine compressor intake. Each fireseal is made of two semicircular sections, which are bolted to the engine fireseal flange and to each other to form a complete fireseal between engine and cowlings. The fireseals also provide location and support for all lines, controls and ducts which pass from one engine fire zone to another.

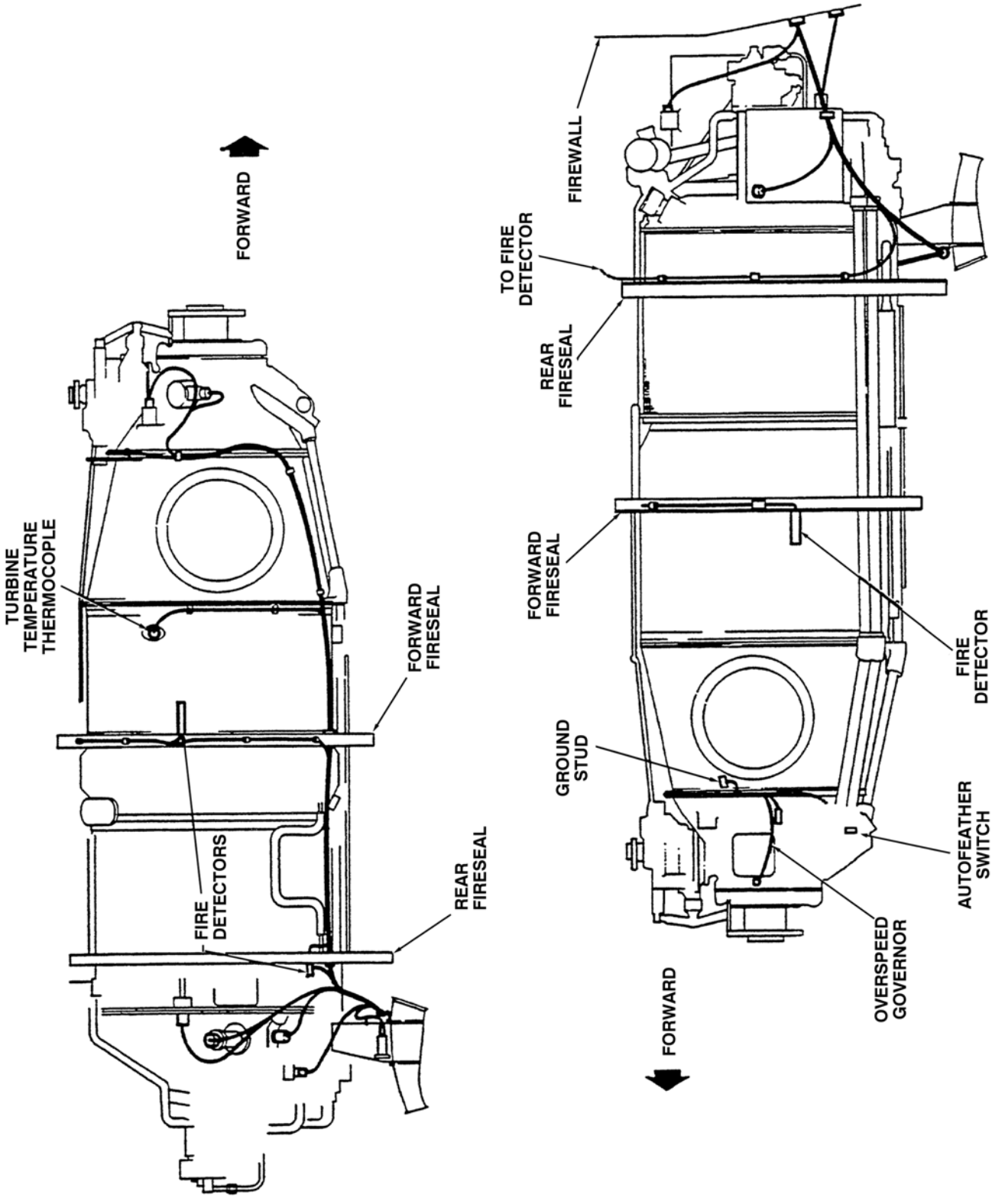
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Electrical Harness – General Data

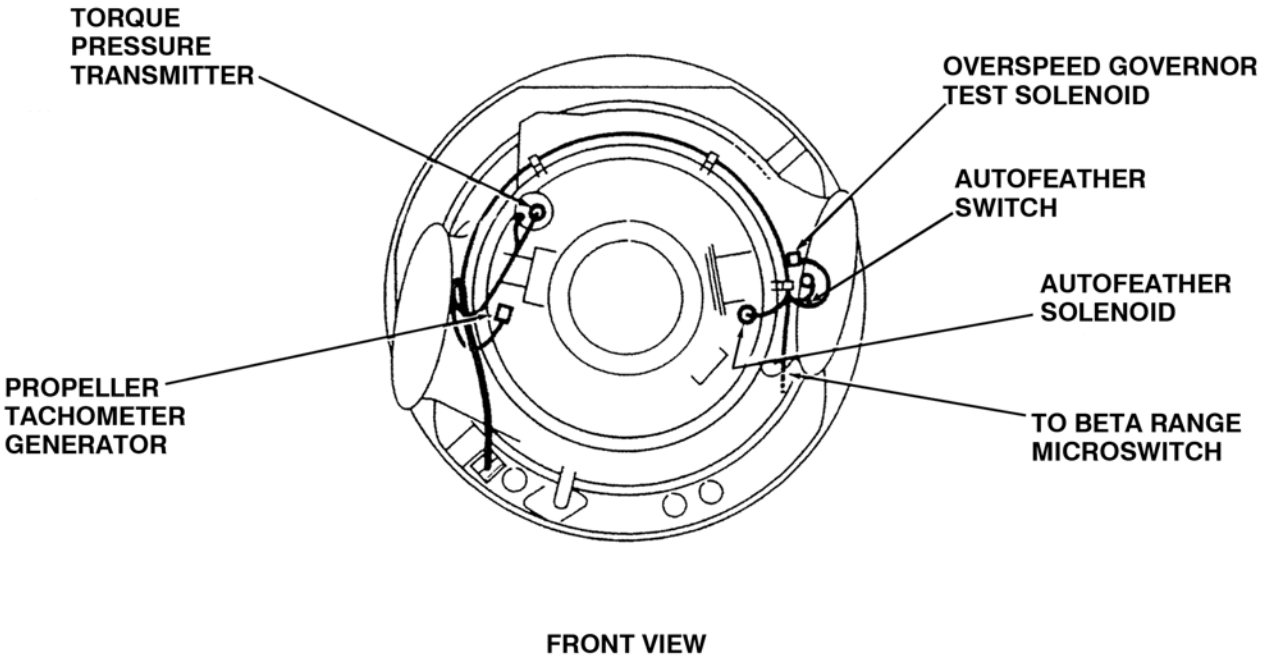
1. General

Refer to Figure 1 for details.

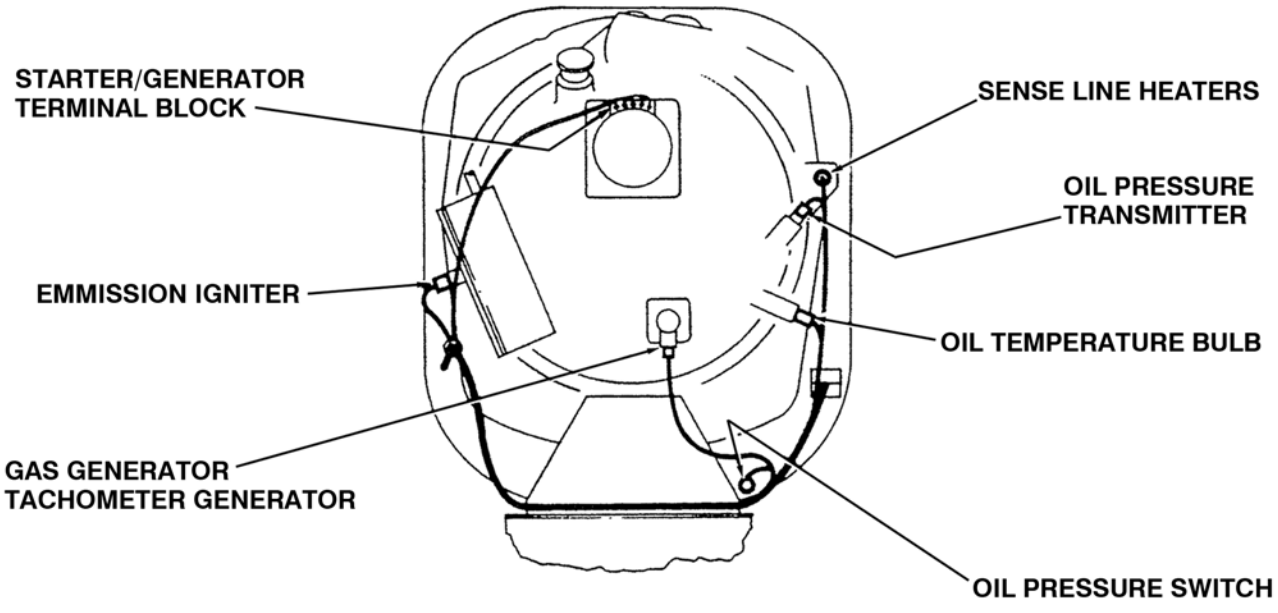
The power plant electrical harness distributes power to or from all engine electrical services, fire detectors and starter generator. The harness is disconnected at the engine firewall by means of two connectors and two terminal lugs for the heavier starter generator cables. The harness is routed from the engine firewall, forward through the rear and forward fireseals to the reduction gear case and propeller electrical systems.



Electrical Harness
Figure 1 (Sheet 1 of 2)



FRONT VIEW



REAR VIEW

Electrical Harness
Figure 1 (Sheet 2 of 2)

2. Removal/Installation

Refer to and Figure 1 for details.

A. Remove Electrical Harness

1) Ensure all electrical power is switched OFF.

2) Disconnect two electrical connectors at engine firewall.

3) Disconnect starter/generator leads at engine firewall.

4) Disconnect leads at engine current regulator.

5) Disconnect leads at fire detectors on rear and forward fireseals.

6) Disconnect leads from all electrical equipment on accessory case and reduction gear case.

7) Remove fairleads from forward and rear fireseals.

8) Disconnect leads from starter/generator.

9) Remove clamps retaining harness in position.

10) Remove harness.

11) With ignition harness remove as follows:

a) Disconnect harness leads at ignition exciter.

b) Disconnect harness leads at forward and rear fireseals.

c) Remove fairleads from forward and rear fireseals.

d) Disconnect harness leads from igniter plugs.

e) Remove harness.

B. Install Electrical Harness

1) Place electrical harness in position around engine and through fireseals.

2) Install harness retaining clamps.

3) Connect leads to starter/generator.

4) Install fairleads to forward and rear fireseals.

5) Connect leads to all electrical equipment on accessory case and reduction gear case.

6) Install leads to fire detectors on rear and forward fireseals.

7) Connect leads to engine current regulator.

8) Connect starter/generator leads to terminal lugs on firewall.

9) Connect electrical connectors at firewall and lockwire.

10) With ignition harness install as follows:

a) Place ignition harness through fireseals.

b) Install harness retaining clamps.

c) Connect leads to igniter plugs.

d) Install fairleads to forward and rear fireseals.

e) Connect leads to ignition exciter.

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Engine Intake Deflector System – General Data

1. General

Refer to Figure 1 for details.

The engine intake deflector system prevents the entry of snow and rain into the engine compressor inlet duct by diverting the particles aft over the oil cooler honeycomb and out through the inertial separator exit duct. Each engine intake is equipped with an identical system which consists of a deflector, two actuators, an air valve, two spring-loaded lock levers, a release lever, microswitches, an exit duct door and an intake deflector switch (common to both systems). The system is controlled by the air intake deflector switch, in the flight compartment, which, when switched to EXTEND, energizes the air valve and allows engine bleed air pressure into the two actuators. The actuators extend fully, lower the attached deflector, and release the tension on the exit duct door cables allowing the spring-loaded exit duct door to open. Spring-loaded lock levers hold the deflector in the extended position. When the INTAKE DEFLECTOR switch is released the air valve is deenergized and relieves the actuators to atmosphere. On retraction, (RETRACT selection) the air valve is energized and the actuators extend relieving the pressure on the lock levers. The release lever solenoid is also energized, rotates a cam to release the locking levers, and operates a microswitch which deenergizes the air valve. The deenergized air valve shuts off bleed air and vents the air from the actuators through the valve, allowing the actuator return spring to retract the deflector. The control circuit is powered from the 28 Volt DC supply and is protected by a 10–ampere INTAKE DEFLECTORS circuit breaker on the main circuit breaker panel. The deflector cannot be selected to an intermediate position. Indication that the intake deflectors have extended and locked in place is provided by display of the text 'A/I' (for anti-ice) on the outboard portion of each engine parameter display. It is operated by two microswitches connected in series, one at the intake deflector and one at the exit duct door. An air line filter is installed in each AN fitting connection at the air valve.

A. Engine Intake Deflector Actuators

The engine intake deflector actuators are screwed into adapters mounted in the inlet duct and are connected by turnbuckle type piston rods to the outer edges of the deflector. Each actuator contains a piston and return spring.

B. Engine Intake Deflector

The engine intake deflector is a louvered plate hinged, at the forward end, to the roof of the intake. The plate is connected to two actuators at the aft end, and mechanically linked by cable to the exit duct door. A striker is attached at the hinge to actuate a microswitch in the indicator circuit.

C. Engine Intake Deflector Air Valve

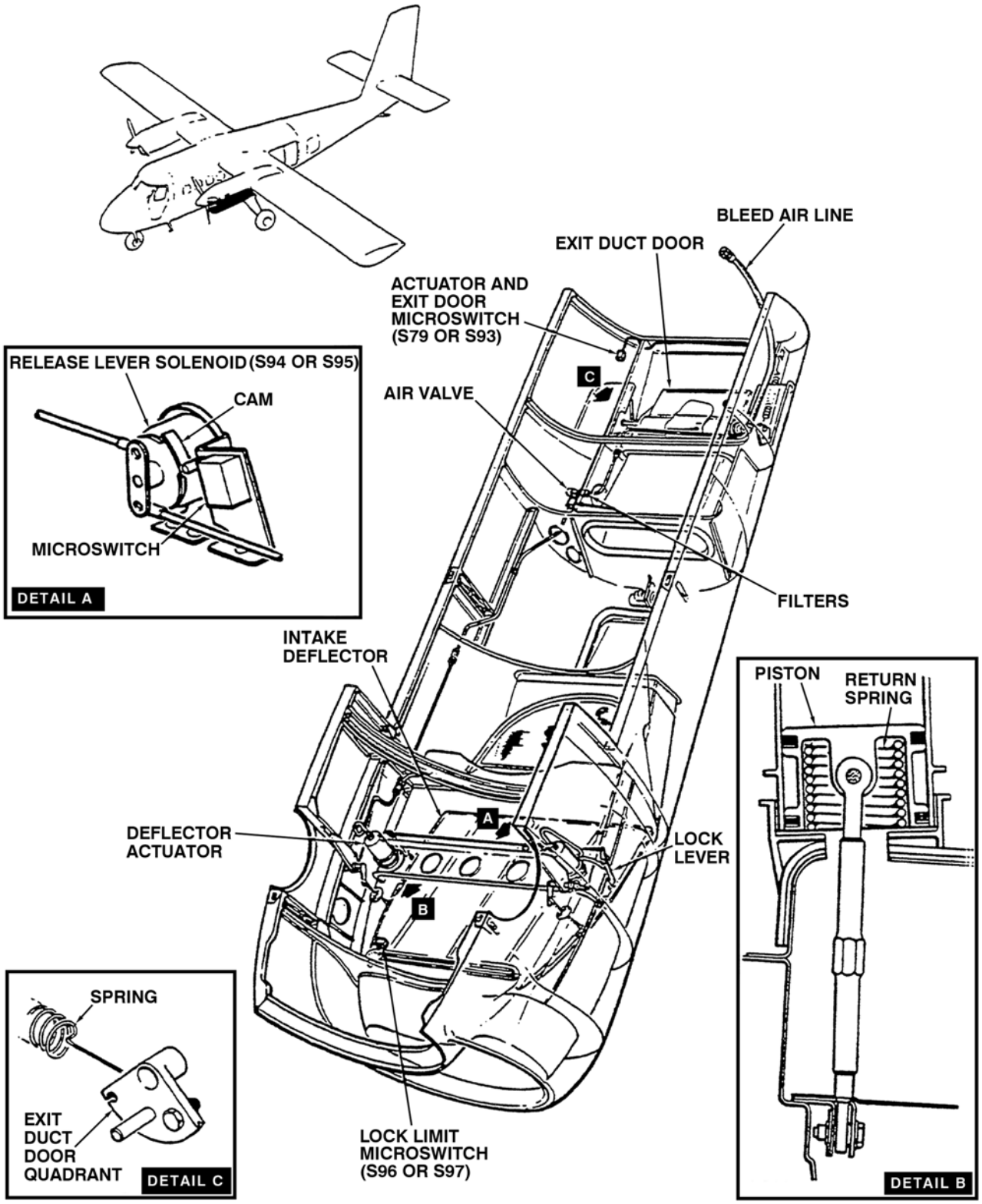
The engine intake deflector air valve is mounted in the lower cowling to the rear of the actuators. The valve is electrically-operated and controls the flow of air pressure in the air line taken from the bleed air system line of the engine.

D. Engine Intake Deflector Locking Levers

The engine intake deflector locking levers are mounted, one on each side of the inlet duct, and protrude through slots into the duct. They are spring-loaded and connected by rods to the release lever which rotates and operates a microswitch to deenergize the air valve, which vents to atmosphere.

E. Engine Intake Deflector Solenoid-Operated Release Lever

The engine intake deflector release lever is mounted above the inlet duct between the locking levers and is connected to them by adjustable rods. The release lever is operated by a solenoid which is controlled by the intake deflector switch.



Engine Intake Deflector System
Figure 1

F. Engine Intake Deflector Switch

The INTAKE DEFLECTOR switch is located on the co-pilot's switch panel and controls the deflectors of both engines. The switch is labeled INTAKE DEFLECTOR and has positions marked RETRACT and EXTEND. It is spring-loaded to the centre “off” position.

G. Engine Intake Deflector Indicators

Indication that the intake deflectors have extended and locked in place is provided by display of the text 'A/I' (for anti-ice) on the outboard portion of each engine parameter display.

2. Troubleshooting

A. Extend Cycle

TROUBLE	POSSIBLE CAUSE	CORRECTION
Intake deflector fails to extend.	Release lever solenoid microswitch not correctly adjusted on its operating cam.	Adjust position of microswitch.
	Pistons jamming in operating cylinders.	Remove pistons and inspect cylinders for signs of scoring and dirt. Ensure each groove in each piston has two piston rings and a corrugated expander; that rings are loose enough to rotate by hand, and that gaps in rings are approximately 180° apart from each other. Lightly lubricate piston rings with silicone grease at assembly.
	Dirt in air valve.	Replace air valve or clean air valve filter.
	Electrical failure.	Loss of voltage on right bus (could be open bus tie switch). Faulty bus tie switch relay. Faulty circuit breaker. Check bonding between lower cowl and airframe.
Snow separator exit duct door not fully open. (With intake deflector in EXTEND position).	Tension springs not hooked up.	Attach tension springs to door operating boss.
	Door control cables adjusted too short.	Adjust cables.
Engine intake deflector position is indicated by A/I (Anti-Ice) displayed in the engine parameter window on the pilot's and co-pilot's PFD.	Deflector position and exit door microswitches require adjustment.	Position both deflector and exit door microswitches so that they are actuated with intake deflector in EXTEND position.
	Open circuit to indicator.	Rectify wiring as necessary.
	Faulty indicator.	Replace indicator.

B. Retract Cycle

TROUBLE	POSSIBLE CAUSE	CORRECTION
Intake deflector fails to retract.	Screws securing cam to release lever solenoid protruding through solenoid mounting plate.	Replace any screws restricting solenoid rotational travel with shorter ones.
	Lock levers incorrectly adjusted or sticking.	Ensure locking levers are correctly adjusted.To adjust the locking lever position, vary the amount of exposed thread on the threaded rod end. Check that locking levers move freely. Note Ensure adequate threads are visible in inspection hole.
	Defective release lever solenoid.	Replace solenoid if it does not rotate.
		CAUTION DO NOT ACTUATE RETRACT SWITCH FOR MORE THAN 5 SECONDS AT ANY ONE TIME.
	Dirt in air valve.	Replace air valve or clean air valve filter.
	Broken piston spring(s).	Remove pistons and replace broken piston springs.
	Pistons jamming in operating cylinders.	Remove pistons and replace broken piston springs.
	Electrical failure.	Loss of voltage on right bus (could be open bus tie switch). Faulty bus tie switch relay. Faulty circuit breaker.
Exit duct door not fully closed (with intake deflector in RETRACT position).	Too much play in control cables.	Adjust cables.
Intake deflector position indicator is displayed with retract cycle completed.	Deflector position and exit duct door position microswitches not correctly adjusted.	Adjust microswitches.
	Faulty microswitch(es).	Replace microswitches, as necessary.
	Indicator circuit shorted.	Rectify wiring as necessary.
	Faulty indicator.	Replace indicator.

3. Adjustment/Test

Equipment required:

- Pressurized air or nitrogen bottle
- Calibrated Pressure Regulator
- Shut-off valve
- Flexible tubing
- MS24392D4 Union (or equivalent) and
- External Electrical Power Supply

A. Adjustment

Refer to Figure 1 for details.

1) Remove Exit Duct Door Microswitch (S79 or S93) and Actuator from the lower cowling.

2) Bend up end of Actutator Arm to 45 degrees at 0.25” to clear the forward end of the operating arm.

3) Reinstall the Exit Duct Door Microswitch (S79 or S93) and Actuator on the lower cowling.

4) Ensure that Exit Duct Door Actutator Arm clears operating lever.

5) Disconnect supply line downstream of the nacelle air valve, remove in-line air filter, and attach shop air using MS24392D4 union and regulator.

6) Set regulated air pressure to 35±5 psi and ensure intake deflector moves smoothly to an extend position.

7) Reduce air pressure until intake deflector is locked in fully extended position.

8) Disconnect shop air.

9) Set Lock Limit Microswitch (S96 or S97) to actuate with intake deflector in the fully extended and locked position.

CAUTION **MAKE SURE THAT MICROSWITCH ACTUATOR TAB HAS SUFFICIENT TRAVEL IN THIS POSITION TO PREVENT DAMAGE.**

10) Reconnect the shop air using MS24392D4 union and regulator. Slowly increase the air pressure until the lock levers are unloaded (air screen fully extended).

11) Verify exit duct door is fully open.

CAUTION **MAKE SURE EXIT DUCT DOOR DOES NOT GET STUCK INSIDE DUCT WHEN FULLY OPEN.**

12) Manually set the Release Lever Solenoid Limit Microswitch (S94 or S95) to actuate with intake deflector in fully extended position and levers in the retracted position..

13) With Lock Limit Microswitch (S96 or S97) Exit Duct Door Microswitch (S79 or S93) in correct rigging position, check AI indication in PFD.

CAUTION **MAKE SURE THAT MICROSWITCH ACTUATOR TAB HAS SUFFICIENT TRAVEL IN THIS POSITION TO PREVENT DAMAGE.**

14) Observe that door is clear of locking levers then manually rotate control solenoid to stow levers in unlocked position.

15) Hold the Release Solenoid Limit switch (S94 or S95) open, reduce the shop air. Check that door fully retracts.

16) Disconnect shop air, remove MS24392D4 union, reinstall in-line air filter and the supply line at the Bleed Air Solenoid Valve.

17) Carry out leak check of installed tube fittings and door actuators.

18) Do a function test as per Para 3.B.

B. Function Test

Equipment required:

- Pressurized air or nitrogen bottle
- Calibrated Pressure Regulator
- Shut-off valve
- Flexible tubing

1) Extend Cycle

a) Lower bottom engine cowl.

b) Disconnect intake deflector air pressure line at firewall bulkhead, cap firewall to engine line.

c) Route flexible line from pressure bottle through access hole in top cowl and connect to intake deflector air pressure line.

d) Raise lower engine cowl and secure latches.

e) With regulator closed, open pressurized bottle shut-off valve and set regulator to 35 ± 5 psi.

f) Connect external electrical supply to aircraft.

g) Ensure INTAKE DEFLECTOR circuit breaker is engaged.

h) Hold INTAKE DEFLECTOR switch in EXTEND position until intake deflector displays 'A/I' (Anti-Ice) on the PFD.

i) Ensure intake deflector moves smoothly to extend position and locks. Time 3 seconds approximately.

j) Check exit duct door is fully open.

2) Retract Cycle

CAUTION **DO NOT ACTUATE RETRACT SWITCH FOR MORE THAN 5 SECONDS AT ANY ONE TIME.**

a) Hold INTAKE DEFLECTOR switch in RETRACT position until intake deflector displays 'A/I' (anti-ice).

b) Ensure intake deflector becomes unlocked and retracts flush with top of intake duct.

c) Check exit duct door is fully closed.

d) Switch off and disconnect electrical power.

e) Close off pressurized bottle shut-off valve and disconnect from intake deflector line.

f) Remove blanking cap and connect intake deflector air pressure line at firewall bulkhead.

C. Secure Intake Door Locking Lever Rod

1) Do function test, as per Para 3.B.

2) After function test, to prevent rotation of locking lever actuating rod apply Grade M Loctite to threads at the end of intake door locking lever actuating rods.

3) Repeat function test, as per Para 3.B, to confirm proper operation.

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Exit Duct Door – Maintenance Practices

1. Adjustment/Test

   A. Rig Exit Duct Door

      1) Ensure intake deflector is in fully retracted position.

      2) Push exit duct door to closed position and hold.

      3) Thread cable through hole in bolt at exit duct door quadrant on each side.

      4) Tension cables and tighten nuts.

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Airline Filters – Maintenance Practices

1. Servicing

   A. Clean Air Line Filters

      1) Lower engine bottom cowling.

      2) Disconnect air lines from air valve AN fitting and remove filters.

      3) Check condition of each filter mesh. If serviceable, clean with dry cleaning solvent (P–D–680) and dry with compressed air.

      4) Install filters in air valve AN fitting.

      5) Connect air lines to air valve AN fitting.

      6) Raise and secure engine bottom cowling.

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Engine Drains – General Data

1. General

Refer to Figure 1 for details.

The area surrounding the power plant installation is purged of inflammable vapors by the cooling airflow within the cowlings, but to prevent accumulation of liquids capable of creating a fire hazard, a drain system is provided. The drain system comprises an automatic fuel control (FCU) purge valve, a fuel flow divider/dump valve drain line with check valve, a FCU seepage drain, a starter/generator seal drain and two combustion chamber drains. Waste fuel from the automatic FCU purge valve and fuel flow divider/dump valve is routed through lines and flexible hoses into the fuel system vent line in the wing. Fluid from the FCU seepage drain, starter/generator seal drain and combustion chamber drains is drained overboard through pipes which protrude through the engine lower cowl.

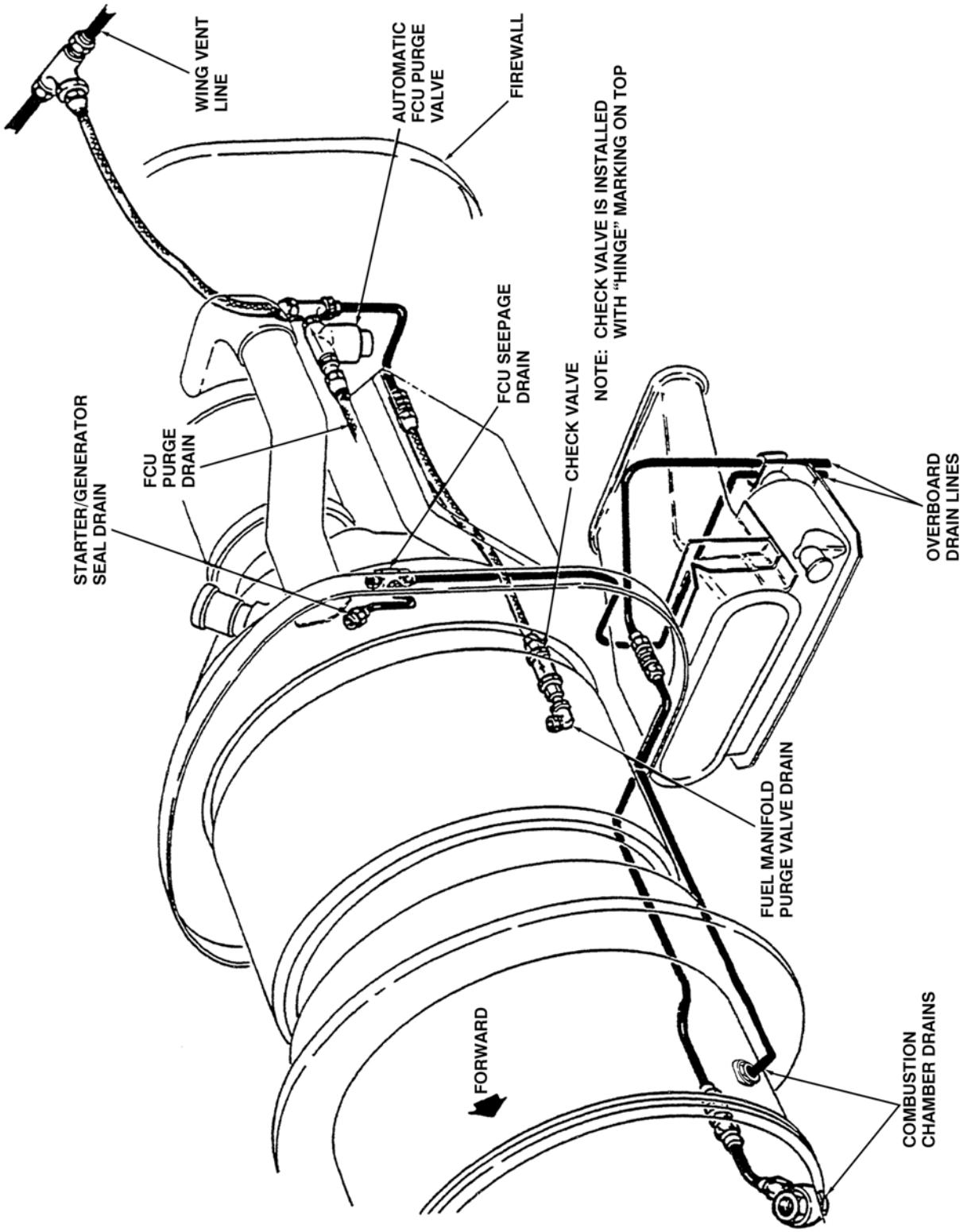
A. Automatic FCU Purge Valve

An automatic FCU purge valve is provided for each engine and is mounted in the engine nacelle aft of the firewall. The valve is energized during the starting cycle to provide a means of purging trapped air from the fuel control unit so that the metering of fuel will not be affected. On completion of the cycle the valve is deenergized (closed).

B. Flow Divider and Dump Valve

The flow divider and dump valve divides the fuel flow from the fuel control unit between the primary and secondary fuel manifolds during engine start-up and operation, and dumps residual fuel from the manifolds at engine shutdown.

When the fuel cut-off valve in the FCU closes during engine shutdown, an operating spring in the flow divider and dump valve overcomes the fuel inlet pressure and moves a piston within the unit body to block the fuel inlet port. This movement of the piston connects both primary and secondary manifolds to the dump valve port and allows residual fuel in the manifold to drain to the fuel system vent in the wing.



Engine Drains
Figure 1

CHAPTER 72

ENGINE

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CHAPTER 72 – ENGINE

LIST OF EFFECTIVE PAGES

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Combustion Section		1
Turbine Section		1
Accessory Section		1

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Engine – General Data

For the following subjects, refer to PT6A–34 Engine Maintenance Manual for information applicable to these subjects.

1. Engine – General
2. Reduction Gear and Shaft Section
3. Air Inlet Section
4. Compressor Section
5. Combustion Section
6. Turbine Section
7. Accessory Section

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CHAPTER 73

ENGINE FUEL AND CONTROL

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CHAPTER 73 – ENGINE FUEL AND CONTROL

LIST OF EFFECTIVE PAGES

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Engine Fuel and Control – General Data

1. Engine Fuel and Control – General

   Refer to PT6A–34 Engine Maintenance Manual for information applicable to this subject.

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Distribution – General Data

1. General

Fuel from the firewall mounted shut-off valve is routed through an external supply line up to the engine feed port mounted on the fuel heater (refer to PT6A–34, Engine Maintenance Manual). From the fuel heater, fuel is routed through an external line down the right side of accessory gear case to the base of the engine fuel pump, then from the pump through another external line to the fuel control unit (FCU). For further details of engine fuel distribution refer to PT6A–34, Engine Maintenance Manual.

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Controlling – General Data

1. Controlling

   Refer to PT6A–34 Engine Maintenance Manual for information applicable to this subject.

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Indicating – General Data

1. General

Engine fuel indicating consists of flow indication only. The fuel flow is measured in pounds per hour displayed on the upper RH corner of the primary flight display, on the engine display window. For aircraft configured for SI units, the fuel flow will be displayed in kilograms per hour.

A. Fuel Flow Indicating System

A fuel flow transmitter is installed in each engine nozzle fuel feed line to indicate the rate at which fuel is being delivered to the engine in pounds per hour or optionally kilograms per hour. The system consists of a fuel flow transmitter located in each engine line from the FCU to the manifold, which are connected to the Data Acquisition Units (DAUs); indication is displayed on the Primary Flight Display (PFD) – engine display window.

B. Fuel Flow Indication

Fuel flow indication is displayed in Lb/H (kg/H) the engine display window.

C. Fuel Flow Transmitter

A fuel flow transmitter is located on each engine in the main nozzle feed line. The transmitter is inserted in the engine main fuel feed line, so that fuel flows through the body of the transmitter causing rotation of a turbine that sends pulses to the DAU, that are translated into fuel flow.

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CHAPTER 74

IGNITION

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CHAPTER 74 – IGNITION

LIST OF EFFECTIVE PAGES

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General		1

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Ignition – General Data

1. General

The ignition system provides electrical power to each engine from the left DC bus. The ignition system consists of an ignition, selector switch, engine igniter switches and two circuit breakers, installed in the flight compartment. Ignition is initiated automatically through the starter system during ground starting of the engine. For air relights and ignition system ground testing only, the ignition may be operated manually through the ignition selector switch. Engine ignition is accomplished by spark igniters. For information regarding the items installed on the engine, refer to PT6A–34, Engine Maintenance Manual.

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Spark Ignitor Ignition - Description – General Data

1. General

The spark igniter system consists of an ignition exciter, two spark igniter plugs and shielded cables located on the engine, an igniter selector switch, a common ignition switch and two circuit breakers.

A. Ignition Exciter Unit

The ignition exciter unit, mounted on the engine accessory gear case, transforms the DC input to a pulsed, high voltage output through solid state circuitry, a transformer and diodes. When the unit is energized, a capacitor on the high voltage side of the output transformer is progressively charged until the energy stored is sufficient to ionize a spark gap in the unit and discharge the capacitor across the two spark igniters through a dividing and step-up transformer network. The network also enables the capacitor to discharge automatically in the event of either or both igniters becoming inoperative or input voltage is switched OFF.

B. Ignition Switch

The IGNITION switch is mounted on the overhead console and is marked, MANUAL and NORMAL. When the IGNITION switch is set to NORMAL the ignition circuit is controlled by the starter circuit, so that when the START switch is set to the LEFT or RIGHT position, power is automatically supplied to the selected spark igniters. When the IGNITION switch is set to MANUAL, power is supplied to the spark igniters, independently of the START switch position.

C. Start Switch

The START switch engages the starter of the selected engine and, unless the IGNITION switch is in MANUAL, activates the ignitor of the selected engine (refer to 80-00-00, Starting – General Data).

D. Ignition System Circuit Breakers

Two ignition circuit breakers are incorporated in the system, one to protect each circuit. Both are push-to-reset types marked IGNITION L and IGNITION R and are mounted on the main circuit breaker panel to the rear of the pilot's door. The circuit breakers are 5–amperes.

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CHAPTER 76

ENGINE CONTROLS

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CHAPTER 76 – ENGINE CONTROLS

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

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CHAPTER 76 – ENGINE CONTROLS

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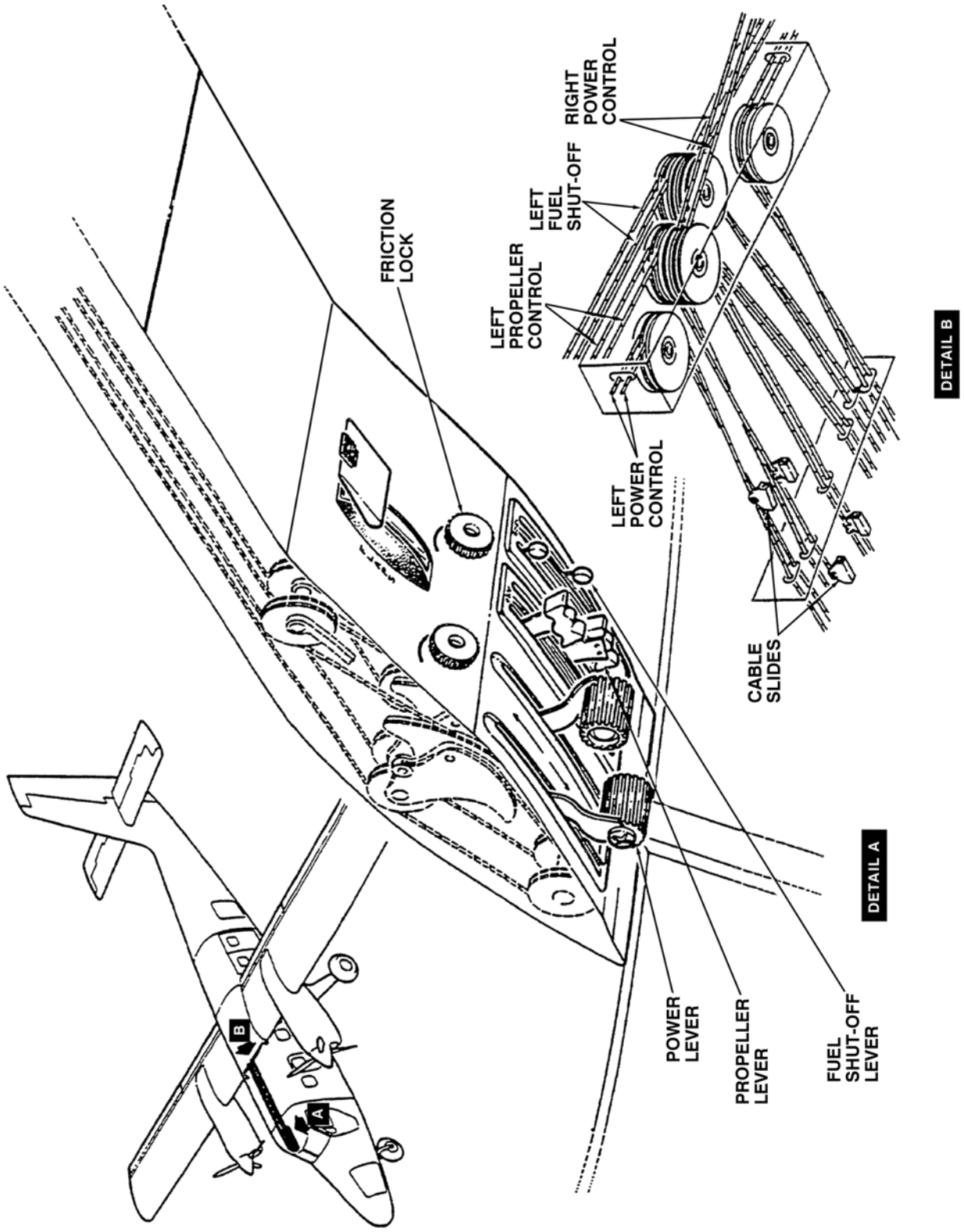
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Engine Controls – General Data

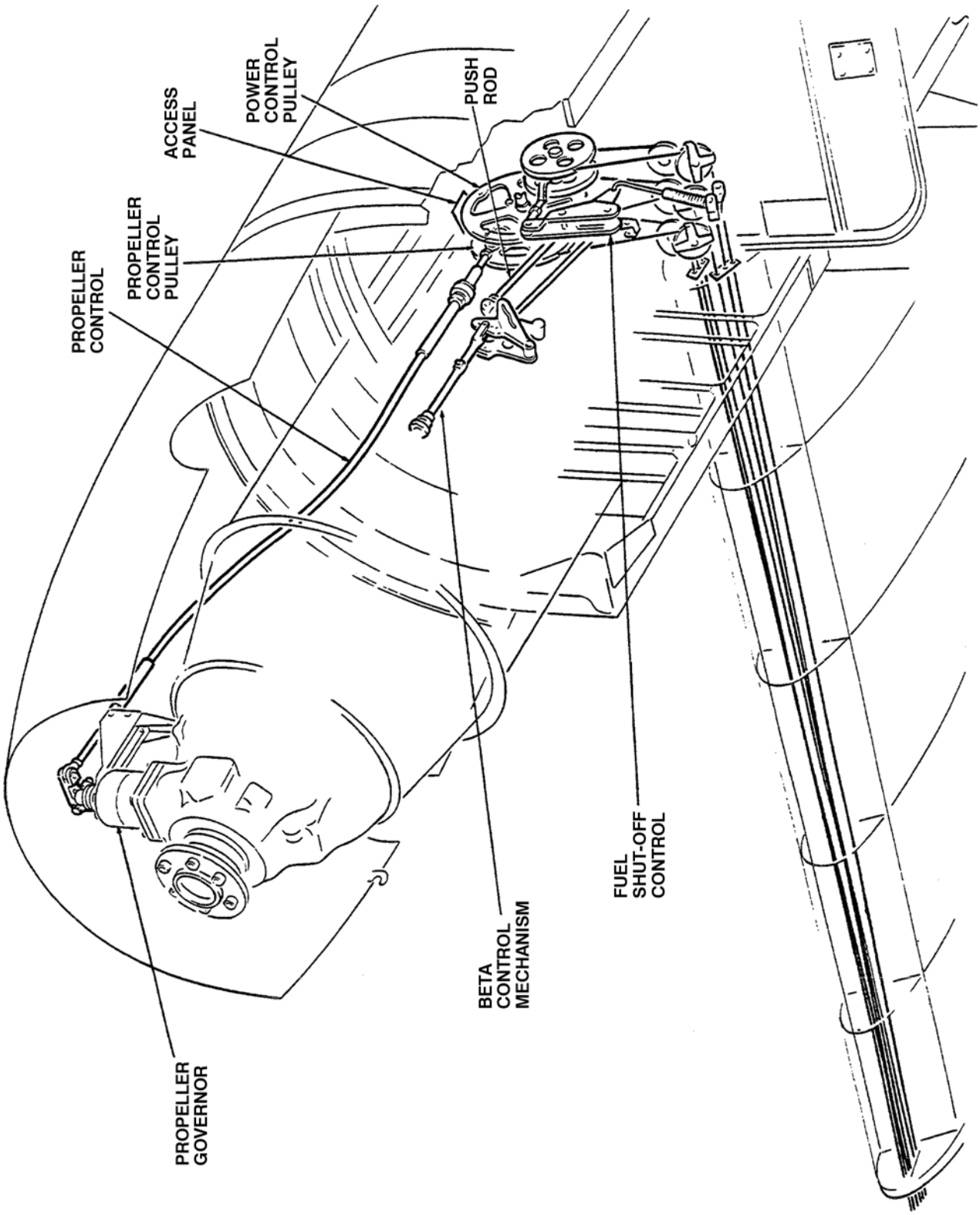
1. General

Refer to Figure 1 for details.

The engine controls consist of three main systems, power, propeller and fuel. Each system is basically of the cable and pulley type, with the addition of cable slides in the power and propeller control systems, to operate microswitches in relation to control lever positions. The control levers are mounted in the flight compartment overhead console. Provision is made for the use of rigging pins to facilitate the rigging of each system. Friction dampers are provided in the overhead console to hold the power and propeller levers in any selected position.



Engine Controls
Figure 1 (Sheet 1 of 2)



Engine Controls
Figure 1 (Sheet 2 of 2)

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Power Control – Maintenance Practices

1. General

Control of an engine is achieved through the operation of the power lever control system, the fuel shut-off control system and the propeller control system. The control levers for each system are mounted in the flight compartment overhead console. A cable attached to each lever quadrant runs along the aircraft roof, wing leading edge, and engine nacelle to its respective control pulley on the engine firewall. From the control pulleys, a rod or teleflex is used to actuate the fuel control unit, fuel shut-off valve and the propeller governor. Both engines are controlled by identical systems.

CAUTION DO NOT SELECT REVERSE POWER WITH ENGINE STOPPED, IF BETA CONTROL IS CONNECTED.

2. Rig Power Lever Controls

Refer to Figure 1 for details.

Note Slotted adjustment points utilize serrated washers. Ensure serrations in vernier washer and lever are properly aligned before tightening bolts.

A. Adjust Linkage and Controls at Rear of Engine

Adjust the linkage and controls at the rear of the engine as follows:

1) Disconnect the rear teleflex control rod at the beta control cam. If the engine is installed, ensure that the power pushrod is disconnected at the cam follower lever.

2) Disconnect the FCU control rod from the FCU arm, on the FCU assembly.

3) Remove the nut, washer and cotter pin from the FCU spindle shaft.

4) Install rigging gauge SD12556 so that the maximum power stop bolt is hard against the rigging face and the gauge bush screws onto the FCU spindle shaft, but allowing the serrated vernier washer and the FCU arm to rotate freely on the spindle.

Note Ensure that the vernier washer is positioned with the 25–serration side toward the FCU arm.

5) Position the FCU arm behind the hole in the rigging gauge; insert the gauge rigging pin through the hole and FCU arm.

6) With the maximum power stop bolt hard against the rigging gauge face, rotate the vernier washer until correct meshing with the FCU arm and spindle is obtained. Mark the relative position between the FCU arm, vernier washer and FCU spindle with permanent marker.

7) Adjust the low-idle set screw to contact the FCU stop lever, and lock it in position.

8) Remove the rigging gauge. Ensure that the low-idle set screw contacts the FCU stop lever and the alignment marking obtained in Para 2.A.6) is in line. Install the castilated nut and cotter pin on the FCU spindle shaft.

9) Connect the FCU control rod to the midslot position on the FCU arm.

10) With the low-idle set screw contacting FCU stop lever and cam follower pin contacting the FCU actuating lever, adjusting the length of the FCU control rod until the index marks are in line, between cam follower pin and FCU actuating lever.

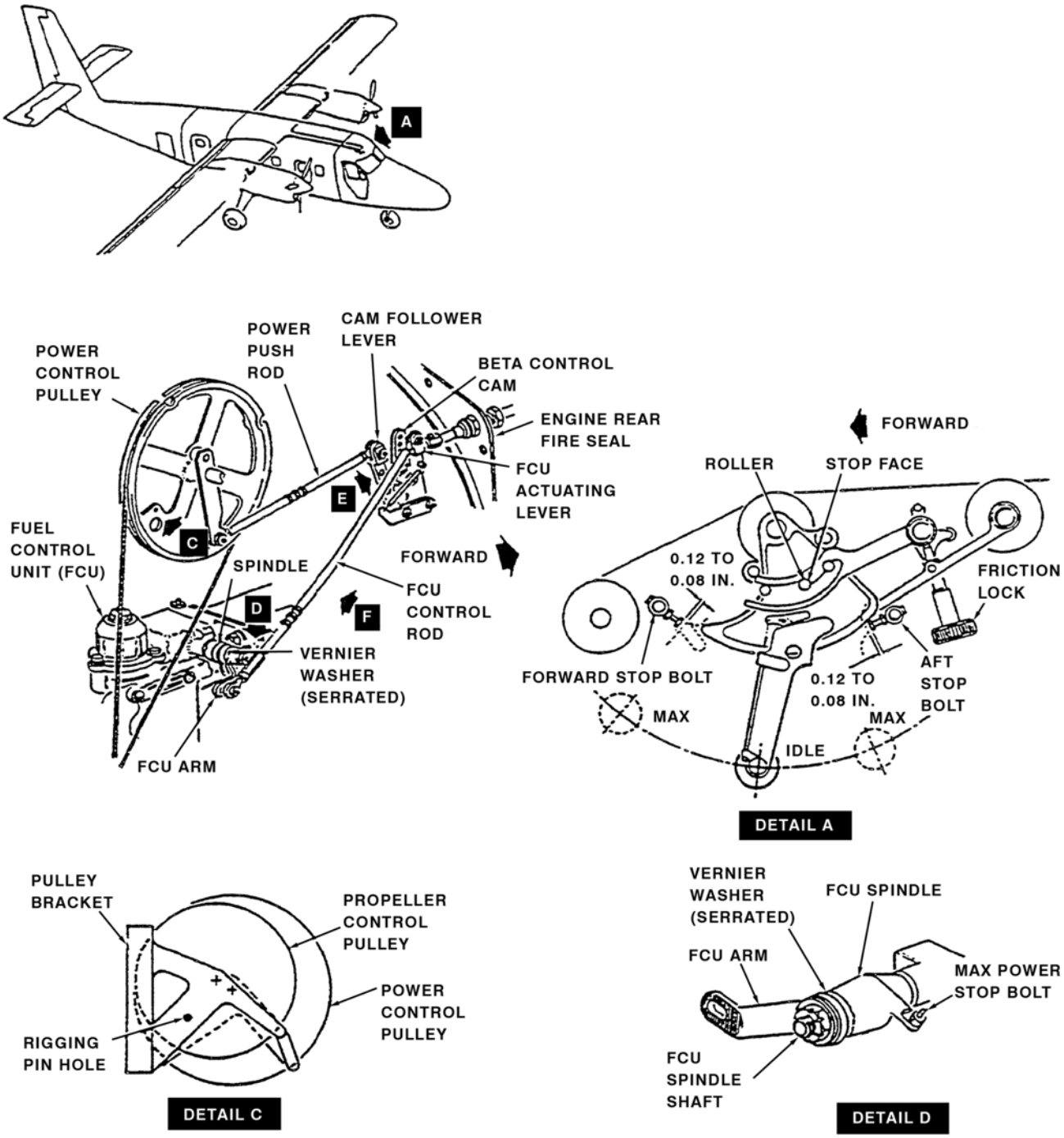
11) With the FCU reversing lever contacting the base of the FCU actuating lever and the cam index marks in line, set the stop bolt on the reversing lever to give a 0.35 inch gap between the bolt head and follower pin.

12) Verify that marked position Para 2.A.6) has remained in alignment, if not repeat procedures from Para 2.A.6) through Para 2.A.12).

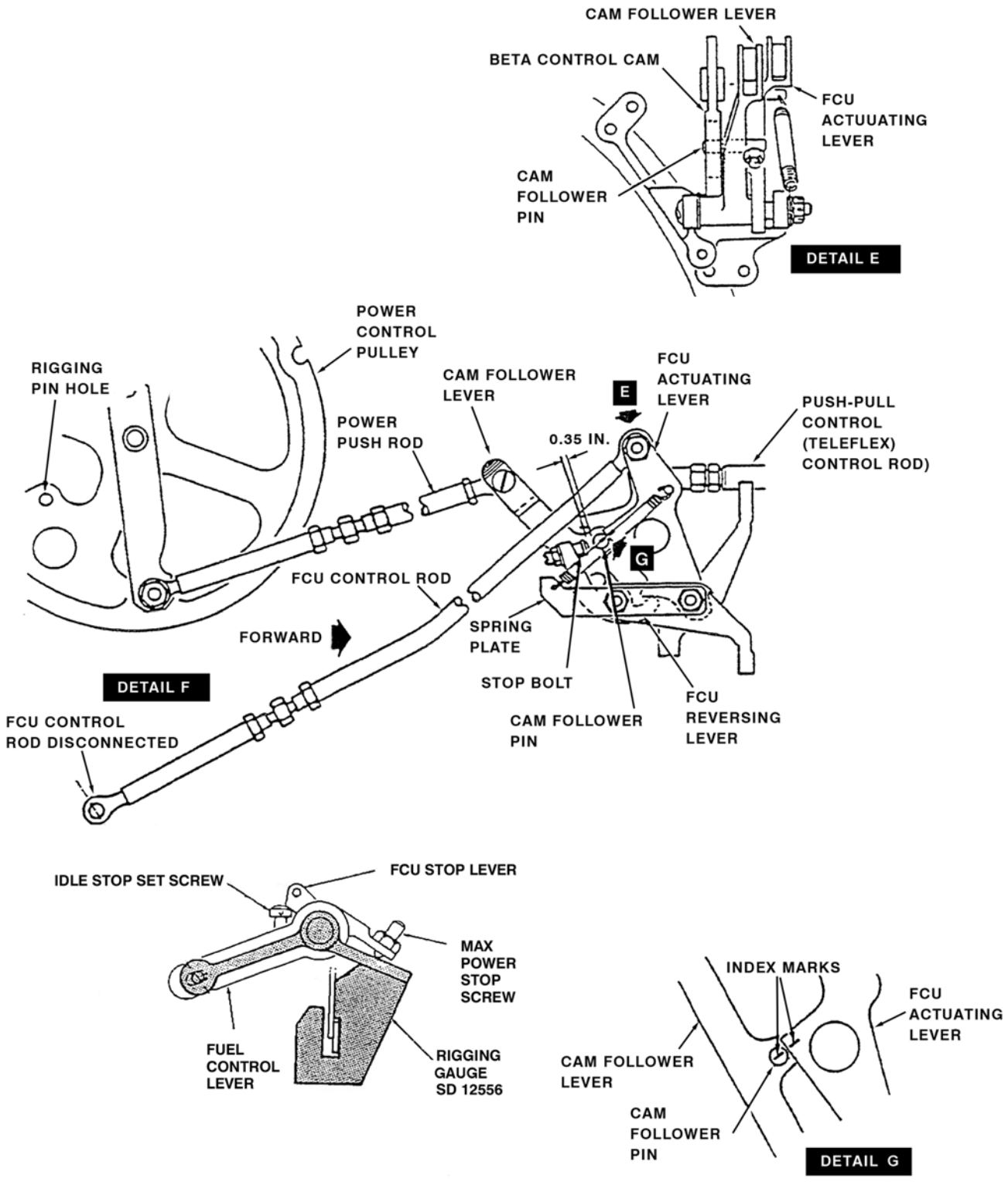
13) Reinstall rear teleflex control rod to beta control cam and secure all applicable hardware.

14) Ensure all applicable hardware is secured.

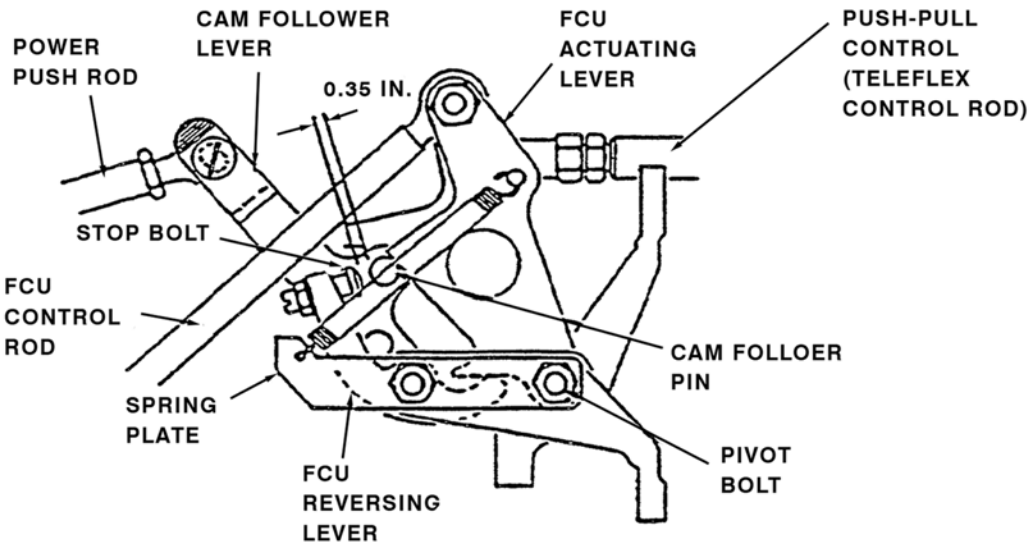
Note Beta Controls to be reconnected and adjusted as necessary to Rig Beta Controls System (refer to Para 3.A).



Power Lever Control System Rigging
Figure 1 (Sheet 1 of 3)



Power Lever Control System Rigging
Figure 1 (Sheet 2 of 3)



Power Lever Control System Rigging
Figure 1 (Sheet 3 of 3)

B. Adjust Controls in Airframe

With the power push rod disconnected at cam follower lever.

1) Move power lever in flight compartment to IDLE position (roller against stop face). Apply friction lock.

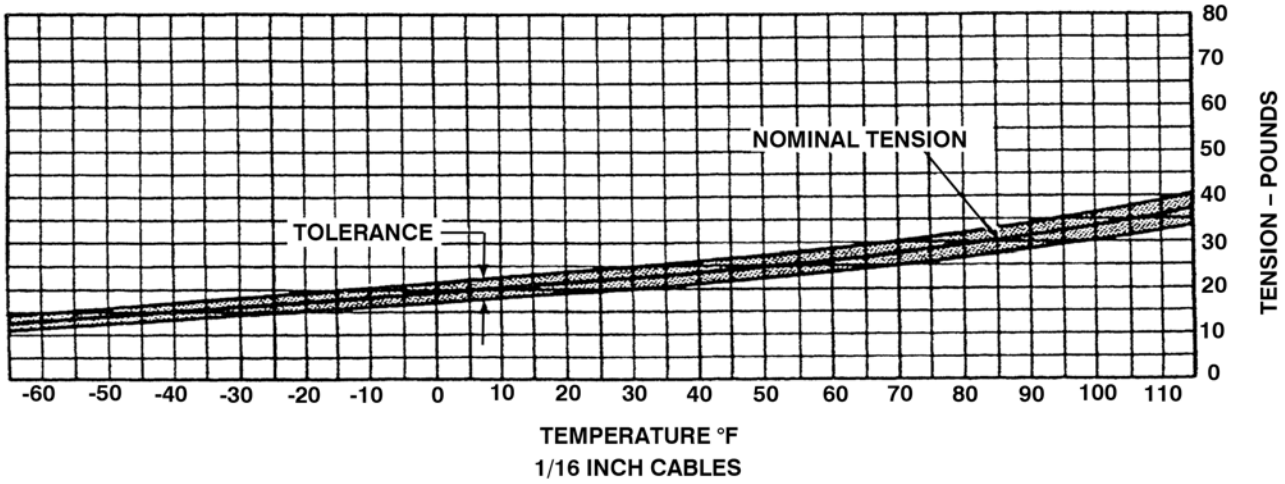
2) Insert rigging pin SD10545–1 through propeller control pulley and into power control pulley in the engine nacelle.

3) Adjust turnbuckles in wing leading edge to obtain cable tension shown in Figure 2.

4) Ensure power lever is not moved from IDLE and rigging pin moves freely in and out of the rigging pin hole of power control pulley.

5) Lock turnbuckles. Remove rigging pin.

6) Operate lever through full range of travel, return to IDLE and ensure that rigging pin can be freely inserted into rigging hole. Readjust turnbuckles if necessary.



Cable Tension Graph
Figure 2

C. Adjust Controls – Airframe to Engine Interconnect

1) With linkage and controls rigged as detailed in Para 2.A, position FCU stop lever to contact low idle set screw.

Note Ensure the rear teleflex cable rod end is disconnected at the beta control cam.

2) With rigging pin SD10545–1 installed in power control pulley and cam follower pin contacting FCU actuating lever, adjust length of power push rod (if necessary) and connect to mid-slot of cam follower lever.

Note Cam follower pin index mark and FCU actuating lever index mark should remain aligned from Para 2.A.10).

3) Remove rigging pin from power control pulley.

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Insert in 76-10-00 after Page 6

POWER CONTROL

REASON FOR ISSUE:

To introduce Mod 6/2476, Propeller Governor Threaded Rod.

ACTION:

In Chapter 76, Section 76-10-00, ignore existing Pages 7 to 16 and insert revised Pages, as shown on Pages 2 to 14 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

4) Advance power lever in flight compartment until max power stop bolt contacts maximum power stop anvil on the fuel control unit.

5) Adjust and lock power lever forward stop bolt in overhead console to give 0.12 to 0.08 inch clearance between stop bolt and power lever (refer to Figure 1, Detail A). Clearance may be checked with setting pin SD12570–3 or feeler gauge.

6) Set propeller lever to full INCREASE and retard power lever to reverse in the flight compartment until max power stop bolt again contacts maximum power stop anvil.

7) Adjust and lock power lever aft stop bolt in overhead console to give 0.12 to 0.08 inch clearance between stop bolt and power lever. Clearance may be checked with setting pin SD12570–3 or feeler gauge.

8) Move power lever through full range of travel and check for complete freedom of movement.

9) Connect rear teleflex push-pull control rod end to beta control cam.

Note On completion of adjustment, a check should be carried out to ensure that FCU actuating lever and cam cluster geometry arrangement look the same on both engines. Any appreciable difference will induce power lever split.

10) Rig Beta controls in accordance with Para 3.A (Pre Mod 6/2476) or Para 3.B (Post Mod 6/2476).

Note Refer to 76-10-11, Overhead Console – General Data for rigging of power/propeller lever interlock, and to Para 6., for rigging of propeller reset and autofeather microswitches.

3. Rig Beta Controls

A. Adjust Engine Front and Rear Linkage (Pre Mod 6/2476)

Refer to Figure 5 for details.

1) Disconnect teleflex push-pull control rod end at beta control cam, at rear of engine.

2) Disconnect fuel governor air bleed link at fuel governor reset arm.

3) Adjust propeller governor beta valve:

a) Loosen low pitch adjuster locknut.

b) With propeller in FEATHER and while holding a forward force on propeller reversing lever, adjust low pitch stop in or out, as necessary, to set the bottom of beta valve clevis slot flush with front face of cap nut.

c) Tighten low pitch adjuster locknut.

d) Disconnect push-pull rod from propeller reversing lever.

Note Ensure bushing is not lost from reversing lever when disconnecting push-pull rod.

e) Check that 1.1 to 1.3 inch travel exists in low pitch adjuster by sliding push-pull rod in and out of adjuster.

f) On satisfactory completion of check, re-assemble control at propeller reversing lever.

Note If 1.1 to 1.3 inch travel cannot be obtained check position of propeller low stop collar; distance from propeller mounting face to forward face of carbon block groove should be 1.26 to 1.27 inches. Adjust low stop rods (three) in collar as necessary (refer to Hartzell Manual).

Wear of low stop collar and of cam follower pin in beta control cam will give a very low N_P at 52% N_G.

4) With linkage still positioned as in Para 3.A.3), move fuel governor reset arm forward against its maximum stop. Adjust length of fuel governor link and connect to inner hole of fuel governor reset arm.

5) Move power lever in overhead console fully forward and apply friction lock.

6) Ensure reversing lever is fully forward and low pitch stop in adjuster is contacted and fuel governor reset arm is set as per (Para 3.A.3)). Adjust length of push-pull control at rear clevis rod end to align with the hole in the beta control cam, then lengthen push-pull control by 1/2-turn of rear clevis rod end to ensure low pitch stop is contacted, and connect to hole in beta control cam.

7) Release power lever friction lock, move power lever through full forward range of travel and check for binding and complete freedom of movement. Do not operate in reverse range.

8) Move power levers of both engines back to idle. Check that push-pull teleflex control in low pitch stop adjuster of both engines move back the same distance (approx. 1/8). If required, adjust push-pull at rear clevis rod end to obtain similar movement on both engines.

B. Adjust Engine Front and Rear Linkage (Post Mod 6/2476)

Refer to Figure 6 for details.

1) Disconnect teleflex push-pull control rod end at beta control cam, at rear of engine.

2) Disconnect fuel governor interconnect rod at fuel governor reset arm.

3) Adjust propeller governor beta valve:

a) Loosen low pitch adjuster locknut.

b) With propeller in FEATHER and while holding a forward force on propeller reversing lever, adjust low pitch stop in or out, as necessary, to set the bottom of beta valve clevis slot flush with front face of cap nut.

c) Tighten low pitch adjuster locknut.

d) Disconnect push-pull rod from propeller reversing lever.

Note Ensure bushing is not lost from reversing lever when disconnecting push-pull rod.

e) Check that 1.1 to 1.3 inch travel exists in low pitch adjuster by sliding push-pull rod in and out of adjuster.

f) On satisfactory completion of check, re-assemble control at propeller reversing lever.

Note If 1.1 to 1.3 inch travel cannot be obtained check position of propeller low stop collar; distance from propeller mounting face to forward face of carbon block groove should be 1.26 to 1.27 inches. Adjust low stop rods (three) in collar as necessary (refer to Hartzell Manual).

Wear of low stop collar and of cam follower pin in beta control cam will give a very low N_P at 52% N_G.

4) Adjust fuel governor interconnect rod:

a) Separate the fuel governor interconnect rod, and install the governor rod end and internal adjuster to the inner hole of the fuel governor reset arm. Secure with bolt, washer, and castellated nut. Tighten the castellated nut and torque to 12 to 18 in–lbs. Lock with a cotter pin. Refer to Detail C of Figure 6.

b) Re-connect the two halves of the fuel governor interconnect rod. With the interconnect rod still positioned fully forward and the beta valve slot flush with the front face of the cap nut, move fuel governor reset arm fully forward against its maximum stop.

c) Adjust the fuel governor interconnect rod to set a 0.100 inch ± 0.010 inch gap between the internal and external adjusters. Refer to Section D–D of Figure 6.

d) Check upper rod end for safety. Separate the fuel governor interconnect rod to verify 0.300 inch minimum engagement of internal adjuster at lower rod end. Refer to Section D–D of Figure 6.

e) Tighten locknuts 32 to 36 in–lbs and fasten with lockwire.

Note For fuel governor interconnect rod lubrication instructions, refer to Pratt & Whitney PT6A-34 Maintenance Manual.

5) Move power lever in overhead console fully forward and apply friction lock.

6) Ensure reversing lever is fully forward and low pitch stop in adjuster is contacted and fuel governor reset arm is set as per (Para 3.B.3)). Adjust length of push-pull control at rear clevis rod end to align with the hole in the beta control cam, then lengthen push-pull control by 1/2-turn of rear clevis rod end to ensure low pitch stop is contacted, and connect to hole in beta control cam.

7) With the power levers at idle, check the fuel governor interconnect rod to ensure the gap between the internal and external adjusters is 0.000–0.003 inch, and ensure fuel governor reset arm is against the reset stop. Refer to Detail C of Figure 6.

8) Release power lever friction lock, move power lever through full forward range of travel and check for binding and complete freedom of movement. Do not operate in reverse range.

9) Move power levers of both engines back to idle. Check that push-pull teleflex control in low pitch stop adjuster of both engines move back the same distance (approx. 1/8). If required, adjust push-pull at rear clevis rod end to obtain similar movement on both engines.

4. Rig Fuel Shut-off Controls

Refer to Figure 3 for details.

A. Adjustment of the Fuel Shut-off Controls in Airframe

1) Disconnect the FCU fuel shut-off rod end at the intermediate lever attachment bolt on the FCU unit.

2) Select the fuel condition lever to the OFF position in the flight compartment.

3) Install the rig pin in the nacelle engine pulley quadrant.

4) Adjust the wing leading edge control cable and turnbuckles for correct cable tension as required (approximately 25 lbs), refer to Figure 2.

Note Ensure that the fuel condition lever remains in the OFF detent during this adjustment.

5) Remove the rig pin in the nacelle.

6) Select the fuel condition lever to the ON position.

7) Ensure engine control lever moves freely with no binding.

8) Safety the engine control cable turnbuckles.

B. Adjustment of the Fuel Shut-off Control Airframe to Engine

1) Ensure the engine fuel control lever and fuel shut-off control system are rigged according to Para 4.A.

2) Move the engine fuel condition lever in flight compartment to the OFF position.

3) Install the rig pin in the nacelle engine pulley quadrant.

4) Position the FCU intermediate lever to maintain contact between the FCU actuating lever and the FCU fuel cut-off stop bolt.

5) In the nacelle, adjust the fuel shut-off control rod to allow the attach bolt to be installed while maintaining the contact between the FCU actuating lever and the FCU fuel cut-off stop bolt.

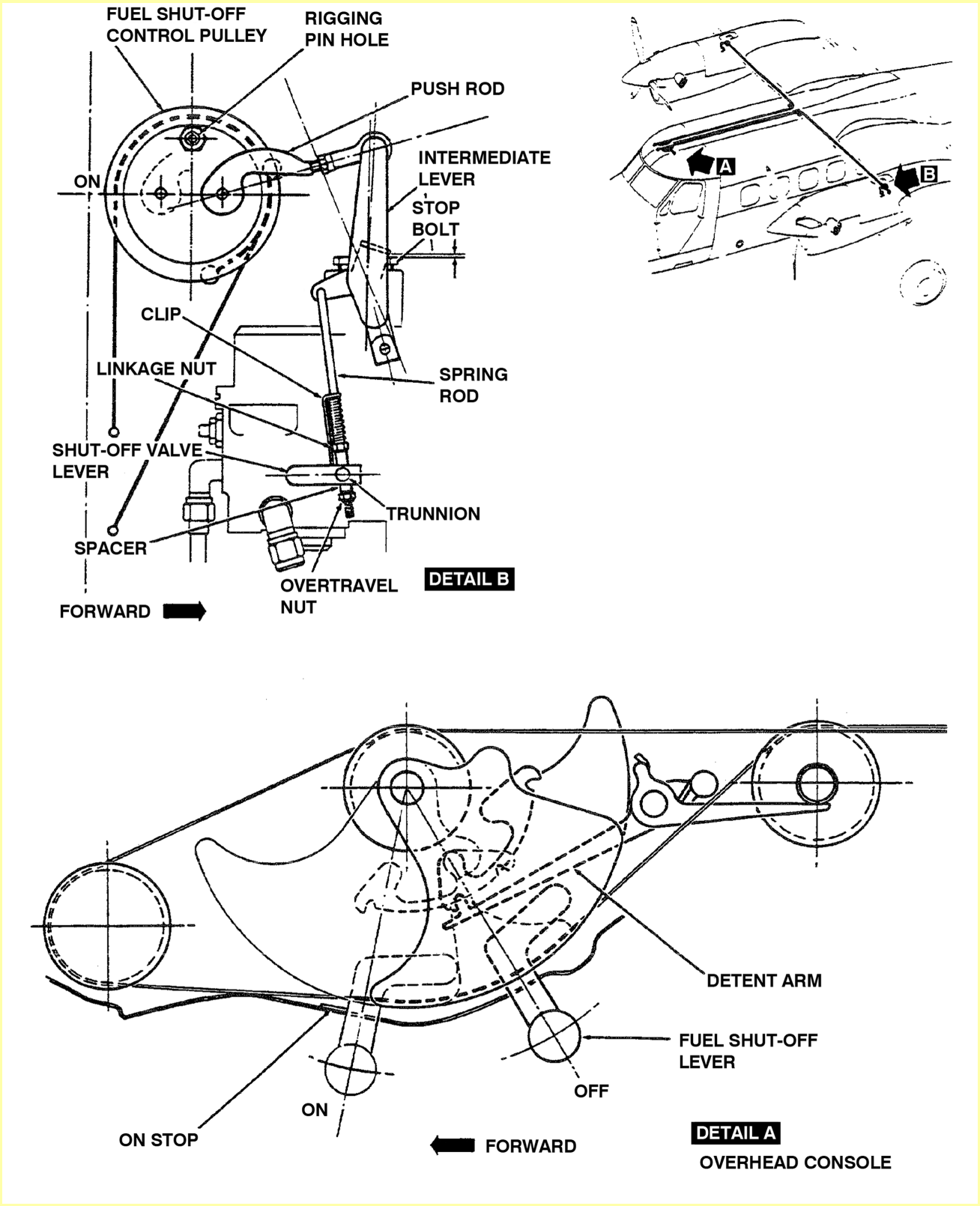
6) Remove rig pin from the nacelle.

7) Move engine fuel condition lever to the ON position.

8) Verify that the FCU intermediate lever is aft of the perpendicular position indicating the FCU shut-off is in the fuel ON position.

9) Ensure fuel shut-off control lever moves freely with no binding.

10) Tighten and safety all control hardware. Carry out function test of the FCU fuel shut-off system.



Rigging Fuel Shut-off Controls
Figure 3

5. Rig Propeller Controls

Refer to Figure 4 for details.

A. Adjust Controls in Airframe

1) Disconnect push-pull control at speed select lever on propeller governor unit.

2) Move propeller lever in flight compartment to minimum rpm position detent (roller against stop face). Apply friction lock.

3) Insert rigging pin SD10545–1 in propeller control pulley.

4) Adjust turnbuckles in wing leading edge to obtain cable tension shown in Figure 2.

5) Ensure propeller lever is not moved from minimum rpm position detent and rigging pin moves freely in rigging pin hole of propeller control pulley.

6) Lock turnbuckles.

B. Adjust Controls – Airframe to Engine Interconnect

1) Install rigging pin SD10545–1 in propeller control pulley bracket in engine nacelle.

2) Ensure that propeller governor unit speed select lever is set at minimum rpm position.

3) Adjust sliding rod end so as to achieve a gap of 0.230 inch between plate and feather stop bolt (refer to Detail A of Figure 4).

Note Attachment bolt to be in mid slot position on speed select lever (refer to Detail A of Figure 4).

4) Remove rigging pin from propeller control pulley.

5) Ensure friction lock knob is loosened to permit control lever to move freely.

6) Move control lever fully forward until plate on speed select lever is hard against maximum rpm stop bolt (refer to Detail A of Figure 4).

7) Adjust forward stop bolt in overhead propeller lever console so as to have a gap of 0.120 to 0.080 inch.

Note Clearance may be checked with setting pin SD12570–3 or feeler gauges.

8) Check that propeller levers align within 0.100 inch at knobs. Relocate attachment bolt in slot of speed select lever, on the Constant Speed Unit (CSU) as required.

9) With propeller lever forward apply light pressure on lever. Check gap of forward stop bolt and if necessary readjust to achieve 0.120 to 0.080 inch gap.

10) Retard propeller lever until the speed select lever plate contacts the feather stop bolt on governor.

11) With propeller lever fully aft apply light pressure on lever. Check gap of aft stop bolt and if necessary readjust to achieve 0.120 to 0.080 inch gap.

12) Move propeller lever through full range of travel and check for complete freedom of movement.

13) Safety all attachments and stop bolts as required.

14) If a new cable has been installed, adjust the length as follows:

a) Install rigging pin SD10545–1 in propeller control pulley.

b) Ensure that propeller governor unit is at minimum rpm position.

c) Adjust sliding rod end so as to achieve a gap of 0.230 inch between plate and feather stop bolt (refer to Detail A of Figure 4).

15) If the new cable is too long, trim the cable as follows:

CAUTION **THE FOLLOWING PROCEDURE FOR TRIMMING CABLE IS FOR P/N C6CE1351–31 ONLY.**

a) Cut the cable with an abrasive cutoff wheel.

Note The cable wires are stress relieved and will not unravel when cut.

b) Grind or file the cable to shape a rounded end. Grind or file the cable end in the direction of the cable winding.

CAUTION **DO NOT USE A SOLVENT ON THE CABLE ASSEMBLIES. THE SOLVENT WILL DESTROY THE INTERNAL LUBRICANT OF THE CABLE AND WILL MAKE THE CABLE UNFIT FOR USE.**

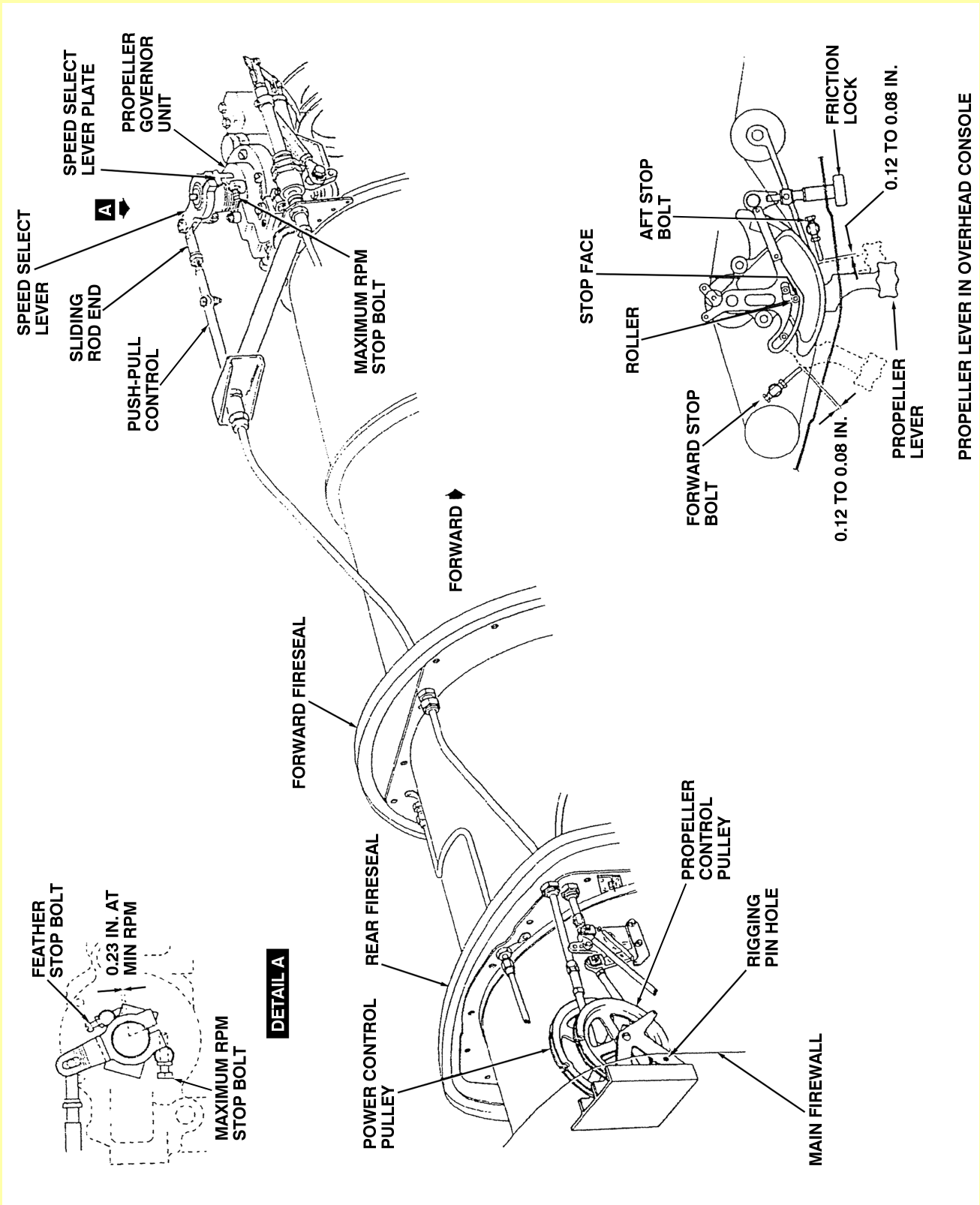
c) Clean the end of the cable by brushing at right angles to the centerline of the cable with a hard fiber, plastic or brass bristle brush. Do not use a steel brush.

d) Using dry, oil free compressed air at no more than 25 psi, blow the loose dirt particles from the cable.

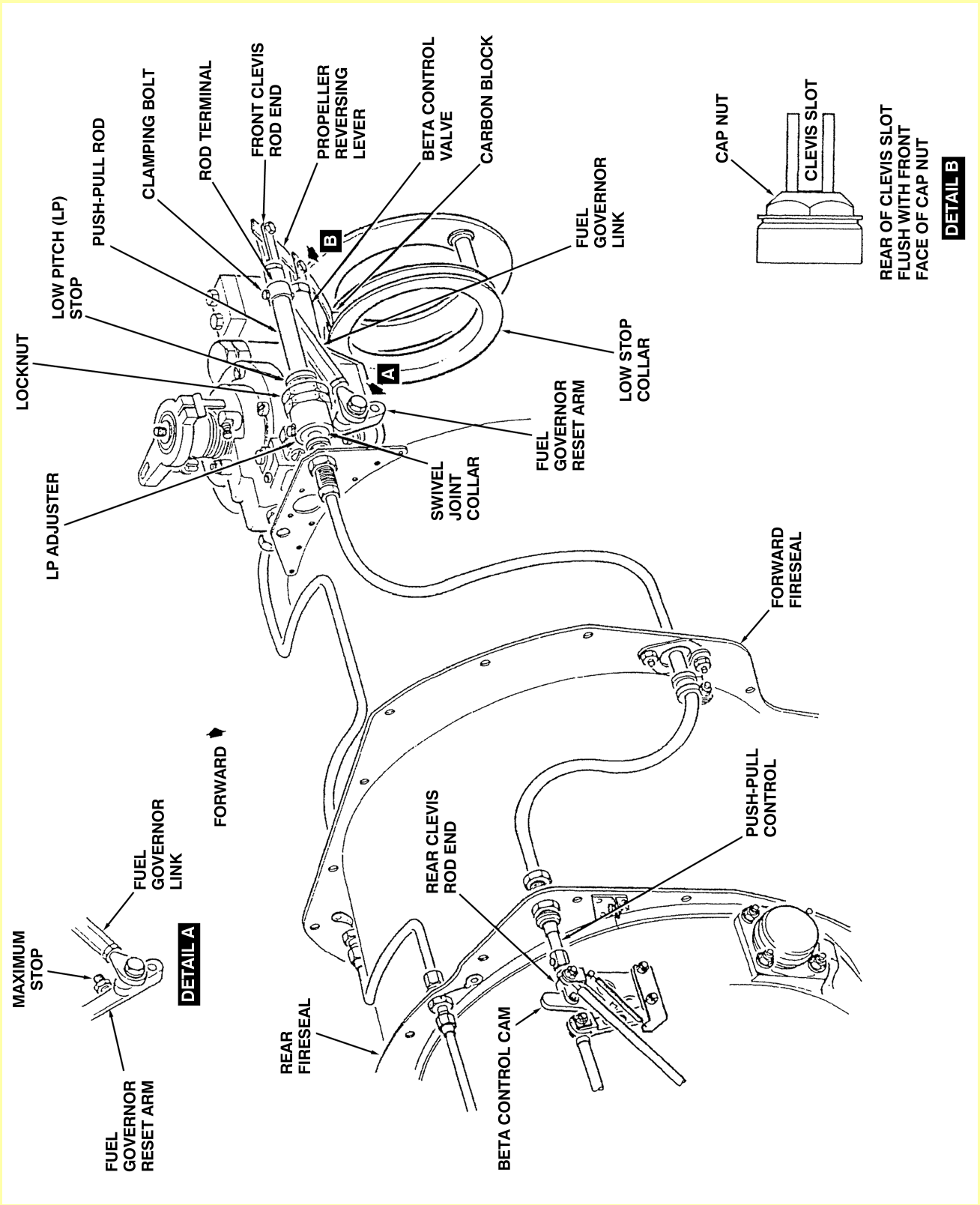
e) Wipe the cable clean with a dry lint free cloth.

Note It is not necessary to apply any protection to the cut cable end to prevent corrosion, fraying and unwinding.

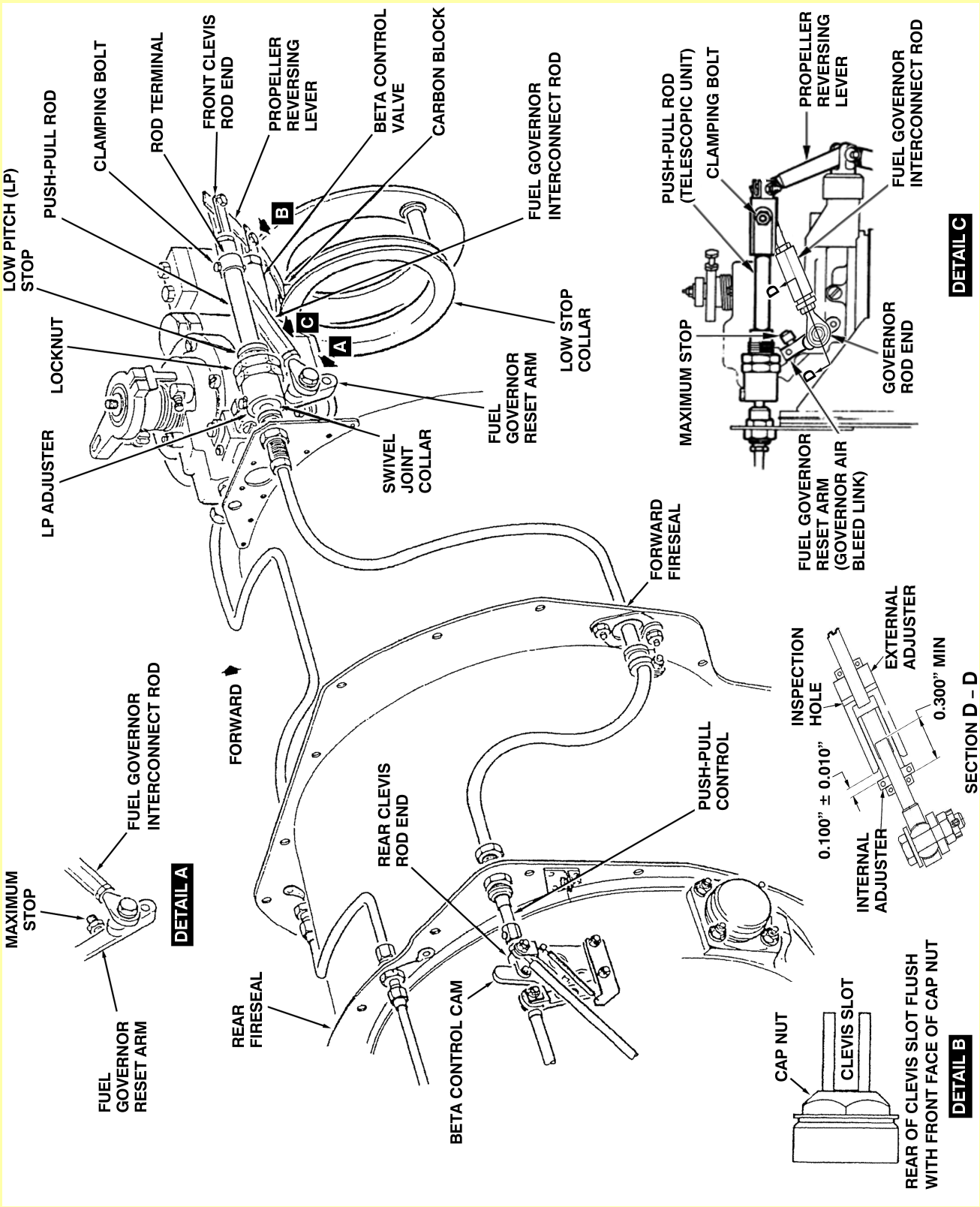
f) Do a visual inspection on the cable for serviceability.



Rigging Propeller Control System
Figure 4



Rigging Beta Control System (Pre Mod 6/2476)
Figure 5



Rigging Beta Control System (Post Mod 6/2476)
Figure 6

6. Rig Cable Slides

Refer to Figure 7 for details.

A. Adjust Cable Slides (Autofeather)

1) Ensure propeller levers are at full INCREASE and with teleflex push-pull control (refer to Figure 5 (Pre Mod 6/2476) or Figure 6 (Post Mod 6/2476)) disconnected at beta control cam, set both power levers to MAX REVERSE. Apply friction lock.

2) Remove upholstery from cabin roof in vicinity of stations 147 to 157.

3) Adjust position of cable slides, if necessary, by loosening bolts securing slides to cable. Move slides so that front chamfers of cable slides are 2.6 inches from rear end of slide tubes as shown in Figure 7.

4) Secure glides by tightening bolts.

5) Release power lever friction lock, move power levers through full range of travel and check for complete freedom of movement.

6) Connect push-pull control to beta control cam.

B. Adjust Cable Slides (Propeller Reset Caution)

1) Move right power lever to IDLE position and apply friction lock.

2) Move left propeller lever to FEATHER position and apply friction lock.

3) Remove upholstery from cabin roof in vicinity of stations 140 to 148.

4) Adjust position of cable slides, if necessary, by loosening bolts securing slides to cables.

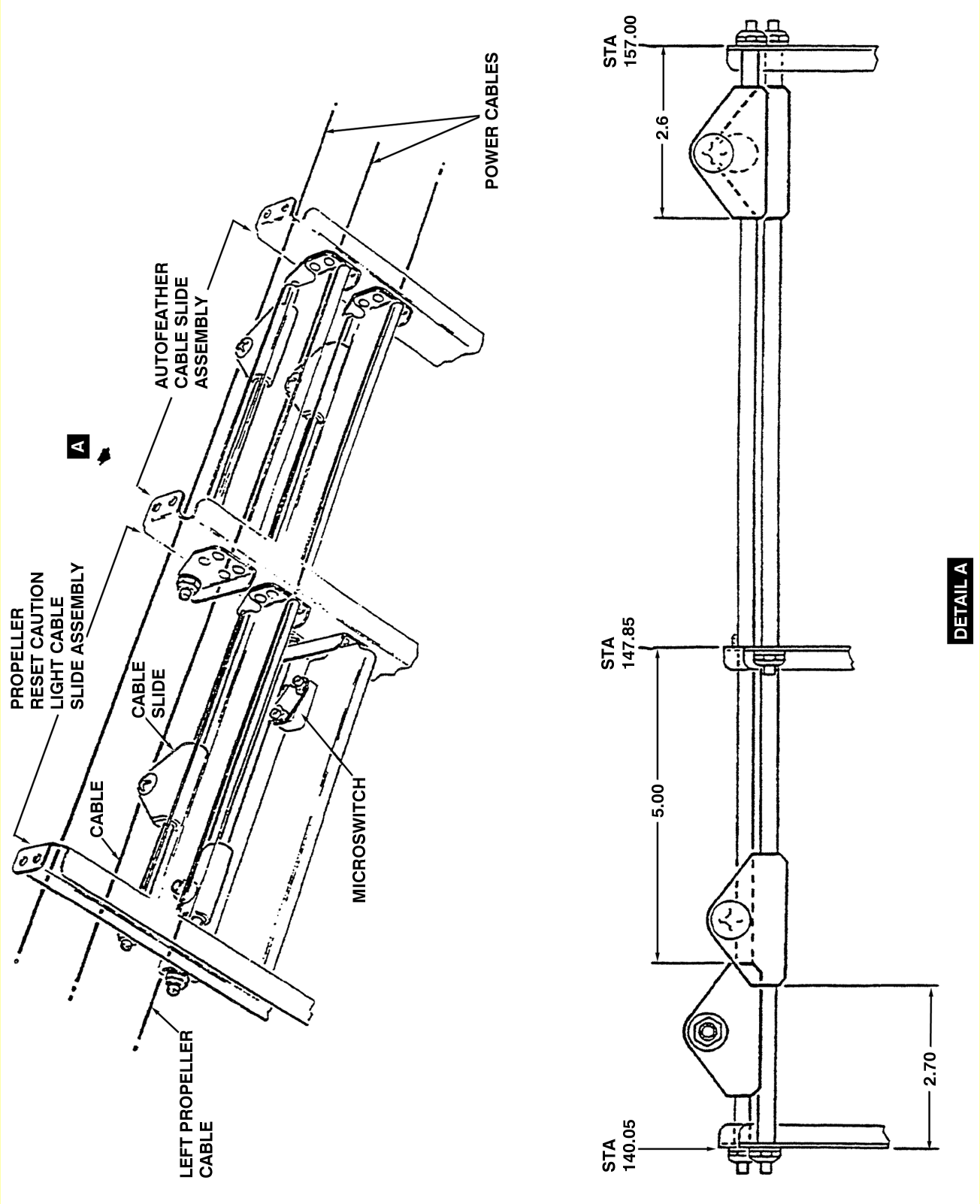
5) Move propeller cable slide (right side) so that rear end of slide is 5.00 inches from rear end of slide tube as shown in Figure 7.

6) Secure propeller cable slide by tightening bolt.

7) Move cable slide (left side) so that front end of slide is 2.7 inches from front end of slide tube, as shown in Figure 7.

8) Secure power cable slide by tightening bolt.

9) Release power and propeller lever friction locks. Check freedom of controls by moving propeller levers through full range of travel and power levers through forward range.



Rigging of Cable Slides
Figure 7

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4) Advance power lever in flight compartment until max power stop bolt contacts maximum power stop anvil on the fuel control unit.

5) Adjust and lock power lever forward stop bolt in overhead console to give 0.12 to 0.08 inch clearance between stop bolt and power lever (refer to Figure 1, Detail A). Clearance may be checked with setting pin SD12570–3 or feeler gauge.

6) Set propeller lever to full INCREASE and retard power lever to reverse in the flight compartment until max power stop bolt again contacts maximum power stop anvil.

7) Adjust and lock power lever aft stop bolt in overhead console to give 0.12 to 0.08 inch clearance between stop bolt and power lever. Clearance may be checked with setting pin SD12570–3 or feeler gauge.

8) Move power lever through full range of travel and check for complete freedom of movement.

9) Connect rear teleflex push-pull control rod end to beta control cam.

Note On completion of adjustment, a check should be carried out to ensure that FCU actuating lever and cam cluster geometry arrangement look the same on both engines. Any appreciable difference will induce power lever split.

10) Rig Beta controls in accordance with Para 3.A.

Note Refer to 76-10-11, Overhead Console – General Data for rigging of power/propeller lever interlock, and to Para 6., for rigging of propeller reset and autofeather microswitches.

3. Rig Beta Controls

A. Adjust Engine Front and Rear Linkage

Refer to Figure 5 for details.

1) Disconnect teleflex push-pull control rod end at beta control cam at rear of engine.

2) Disconnect fuel governor air bleed link at fuel governor reset arm.

3) Adjust propeller governor beta valve:

a) Loosen low pitch adjuster locknut.

b) With propeller in FEATHER and while holding a forward force on propeller reversing lever, adjust low pitch stop in or out as necessary to set the bottom of beta valve clevis slot flush with front face of cap nut.

c) Tighten low pitch adjuster locknut.

d) Disconnect push-pull rod from propeller reversing lever.

Note Ensure bushing is not lost from reversing lever when disconnecting push-pull rod.

e) Check that 1.1 to 1.3 inch travel exists in low pitch adjuster by sliding push-pull rod in and out of adjuster.

f) On satisfactory completion of check, reassemble control at propeller reversing lever.

Note If 1.1 to 1.3 inch travel cannot be obtained check position of propeller low stop collar; distance from propeller mounting face to forward face of carbon block groove should be 1.26 to 1.27 inches. Adjust low stop rods (three) in collar as necessary (refer to Hartzell Manual).

Wear of low stop collar and of cam follower pin in beta control cam will give a very low N_P at 52% N_G.

4) With linkage still positioned as in Para 3.A.3), move fuel governor reset arm forward against its maximum stop. Adjust length of fuel governor link and connect to inner hole of fuel governor reset arm.

5) Move power lever in overhead console fully forward and apply friction lock.

6) Ensure reversing lever is fully forward and low pitch stop in adjuster is contacted and fuel governor reset arm is set as per (Para 3.A.3)). Adjust length of push-pull control at rear clevis rod end to align with hole beta control cam, then lengthen push-pull control by 1/2-turn of rear clevis rod end to ensure low pitch stop is contacted, and connect to hole in beta control cam.

7) Release power lever friction lock, move power lever through full forward range of travel and check for binding and complete freedom of movement. Do not operate in reverse range.

8) Move power levers of both engines back to idle. Check that push-pull teleflex control in low pitch stop adjuster of both engines move back the same distance (approx. 1/8). If required, adjust push-pull at rear clevis rod end to obtain similar movement on both engines.

4. Rig Fuel Shut-off Controls

Refer to Figure 3 for details.

A. Adjustment of the Fuel Shut-off Controls in Airframe

1) Disconnect the FCU fuel shut-off rod end at the intermediate lever attachment bolt on the FCU unit.

2) Select the fuel condition lever to the OFF position in the flight compartment.

3) Install the rig pin in the nacelle engine pulley quadrant.

4) Adjust the wing leading edge control cable and turnbuckles for correct cable tension as required (approximately 25 lbs), refer to Figure 2.

Note Ensure that the fuel condition lever remains in the OFF detent during this adjustment.

5) Remove the rig pin in the nacelle.

6) Select the fuel condition lever to the ON position.

7) Ensure engine control lever moves freely with no binding.

8) Safety the engine control cable turnbuckles.

B. Adjustment of the Fuel Shut-off Control Airframe to Engine

1) Ensure the engine fuel control lever and fuel shut-off control system are rigged according to Para 4.A.

2) Move the engine fuel condition lever in flight compartment to the OFF position.

3) Install the rig pin in the nacelle engine pulley quadrant.

4) Position the FCU intermediate lever to maintain contact between the FCU actuating lever and the FCU fuel cut-off stop bolt.

5) In the nacelle, adjust the fuel shut-off control rod to allow the attach bolt to be installed while maintaining the contact between the FCU actuating lever and the FCU fuel cut-off stop bolt.

6) Remove rig pin from the nacelle.

7) Move engine fuel condition lever to the ON position.

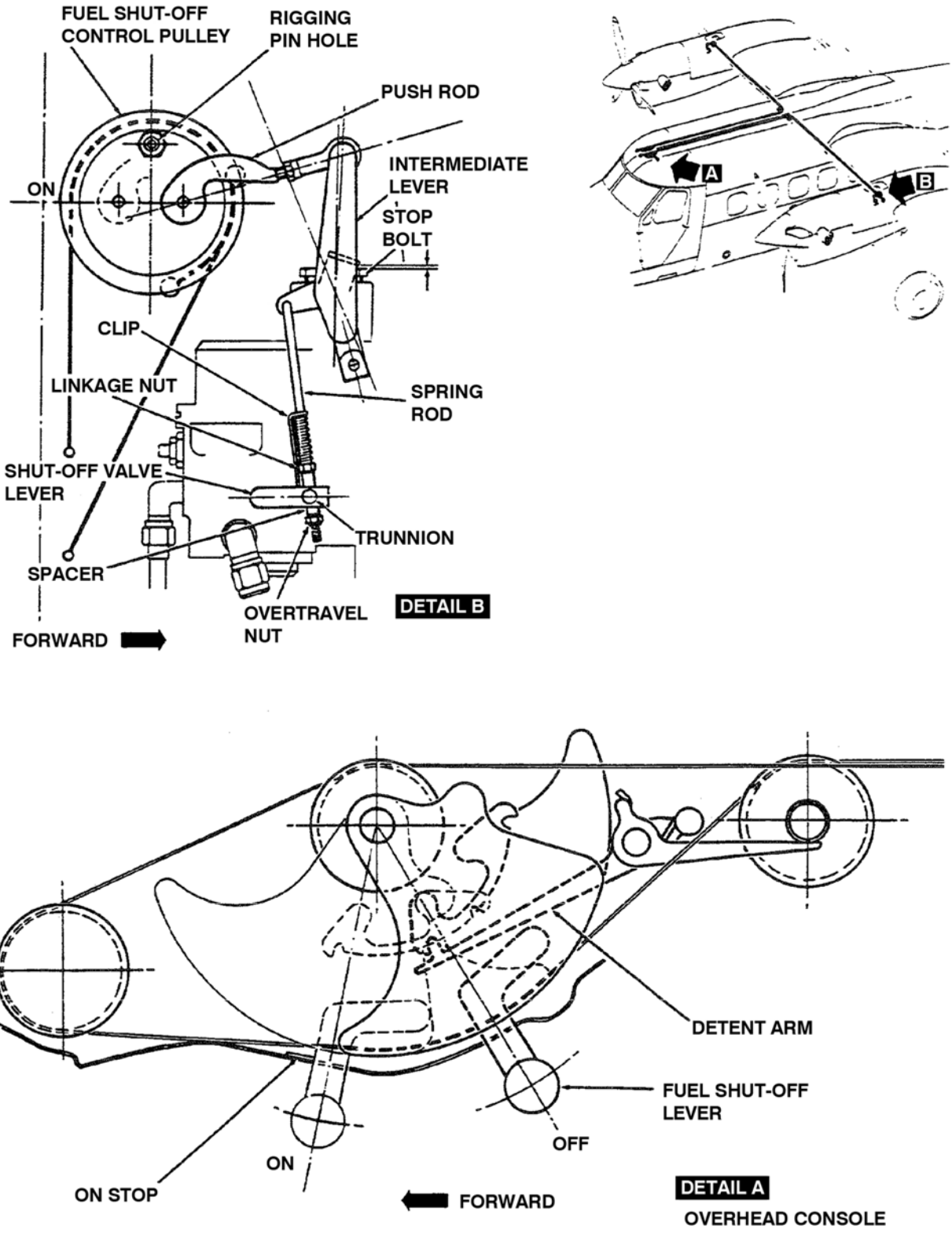
NOTE: Please see the TEMPORARY REVISION that revises this page.

8) Verify that the FCU intermediate lever is aft of the perpendicular position indicating the FCU shut-off is in the fuel ON position.

9) Ensure fuel shut-off control lever moves freely with no binding.

10) Tighten and safety all control hardware. Carry out function test of the FCU fuel shut-off system.

NOTE: Please see the TEMPORARY REVISION that revises this page.



Rigging Fuel Shut-off Controls
Figure 3

5. Rig Propeller Controls

Refer to Figure 4 for details.

A. Adjust Controls in Airframe

1) Disconnect push-pull control at speed select lever on propeller governor unit.

2) Move propeller lever in flight compartment to minimum rpm position detent (roller against stop face). Apply friction lock.

3) Insert rigging pin SD10545–1 in propeller control pulley.

4) Adjust turnbuckles in wing leading edge to obtain cable tension shown in Figure 2.

5) Ensure propeller lever is not moved from minimum rpm position detent and rigging pin moves freely in rigging pin hole of propeller control pulley.

6) Lock turnbuckles.

B. Adjust Controls – Airframe to Engine Interconnect

1) Install rigging pin SD10545–1 in propeller control pulley bracket in engine nacelle.

2) Ensure that propeller governor unit speed select lever is set at minimum rpm position.

3) Adjust sliding rod end so as to achieve a gap of 0.230 inch between plate and feather stop bolt (refer to Detail A of Figure 4).

Note Attachment bolt to be in mid slot position on speed select lever (refer to Detail A of Figure 4).

4) Remove rigging pin from propeller control pulley.

5) Ensure friction lock knob is loosened to permit control lever to move freely.

6) Move control lever fully forward until plate on speed select lever is hard against maximum rpm stop bolt (refer to Detail A of Figure 4).

7) Adjust forward stop bolt in overhead propeller lever console so as to have a gap of 0.120 to 0.080 inch.

Note Clearance may be checked with setting pin SD12570–3 or feeler gauges.

8) Check that propeller levers align within 0.100 inch at knobs. Relocate attachment bolt in slot of speed select lever, on the Constant Speed Unit (CSU) as required.

9) With propeller lever forward apply light pressure on lever. Check gap of forward stop bolt and if necessary readjust to achieve 0.120 to 0.080 inch gap.

10) Retard propeller lever until the speed select lever plate contacts the feather stop bolt on governor.

11) With propeller lever fully aft apply light pressure on lever. Check gap of aft stop bolt and if necessary readjust to achieve 0.120 to 0.080 inch gap.

12) Move propeller lever through full range of travel and check for complete freedom of movement.

13) Safety all attachments and stop bolts as required.

14) If a new cable has been installed, adjust the length as follows:

a) Install rigging pin SD10545–1 in propeller control pulley.

NOTE: Please see the TEMPORARY REVISION that revises this page.

b) Ensure that propeller governor unit is at minimum rpm position.

c) Adjust sliding rod end so as to achieve a gap of 0.230 inch between plate and feather stop bolt (refer to Detail A of Figure 4).

15) If the new cable is too long, trim the cable as follows:

CAUTION **THE FOLLOWING PROCEDURE FOR TRIMMING CABLE IS FOR P/N C6CE1351–31 ONLY.**

a) Cut the cable with an abrasive cutoff wheel.

Note The cable wires are stress relieved and will not unravel when cut.

b) Grind or file the cable to shape a rounded end. Grind or file the cable end in the direction of the cable winding.

CAUTION **DO NOT USE A SOLVENT ON THE CABLE ASSEMBLIES. THE SOLVENT WILL DESTROY THE INTERNAL LUBRICANT OF THE CABLE AND WILL MAKE THE CABLE UNFIT FOR USE.**

c) Clean the end of the cable by brushing at right angles to the centerline of the cable with a hard fiber, plastic or brass bristle brush. Do not use a steel brush.

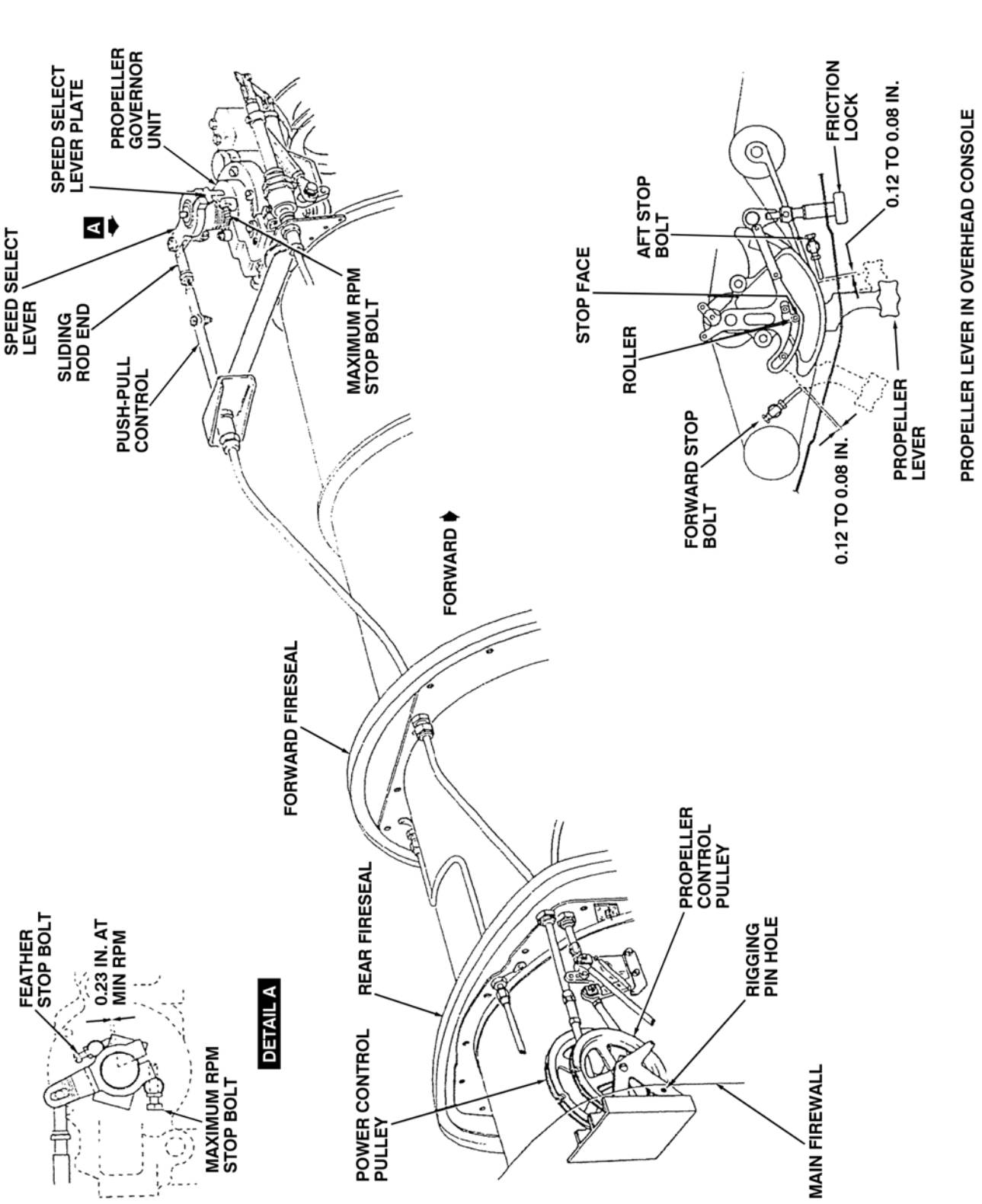
d) Using dry, oil free compressed air at no more than 25 psi, blow the loose dirt particles from the cable.

e) Wipe the cable clean with a dry lint free cloth.

Note It is not necessary to apply any protection to the cut cable end to prevent corrosion, fraying and unwinding.

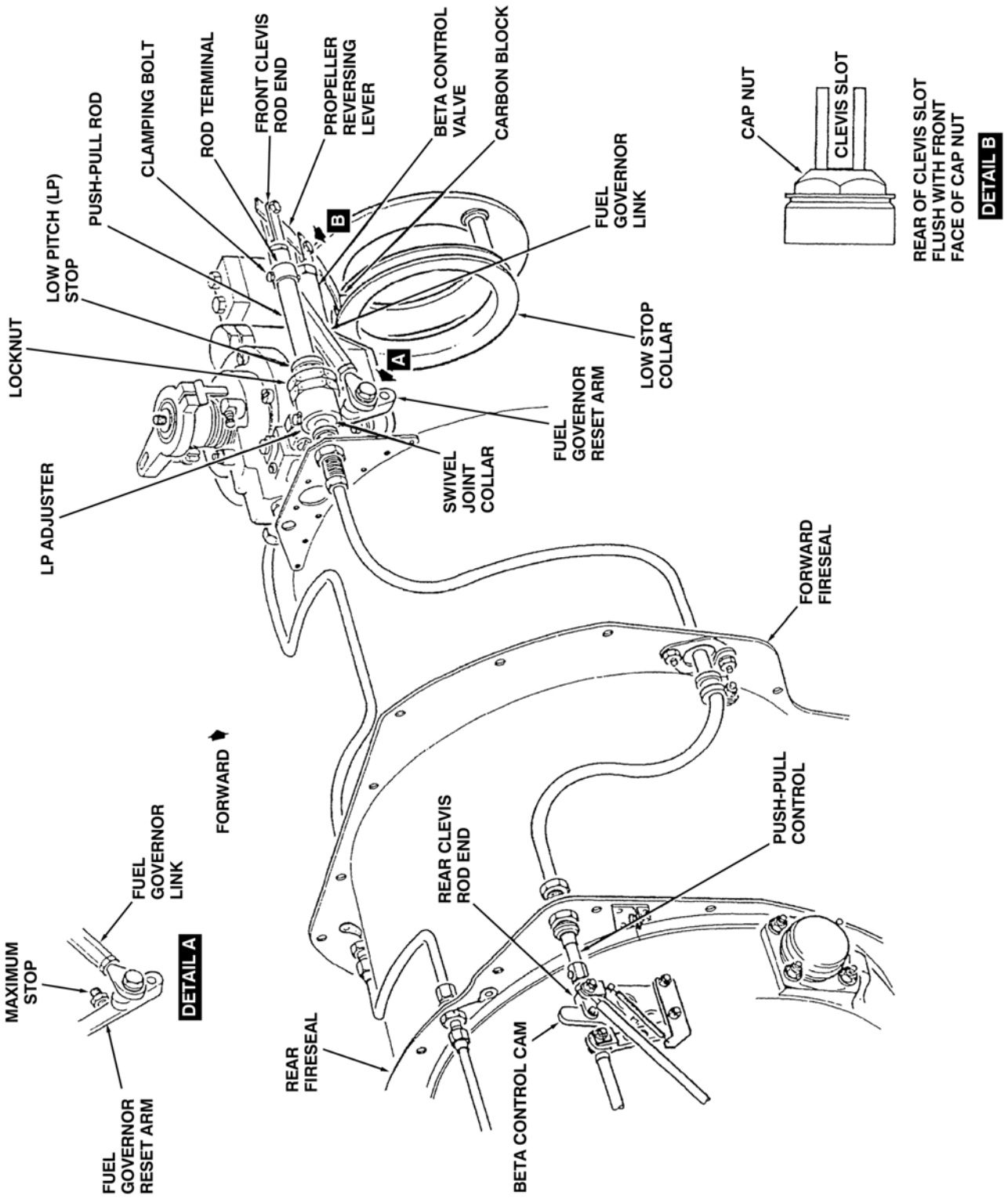
f) Do a visual inspection on the cable for serviceability.

NOTE: Please see the TEMPORARY REVISION that revises this page.



Rigging Propeller Control System
Figure 4

NOTE: Please see the TEMPORARY REVISION that revises this page.



Rigging Beta Control System
Figure 5

NOTE: Please see the TEMPORARY REVISION that revises this page.

6. Rig Cable Slides

Refer to Figure 6 for details.

A. Adjust Cable Slides (Autofeather)

1) Ensure propeller levers are at full INCREASE and with teleflex push-pull control (refer to Figure 5) disconnected at beta control cam, set both power levers to MAX REVERSE. Apply friction lock.

2) Remove upholstery from cabin roof in vicinity of stations 147 to 157.

3) Adjust position of cable slides, if necessary, by loosening bolts securing slides to cable. Move slides so that front chamfers of cable slides are 2.6 inches from rear end of slide tubes as shown in Figure 6.

4) Secure glides by tightening bolts.

5) Release power lever friction lock, move power levers through full range of travel and check for complete freedom of movement.

6) Connect push-pull control to beta control cam.

B. Adjust Cable Slides (Propeller Reset Caution)

1) Move right power lever to IDLE position and apply friction lock.

2) Move left propeller lever to FEATHER position and apply friction lock.

3) Remove upholstery from cabin roof in vicinity of stations 140 to 148.

4) Adjust position of cable slides, if necessary, by loosening bolts securing slides to cables.

5) Move propeller cable slide (right side) so that rear end of slide is 5.00 inches from rear end of slide tube as shown in Figure 6.

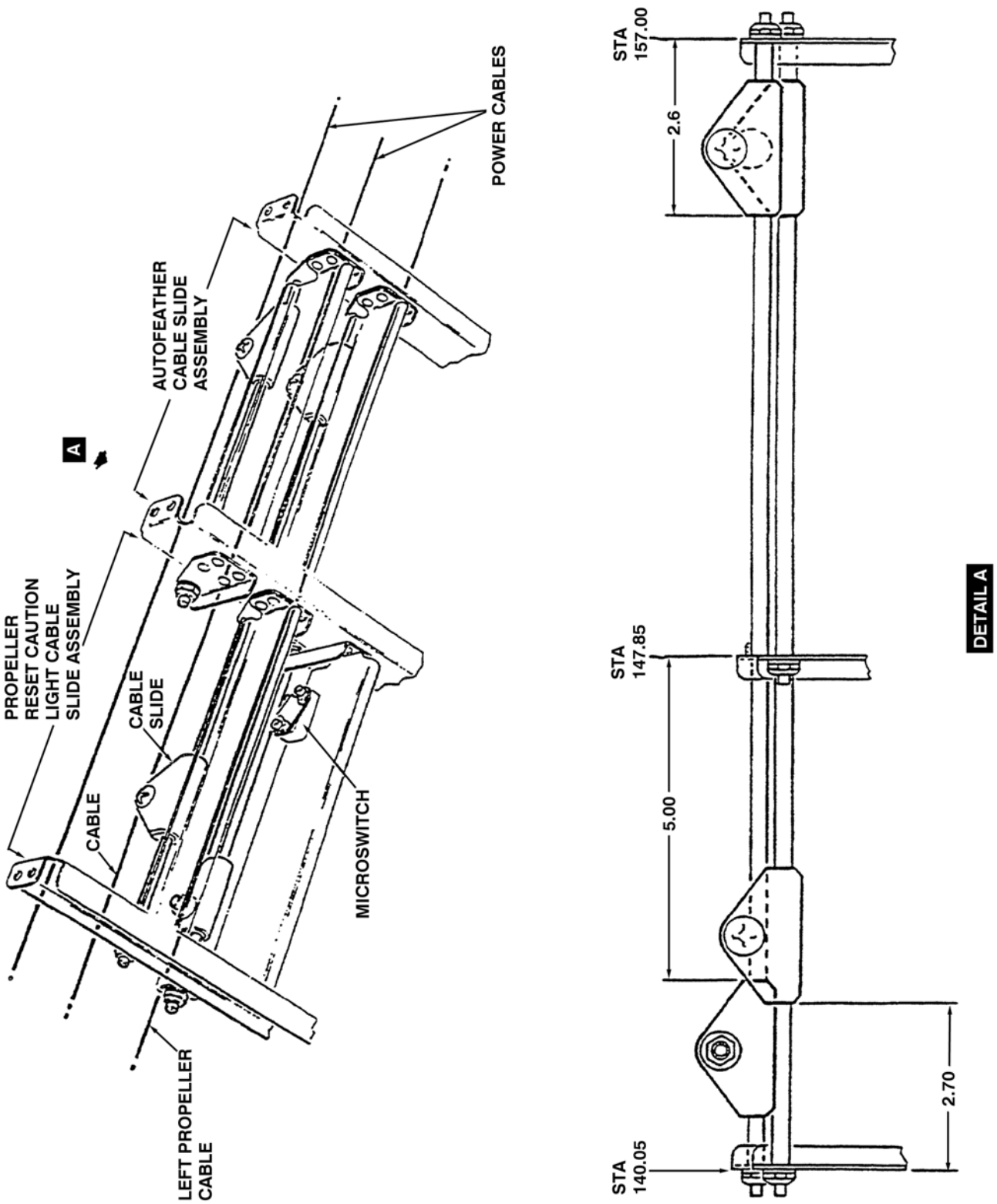
6) Secure propeller cable slide by tightening bolt.

7) Move cable slide (left side) so that front end of slide is 2.7 inches from front end of slide tube, as shown in Figure 6.

8) Secure power cable slide by tightening bolt.

9) Release power and propeller lever friction locks. Check freedom of controls by moving propeller levers through full range of travel and power levers through forward range.

NOTE: Please see the TEMPORARY REVISION that revises this page.



Rigging of Cable Slides
Figure 6

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR76-2**

Insert in 76-10-05 after 76-10-00

FUEL CONTROL UNIT PUSH ROD END – INSPECTION AND LUBRICATION

REASON FOR ISSUE:

To add inspection and lubrication instructions for the Fuel Control Unit Push Rod End.

ACTION:

In Chapter 76, insert new Section 76-10-05 after 76-10-00, Power Control – Maintenance Practices, as shown on Pages 3 to 6 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

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Fuel Control Unit Push Rod End – Inspection and Lubrication

1. General

The Fuel Control Unit (FCU) push rod assembly connected to the FCU power lever contains a rod end bearing at each end; this allows for the free motion of the linkage and resolves any misalignment between the ends.

The Pre Mod 6/2347 FCU push rod assemblies use rod end bearings (ref P/N DSC4–3A) that have an inner race with a fabric-PTFE liner. The Post Mod 6/2347 FCU push rod assemblies use rod end bearings (ref P/N VSC30–3A) that are fabricated with a metal inner race and a dry film lubricant.

Note Post Mod 6/2347 FCU push rod assembly bearings should not be lubricated, except on aircraft operating in high salt environments between the tropics of Cancer and Capricorn, or where the operator has previously observed corrosion on Post Mod 6/2347 FCU push rod assembly bearings. When lubrication is required for corrosion prevention, lubricate in accordance with Para 3. and Table 1 — Lubrication of Engine Control Rod Ends.

2. Inspection of Fuel Control Unit Rod End Bearings

WARNING THE PROCEDURE USES MATERIAL THAT MAY BE DANGEROUS TO THE PERSONNEL'S HEALTH AND/OR THE ENVIRONMENT. SUFFICIENT PRECAUTIONS MUST BE TAKEN AT ANY STEP OF USE. REFER TO THE MATERIAL SAFETY DATA SHEET AND ANY LOCAL REGULATIONS REGARDING EXPOSURE, USE, AND DISPOSAL PRIOR TO COMMENCING WORK.

A. Inspection of Fuel Control Unit Rod End Bearings

1) Switch OFF aircraft electrical power (refer to 24-00-00, Electrical Power – General Data).

2) Lower the engine cowlings.

3) Remove panel to gain access to the Fuel Control Unit.

4) Use a fine tip permanent maker to draw a line across the edge of the serrated washer and the applicable mating parts to ensure that the washer to lever matching remains undisturbed.

Note Slotted adjustment points utilize serrated washers. Ensure serrated washers, where applicable, are properly marked for alignment with the mating parts before disconnecting.

5) Disconnect the FCU control rod from the FCU arm; remove the cotter pins, nuts, bolts and washers.

6) Clean each FCU rod end bearing in accordance with Para 2.B.

7) Identify the rod end bearing type as per Table 1 — Lubrication of Engine Control Rod Ends.

8) Inspect each rod end bearing ball:

a) Manually turn the ball and feel for signs of roughness or excessive drag in rotation. The bearing ball must be easily moveable with the fingers. If the rod end bearing ball does not move easily and smoothly, replace.

b) Check the rod end bearing ball for excessive side and radial play. Replace rod ends with excessive side or radial play.

c) Examine the rod end housing and bearing ball for corrosion. Replace rod ends with corrosion on the bearing ball. Remove light corrosion on the housing only with a coarse weave cloth or fibre brush.

Note If light corrosion was removed from the rod end housing, clean rod end bearing in accordance withPara 2.B.

d) Examine the rod end bearing for damage, including:

- scratches;
- scoring;
- cracks;
- flaked plating;
- fretting against the edges of the race.

e) Examine the rod end race for damage, including evidence of:

- excessive wear as indicated by roughness, or excessive axial or radial play;
- liner cracking (fabric-PTFE lined rod ends only);
- detached liner as indicated by the liner being loose or shifted to one side (fabric-PTFE lined rod ends only);
- evidence of corrosion between the housing and the liner or dry-film lubricating coating;
- sharp protrusions and/or chips.

9) Replace rod ends if damage is observed in Para 2.A.8).

10) Lubricate rod end bearing using the lubricant specified in Table 1 — Lubrication of Engine Control Rod Ends and in accordance with Para 3.

11) Re-install the FCU control rod; install the cotter pins, nuts, bolts and washers. Replace damaged or corroded hardware.

Note Ensure that the marked position of each serrated washer and mating part has remained in alignment. If alignment is not maintained or cannot be confirmed, check rigging as per 76-10-00, Power Control – Maintenance Practices.

12) Check free movement of the linkage.

13) With power levers at IDLE position.

a) Confirm that the FCU power lever is in the idle position (against idle stop).

b) Push FCU push rod forward and release.

c) FCU power lever should immediately return to idle position.

14) Reinstall panel.

15) Close engine cowls.

16) Do an engine run in accordance with 71-00-00, Power Plant (Mod 6/2012) – General Data.

B. Cleaning

1) Materials and Equipment

a) Materials

- Cleaning solvent DS108, Diestone DLS, or Diestone HFP

b) Equipment

- Cloth, coarse weave
- Cloth, lint-free

2) Safety Precautions

a) Sufficient ventilation must be provided when using solvent in confined areas. Wear protective respiratory equipment when applying solvent.

b) Wear protective gloves to avoid skin contact with solvents. If contact occurs, wash the area thoroughly with soap and water. Wash or discard contaminated clothing as recommended by the Material Safety Data Sheet (MSDS).

c) Refer to the applicable manufacturer's MSDS for specific safety data on any of the materials specified herein.

3) Cleaning Procedure

CAUTION **EXCESSIVE SOLVENT MAY DAMAGE OR LOOSEN FABRIC-PTFE LINER OR DRY FILM SELF-LUBRICATING COATING. DO NOT USE EXCESSIVE SOLVENT TO CLEAN ROD END BALL. DO NOT SOAK ROD END.**

CAUTION **WATER MAY DAMAGE, SWELL OR LOOSEN FABRIC-PTFE LINER. DO NOT USE WATER TO CLEAN ROD END BALL. DO NOT SOAK ROD END.**

a) Clean the rod end housing and/or ball using a clean, dry cloth. Remove all residual lubricant, dust and dirt.

b) When necessary, clean with a lint-free cloth lightly dampened with cleaning solvent in accordance with Para 2.B.3).a).

3. Lubrication

Lubrication of certain rod ends is recommended after inspection and, on condition, after wash-out of the engine externals in the area of the power control linkage.

WARNING **DO NOT LUBRICATE DRY FILM COATED ROD ENDS. EXCEPT AS NOTED IN Table 1 — Lubrication of Engine Control Rod Ends. DAMAGE TO THE COATING AND/OR ROD END BINDING MAY RESULT.**

CAUTION **DO NOT USE LUBRICANTS ON FABRIC-PTFE LINED ROD ENDS OTHER THAN THOSE SPECIFIED IN Table 1 — Lubrication of Engine Control Rod Ends. DAMAGE TO THE LINER AND/OR EXCESSIVE BREAK-OUT TORQUE MAY RESULT.**

Table 1 Lubrication of Engine Control Rod Ends

Rod End Type	Liner or Coating Type	Liner or Coating Appearance	Lubrication Requirement	
Pre Mod 6/2347	Fabric-PTFE Liner	Amber to dark orange bearing race liner. Edge visible around the ball.	LUBRICATE	MIL-L-7870 (Royco 363) OR MIL-PRF-81322 (AeroShell Grease 22)
Post Mod 6/2347	Dry Film Coating	Dark grey to black coating, applied to the inside of the bearing race. Note The dry film lubricant layer is thin, with no edge visible. Rotate the ball and use a bright light to view the inside of the race.	DO NOT LUBRICATE EXCEPT, lubricate rod end bearings on aircraft operating in high salt environments between the tropics of Cancer and Capricorn, or where the operator has previously observed corrosion on Post Mod 6/2347 FCU push rod assembly bearings.	

Note If there is difficulty identifying the rod end type contact Viking Technical Support.

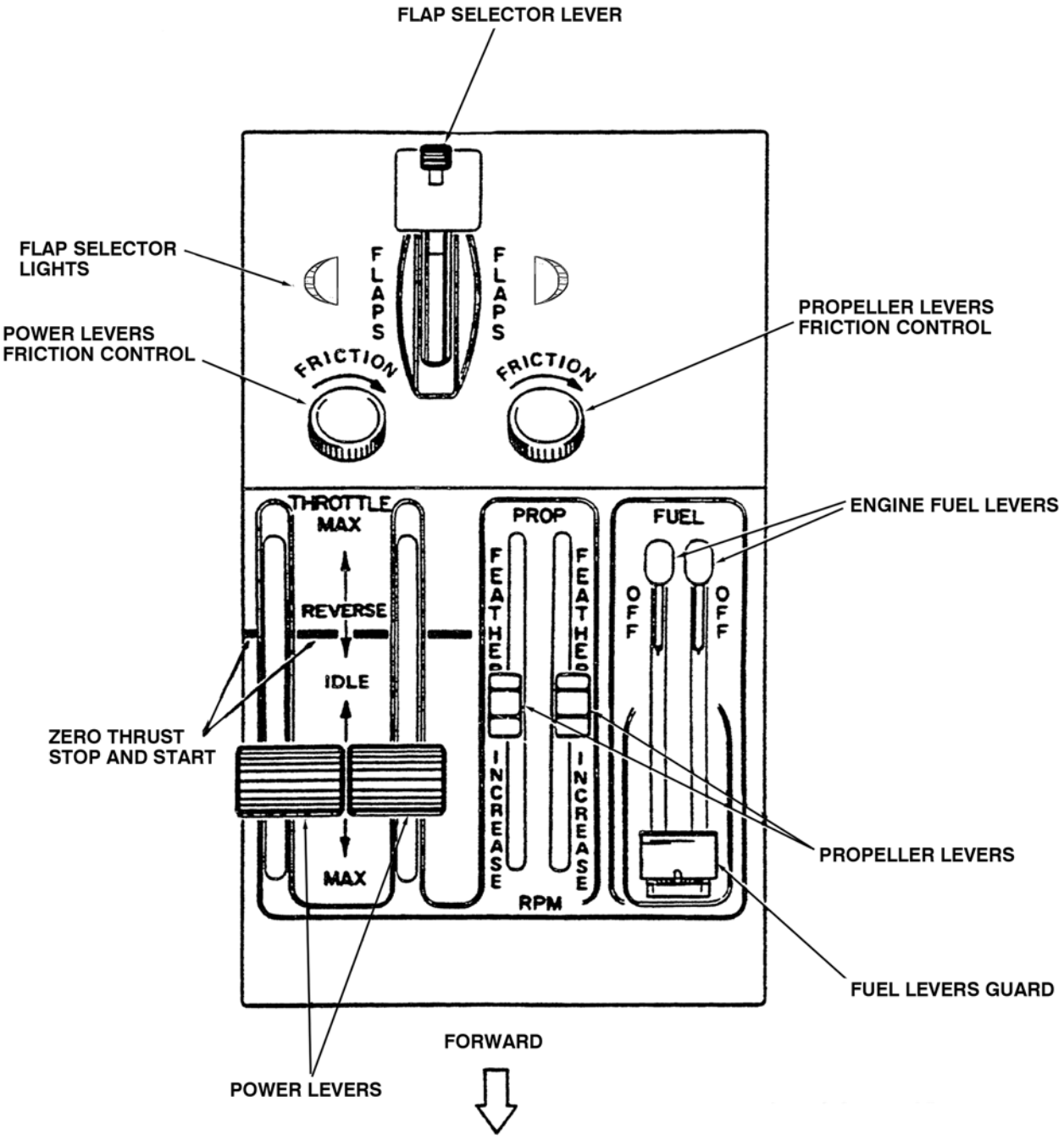
Overhead Console – General Data

1. General

Refer to Figure 1 for details.

The overhead console contains the power, propeller, fuel and flap control levers, together with associated quadrants, pulleys and control cables. The power and propeller levers are fitted with friction locks and positioning detents. Incorporated in the overhead console is a power/propeller lever interlock which prevents reverse power lever selection at propeller speeds below 91% rpm. There are five microswitches mounted in the overhead console, four are operated by the fuel control levers in the OFF position and one operated by the power levers in the reverse range. The fuel lever microswitches are operated during the start cycle to change the starter from a compound motor to a series motor, and provide fuel off indication to the CAS. The power lever microswitch is operated in the reverse range to detect the command of reverse power.

The console also has a spring-loaded hinged guard, fitted on the front of the console to prevent inadvertent movement of the fuel levers when they are in the ON position.



Overhead Console
Figure 1

2. Adjustment/Test

Refer to Figure 2 for details.

A. Rig Power/Propeller Lever Interlock

1) Disconnect power push rods from power control pulleys (refer to 76-10-00, Power Control – Maintenance Practices) and remove left side fairing and lower panel from overhead console.

2) Raise latch lever to main quadrant shaft and support in this position with an 0.06 inch separator between latch lever and main quadrant shaft.

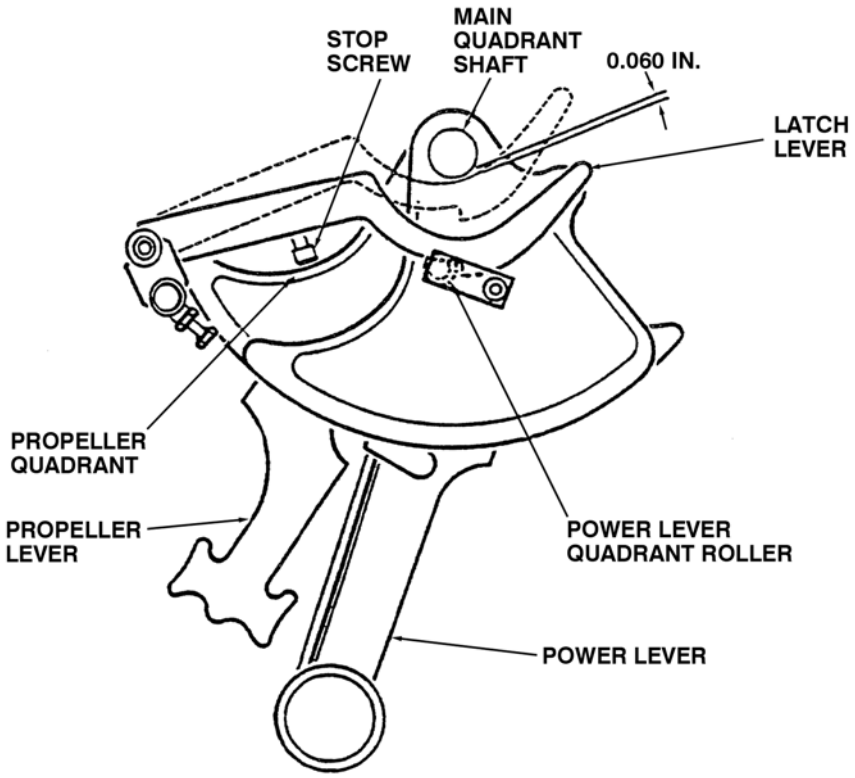
3) Set both power levers to the MAX REVERSE position.

4) Advance propeller lever to maximum rpm position and adjust stop screw, on latch lever shaft, to contact propeller quadrant. Retard propeller lever to mid range.

5) Repeat Para 2.A.4) for opposite propeller lever.

6) Release latch lever and set both power levers to the IDLE position.

7) Check that power levers cannot be moved into REVERSE until either propeller lever is in last inch (approximately) of forward travel. Connect power push rods to power control pulleys.



Rigging Power/Propeller Lever Interlock
Figure 2

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Fuel Control Unit – Maintenance Practices

The Fuel Control Unit (FCU), as received from the OEM or overhaul facility, must be reconfigured to function correctly on the Twin Otter, Series 400.

1. Removal/Installation

CAUTION ANY DAMAGED OR UNACCEPTABLE HARDWARE SHOULD BE REPLACED PRIOR TO INSTALLATION ON NEW FCU.

A. To remove power plant, refer to 71-00-00, Power Plant (Mod 6/2012) – General Data.

B. To remove existing FCU, refer to P&WC PT6A–34 Engine Maintenance Manual, Chapter 73-20-00.

C. Remove from existing FCU the following, refer to Figure 1.

1) Remove the FCU Arm, washer and nut off of the FCU Spindle Shaft. Retain for installation on new FCU.

2) Remove the screws (2), washers (2) and nuts (2) from the bottom end of the FCU Shutoff Control Levers.

3) Remove the cotter pins from the FCU Intermediate (shaft) Pin, and washers from the outside of the FCU Shutoff Control Levers. Retain hardware for installation on new FCU.

4) Remove the nut, washers (2), bolt and spacer that keeps the FCU Shutoff Control Levers aligned. Retain hardware for installation on new FCU.

5) Remove the FCU Shutoff Control Levers from the right and left hand sides of the FCU Intermediate Lever. Retain FCU Shutoff Control Levers, FCU Intermediate (shaft) Pin and associated hardware for reinstallation on new FCU.

D. Rework new FCU as follows:

1) Cut off safety cable, support Pratt & Whitney supplied FCU arm and remove the Pratt & Whitney supplied extension nut. Remove Pratt & Whitney supplied FCU arm and retain with the Pratt & Whitney supplied extension nut to be reinstalled on old FCU.

CAUTION DO NOT REMOVE SERRATED SPACER FROM THE FCU SPINDLE SHAFT.

2) Temporarily install FCU arm, washer and nut removed from the old FCU, refer to Para 1.C.1).

3) Remove the Pratt & Whitney supplied stop bolt from new FCU. Retain to be reinstalled on old FCU.

4) Remove the cotter pins (2) from the FCU Intermediate Lever (shaft) Pin.

5) Remove the Pratt & Whitney supplied nut and spacer from High Idle Rod. Retain to be reinstalled on old FCU.

6) Carefully withdraw the FCU Intermediate (shaft) Pin until R/H side of FCU Intermediate Lever can be slid forward to free the High Idle Rod from the Intermediate Lever.

7) Reposition FCU Intermediate Lever and reinsert the FCU Intermediate Lever (shaft) Pin. Remove cotter pin from base of High Idle Rod, remove High Idle Rod. Retain to be reinstalled on old FCU.

8) Hold FCU Intermediate Lever in position and remove the FCU Intermediate Lever (shaft) Pin. Install the FCU Shutoff Control Lever on left side of Intermediate Lever with flat side to the rear and reinsert the FCU Intermediate Lever (shaft) Pin. Install other FCU Shutoff Control Lever on right side of Intermediate Lever with flat side facing forward, refer to Figure 1.

9) Using a 3/16” bolt or drill bit to help align the two FCU Shutoff Control Levers, install spacer, bolt, washer and nut.

10) Install screws (2), washers (2), nuts (2) in the bottom hole of each FCU Shutoff Control Lever.

Note The flat side of the FCU Shutoff Control Levers are facing forward.

11) Install cotter pins (2) in FCU Intermediate Lever (shaft) Pin.

12) Remove alignment bolt/bit drill.

13) Install screw and nut for new stop bolt.

14) Attach rigging gauge SD12556 using a temporary washer in between gauge and lever, to maintain alignment. Adjust FCU arm so that rig pin on gauge slides freely through the hole on the lever.

Note There should be no friction on the rig pin.

15) Readjust serrated spacer as required to eliminate any friction. Once adjusted, mark an index line across FCU shaft, serrated spacer and lever for ground run rigging reference.

16) Tighten nut and install MS24665–138 cotter pin (may require removal for ground runs).

17) Hold Intermediate Lever in closed position and adjust stop bolt until light contact with lever occurs. Adjust “null” dimension between stop bolt and throttle pick-up position so that 0.020” gap is achieved for static rigging ONLY.

18) Tighten jam nut.

E. Reinstall retained parts on the old FCU as follows:

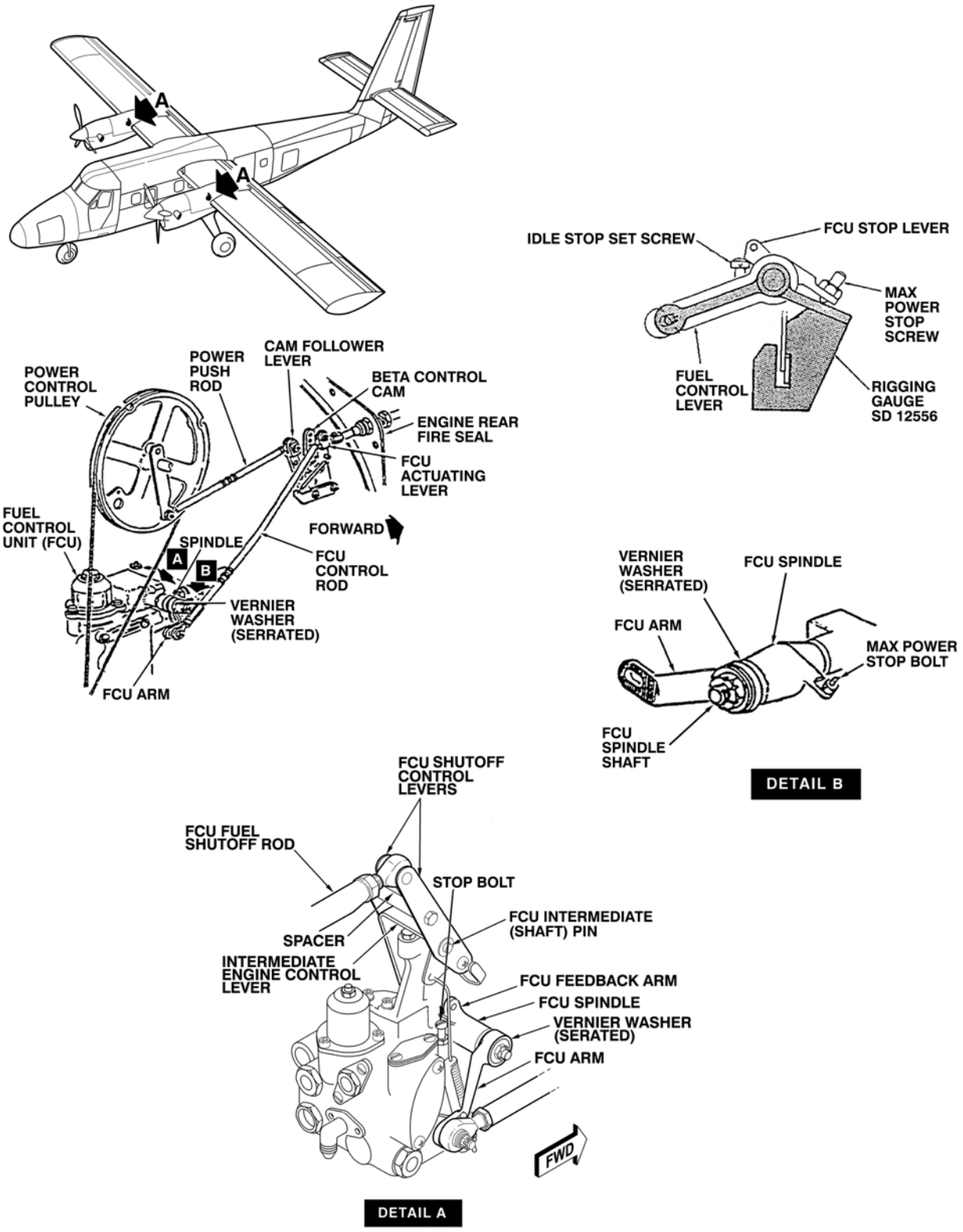
1) Install the Pratt & Whitney supplied FCU arm and supplied extension nut on old FCU.

2) Install the Pratt & Whitney supplied stop bolt on old FCU.

3) Install the High Idle Rod and cotter pins (2) on old FCU.

F. To install new FCU, refer to P&WC PT6A–34 Engine Maintenance Manual, Chapter 73-20-00.

G. To install power plant, refer to 71-00-00, Power Plant (Mod 6/2012) – General Data.



Power Lever Control Rigging
Figure 1

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Emergency Shutdown – General Data

1. General

As the oil system is an integral part of the engine, only the fuel system is provided with an emergency shut-off control. The fuel system emergency shut-off consists essentially of two emergency shut-off switches, one for each engine, mounted on the emergency panel and two firewall mounted shut-off valves (refer to 28-20-25, Emergency Shut-off Valve – Maintenance Practices). Each switch is labeled FUEL with marked OFF and NORMAL positions. The emergency shut-off circuits are protected by circuit breakers, mounted on the main circuit breaker panel, labeled FUEL SHUTOFF L and FUEL SHUTOFF R.

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CHAPTER 77

ENGINE INDICATING

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CHAPTER 77 – ENGINE INDICATING

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Engine Indicating – General Data

1. General

The engine indicating systems comprise a torque pressure indicating system, a gas generator tachometer indicating system and a turbine temperature indicating system. For information on the propeller tachometer indicating system refer to 61-40-00, Propeller Indicating – General Data.

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Torque Pressure Indicating System – General Data

1. General

The torque pressure indicating system consists essentially of a pressure transmitter mounted on the engine reduction gear case. Torque pressure indication is displayed in the engine parameters display, provided in the upper inboard 1/6th of the Primary Flight Display. The power supply for the indicating system is 28 Volt DC with 5–ampere circuit breakers, marked TORQUE PRESSURE L and TORQUE PRESSURE R.

A. Torque Pressure Indication

Torque pressure (TRQ PSI) is indicated with a pointer type display with integer values beneath each display. Units are displayed in psi.

B. Pressure Transmitter

The torque pressure transmitter is mounted on the engine reduction gear case and is used to transmit a pressure differential, obtained from two tappings on the reduction gear case, which is supplied to the DAU for conversion to digital data for the Apex system.

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Gas Generator Tachometer Indicating System – General Data

1. General

The gas generator tachometer indicating system consists essentially of a tachometer generator, mounted on the engine accessory case. Indication is displayed in the engine parameters display provided in the upper inboard 1/6th portion of the Primary Flight Display.

A. Gas Generator Tachometer Indication

Tachometer indication (N_G) is displayed in percentage values within the engine parameters display.

B. Gas Generator Tachometer Generator

The tachometer generator provides a sinusoidal frequency output, proportional to gas generator rpm, which is supplied to the DAU for conversion to digital data for display on the engine parameters display. The tachometer generator is mounted on the engine accessory case and is driven from the internal scavenge pump drive. Rotation is counterclockwise with a drive ratio of 0.112 : 1.

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Gas Generator Tachometer Generator – Maintenance Practices

1. Adjustment/Test

A. Bench Test Tachometer Generator

Refer to for details.

Equipment required:

- Variable speed motor
- Drive speed indicator
- Tachometer generator output indicator (percent rpm)
- Mounting fixture and wiring
- Motor control switch
- HI–RPM–LOAD (Three 40 ohm, Y connected resistances)
- LO–RPM–LOAD (Three 20 ohm, Y connected resistances)
- Load ON–OFF switch (Two required)
- Multimeter

1) Continuity Test:

a) Set multimeter to OHMS R x 1.

b) Measure the continuity resistance between each pair of stator windings A–B, A–C and B–C (C = Case). The resistance between each pair of windings must be less than 50 ohms and must be within 2.0 ohms of each other.

2) Rotation Test:

a) Connect tachometer generator to output indicator as shown in Figure 1.

b) Operate motor and check output indicator pointer moves in a clockwise direction as motor speed is increased.

Note Tachometer generator is rotated in a counterclockwise direction when viewed from shaft end.

c) Shutdown motor.

3) Voltage Output Test:

a) Disconnect wiring from output indicator (refer to Para 1.A.2).a)) and connect to test point terminals as shown in Figure 1.

b) Ensure LO–RPM–LOAD switch is OFF, and select HI–RPM–LOAD switch to ON.

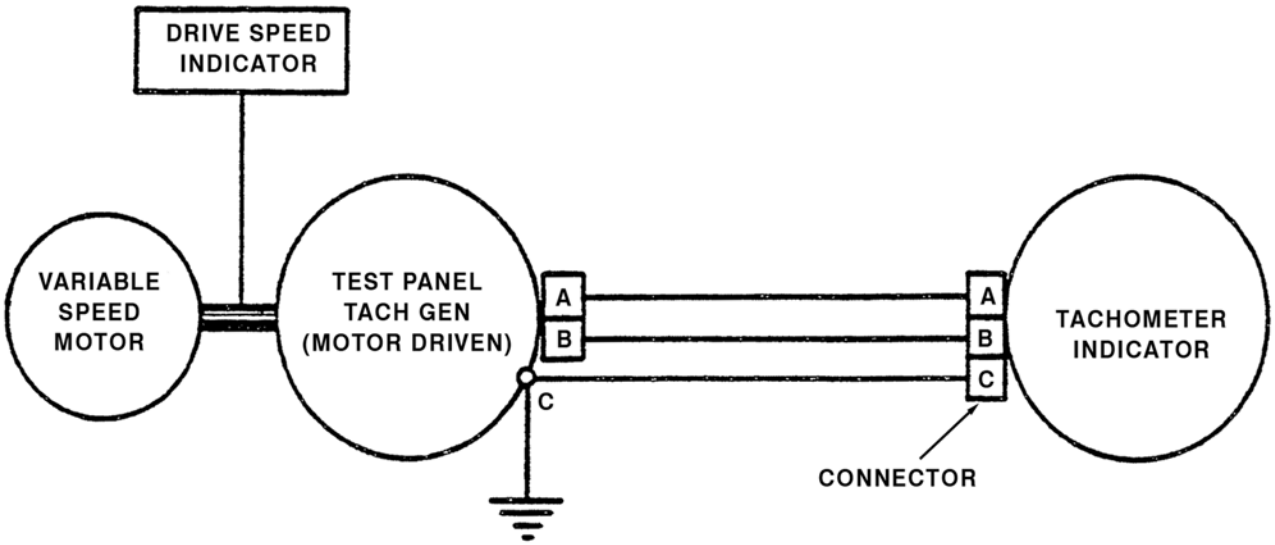
c) Operate motor and increase speed to 4200 rpm (drive speed indicator).

d) Set multimeter to 30 volts AC range and measure voltage outputs at the test point terminals across each of the phases A–B, A–C and B–C. The voltage across each phase must be 21 ± 0.5 volts AC.

e) Decrease motor speed to 1000 rpm, select HI–RPM–LOAD switch to OFF and LO–RPM–LOAD switch to ON.

f) Repeat Para 1.A.3).d) and check that phase voltages are not less than 3.5 volts AC.

g) Remove tachometer generator from test bench.



NOTE: TACHOMETER GENERATOR (TACH GEN) IS ROTATED IN A CLOCKWISE DIRECTION WHEN VIEWED FROM THE SHAFT END

Test Rig Layout Schematic
Figure 1

Turbine Temperature Indicating System – General Data

1. General

The turbine temperature (T_5) indicating system provides an accurate indication of engine operating temperatures taken at a point between the two turbines. The system consists essentially of a thermocouple assembly, mounted around the power turbine stator housing, and a turbine temperature indication displayed in the engine parameters display provided in the upper inboard 1/6th portion of the primary flight display.

A. Turbine Temperature Indication

Turbine temperature (T_5) is displayed with a pointer type display with integer values beneath each display. Units are displayed in °C.

B. Thermocouple Assembly

The thermocouple assembly consists of 8 individual chromel-alumel thermocouple probes connected in parallel, two bus-bars, twin leads and a connector. Each probe protrudes through a threaded boss into the turbine stator housing between the two turbines and may be removed and replaced individually.

2. Adjustment/Test

A. Calibration Check

CAUTION **EXERCISE CARE WHEN REMOVING OR INSTALLING TERMINAL BOLTS. EXCESSIVE PRESSURE ON LUGS COULD RESULT IN DAMAGE TO INSULATION.**

Equipment required:

- Calibrated thermocouple source
- Multimeter – Avo 8 (or equivalent)

1) Disconnect turbine temperature leads from thermocouple harness terminals at engine.

2) Using a multimeter, check insulation and resistance limits of T_5 thermocouple harness at engine thermocouple harness terminals.

   - Insulation (terminals to ground) – not less than 50,000 ohms
   - Resistance (between terminals) – between 0.58 and 0.74 ohm

3) Using the thermocouple source, connect to the disconnected thermocouple leads on the aircraft side.

   a) Set thermocouple source output to 0°C (0µV) and record the output temperature displayed on Apex.

   b) Set thermocouple source output to 600°C (24901.6µV) and record the output temperature displayed on Apex.

   c) Set thermocouple source output to 700°C (29128.3µV) and record the output temperature displayed on Apex.

   d) Set thermocouple source output to 800°C (33276.8µV) and record the output temperature displayed on Apex.

      Ensure readings are within the following accuracies:

      - –55°C to 300°C ± 36°C
      - 300°C to 800°C ± 15°C
      - 700°C to 800°C ± 10°C
      - 800°C to 1200°C –5°C/+10°C

4) Reconnect turbine temperature leads to thermocouple harness terminals at engine.

CHAPTER 78

EXHAUST

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CHAPTER 78 – EXHAUST

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Exhaust – General Data

1. General

The exhaust system consists of two exhaust nozzles, one on each side of the engine exhaust case. The nozzles protrude through the engine cowlings, at a point between the upper and lower cowlings, then curve rearwards to direct the exhaust gases to the side of the engine and over the wing. To assist in the direction change of the exhaust gases, two curved vanes are fitted inside each exhaust nozzle near the outlet end. Each nozzle is secured to the engine exhaust case flange with nuts, bolts and washers. The cowlings have fairings around the nozzle openings.

2. Removal/Installations

A. Remove Exhaust Nozzle

1) Open lower engine cowling and remove upper engine cowling. (Refer to 71-10-00, Cowlings – General Data).

2) Remove nuts, bolts, and washers, securing nozzle to engine exhaust duct flange and remove nozzle.

B. Install Exhaust Nozzle

1) Position nozzle on engine exhaust duct flange and secure with nuts, bolts, and washers.

2) Install upper cowling and close lower engine cowling. (Refer to 71-10-00, Cowlings – General Data).

3. Approved Repairs

A. Weld Cracked Exhaust Nozzle

1) Refer to Para 2.A and remove nozzle.

2) Clean nozzle with a suitable cleaning solvent.

3) Stop-drill crack using a No. 36 drill and carefully deburr.

4) Clean weld area with a clean stainless steel brush or fine abrasive paper.

5) Carry out weld repair by T.I.G. welding method using Inconel 625 filler wire. Preheating is not required.

CAUTION DO NOT CARRY OUT LOCAL STRESS RELIEVING BY TORCH METHOD

6) Stand nozzle on flanged end and stress relieve at 1600 ± 25°F for one hour then air cool in still dry air.

Note Small repairs (1/4 inch maximum) do not require relief after welding.

7) Refer to Para 2.B and install nozzle.

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CHAPTER 79

OIL

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CHAPTER 79 – OIL

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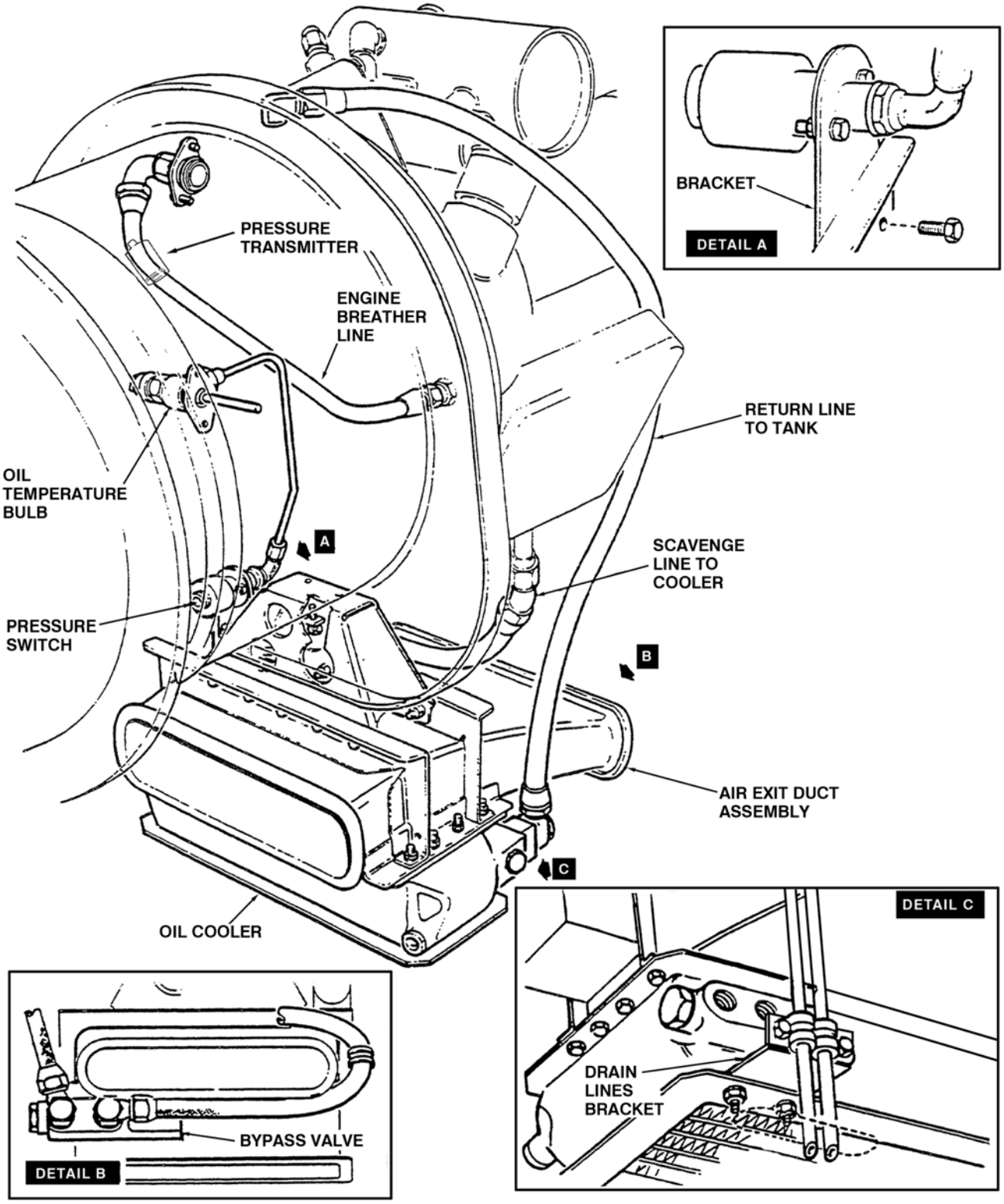
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Oil – General Data

1. General

Refer to Figure 1 for details.

The oil system is an integral part of the engine and is fully described in the PT6A–34 Engine Maintenance Manual. However, external lines are used to transfer scavenge oil from the engine to the cooler, and from the cooler to the tank. An engine breather line, connected to the upper right of the accessory case, passes through the firewall and vents overboard through the upper trailing edge of the wing strut. The system also consists of an engine mounted oil pressure transducer and oil temperature probe for each engine. There is an oil pressure switch mounted on top of the oil cooler.



Oil System
Figure 1

Distribution – General Data

1. General

The distribution portion of the oil system consists of two external lines, an oil cooler and bypass valve. Scavenge oil is pumped from the engine accessory case through an external line to the oil cooler bypass valve, which either allows oil through the cooler or bypasses oil to the return line if the temperature is low or cooler is restricted. The external return line transfers the oil from the cooler or bypass valve into the base of the fuel heater, mounted on top of the tank in the engine accessory case.

A. Oil Cooler

The oil cooler is located below the engine accessory case and attached to the bottom of an air exit duct assembly by eight bolts, washers and nuts. The oil cooler is of the honeycomb type and fitted with a bypass valve and drain plug.

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Oil Cooler (Mod 6/1438 and Mod 6/1481) – Maintenance Practices

1. Removal/Installation

Refer to Figure 1 for details.

A. Remove Oil Cooler

1) Lower engine bottom cowling.

2) Have container in position to receive oil spillage.

3) Disconnect inlet and outlet lines at oil cooler. Cap open lines.

4) Remove bolts securing drain lines bracket to lower rear flange of oil cooler. Disconnect clips from drain lines and remove bracket.

5) Remove bolts securing oil cooler to duct assembly. Remove oil cooler by sliding cooler forward and down.

B. Install Oil Cooler

1) Position oil cooler to duct assembly, center on duct opening and install eight retaining bolts, washers and nuts.

2) Clip drain lines to drain lines bracket and position bracket on lower rear flange of oil cooler.

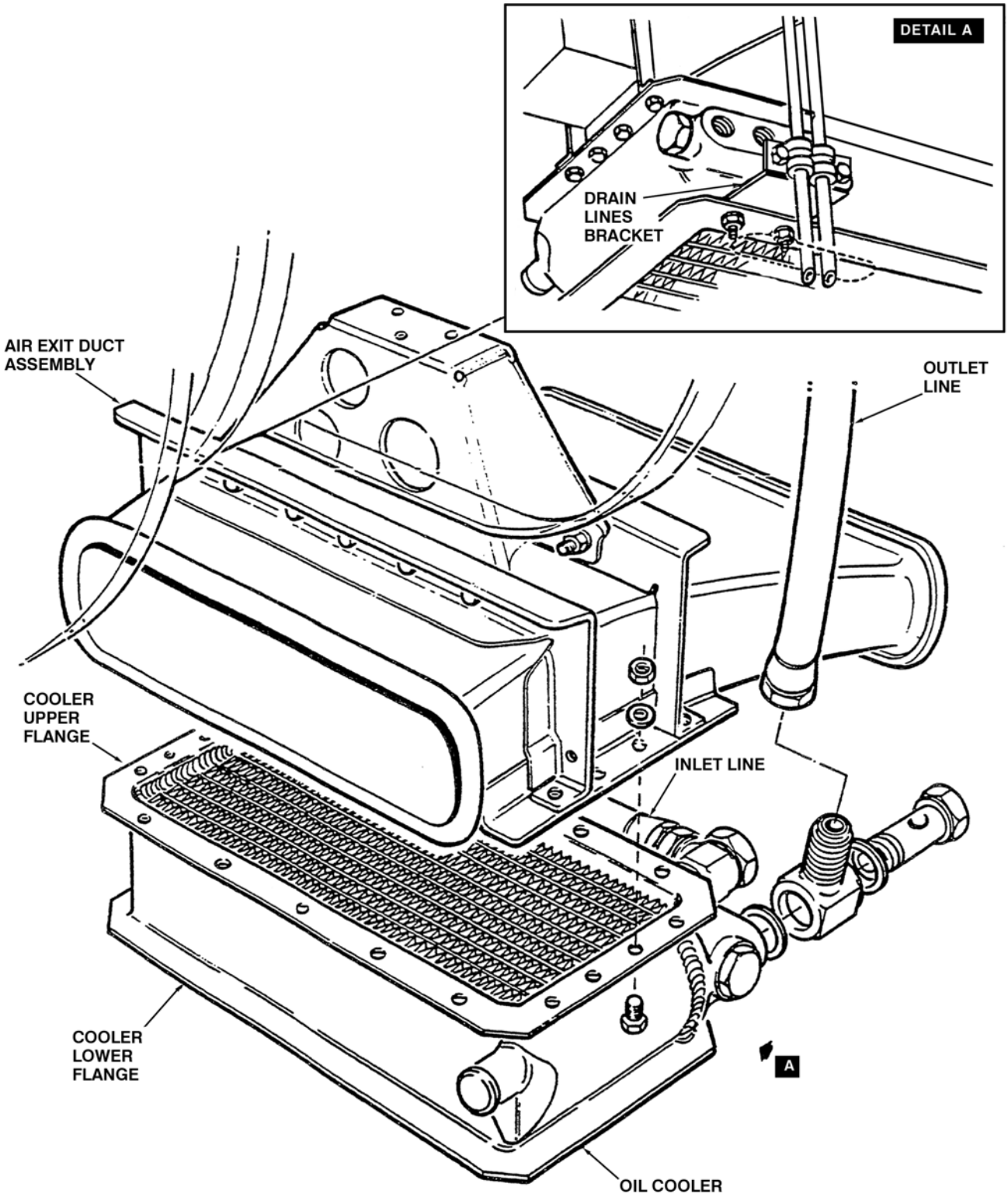
3) Raise engine bottom cowling and ensure drain lines protrude through holes in cowling. Lower cowling and install two bracket retaining screws, washers and nuts.

Note If installing new oil cooler, drill cooler upper and lower flanges to match holes in duct (Para 1.B.1)) and drain lines bracket (Para 1.B.3)).

4) Connect inlet and outlet lines to oil cooler.

5) Fill oil tank. (Refer to 12-10-25, Oil System – General Data).

6) Check for oil leaks on subsequent engine run. Top up oil tank after engine run.



Oil Cooler Installation
Figure 1

2. Adjustment/Test

A. Bench Test Oil Cooler

Equipment required:

- Liquid bath
- Regulated pressure air source
- Hot water supply
- Test setup as shown in Figure 2

1) Plug cooler outlet and attach air hose to inlet.

2) Attach other end of air hose to regulated pressure air source and pressurize cooler to 12 psi.

3) Submerge cooler in water at 140° – 180°F.

4) After five minutes of immersion, increase air pressure to 100 psi. There must be no evidence of leakage.

WARNING INCREASE AIR PRESSURE GRADUALLY TO AVOID BURNS FROM HOT WATER SPLASHING, IF LARGE LEAKS OCCUR.

5) Slowly vent air pressure, remove cooler from bath and disconnect air hose.

6) Install cooler in test setup as shown in Figure 2.

7) Apply room temperature water at 150 psi (max) to cooler and hold.

8) After retaining pressure for 10 minutes, check cooler for blown or bulged plates.

9) Slowly release pressure and remove cooler from test setup.

10) Dry cooler thoroughly with compressed air.

11) Test Mod 6/1481 control valve function as follows:

a) Submerge valve in water at 160 ± 5°F for five minutes.

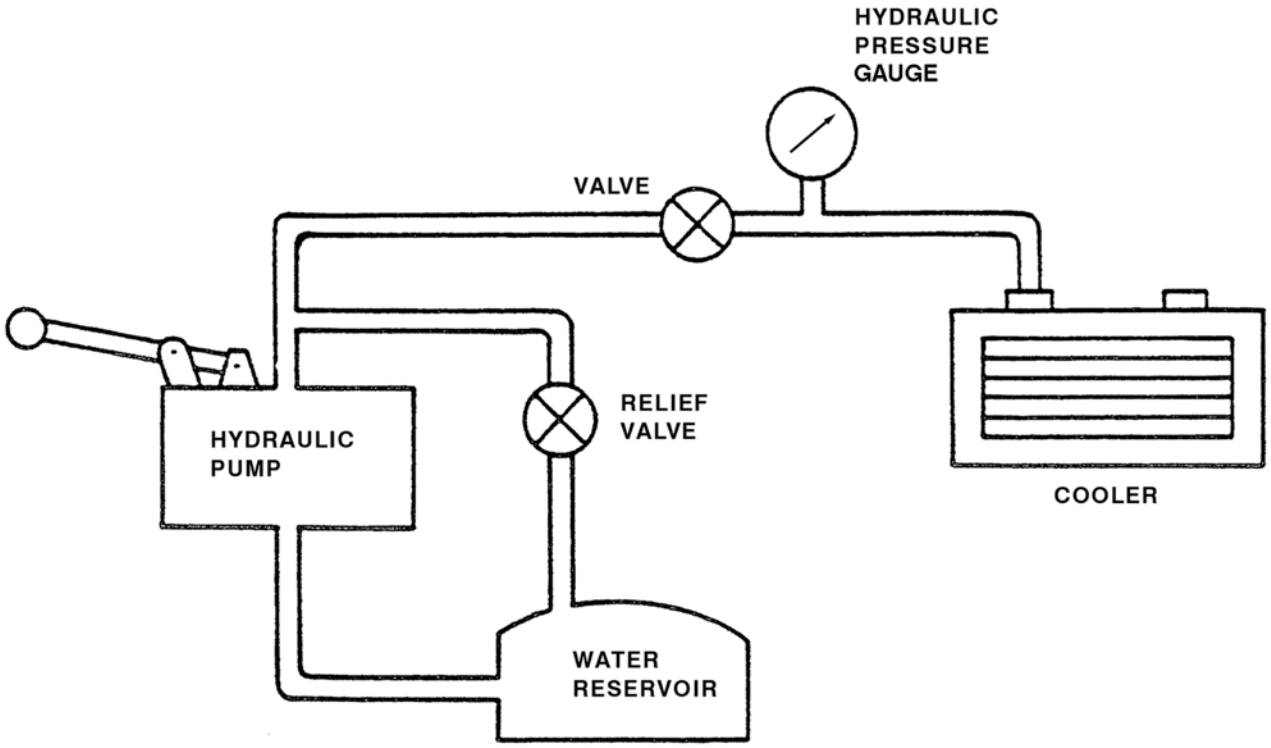
b) Remove valve, measure and record length.

c) Submerge valve in water at 185 ± 5°F for five minutes.

d) Remove valve, measure and record length.

e) Check difference in lengths recorded in Para 2.A.11).b) and Para 2.A.11).d) is not less than 0.170 inch.

12) Dry valve and cooler thoroughly with compressed air.



Hydrostatic Test Setup
Figure 2

Indicating – General Data

1. General

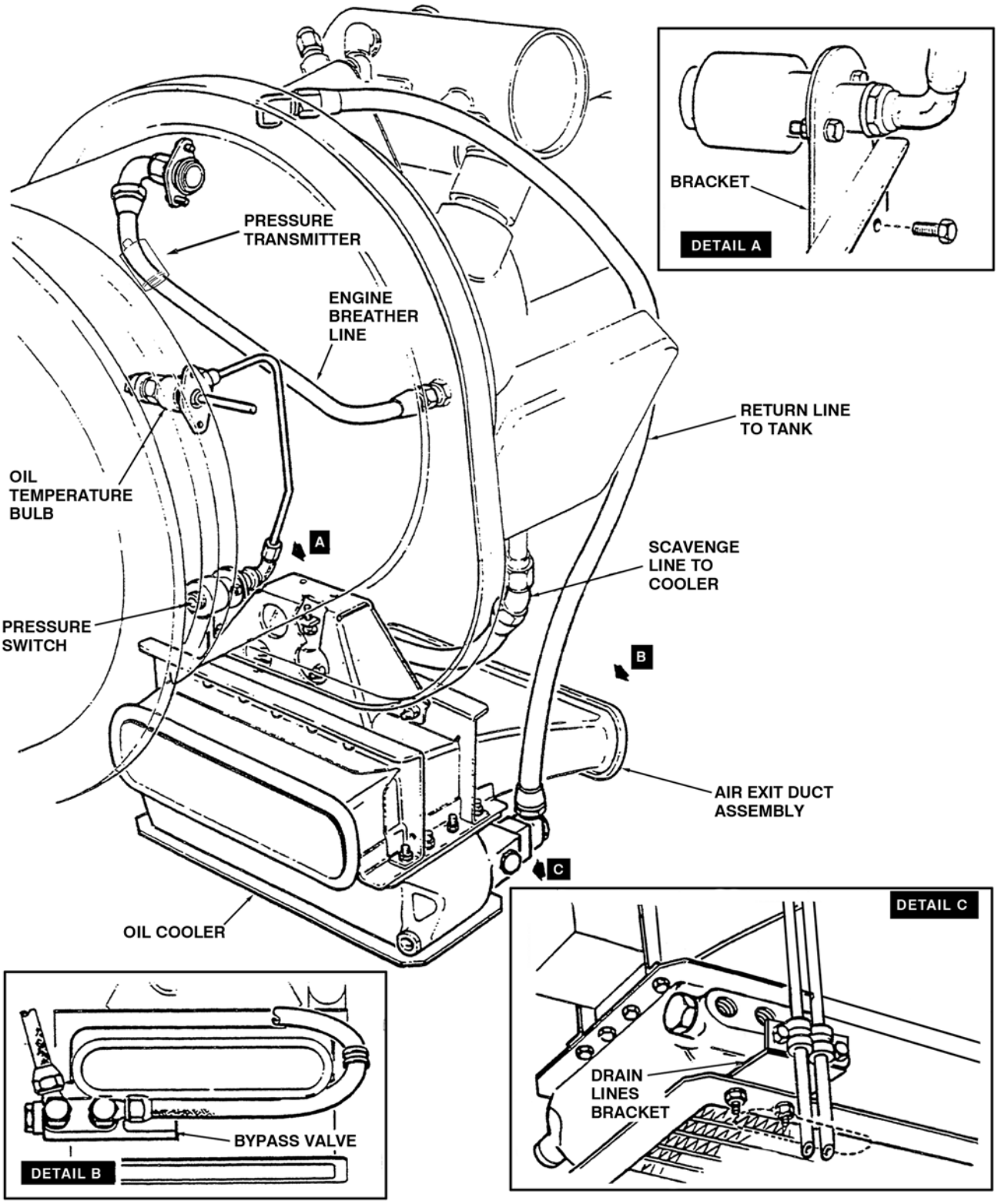
Oil system indicating consists of three systems, a pressure indicating system, a temperature indicating system and a low oil pressure indicating system.

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Oil Pressure Indicating System – General Data

1. General

The oil pressure indicating system for each engine consists of a pressure transducer and a low pressure switch. The oil pressure transducer is threaded into the appropriate engine port, with the wiring connected through the DAU to APEX (refer to Figure 1). The DAU does the required logic to trigger the oil pressure warning and caution messages related to engine oil pressure limits. Excitation power of 28 Volt DC for the oil pressure transducers is provided through the OIL PRESSURE L and OIL PRESSURE R circuit breakers on the main circuit breaker panel (refer to 24-30-00, DC Generation – General Data). Oil pressure is valid over the range of 0 – 200 psi.

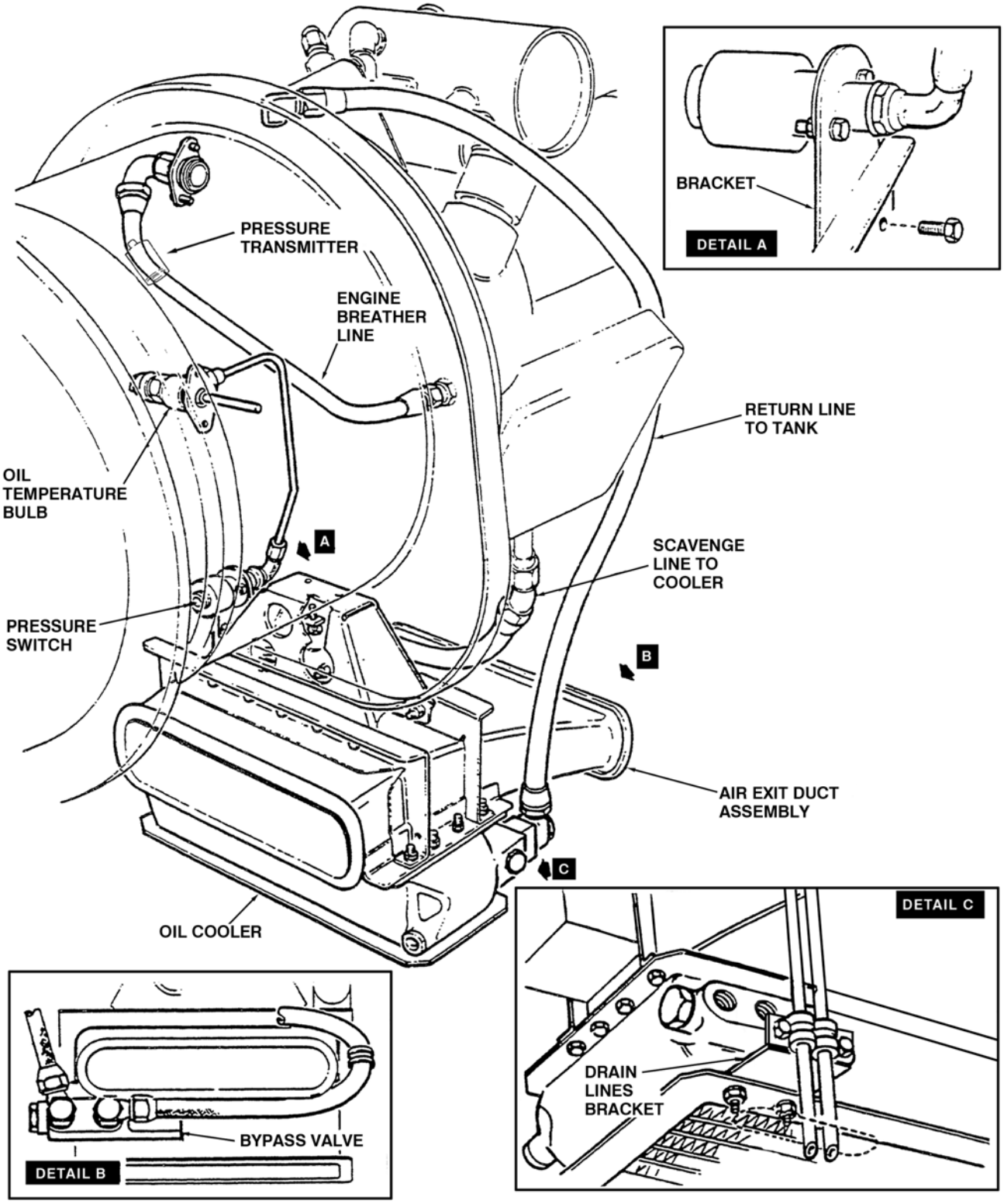


Oil System
Figure 1

Oil Temperature Indicating System – General Data

1. General

The oil temperature indicating system for each engine consists of an oil temperature bulb that provides temperature information via the DAU for display on the CAS.



Oil System
Figure 1

Low Oil Pressure Indicating System – General Data

1. General

The oil pressure switch does a dual function, as it is integral to the optional propeller de-ice system (when aircraft is equipped) and as a redundant low oil pressure sensor. If the engine is running and there is a low pressure condition (oil pressure drops to less than 40 – 42 psi) sensed at the oil pressure switch, then this will trigger a Left/Right Oil Pressure caution CAS message. The switch will de-activate when engine oil pressure rises above 44 – 46 psi.

2. Adjustment/Test

A. Function Test

Equipment required:

- Pressure tester SD5688–3, or
- Pressure tester, Barfield Model 2311F with 0 – 160 psi test gauge, nipple AN816–4 and hose Aeroquip 601000–4–0240 and a digital multimeter.

1) Disconnect oil line to pressure switch.

2) Connect hose from tester to pressure switch.

3) Connect a multimeter to the pressure switch pins and monitor the output impedance from the unit.

4) Using handpump (Tester SD5688) or screw on (Tester Barfield) tester, apply pressure to pressure switch and check that impedance shows high (open circuit) when the applied pressure is between 40 to 45 psi.

5) Slowly decrease pressure to pressure switch by carefully opening bleed valve (Tester SD5688) or by counterclockwise rotation of screw (Tester Barfield) and check that the impedance goes low (closed circuit) when the applied pressure is at 41 ± 2 psi.

6) Disconnect tester and reconnect line to pressure switch.

7) Repeat test for low oil pressure switch on other engine.

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CHAPTER 80

STARTING

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CHAPTER 80 – STARTING

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Starting – General Data

1. General

Engine starting is achieved by a starter/generator in conjunction with a START switch, control relays and the ignition system (refer to 74-00-00, Ignition – General Data). Electrical power for starting is provided by the aircraft battery (main) or, if available, an external power source. DC power from the aircraft battery or external power source, through the reverse current relay, drives the starter/generator which rotates the gas generator turbine up to approximately 12% rpm, when the fuel control lever is operated to permit light up to occur. When the engine has accelerated to a self-sustaining speed, and the START switch is released, the starter/generator may be selected "on-line" for DC generation (refer to 24-30-00, DC Generation – General Data). An automatic fuel purge valve (refer to 71-70-00, Engine Drains – General Data) is provided for each engine. When the relevant engine start circuit is energized, the valve is energized to provide a means of purging trapped air from the appropriate engine fuel control unit; the valve is de-energized when the START switch is released. For engine starting procedures refer to 71-00-00, Power Plant (Mod 6/2012) – General Data.

Aircraft with Mod 6/2315 also include a small 24 volt avionics start battery. The battery is used to supplement electrical power supply to certain avionics systems during an engine start. Further information on the avionics start battery is provided in 24-30-32, Avionics Start Battery (Mod 6/2315) – General Data.

2. Description

The starter system consists of a starter/generator mounted on the accessory case of each engine, a START switch marked LEFT–RIGHT located on the overhead console in the flight compartment and control relays (left and right start relays, and left and right No. 1/No. 2 control relays) located in the main distribution box. Circuit protection is provided by two 5 ampere circuit breakers, labelled START L and START R, on the main circuit breaker panel.

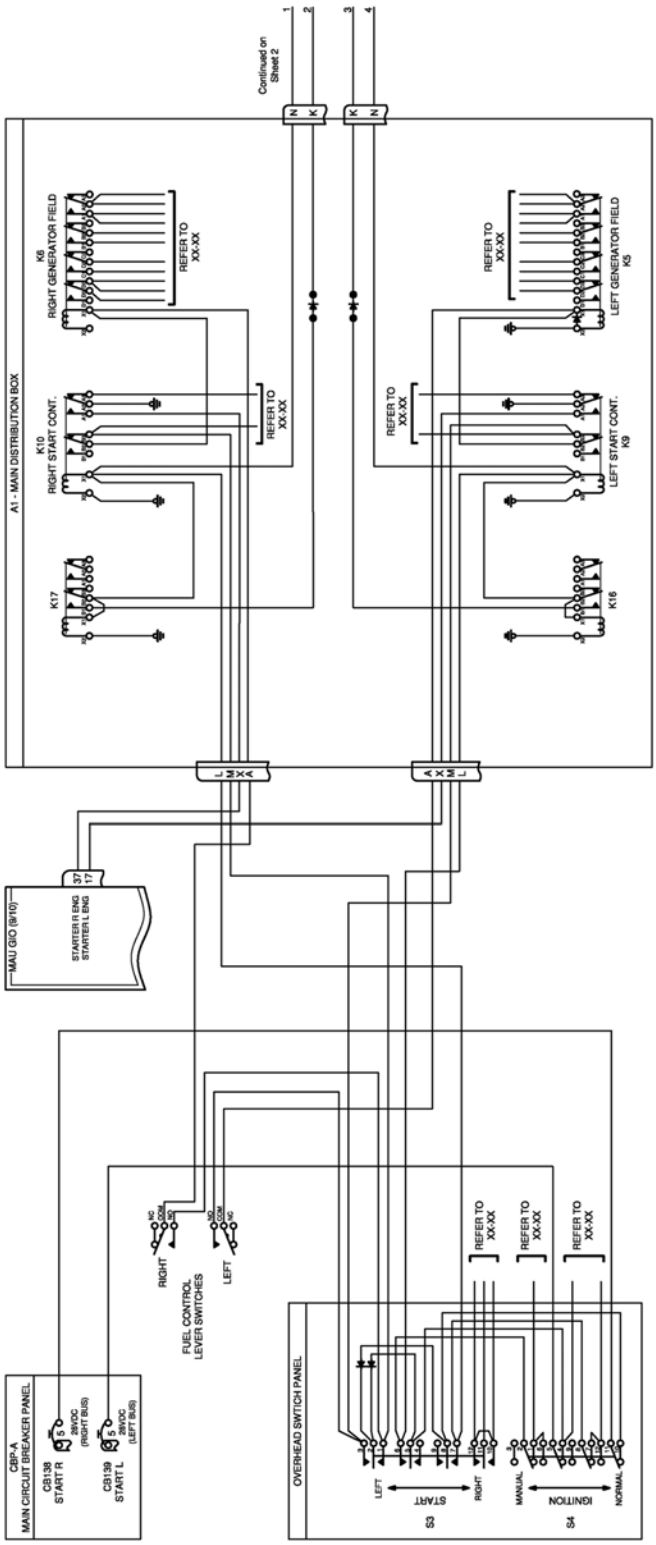
3. Operation

Refer to Figure 1 for details.

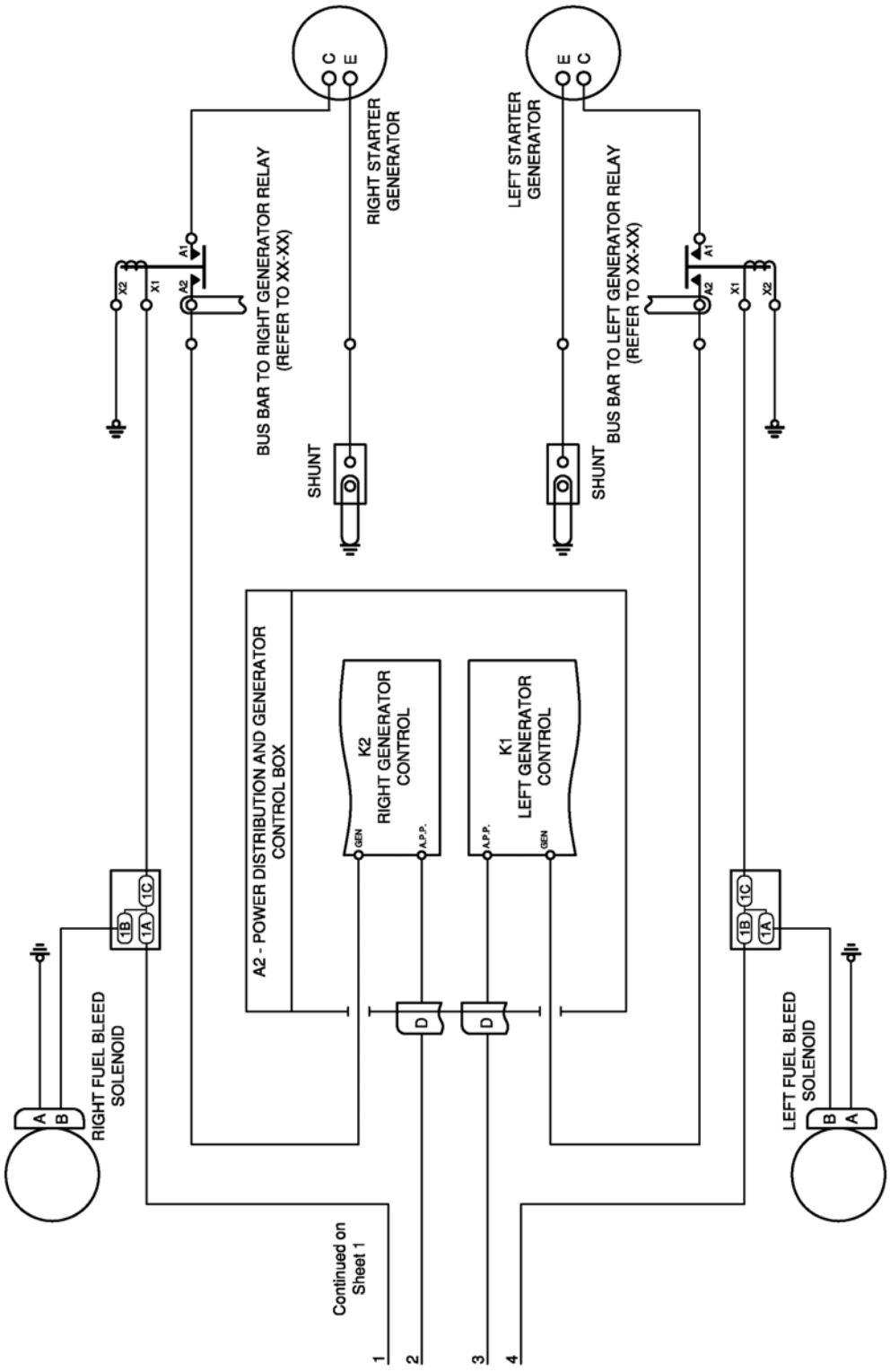
With the MASTER switch set to ON, and the POWER SOURCE switch selected to BATTERY or EXTERNAL (Pre Mod 6/2314) or with EXTERNAL or BATTERY switches ON (Post Mod 6/2314), depending on power supply available, the main battery relays (or external power relays) are energized, connecting battery or external power to the DC buses.

With the IGNITION switch set to NORMAL, and the START switch held to LEFT or RIGHT, DC power from the battery bus is fed through the IGNITION and START switches to energize the start relay, generator field relay (through the fuel control lever limit switch), ignition and start relay, start No. 1 and No. 2 control relays and the fuel purge valve solenoid. With the start No. 2 control relay energized, power is connected to the APP terminal of the reverse current relay, energizing the main relay to connect main battery (or external) power to the starter/generator (through the reverse current relay main contacts and the closed contacts of the start relay), which operates to rotate the gas generator turbine. (The START switch overrides the generator field relays to hold the external power relay on).

When the gas generator turbine rpm stabilizes at approximately 12% and, in accordance with the starting procedure (refer to 71-00-00, Power Plant (Mod 6/2012) – General Data) the fuel control lever is advanced to ON to provide fuel to permit light-up, the fuel control lever limit switch opens. Opening the limit switch de-energizes the generator field relay which changes the starter/generator from a compound to a series motor for faster acceleration.



Engine Starting – Electrical Schematic
Figure 1 (Sheet 1 of 2)



Engine Starting – Electrical Schematic
Figure 1 (Sheet 2 of 2)

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Starter/Generator – General Data

1. General

The starter/generator is mounted on the upper rear face of the engine accessory gear case with a quick-disconnect kit. The kit consists of a drive end pad and a clamp. The pad is secured to the accessory gear case with four studs, washers and nuts, and the clamp is used to attach the starter/generator to the drive end pad. The starter generator is a wet spline drive shaft with an O-ring seal installed on the shaft. The starter/generator is cooled by air drawn from the engine intake area forward of the engine rear fire seal. The cooling air is ducted to the rear of the starter/generator to a duct assembly which is clamped to the starter/generator. The cooling air flow travels forward through the starter/generator, augmented by an integral four-bladed fan, and is exhausted through a screen at the drive end. For further maintenance details refer to the starter-generator Component Maintenance Manual (CMM). Maintenance of the LRU beyond that described below should be done at an accredited maintenance depot.

2. Removal/Installation

Note For unscheduled starter/generator removal, refer to Pratt and Whitney Engine Maintenance Manual, Chapter 72-00-00, Unscheduled Inspection, Starter-Generator Replacement.

Refer to Figure 1 for details.

A. Remove Starter/Generator

1) Ensure electrical power is OFF.

2) Gain access to starter/generator through engine nacelle.

a) Remove clamp securing cooling air duct hose to duct assembly. Disengage cooling air duct and remove air duct.

b) Loosen clamp securing duct assembly to rear of starter/generator fan cover.

c) Loosen clamp securing duct assembly to rear of starter/generator fan cover and remove duct assembly.

d) Remove four screws securing fan cover and carefully remove fan cover.

3) Remove electrical leads from terminal block.

CAUTION STARTER/GENERATOR COOLING FAN BLADES MAY BE DAMAGED OR BENT ON REMOVAL.

4) Support starter/generator and remove retaining clamp from drive end pad.

5) Slide starter/generator rearward to disengage drive. Remove starter/generator.

6) Remove felt pad from starter/generator.

Note The starter/generator drive end pad need only be removed when changing an engine. (Refer to 71-00-00, Power Plant (Mod 6/2012) – General Data).

B. Install Starter/Generator

1) Check starter/generator drive gearshaft splines for wear in accordance with procedures in the CMM.

2) Ensure electrical power is OFF.

3) Position felt pad around rear of starter/generator.

4) Install O-ring seal on starter generator shaft.

CAUTION **STARTER/GENERATOR MAY BE DAMAGED OR BENT ON INSTALL.**

STARTER/GENERATOR DRIVE GEAR CARBON SEAL CAN BE EASILY DAMAGED IF NOT CAREFUL ON INSTALLATION.

5) Carefully position starter/generator drive end to drive end pad, with terminal block at 12 o'clock position. Engage drive.

6) Secure starter/generator to drive end pad with clamp. Torque clamp to 50 pound–inches.

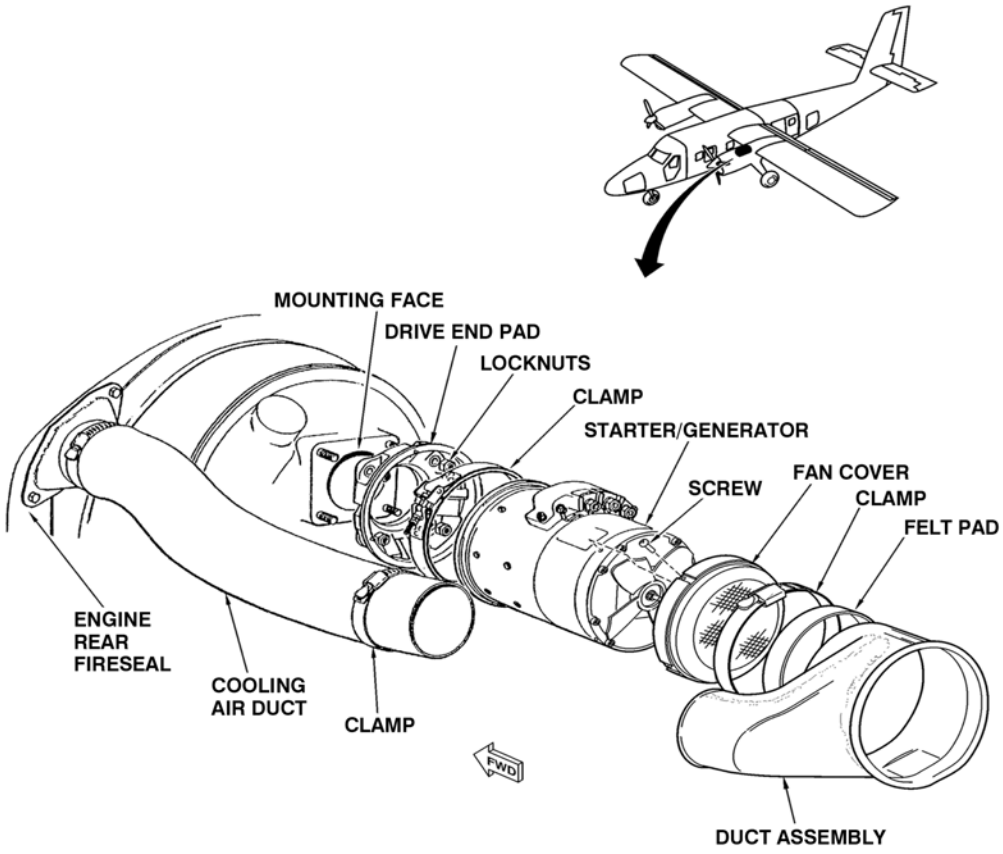
7) Secure electrical leads to terminal block.

8) Install fan cover with four screws, install air duct assembly.

9) Ensure duct assembly is against firewall. Install duct assembly retaining clamp and torque to 25 pound–inches.

10) Install cooling air duct hose to duct assembly and secure with clamp.

11) Function test starter/generator. (Refer to 24-30-00, DC Generation – General Data).



Starter/Generator Removal and Installation
Figure 1

3. Adjustment/Test

   A. Bench Test Starter/Generator

      Bench testing of the LRU should be done in accordance with the CMM and at an accredited maintenance depot.

   B. Adjust Brush Seating and Brush Run-in Starter/Generator

      Refer to Goodrich Corporation Standard Practices Document SPD 1006.

   C. Flashing the Generator Field (Bench Test)

      Do in accordance with the CMM and at an accredited maintenance depot.

4. Inspection/Check

   A. Check Brush Wear Starter/Generator

      1) Gain access to starter/generator.

      2) Inspect brushes and brush holders in accordance with the CMM.

      3) Replace brushes if remaining allowable wear is liable to be exceeded before next inspection.

         Note To facilitate brush seating adjustment and brush run-in (refer to Para 3.B) when brushes are replaced, the starter/generator should be removed (refer to Para 2.A).

   B. Check Brush Spring Tension

      Check brush spring tension in accordance with the CMM.

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CHAPTER 91

CHARTS

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CHAPTER 91 – CHARTS

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

REFERENCE	PAGE	DATE	REFERENCE	PAGE	DATE
91	* 1	Sep. 28/18		* 4	Sep. 28/18
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	* 3	Sep. 28/18		* 6	Sep. 28/18
	* 4	Blank Page		* 7	Sep. 28/18
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91-24-00	* 1	Sep. 28/18		* 16	Sep. 28/18
	* 2	Sep. 28/18	91-32-00	* 1	Sep. 28/18
	* 3	Sep. 28/18		* 2	Sep. 28/18
	* 4	Sep. 28/18	91-76-00	* 1	Sep. 28/18
	* 5	Sep. 28/18		* 2	Sep. 28/18
	* 6	Sep. 28/18		* 3	Sep. 28/18
91-27-00	* 1	Sep. 28/18		* 4	Sep. 28/18
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CHAPTER 91 – CHARTS

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Charts – General Data

1. General

This chapter contains miscellaneous charts associated with the various systems. As an aid to quick reference, the various sections are arranged in chapter order. For example, the second element of the three element number is (in the case of FLIGHT CONTROLS) 27, therefore, pages relating to charts associated with flight controls, will bear the three element number 91–27–00.

A. Cable Charts

The left and right end terminations for all cables listed in the various sections of this chapter, are shown on Figure 1 and Figure 2, in order to show the measuring points, when referring to the cable length column in the particular cable chart. Thus, if the length column states A7 to B1 69.120 inches, the length between the left-hand termination A7 shown on Figure 1, and right-hand termination B1 shown on Figure 2, will be 69.120 inches. Where a third or center terminal is fitted to a cable, a third reference is given. For example $A7_L$ = 15.400 inches shown in length column, indicates that dimension $A7_L$ on Figure 1, is 15.400 inches. Comprehensive data for each cable is given on the cable charts; termination numbers on Figure 1 and Figure 2, are for reference only.

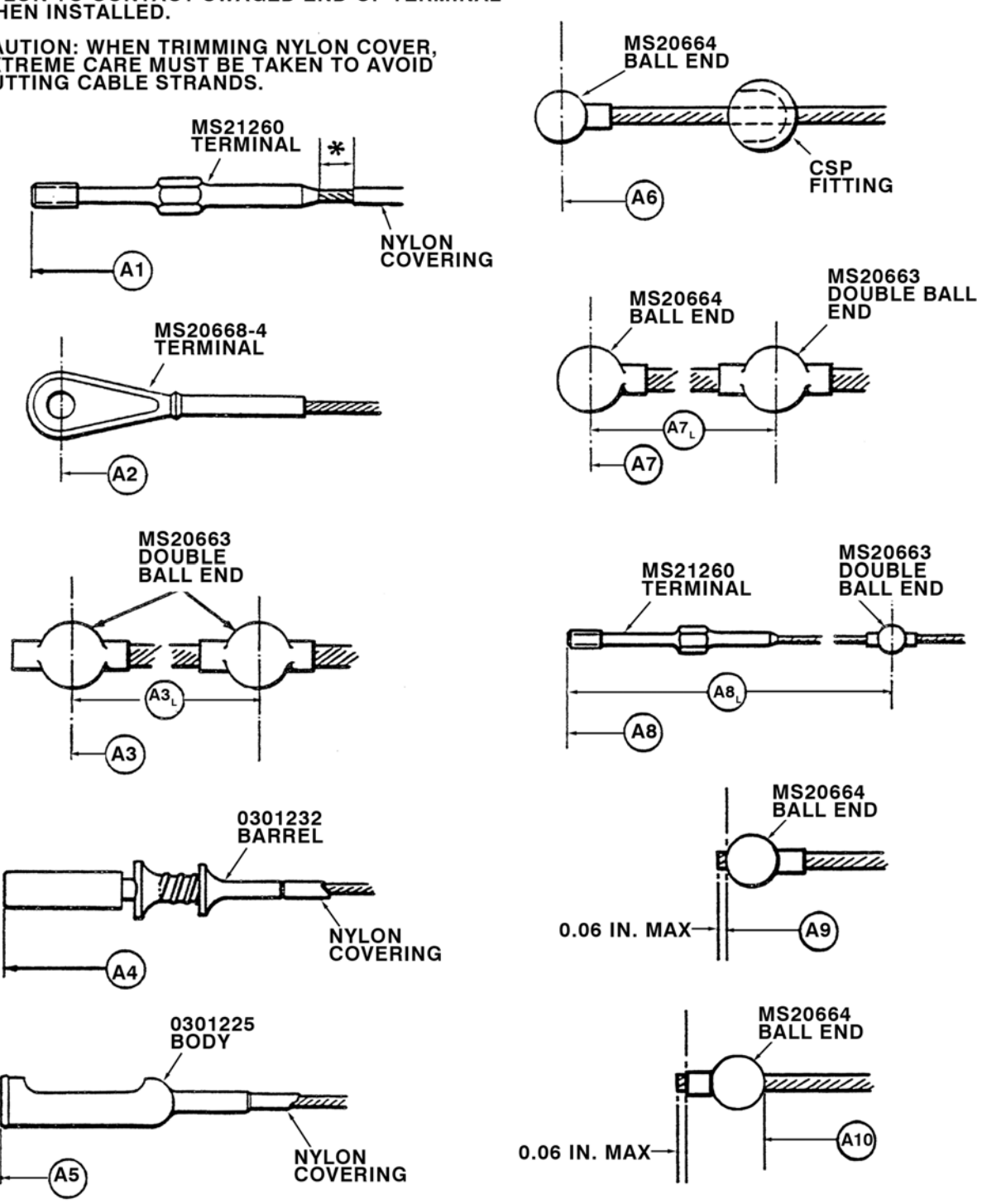
If cables are nylon covered, it is essential that the nylon be trimmed back before installation of the terminal. When the nylon must be trimmed back clear of a terminal shoulder, the appropriate dimension is indicated in the relevant cable chart termination column by an asterisk (*) followed by the dimension in inches. This is also shown on Figure 1 and Figure 2. Where no dimension is given, trim nylon so that after installation, nylon covering butts terminal shoulder.

B. Electrical Load Charts

The electrical load charts provide a description and quantity of each unit of electrical equipment in the aircraft, as well as the normal load drawn by each unit.

*WHEN CABLES ARE NYLON COVERED, TRIM NYLON BACK TO MEASUREMENT GIVEN IN LEFT TERMINATION COLUMN ON APPROPRIATE CABLE CHART WHERE NO MEASUREMENT IS GIVEN, TRIM NYLON TO CONTACT SWAGED END OF TERMINAL WHEN INSTALLED.

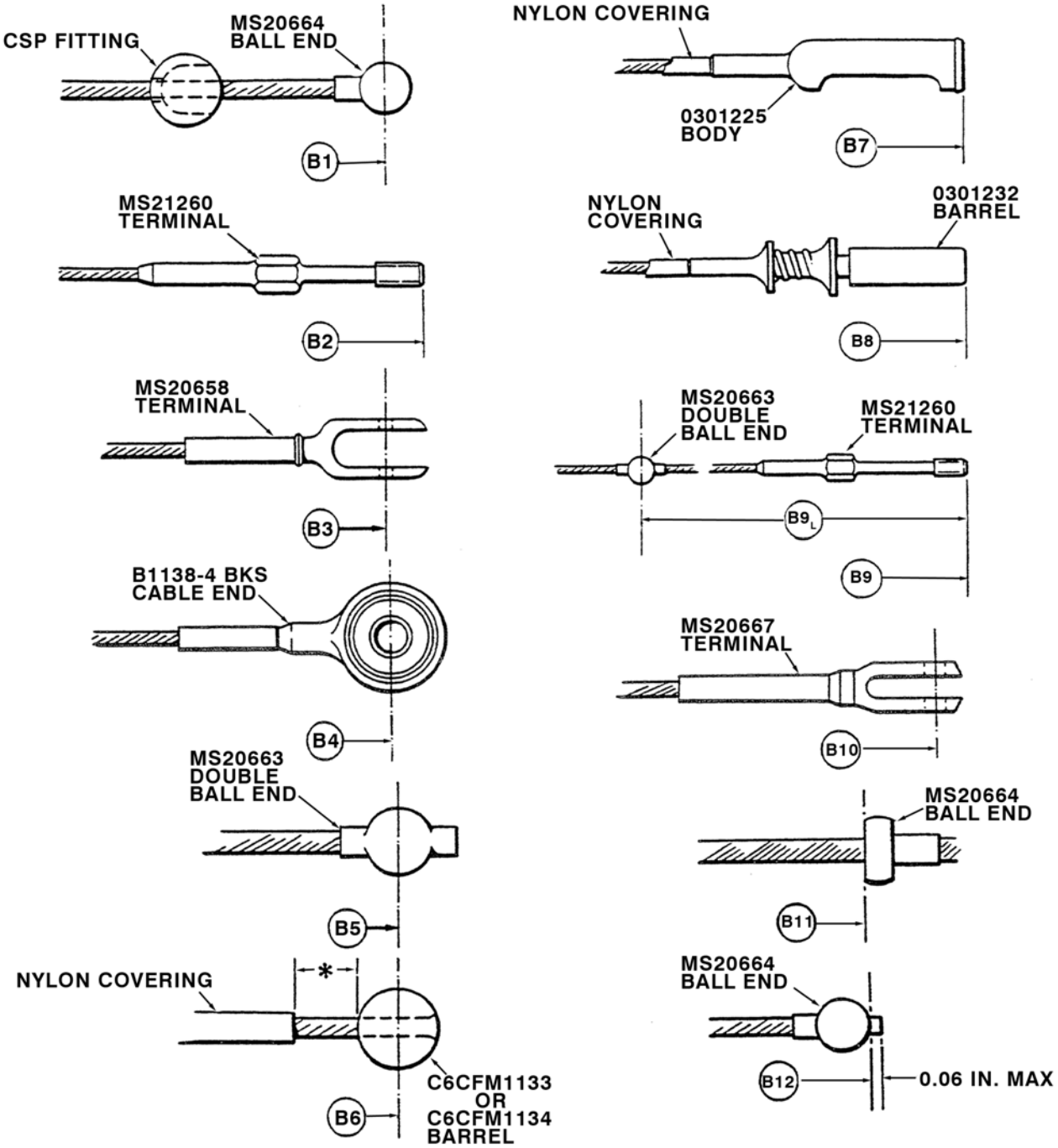
CAUTION: WHEN TRIMMING NYLON COVER, EXTREME CARE MUST BE TAKEN TO AVOID CUTTING CABLE STRANDS.



Left-hand Cable Terminations
Figure 1

***WHEN CABLES ARE NYLON COVERED, TRIM NYLON BACK TO MEASUREMENT GIVEN IN RIGHT TERMINATION COLUMN ON APPROPRIATE CABLE CHART. WHERE NO MEASUREMENT IS GIVEN, TRIM NYLON TO CONTACT SWAGED END OF TERMINAL WHEN INSTALLED.**

CAUTION: WHEN TRIMMING NYLON COVER; EXTREME CARE MUST BE TAKEN TO AVOID CUTTING CABLE STRANDS.



Right-hand Cable Terminations
Figure 2

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Auto Flight Charts (S.O.O. 6221) – General Data

1. General

The charts given in this section relate to the various components associated with auto flight.

A. Cable Charts – Honeywell Autopilot (S.O.O. 6221)

Table 1 — Autopilot Servo Actuator Control – Cable Chart gives the Part No. end terminations, cable size, length, material and cable proof load figures for the autopilot servo actuator cables. When using the chart, refer to 91-00-00, Charts – General Data, Cable Charts.

Table 1 Autopilot Servo Actuator Control – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6R1806–7 Aileron servo actuator cable.	MS20664C3 (Swaged)	MS20664C3 (Swaged)	3/32 inch 7 X 7	A9 to B11 20.50 ± 0.120	Carbon Steel MIL–W–83420 TY1 COB	550
C6R1806–9 Elevator servo actuator cable.	MS20664C3 (Swaged)	MS20664C2 (Swaged)	3/32 inch 7 X 7	A9 to B11 50.50 ± 0.120	Carbon Steel MIL–W–83420 TY1 COB	550
C6R1806–11 Rudder servo actuator cable.	MS20664C3 (Swaged)	MS20664C3 (Swaged)	3/32 inch 7 X 7	A9 to B11 49.75 ± 0.120	Carbon Steel MIL–W–83420 TY1 COB	550
C6R1807–9 Aileron servo actuator cable. (1) Rudder servo actuator cable. (2)	MS21260–53LH	MS20664C3 (Swaged)	3/32 inch 7 X 7	A1 to B11 4.80 ± 0.120	Carbon Steel MIL–W–83420 TY1 COB	550
C6R1807–11 Aileron servo actuator cable.	MS21260–53RH	MS20664C3 (Swaged)	3/32 inch 7 X 7	A1 to B12 26.50 ± 0.120	Carbon Steel MIL–W–83420 TY1 COB	550
C6R1807–13 Elevator servo actuator cable.	MS21260–53RH	MS20664C3 (Swaged)	3/32 inch 7 X 7	A1 to B12 34.50 ± 0.120	Carbon Steel MIL–W–83420 TY1 COB	550
C6R1807–15 Rudder servo actuator cable.	MS21260–53RH	MS20664C3 (Swaged)	3/32 inch 7 X 7	A1 to B12 34.75 ± 0.120	Carbon Steel MIL–W–83420 TY1 COB	550
C6R2042–1 Pitch Trim bridle cable.	MS20664C2 (Swaged) with sleeve VS100–S3 and sleeve spacer C6R2042–29	MS20664C2 (Swaged)	1/16 inch 7 X 7	A10 to B11 57.50 ± 0.030	Carbon Steel MIL–W–83420 TY1 COB	290
C6R2043–1 Pitch Trim bridle adjust cable.	MS20664C3 (Swaged)	MS20664C3 (Swaged) with crosspiece VS100–C3 and nut VS 100–N3	3/32 inch 7 X 19	A10 to B11 2.50 ± 0.030	Carbon Steel MIL–W–83420 TY1 COB	550

Electrical Load Charts – General Data

1. General

The charts given in this section relate to loading of a typical aircraft electrical system.

Table 1 28 Volt DC Load Chart – Left Bus

CB #	CB Grid	Load Equipment Left Bus	CB Rating (Amps)	Rated Current (Amps)
105	**A-03**	GEN FAIL R	5.00	0.20
107	**A-04**	CABIN HEAT	5.00	0.45
109	**A-05**	WINDSHIELD WIPERS	10.00	4.80
111	**A-06**	PITOT HT LEFT AND STALL SENSOR	7.50	7.30
113	**A-07**	PY (FUEL CONT SENSOR) HEATER LEFT	5.00	3.90
115	**A-08**	FUEL PUMP FWD #1	15.00	6.50
117	**A-09**	FUEL PUMP AFT #1	15.00	6.50
119	**A-10**	BOOST PUMP PRESSURE 1	5.00	0.04
121	**A-11**	TORQUE PRESS LEFT	5.00	0.01
123	**A-12**	OIL PRESS IND LEFT	5.00	0.01
125	**A-13**	WING FUEL CONTROLS L	7.50	5.80
127	**A-14**	FLIGHT CONTROLS INDICATION	5.00	0.05
	A-15	RESERVED		
131	**A-16**	MAP LIGHTS & PASS WARN SIGN LTS	5.00	0.22
133	**A-17**	CABIN LTS BRIGHT	5.00	0.93
	A-18	RESERVED		
137	**C-01**	WINDSHIELD HEAT L	5.00	0.36
1	**NA**	WINDSHIELD HEAT L	30.00	26.00
141	**C-03**	ENGINE IGNITORS LEFT	5.00	1.20
143	**C-04**	VALVE BLEED AIR LEFT	5.00	0.76
145	**C-05**	PROPELLER DEICE	20.00	18.00
147	**C-06**	HYD OIL PUMP (INCLUDES K11 RELAY)	35.00	30.00
149	**C-07**	PROPELLER AUTOFEATHER	5.00	0.10
151	**C-08**	FUEL SOV LEFT	5.00	
153	**C-09**	FUEL QTY IND AFT/LEFT	5.00	0.03
155	**C-10**	FIRE EXT BOTTLE	5.00	
157	**C-11**	FIRE DETECTION LEFT	5.00	0.01
	C-12	RESERVED		
161	**C-13**	FIRE DET FAULT IND L	5.00	0.01

Table 1 28 Volt DC Load Chart – Left Bus (continued)

CB #	CB Grid	Load Equipment Left Bus	CB Rating (Amps)	Rated Current (Amps)
	C-14	RESERVED		
165	**C-15**	BEACON LT (PRE MOD 6/2240)	5.00	0.19
165	**C-15**	BEACON LT (POST MOD 6/2240)	5.00	0.67
167	**C-16**	LANDING LT LEFT (PRE MOD 6/2285)	7.50	2.20
167	**C-16**	LANDING LT LEFT (POST MOD 6/2285)	5.00	1.50
169	**C-17**	WING POS LTS (INCL TAIL POS LT)	5.00	0.52
171	**C-18**	AIRFRAME DEICE AUTO	5.00	3.20
201	**E-01**	MAU POWER SUPPLY MODULE 1A	20.00	1.30
221	**E-03**	AUDIO/MARKER PANEL PILOT	5.00	1.53
241	**E-05**	DATA ACQUISITION UNIT 1A	5.00	0.30
261	**E-07**	AVIONICS FAN	1.00	0.22
203	**F-01**	MAU POWER SUPPLY MODULE 2A	20.00	1.30
223	**F-03**	MULTI-MODE DIGITAL RADIO 1	7.50	4.40
243	**F-05**	DATA ACQUISITION UNIT 2A	5.00	0.40
	F-07	RESERVED (VIDEO)		
205	**G-01**	AURAL ALERT 1	1.00	0.01
225	**G-03**	MODE S APEX TRANSPONDER 1	5.00	1.64
245	**G-05**	ENHANCED GROUND PROXIMITY COMPUTER	3.00	0.28
	G-07	RESERVED (SATELLITE TRACKING)		
207	**H-01**	PILOT PFD, FLAT PANEL DISPLAY, 10.4” DIA	5.00	0.50
227	**H-03**	DME 1	3.00	0.55
247	**H-05**	TCAS I	5.00	1.65
247	**H-05**	TCAS II	5.00	1.02
	H-06	RESERVED		
267	**H-07**	GEAR INDICATION	5.00	0.04
209	**J-01**	UPPER MFD, FLAT PANEL DISPLAY, 10.4” DIA	5.00	0.50
229	**J-03**	GPS SENSOR UNIT 1	3.00	0.25
249	**J-05**	RADAR ALTIMETER 1	3.00	0.50
	J-07	RESERVED (GEAR ADVISORY)		
211	**K-01**	ADAHRS 1A	3.00	0.36
231	**K-03**	AR-120 CVR SYSTEM	5.00	0.28
251	**K-05**	C406–N ELT NAV INTERFACE	1.00	0.06
	K-07	RESERVED		

Table 1 28 Volt DC Load Chart – Left Bus (continued)

CB #	CB Grid	Load Equipment Left Bus	CB Rating (Amps)	Rated Current (Amps)
213	**M-01**	FLIGHT CONTROLLER 1A	3.00	0.09
233	**M-03**	AR-256 FDR SYSTEM (PRE MOD 6/2332)	5.00	0.11
233	**M-03**	FA2100 FDR SYSTEM (512) (POST MOD 6/2332)	5.00	0.34
253	**M-05**	NAT 245 CABIN PA AMPLIFIER SYSTEM	2.50	0.60
273	**M-07**	AVIONICS STRT BATT – CHARGE (POST MOD 6/2315)	30.00	2.00
215	**N-01**	PITCH TRIM (S.O.O. 6221)	5.00	1.10
235	**N-03**	CAUTION LIGHTS NORMAL	2.00	0.26
255	**N-05**	ESIS	15.00	5.47
275	**N-07**	AVIONICS STRT BATT – CTRL (POST MOD 6/2315)	2.00	0.35
NA	**NA**	GENERATOR FIELD RELAY		0.35
NA	**NA**	GENERATOR RELAY		0.40
NA	**NA**	BATTERY RELAY		0.40
NA	**NA**	GENERATOR OVERVOLT RELAY		0.16
NA	**NA**	GENERATOR REVERSE CURRENT RELAY		0.37
NA	**NA**	MAIN BATT CHARGE		48.00
217	**P-01**	AP SERVOS (S.O.O. 6221)	7.50	2.00
216	**P-02**	RESERVED (CCD CONTROL)		
237	**P-03**	WX RADAR	5.00	2.04
257	**P-05**	ESIS INSTRUMENT	3.00	0.00
277	**P-07**	PANEL LIGHTS L	5.00	0.69
219	**Q-01**	DISPLAY CONTROLLER PILOT	3.00	0.13
239	**Q-03**	MFD CONTROLLER	3.00	0.20
259	**Q-05**	ESIS MAGNETOMETER	1.00	0.00
279	**Q-07**	FLOOD LIGHTS	5.00	0.50
21	**R-01**	BAGGAGE COMP LT FWD	5.00	0.11
21	**R-1**	BAGGAGE COMP LT REAR		0.22
21	**R-01**	ENTRANCE LT		0.11
26	**R-02**	POWER AMP (HF COMMUNICATION SYSTEM)	30.00	13.00
	R-03	RESERVED		
22	**S-01**	DOME LIGHT AND CLOCK	5.00	0.24
25	**S-02**	RX/EX ANT (HF COMMUNICATION SYSTEM)	5.00	4.00
	S-03	RESERVED		
23	**T-01**	BATTERY VOLTIMETER	1.00	0.01

Table 1 28 Volt DC Load Chart – Left Bus (continued)

CB #	CB Grid	Load Equipment Left Bus	CB Rating (Amps)	Rated Current (Amps)
27	**T-02**	EMERGENCY LIGHTS BATTERY	10.00	2.00
	T-03	RESERVED		
24	**U-01**	ELT SWITCH	1.00	0.00
28	**U-02**	DAU COOLING FAN	2.00	0.40
	U-03	RESERVED		

Table 2 28 Volt DC Load Chart – Right Bus

CB #	CB Grid	Load Equipment Right Bus	CB Rating (Amps)	Rated Current (Amps)
	B-01	RESERVED		
104	**B-03**	GEN FAIL L	5.00	0.20
106	**B-04**	VENT FAN	20.00	18.00
108	**B-05**	INTAKE DEFLECTORS	10.00	5.30
110	**B-06**	PITOT HT RIGHT	7.50	5.30
112	**B-07**	PY (FUEL CONT SENSOR) HEATER RIGHT	5.00	3.92
114	**B-08**	FUEL PUMP FWD #2	15.00	6.05
116	**B-09**	FUEL PUMP AFT #2	15.00	6.24
118	**B-10**	BOOST PUMP PRESSURE 2	5.00	0.04
120	**B-11**	TORQUE PRESS IND RIGHT	5.00	0.01
122	**B-12**	OIL PRESS IND RIGHT	5.00	0.01
124	**B-13**	WING FUEL CONTROLS R	7.50	5.80
	B-14	RESERVED		
128	**B-15**	READING LIGHTS	5.00	1.10
130	**B-16**	TAXI LIGHT	5.00	1.50
132	**B-17**	DOORS UNLOCKED	5.00	0.16
134	**B-18**	ACTUATOR AILERON TRIM	5.00	0.10
136	**D-01**	WINDSHIELD HEAT R	5.00	0.36
2	**NA**	WINDSHIELD HEAT R	30.00	26.00
18	**NA**	STN 332 CABIN 28V DC PLUG (S.O.O. 6283)	5.00	5.00
18	**NA**	STN 332 CABIN 14V DC PLUG (S.O.O. 6272)	5.00	5.00
140	**D-03**	ENGINE IGNITOR RIGHT	5.00	1.26
142	**D-04**	VALVE BLEED AIR RIGHT	5.00	1.00
144	**D-05**	PROPELLER OVERSPEED GOVERNOR	5.00	0.76

Table 2 28 Volt DC Load Chart – Right Bus (continued)

CB #	CB Grid	Load Equipment Right Bus	CB Rating (Amps)	Rated Current (Amps)
146	**D-06**	HYDRAULIC PRESSURE INDICATOR	5.00	0.02
148	**D-07**	FUEL CROSS-FEED SOV	5.00	2.01
150	**D-08**	FUEL SOV RIGHT	5.00	0.00
152	**D-09**	FUEL QTY IND FWD/RIGHT	5.00	0.03
154	**D-10**	FIRE EXT BOTTLE	5.00	0.00
156	**D-11**	FIRE DETECTION RIGHT	5.00	0.01
	D-12	RESERVED		
158	**D-13**	FIRE DET FAULT IND R	5.00	0.00
	D-14	RESERVED		
164	**D-15**	STROBE LIGHTS	5.00	2.30
166	**D-16**	LANDING LIGHT RIGHT (PRE MOD 6/2285)	7.50	2.20
166	**D-16**	LANDING LIGHT RIGHT (POST MOD 6/2285)	5.00	1.25
168	**D-17**	WING INSPECTION LIGHT	5.00	0.44
170	**D-18**	AIRFRAME DEICE MANUAL	5.00	0.00
200	**E-02**	MAU POWER SUPPLY MODULE 1B	20.00	1.60
220	**E-04**	AUDIO/MARKER PANEL COPILOT	5.00	0.33
240	**E-06**	DATA ACQUISITION UNIT 1B	5.00	0.29
202	**F-02**	MAU POWER SUPPLY 2B	20.00	1.60
222	**F-04**	MULTI-MODE DIGITAL RADIO 2	7.50	4.4
242	**F-06**	DATA ACQUISITION UNIT 2B	5.00	0.4
204	**G-02**	AURAL ALERT 2	1.00	0.01
224	**G-04**	MODE S APEX TRANSPONDER 2	5.00	1.64
244	**G-06**	HEADSETS	2.00	0.05
	H-06	RESERVED		
206	**H-02**	COPILOT PFD, FLAT PANEL DISPLAY, 10.4” DIA	5.00	0.50
226	**H-04**	DME 2	3.00	0.55
208	**J-02**	LOWER MFD, FLAT PANEL DISPLAY, 10.4” DIA	5.00	0.50
228	**J-04**	GPS SENSOR UNIT 2	3.00	0.25
248	**J-06**	RADAR ALTIMETER 2	3.00	2.0
210	**K-02**	ADAHRS 1B	3.00	0.36
230	**K-04**	SATELLITE WEATHER	5.00	0.22
250	**K-06**	RESERVED (AUTO BRIEF)		
212	**M-02**	FLIGHT CONTROLLER 1B	3.00	0.1

Table 2 28 Volt DC Load Chart – Right Bus (continued)

CB #	CB Grid	Load Equipment Right Bus	CB Rating (Amps)	Rated Current (Amps)
232	**M-04**	RESERVED (STORMSCOPE OPTION)	3.00	0.22
252	**M-06**	PWR CONV 28V INPUT	5.00	0.04
214	**N-02**	CAUTION LTS BYPASS	2.00	
234	**N-04**	SAT COM (S.O.O. 6298)	3.00	0.50 CONT. 0.80 MAX
236	**P-04**	HF COMMUNICATION SYSTEM CTRL	2.00	0.12
	P-06	RESERVED		
218	**Q-02**	DISPLAY CONTROLLER COPILOT	3.00	0.13
238	**Q-04**	ECTM/FOQA (S.O.O. 6288)	3.00	0.15
258	**Q-06**	VHF-FM (S.O.O. 6296)	3.00	0.70

TEMPORARY REVISION

PSM: 1-64-2 **AIRCRAFT: DHC-6** **SERIES: 400**

PUBLICATION: MAINTENANCE MANUAL **TEMP. REVISION NO.: TR91-5**

Insert in 91-27-00 before Page 1

FLIGHT CONTROL CHARTS

REASON FOR ISSUE:

To introduce Mod 6/2483, Flight Control Cables – Improved Manufacturing Control which removes Section 91-27-00, Flight Control Charts from PSM 1-64-2.

ACTION:

In Chapter 91, ignore existing Section 91-27-00 and replace with Page 2 of this Temporary Revision.

-Continued overleaf-

RECORDING INSTRUCTIONS:

Record this TEMPORARY REVISION NO. and the relevant information in the appropriate columns of the TEMPORARY REVISION INDEX provided at the front of the manual.

Flight Control Charts – General Data

1. Deleted.

NOTE: Please see the TEMPORARY REVISION that revises this page.

Flight Control Charts – General Data

1. General

The charts given in this section relate to the various components associated with flight controls.

A. Cable Charts

Table 1 — Aileron Control – Cable Chart to Table 7 — Flap/Elevator Interconnect Trim – Cable Chart, give the P/N, end terminations, cable size, length, material and cable proof load figures for all flight control cables. Each table contains the charts for a particular sub-system. When using these charts, refer to 91-00-00, Charts – General Data, Cable Charts.

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 1 Aileron Control – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
NAS304–37–1470 Control column to station 150.0, left-hand side.	– – – – – – – –	– – – – – – – –	– – – –	– – – –	– – – – – – –	– – – – –
NAS304R37–1470 Control column to station 150.0, left-hand side.	– – – – – – – –	– – – – – – – –	– – – –	– – – –	– – – – – – –	– – – – –
C6CF1124–1 Aileron quadrant, lower groove, left-hand forward.	MS21260S4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–2	1/8 inch 7 X 19	A1 to B1 19.700 ± 0.120	Carbon Steel MIL–W–1511	1200
C6CF1452–1 Aileron quadrant, lower groove, left-hand forward. (Mod 6/1435).	MS21260S4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–12	1/8 inch 7 X 19	A1 to B1 19.700 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–16 Aileron quadrant, lower groove, left-hand forward. (Mod 6/1435)	MS21260S4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–12	1/8 inch 7 X 19	A1 to B1 19.700 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1124–3 Aileron quadrant, upper groove, right-hand forward.	MS21260S4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–2	1/8 inch 7 X 19	A1 to B1 19.875 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–13 Aileron quadrant, upper groove, right-hand forward.	MS21260S4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–2	1/8 inch 7 X 19	A1 to B1 19.875 ± 0.120	Stainless Steel MIL–W–83420	1055

Note If aileron circuit is changed in part or in whole, ailerons must be balanced per the procedure used for balancing aircraft fitted with wing fuel tanks (S.O.O. 6095).

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 1 Aileron Control – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1125–1 Aileron quadrant, lower groove, left-hand rear.	MS21260S4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 20.125 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–14 Aileron quadrant, lower groove, left-hand rear.	MS21260S4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 20.125 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1125–3 Aileron quadrant, upper groove, right-hand rear.	MS21260S4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 20.125 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–15 Aileron quadrant, upper groove, right-hand rear.	MS21260S4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 20.125 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1152–1 Control column to station 150.0, right-hand side.	C6CF1146–29 made from MS20668–4 hole enlarged to 0.257 dia with letter F drill (Swaged)	MS21260S4 RH (Swaged)	1/8 inch 7 X 19	A2 to B2 147.00 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–10 Control column to station 150.0, right-hand side.	C6CF1146–29 made from MS20668–4 hole enlarged to 0.257 dia with letter F drill (Swaged)	MS21260S4 RH (Swaged)	1/8 inch 7 X 19	A2 to B2 147.00 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1153–3 Station 150.0 to aileron quadrant, right-hand side.	MS21260S4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 59.250 ± 0.120	Carbon Steel MIL–W–1511	1200

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 1 Aileron Control – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
EO69053–11 Station 150.0 to aileron quadrant, right-hand side.	MS21260S4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 59.250 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1155–3 Station 150.0 to aileron quadrant, left-hand side.	MS21260S4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–2	1/8 inch 7 X 19	A1 to B1 59.250 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–12 Station 150.0 to aileron quadrant, left-hand side.	MS21260S4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–2	1/8 inch 7 X 19	A1 to B1 59.250 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CW1031–1 Left-hand wing forward, and right-hand wing rear.	MS20664C4 (Swaged) with fitting CSP–87–1	MS21260S4 LH (Swaged)	1/8 inch 7 X 19	A6 to B2 255.280 ± 0.200	Carbon Steel MIL–W–1511	1200
EO69053–17 Left-hand wing forward, and right-hand wing rear.	MS20664C4 (Swaged) with fitting CSP–87–1	MS21260S4 LH (Swaged)	1/8 inch 7 X 19	A6 to B2 255.280 ± 0.200	Stainless Steel MIL–W–83420	1055
C6CW1032–1 Left-hand wing rear, and right-hand wing forward.	MS20664C4 (Swaged) with fitting CSP–87–2	MS21260S4 RH (Swaged)	1/8 inch 7 X 19	A6 to B2 255.540 ± 0.200	Carbon Steel MIL–W–1511	1200
EO69053–18 Left-hand wing rear, and right-hand wing forward.	MS20664C4 (Swaged) with fitting CSP–87–2	MS21260S4 RH (Swaged)	1/8 inch 7 X 19	A6 to B2 255.540 ± 0.200	Stainless Steel MIL–W–83420	1055

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 2 Rudder Control – Cable Chart

CABLE PART NO. LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
NAS 305–34–4201 Rudder quadrant inboard groove to station 455.0.	– – – – – – –	– – – – – – –	– – – –	– – – –	– – – – – – –	– – – – –
NAS 305R34–4201 Rudder quadrant inboard groove to station 455.0.	– – – – – – –	– – – – – – –	– – – –	– – – –	– – – – – – –	– – – – –
NAS 305–35–4266 Rudder quadrant inboard groove to station 455.0.	– – – – – – –	– – – – – – –	– – – –	– – – –	– – – – – – –	– – – – –
NAS 305R35–4266 Rudder quadrant inboard groove to station 455.0.	– – – – – – –	– – – – – – –	– – – –	– – – –	– – – – – – –	– – – – –
C6CF1150–1 Station 455.0 to right-hand side rudder control attachment.	MS21260L5 LH (Swaged)	MS20658–5 (Swaged)	5/32 inch 7 X 19	A1 to B3 83.600 ± 0.120	Carbon Steel MIL–W–1511	1680
EO69053–6 Station 455.0 to right-hand side rudder control attachment.	MS21260L5 LH (Swaged)	MS20658–5 (Swaged)	5/32 inch 7 X 19	A1 to B3 83.600 ± 0.120	Stainless Steel MIL–W–83420	1440
C6CF1151–1 Station 455.0 to left-hand side rudder control attachment.	MS21260L5 RH (Swaged)	B1138–4BKS (Airite Products) with bearing DSC69–1 (Swaged)	5/32 inch 7 X 19	A1 to B4 83.600 ± 0.120	Carbon Steel MIL–W–1511	1680
EO69053–7 Station 455.0 to left-hand side rudder control attachment.	MS21260L5 RH (Swaged)	B1138–4BKS (Airite Products) with bearing DSC69–1 (Swaged)	5/32 inch 7 X 19	A1 to B4 83.600 ± 0.120	Stainless Steel MIL–W–83420	1440

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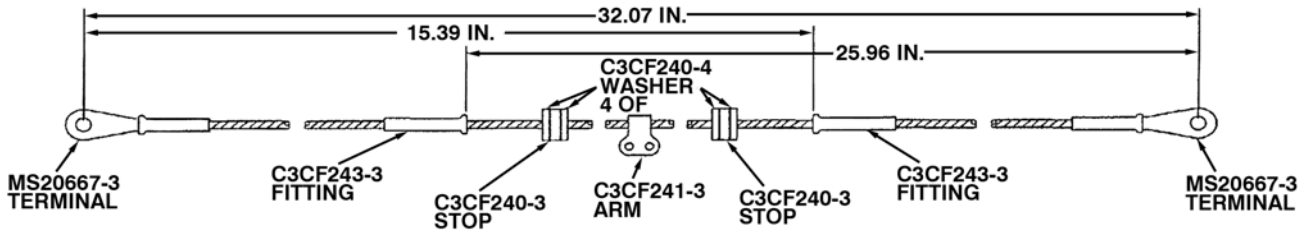
Table 3 Elevator Control – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1146–1 Front of elevator lever to stations 426.75.	C6CF1146–29 made from MS20668–4 hole enlarged to 0.257 in. dia with letter F drill. (Swaged)	MS21260L4 RH (Swaged)	1/8 inch 7 X 19	A2 to B2 395.500 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–1 Front of elevator lever to stations 426.75.	C6CF1146–29 made from MS20668–4 hole enlarged to 0.257 in. dia with letter F drill. (Swaged)	MS21260L4 RH (Swaged)	1/8 inch 7 X 19	A2 to B2 395.500 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1147–1 Rear of elevator lever to stations 426.75.	C6CF1146–29 made from MS20668–4 hole enlarged to 0.257 in. dia with letter F drill. (Swaged)	MS21260L4 LH (Swaged)	1/8 inch 7 X 19	A2 to B2 366.850 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–2 Rear of elevator lever to stations 426.75.	C6CF1146–29 made from MS20668–4 hole enlarged to 0.257 in. dia with letter F drill. (Swaged)	MS21260L4 LH (Swaged)	1/8 inch 7 X 19	A2 to B2 366.850 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1148–1 Upper cable, station 426.75 to elevator quadrant.	MS21260L4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 108.600 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–3 Upper cable, station 426.75 to elevator quadrant.	MS21260L4 LH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–1	1/8 inch 7 X 19	A1 to B1 108.600 ± 0.120	Stainless Steel MIL–W–83420	1055

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Table 3 Elevator Control – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1149–1 Upper cable, station 426.75 to elevator quadrant.	MS21260L4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–2	1/8 inch 7 X 19	A1 to B1 111.800 ± 0.120	Carbon Steel MIL–W–1511	1200
EO69053–4 Upper cable, station 426.75 to elevator quadrant.	MS21260L4 RH (Swaged)	MS20664C4 (Swaged) with fitting CSP–87–2	1/8 inch 7 X 19	A1 to B1 111.800 ± 0.120	Stainless Steel MIL–W–83420	1055
C6CF1100–11 Elevator lever stop cable.	Refer to Detail A	Refer to Detail A	3/32 inch 7 X 19	Refer to Detail A	Carbon Steel MIL–W–1511	550
EO69053–5 Elevator lever stop cable.	Refer to Detail A	Refer to Detail A	3/32 inch 7 X 19	Refer to Detail A	Stainless Steel MIL–W–83420	550



DETAIL A

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 4 Flap Control – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
NAS312–17–0480 Flap selector valve lower cable.	– – – – – – –	– – – – – – –	– – – –	– – – –	– – – – – –	– – – – –
NAS312R17–0480 Flap selector valve lower cable.	– – – – – – –	– – – – – – –	– – – –	– – – –	– – – – – –	– – – – –
C6CF1038–1 Flap indicator cable, forward (Pre Mods 6/1254 and 6/1404).	MS21260S2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–A11	1/16 inch 7 X 7	A1 to B1 106.900 ± 0.120	Carbon Steel MIL–W–1511	216
C6CF1039–1 Flap indicator cable, rear.	MS21260S2 LH (Swaged)	MS20663C2 (Swaged)	1/16 inch 7 X 7	A1 to B5 57.250 ± 0.120	Carbon Steel MIL–W–1511	288
C6CF1040–1 Flap selector lever to station 162.0.	MS21260S2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 100.875 ± 0.120	Carbon Steel MIL–W–1511	288
EO69053–19 Flap selector lever to station 162.0.	MS21260S2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 100.875 ± 0.120	Stainless Steel MIL–W–83420	288
C6CF1042–1 Flap selector lever to station 162.0.	MS21260S2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 66.250 ± 0.120	Carbon Steel MIL–W–1511	288
EO69053–20 Flap selector lever to station 162.0.	MS21260S2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 66.250 ± 0.120	Stainless Steel MIL–W–83420	288
C6CF1122–1 Flap selector valve follow up cable. (Pre Mod 6/1877).	MS20663C2 2 OFF (Swaged)	MS20664C2 (Swaged)	1/16 inch 7 X 7	A3 to B1 38.860 ± 0.120 A3L= 14.470 ± 0.030	Carbon Steel MIL–W–1511	288

NOTE: Please see the TEMPORARY REVISION that revises this page.

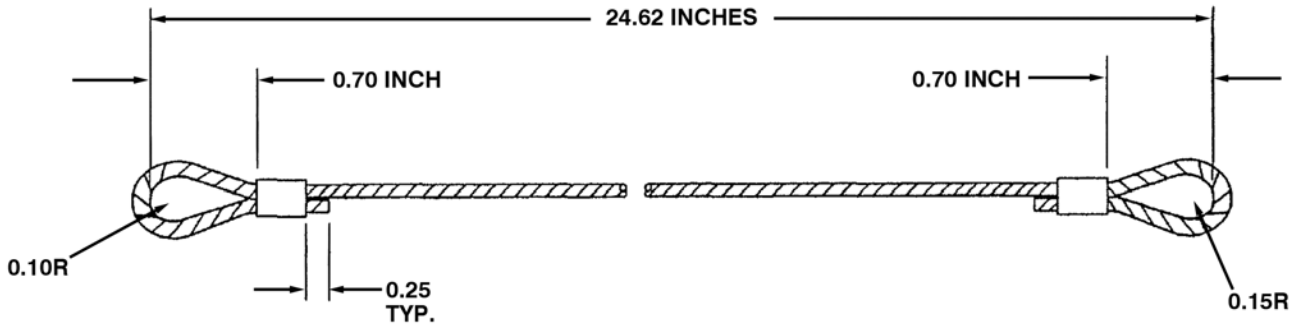
Table 4 Flap Control – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
EO69053–22 Flap selector valve follow up cable.	MS20663C2 2 OFF (Swaged)	MS20664C2 (Swaged)	1/16 inch 7 X 7	A3 to B1 38.860 ± 0.120 A3L= 14.470 ± 0.030	Stainless Steel MIL–W–83420	288
C6CF1122–3 Flap selector valve follow up cable. (Post Mod 6/1877).	MS20663C2 2 OFF (Swaged)	MS20664C2 (Swaged)	1/16 inch 7 X 19	A3 to B1 38.860 ± 0.120 A3L= 14.470 ± 0.030	Carbon Steel MIL–W–83420 TYPE 1 COMP A	288
EO69053–27 Flap selector valve follow up cable.	MS20663C2 2 OFF (Swaged)	MS20664C2 (Swaged)	1/16 inch 7 X 19	A3 to B1 38.860 ± 0.120 A3L= 14.470 ± 0.030	Stainless Steel MIL–DTL–83420	288
C6CF1123–3 Flap selector valve upper cable.	MS21260S2 RH (Swaged)	MS20663C2 (Swaged)	1/16 inch 7 X 7	A1 to B5 66.000 ± 0.120	Carbon Steel MIL–W–1511	288
EO69053–21 Flap selector valve upper cable.	MS21260S2 RH (Swaged)	MS20663C2 (Swaged)	1/16 inch 7 X 7	A1 to B5 66.000 ± 0.120	Stainless Steel MIL–W–83420	288
C6CF1451–11 Flap indicator cord, forward cable assembly. (Mod 6/1404).	DSC51–4 parallel connector	DSC51–4 parallel connector		Refer to Detail A	Double braided Dacron cord. D.H.M.S. P1 – 10 Heat seal ends to prevent fraying Pre-stretch to remove kinks.	30

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 4 Flap Control – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1451–13 Flap indicator cable, forward cable assembly. (Mod 6/1404).	MS21260S2 RH (Swaged)	MS20667–2 (Swaged	1/16 inch 7 X 7	A8 to B10 81.98	Carbon Steel MIL–W–1511	288



DETAIL A

Table 5 Rudder Trim – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CT1023–1 From cable C6CF1165–1 quick disconnect, to right and aft of rudder trim tab screw jack drum. (Pre Mod 6/1219).	0301232 Barrel (Quick disconnect) D7–1001–2–1 A Pacific Sc. Co. (Swaged)	C6CFM 1133–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A4 to B6 48.350 ± 0.120	Corr Res Steel MIL–C–5424 Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CT1024–1 From cable C6CF1167–1 quick disconnect, to left and forward of rudder trim tab screw jack drum. (Pre Mod 6/1219).	0301232 Body (Quick disconnect) D7–1001–2–1 A Pacific Sc. Co. (Swaged)	C6CFM 1133–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A5 to B6 48.350 ± 0.120	Corr Res Steel MIL–C–5424 Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CT1026–1 From station 406.0 to right and aft of rudder trim tab screw jack drum. (Mod 6/1219).	MS21260L2 LH (Swaged) *0.5 inch	C6CFM 1133–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A1 to B6 251.100 ± 0.120	Corr Res Steel MIL–C–5424 Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CT1027–1 From station 465.0 to left and forward of rudder trim tab screw jack drum. (Mod 6/1219).	MS21260L2 RH (Swaged) *0.5 inch	C6CFM 1133–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A1 to B6 192.000 ± 0.120	Corr Res Steel MIL–C–5424 Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CF1064–3 From right side of rudder trim handwheel drum to station 406.0.	MS21260L2 RH (Swaged)	C6CFM 1133–27 (Silver induction Brazed)	1/16 inch 7 X 7	A1 to B6 362.250 ± 0.200	Corr Res Steel MIL–C–5424	288

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 5 Rudder Trim – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
EO69053–23 From right side of rudder trim handwheel drum to station 406.0.	MS21260L2 LH (Swaged)	C6CFM 1134–27 (Silver induction Brazed)	1/16 inch 7 X 7	A1 to B6 362.250 ± 0.200	Stainless Steel MIL–C–83420	288
C6CF1065–3 From left side of rudder trim handwheel drum to station 465.0. (Pre Mod 6/1219).	MS21260L2 LH (Swaged)	C6CFM 1134–27 (Silver induction Brazed)	1/16 inch 7 X 7	A1 to B6 424.880 ± 0.200	Corr Res Steel MIL–C–5424	288
C6CF1065–5 From left side of rudder trim handwheel drum to station 465.0. (Mod 6/1219).	MS21260L2 LH (Swaged)	C6CFM 1134–27 (Silver induction Brazed)	1/16 inch 7 X 7	A1 to B6 420.880 ± 0.200	Corr Res Steel MIL–C–5424	288
EO69053–24 From left side of rudder trim handwheel drum to station 465.0. (Mod 6/1219).	MS21260L2 LH (Swaged)	C6CFM 1134–27 (Silver induction Brazed)	1/16 inch 7 X 7	A1 to B6 420.880 ± 0.200	Stainless Steel MIL–C–83420	288
C6CF1165–1 From station 406.0 to quick disconnect cable C6CT1023–1. (Pre Mod 6/1219).	MS21260L2 LH (Swaged)	0301232 Body D7–1001–2–1A Pacific Sc. Co. (Swaged)	1/16 inch 7 X 7	A1 to B7 201–250 ± 0.120	Carbon Steel MIL–W–1511	288
C6CF1167–1 From station 465.0 to quick disconnect cable C6CT1024–1. (Pre Mod 6/1219).	MS21260L2 RH (Swaged)	0301232 Barrel D7–1001–2–1A Pacific Sc. Co. (Swaged)	1/16 inch 7 X 7	A1 to B8 143.750 ± 0.120	Carbon Steel MIL–W–1511	288

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 6 Elevator Trim – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1066–3 From upper side of elevator trim tab handwheel drum to station 465.0.	MS21260L2 LH (Swaged)	C6CFM 1133–27 (Silver induction Brazed)	1/16 INCH 7 X 7	A1 to B6 439.370 ± 0.200	Corr Res Steel MIL–C–5424	288
EO69053–25 From upper side of elevator trim tab handwheel drum to station 465.0.	MS21260L2 LH (Swaged)	C6CFM 1133–27 (Silver induction Brazed)	1/16 INCH 7 X 7	A1 to B6 439.370 ± 0.200	Stainless Steel MIL–W–83420	288
C6CF1067–1 From lower side of elevator trim tab handwheel drum to station 406.0. (Mod 6/1219).	MS21260L2 RH (Swaged)	C6CFM 1134–27 (Silver induction Brazed)	1/16 INCH 7 X 7	A1 to B6 374.870 ± 0.200	Corr Res Steel MIL–C–5424	288
EO69053–26 From lower side of elevator trim tab handwheel drum to station 406.0. (Mod 6/1219).	MS21260L2 RH (Swaged)	C6CFM 1134–27 (Silver induction Brazed)	1/16 INCH 7 X 7	A1 to B6 374.870 ± 0.200	Stainless Steel MIL–W–83420	288
C6CF1067–3 From lower side of elevator trim tab handwheel drum to station 406.0. (Pre Mod 6/1219).	MS21260L2 RH (Swaged)	C6CFM 1134–27 (Silver induction Brazed)	1/16 INCH 7 X 7	A1 to B6 371.070 ± 0.200	Corr Res Steel MIL–C–5424	288
C6CF1404–1 From station 406.0 to forward end of elevator trim tab screw jack drum. (Pre Mod 6/1219).	MS21260L2 LH (Swaged) *0.3 inch	C6CFM 1134–27 (Silver induction Brazed) *0.5 inch	1/16 INCH 7 X 19	A1 to B6 205.100 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 6 Elevator Trim – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1404–3 From station 406.0 to forward end of elevator trim tab screw jack drum. (Mod 6/1219).	MS21260L2 LH (Swaged) *0.3 inch	C6CFM 1134–27 (Silver induction Brazed) *0.5 inch	1/16 INCH 7 X 19	A1 to B6 206.630 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CF1405–1 From station 465.0 to aft end of elevator trim tab screw jack drum. (Pre Mod 6/1219).	MS21260L2 LH (Swaged) *0.5 inch	C6CFM 1133–27 (Silver induction Brazed) *0.5 inch	1/16 INCH 7 X 19	A1 to B6 146.350 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CF1405–3 From station 465.0 to aft end of elevator trim tab screw jack drum. (Mod 6/1219).	MS21260L2 LH (Swaged) *0.5 inch	C6CFM 1133–27 (Silver induction Brazed) *0.5 inch	1/16 INCH 7 X 19	A1 to B6 141.600 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 7 Flap/Elevator Interconnect Trim – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1161–1 From station 443.0 to quick disconnect cable C6CF1401–1. (Pre Mod 6/1219).	MS21260L2 RH (Swaged)	0301225 Body D7–1001–2–1a Pacific Sc. Co. (Swaged)	1/16 inch 7 X 7	A1 to B7 73.120 ± 0.120	Carbon Steel (MIL–W–1511	288
C6CF1163–1 From station 430.0 to quick disconnect cable C6CF1402–1. (Pre Mod 6/1219)	MS21260L2 LH (Swaged)	0301232 Barrel D7–1001–2–1a Pacific Sc. Co. (Swaged)	1/16 inch 7 X 7	A1 to B8 119.750 ± 0.120	Carbon Steel (MIL–W–1511	288
C6CF1400–1 From top inboard end of ball screw jack drum to station 443.0.	MS21260L2 LH (Swaged)	C6CFM1207–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A1 to B6 249.150 ± 0.200	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CF1401–1 From station 443.0 to forward end of interconnect trim tab jack drum. (Pre Mod 6/1219).	0301232 Barrel D7–1001–2–1 A Pacific Sc. Co. (Swaged)	C6CFM1134–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A1 to B6 73.680 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CF1402–1 From station 430.0 to aft end of interconnected trim tab jack drum. (Pre Mod 6/1219).	0301225 Body D7–1001–2–1 A Pacific Sc. Co. (Swaged)	C6CFM1133–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A5 to B6 80.400 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288

NOTE: Please see the TEMPORARY REVISION that revises this page.

Table 7 Flap/Elevator Interconnect Trim – Cable Chart (continued)

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CF1403–1 From lower outboard end of ball screw jack drum to station 430.0.	MS21260L2 RH (Swaged)	C6CFM1133–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A1 to B6 203.750 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CF1412–1 From station 443.0 to forward end of interconnected trim tab screw jack drum. (Mod 6/1219).	MS21260L2 RH (Swaged) *0.5 inch	C6CFM1134–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A1 to B6 149.500 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288
C6CF1413–1 From station 430.0 to aft end of interconnected trim tab screw jack drum. (Mod 6/1219).	MS21260L2 LH (Swaged) *0.5 inch	C6CFM1133–27 (Silver induction Brazed) *0.5 inch	1/16 inch 7 X 19	A1 to B6 196.900 ± 0.120	Corr Res Steel MIL–C–5424 (Except for construction) Nylon covered to 3/32 in. dia (Hackensack Cable Corp.)	288

Landing Gear Charts – General Data

1. General

The charts given in this section relate to the various components associated with landing gear.

A. Cable Charts

Table 1 — Nose Wheel Steering – Cable Chart gives the P/N, end terminations, cable size, length, material and cable proof load figures for all cables associated with the landing gear. When using the chart, refer to 91-00-00, Charts - General – General Data, Cable Charts.

Table 1 Nose Wheel Steering – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
NAS312–17–0180 Right cable to steering valve lever drum.	– – – – – – –	– – – – – – –	– – – –	– – – –	– – – – – –	– – –
C6CF1242–3 Nose wheel steering cable in control column.	MS21260S2 LH (Swaged)	MS21260S2 RH (Swaged)	1/16 inch 7 X 7	A1 to B9 127.440 ± 0.120 $B9_L$ = 70.250 ± 0.120	Carbon Steel MIL–W–1511	288
	CENTER TERMINAL MS20663C2 (Swaged)					
C6CF1251–1 Left cable to steering valve lever drum.	MS20663C2 (Swaged)	MS21260S2 RH (Swaged)	1/16 inch 7 X 7	A3 to B2 28.400 ± 0.120	Carbon Steel MIL–W–1511	288

Engine Control Charts – General Data

1. General

The charts given in this section relate to the various components associated with engine controls.

A. Cable Charts

Table 1 — Power Control (Throttle) – Cable Chart to Table 4 — Control Console – Cable Chart, give the P/N, end terminations, cable size, length, material and cable proof load figures for all power control cables. When using the charts, refer to 91-00-00, Charts – General Data, Cable Charts.

Table 1 Power Control (Throttle) – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CE1243–3 Throttle, left wing. To rear of engine control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 58.300 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1244–3 Throttle, left wing. To front of engine control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 68.200 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1249–3 Throttle, right wing. To rear of engine control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 50.400 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1250–3 Throttle, right wing. To front of engine control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 53.200 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1255–1 Throttle, left-hand lever to wing. Lower cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 162.000 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1256–1 Throttle, left-hand lever to wing. Upper cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 152.870 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1261–1 Throttle, right-hand lever to wing. Lower cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 199.120 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1262–1 Throttle, right-hand lever to wing. Upper cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 183.000 ± 0.120	Carbon Steel MIL–W–1511	288

Table 2 Power Control (Propeller Pitch) – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CE1245–3 Propeller pitch, left wing. To rear of propeller control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 34.300 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1246–3 Propeller pitch, left wing. To front of propeller control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 24.600 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1251–3 Propeller pitch, right wing. To rear of propeller control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 82.100 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1252–3 Propeller pitch, right wing. To front of propeller control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 76.100 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1257–1 Propeller pitch, left-hand lever to wing. Upper cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 196.500 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1258–1 Propeller pitch, left-hand lever to wing. Lower cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 192.000 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1263 Propeller pitch, right-hand lever to wing. Lower cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 166.500 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1264 Propeller pitch, right-hand lever to wing. Upper cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 158.000 ± 0.120	Carbon Steel MIL–W–1511	288

Table 3 Power Control, Fuel Shut-off Valve – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CE1253–1 Fuel shut-off, left-hand lever to wing. Upper cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 193.500 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1254–1 Fuel shut-off, left-hand lever to wing. Lower cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 200.000 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1259–1 Fuel shut-off, right-hand lever to wing. Lower cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 162.250 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1260–1 Fuel shut-off, right-hand lever to wing. Upper cable, fuselage.	MS21260L2 LH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 152.750 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1336–1 Fuel shut-off, left wing. To front of control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 34.300 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1336–3 Fuel shut-off, left wing. To rear of control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 42.750 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1336–5 Fuel shut-off, right wing. To front of control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 75.000 ± 0.120	Carbon Steel MIL–W–1511	288
C6CE1336–7 Fuel shut-off, right wing. To rear of control pulley.	MS21260L2 RH (Swaged)	MS20664C2 (Swaged) with fitting CSP–87–11	1/16 inch 7 X 7	A1 to B1 83.000 ± 0.120	Carbon Steel MIL–W–1511	288

Table 4 Control Console – Cable Chart

CABLE PART NO. & LOCATION	END TERMINATIONS		CABLE SIZE	LENGTH INCHES	MATERIAL & SPEC	PROOF LOAD LBS
	LEFT	RIGHT				
C6CE1315–1 Beta release cable (overhead control console)	C6CEM1068–27 eye end (Swaged) Ensure end of cable is visible through inspection holes.	Soft solder last 0.20 to 0.25 in. of cable to prevent fraying.	1/16 inch 7 X 7	6.00 from attachment hole in eye end, to soldered end of cable.	Corrosion resistant steel (non-magnetic) MIL–C–18375	216

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APPENDIX A

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APPENDIX A

LIST OF EFFECTIVE PAGES

This list shows the effective pages in this chapter, with their latest revision date. An asterisk (*) indicates pages added or replaced at the latest revision. Technical changes to the content are indicated by a black vertical line next to the change or the paragraph that contains the change.

REFERENCE	PAGE	DATE	REFERENCE	PAGE	DATE
APPENDIX A	* 1	Sep. 28/18		* 7	Sep. 28/18
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Appendix A – Miscellaneous

Table 1 Component Maintenance Manual Summary – Appendix A

CHAPTER	MANUFAC-TURER	COMPONENT	CMM OR PART NO.	CMM TITLE
22) FLIGHT CONTROLLER	HONEYWELL	KMC 9200 Flight Controller	071–01597–02 01	Component Maintenance Manual with Illustrated Parts Manual
		KMC 9200 Flight Controller	006–10674–00 00	Installation Manual
		KS 272A Trim Servo	006–05290–00 02	Maintenance Manual
23) COMMUNI-CATIONS	HONEYWELL	AR Series Solid State Memory Voice Recorder (AVR 120 CVR)	980–6023–002	Component Maintenance Manual with Illustrated Part List
		Advance Series Solid State Recorder System (AR-CVR, AR-FDR, AR-COMBI)	012–0708	Installation Manual
		KMA 29 Audio Amplifier / Intercom / Mark Beacon Receiver	065–50678–00 00	Maintenance Manual
		KMA 29 Audio Amplifier / Intercom / Mark Beacon Receiver	006–10678–00 01	Installation and Operation Manual
		KRX 1053 Receiver / Exciter	006–15642–00 01	Component Maintenance Manual
		KHF 1050 HF Communication System	006–10640–00 06	System Installation Manual
		KTR2280 Multi-Mode Digital Radio	006–10654–00 00	Installation and Operation Manual
	Northern Airborne Technologies	NAT 245 PA Amplifier	SM245	Installatin and Operation Manual
		NAT 247 Audio Mixing Amp	247–16–1	Equipment Manual Installation and Operation
24) ELECTRICAL POWER	GOODRICH	DC Starter Generator	23048	DC Starter Generator CMM
		Brush Installation, Seating Lead Information and Run In	1006	Standard Practice Document (SPD)
	Concorde	Aircraft Battery Owner/Operator Manual	DWG–5–0164	Owner/Operator Manual

Table 1 Component Maintenance Manual Summary – Appendix A (continued)

CHAPTER	MANUFAC-TURER	COMPONENT	CMM OR PART NO.	CMM TITLE
		Concorde Valve Regulated Lead-Acid Battery	DWG–5–0171	Owner/Operator Manual
		RG 380E / 60L Valve Regulated Lead-Acid Main Battery	DWG–5–0142	Instructions for Continued Airworthiness, Maintenance Manual Supplement
		RG Series	5–0324	Owner/Operator Manual
	PHOENIX	DC Voltage Regulator	VR1010–24–2A	Maintenance Manual
	L3 Communi-cations	Emergency Power Supply JET Model: PS–850 JET Model: PS–855	TP–470	Installation and Operation Manual
	FTG Aerospace	Lighting Power Supply Interface Internal Specification	112–0002	Installation and Operation Manual
25) EQUIPMENT & FURNISHINGS	ARTEX	ARTEX 406 MHz Emergency Locator Transmitter	570–5060	Description, Operation and Maintenance Manuals
	B/E Aerospace	Seats	3751	JB6 Series Commuter Class Passenger Seats for Viking Air DHC-6 Aircraft
26) FIRE PROTECTION	Walter Kidde /HTL	Discharge Indicators and Connectors	843787 & 843969	Component Maintenance Manual with Illustrated Parts List
27) FLIGHT CONTROLS	PLESSY	ELECTRO-MECHANICAL LINEAR ACTUATOR	L10–20	Overhaul Instructions with Illustrated Parts Breakdown
	DHC	FLAP/ELEVATOR INTERCONNECT BALL SCREW JACK	C6CF1080	Overhaul Manual
		Geared Tab Gearing Mechanism	C6CF1174 & C6CF1174–3	Overhaul Manual
		Screw Jack Assy	C6CT1010	Overhaul Manual
		Trim Control	C6CF1053	Overhaul Manual
		Wing Flap Actuator	C6HF1020	Overhaul Manual
		Syncro Devices	SKA104	Overhaul Instructions
28) FUEL	Airborne	Fuel Pump	1D2–17	Overhaul Instructions with Parts Catalog
	Aircraft Appliance	Check Valve	700 246	M–7016 Overhaul and Maintenance Manual

Table 1 Component Maintenance Manual Summary – Appendix A (continued)

CHAPTER	MANUFAC-TURER	COMPONENT	CMM OR PART NO.	CMM TITLE
	Custom Components	Gage Pressure Switch	8G361–1 (DHCSC70020–5)	Overhaul Manual
	ITT General Controls	Motor Operate Gate Valve	AV16B1864B	
		Actuator Assy Valve DC Motor Motor Operate Gate Valve	106788B107 106788B100 AV16B1762B	Overhaul Instructions with Parts Breakdown
	Parker Aircraft	Level Control Valve	1321–536250	Overhaul Manual
		Swing Check Valve	2650157	Overhaul Manual
	Purolator		E–1097	Operating Instructions Fuel Filter Assy Maintenance Manual
29) HYDRAULIC POWER	DHC	Reservoir Assy	C6HF1005 C6HF1057	Overhaul Manual
	Gar Kenyon	Air Charge Valve and Gage	90869	Manual of Information for Valve and Gauge Assemblies
	Parker Hannifin	Accumulator	1356–58331	Overhaul Instructions with Part Breakdown
	Pneudraulics	Relief Valve	1000–4 (MS28893C4)	Service Instructions with Parts Breakdown
	PTI Technologies	Filter	TM50220	Installation, Operation, and Maintenance Instructions
30) ICE & RAIN PROTECTION	B/E Aerospace	Pneumatic Deicer System	SMR Report No. 97–33–047	Installation and Maintenance Manual
	Rosemount Aerospace	Electrical Windshield Wiper System	2311M95	Component Maintenance Manual with Illustrated Parts List
	Barber Coleman	Windshield Temp Controller	SYLZ50157	Technical Manual - Barber-Coleman Company
	BF Goodrich	Propeller Deicing System	Report 68–471 2C	Maintenance Manual for Electothermal propeller deicing systems
		Brush Block Assembly	Report 59–28K	Maintenance Manual for Installation and Removal of Electothermal propeller deicing system
			Report 68–04–724–I	Overhaul of Brush Block Assemblies for Electothermal propeller deicing system

Table 1 Component Maintenance Manual Summary – Appendix A (continued)

CHAPTER	MANUFAC-TURER	COMPONENT	CMM OR PART NO.	CMM TITLE
	Lucas Aerospace	Ejector	19U–2–631E	Overhaul Instructions for Air Ejector Assemblies
		Sequential Timer	42U–23–692C	Overhaul Manual with Parts List
		Water Separator	44U–2–7512A	Service and Overhaul Instructions with Illustrated Parts Breakdown
		Deicer Distributing Valve	1532U–23–8658	Overhaul Manual with Illustrated Parts List
		Regulating and Relief Valve	38U–23–8511B	Overhaul Manual with Illustrated Parts
31) INSTRU-MENTS	HONEYWELL	IM–950 Aircraft Personality Module	A09–6710–007	Component Maintenance Manual with Illustrated Parts List
		KMC 2210 Display Controller	006–10675–0000	Installation Manual
		KMC 2210 Display Controller	071–01596–0201	Component Maintenance Manual with Illustrated Parts List
		KMC 2220 Multi-Function Display Controller	006–10676–0000	Installation Manual
		KMC 2220 Multi-Function Display Controller	071–01598–0301	Component Maintenance Manual with Illustrated Parts List
		MAU (Modular Avionics Unit) Chassis	A09–6710–012	Component Maintenance Manual with Illustrated Parts List
		Dual Generic I/O Module Assy	A09–6710–018	Component Maintenance Manual with Illustrated Parts List
		Single NIC Module Assy	D200704000034	Component Maintenance Manual with Illustrated Parts List
		Single Generic I/O Module Assy	A09–6710–025	Component Maintenance Manual with Illustrated Parts List
		Weather Data Receiver & Antenna XMD154 & XMRANT–01	XMD157–01–8	Component Maintenance Manual with Illustrated Parts List
		Apex Custom Input/Output Module Assy CIO–AP2 Custom I/O Card	D200705000015	Component Maintenance Manual with Illustrated Parts List

Table 1 Component Maintenance Manual Summary – Appendix A (continued)

CHAPTER	MANUFAC-TURER	COMPONENT	CMM OR PART NO.	CMM TITLE
		Power Supply Module Assy	A09–6710–015	Component Maintenance Manual with Illustrated Parts List
		AGM–200 Advanced Graphics Module	D200703000019	Component Maintenance Manual with Illustrated Parts List
		AIO–F01 Actuator I/O Module	A09–6710–032	Component Maintenance Manual with Illustrated Parts List
	Allied Signal	KA 44/44B KTS156 ADF Antennas	006–05535–0003	Component Maintenance Manual with Illustrated Parts List
	L3 Communi-cations	GH–3100 Electronic Standby Instrument System	TP–559	Installation and Operation Manual
		MAG–3100	0040–21200–01	Magnetometer
		GH–3100 Electronic Standby Instrument Pressure	501–1860–0102	Installation and Operation Manual
		Electronic Standby Instrument System GH–3900.2 Installation Manual	0040–34401–01	Installation Manual
		Flight Data Recorder 512	165E1845–00	Installation and Operation Manual
		Multi-Axial Accelerometer	165E4878–01	Installation and Operation Manual with Illustrated Parts List
32) LANDING GEAR	Cleveland	Wheels & Brakes	AWBCMM001–7/ USA	Wheels & Brakes CMM
			AWBTSG0001–3/ USA	Technician's Service Guide
	DHC	Power Brake Valve Assy	C6HF1036/1037	Overhaul Manual
	VAL	Main Landing Gear Leg Assy	C6U1103	Overhaul Manual
	DHC	Nose Gear Leg Assy	71–100	Overhaul Manual
	DHC	Nose Wheel Steering Actuator and Link Sub–Assy	71–190	Overhaul Manual

Table 1 Component Maintenance Manual Summary – Appendix A (continued)

CHAPTER	MANUFAC-TURER	COMPONENT	CMM OR PART NO.	CMM TITLE
	BF Goodrich	Nose Landing Gear Wheel	3–1197	Maintenance Manual
	Team Jas	Nose Wheel Assembly	PDM3–1197	Instructions for Continued Airworthiness
33) LIGHTS	Whelen Engineering	Taxi Light	05131B	Installation and Maintenance Manual
		Exterior Lights/Beacons	01770900–05 01–0771011–01/–02	Installation Guide
	NAT	LD–12	SM–15	LD–12 Lamp Dimmer MM
34) NAVIGATION	HONEYWELL	KDU–1080 10” Flat Panel MFD / EICAS	K5920078–00	Component Maintenance Manual
		KGS–200 Global Navigation System GPS Receiver	066–15706–0000	Component Maintenance Manual with Illustrated Parts Catalogue
		Dual KGS–200 Global Navigation System Sensor Unit	006–10679–0001	Installation Manual
		KMG 7010 MAGNETOMETER	006–10662–0007	Installation Manual
			006–15663–0000	Component Maintenance Manual with Illustrated Parts Catalogue
		KSG 7200 ADAHRS	006–15662–0000	Component Maintenance Manual with Illustrated Parts Catalogue
		KTR 2280	006–15654–0000	Component Maintenance Manual with Illustrated Parts Catalogue
		KXP 2290/2290A Mode S Transponders	006–15689–0000	Component Maintenance Manual with Illustrated Parts Catalogue
		KXP 2290/2290A Mode S Transponders	006–10689–0001	Installation Manual
		ART 2000 Color Weather Radar Antenna /Receiver/Transmitter	006–05332–0003	Maintenance Manual
		CAS 67A TCAS II System	006–05340–0008	Installation Manual
		KRA 405B Radar Altimeter	006–10536–0006	Component Maintenance Manual
		KRA 405B Radar Altimeter	006–10536–0010	Installation Manual

Table 1 Component Maintenance Manual Summary – Appendix A (continued)

CHAPTER	MANUFAC-TURER	COMPONENT	CMM OR PART NO.	CMM TITLE
		KHM 920 Multi-Hazard Awareness Unit (KTA910 TCAS I Unit)	006–15633–00 01	Component Maintenance Manual
		KN 63 Distance Measuring System	006–05176–00 04	Installation Manual
		KN 63 Distance Measuring System	006–05176–00 01	Maintenance Manual
		Mark V Warning Computer	012–0534–001	
		Mark VI MK VII Enhanced Ground Proximity Warning System (Class A TAWS)	060–4314–150	Installation Design Guide
36) PNEUMATICS	Bendix	Pressure Regulating Valve	38U–23–668B	Overhaul Instructions with Parts Breakdown
	Custom Components	Switch Vacuum Pressure	DHCSC70019–1/17	Maintenance Manual
61) PROPELLERS	Woodward	Governor	3075355–01 8210–304	Maintenance Manual
71) POWERPLANT	DHC	Intake Deflector Actuator	C6EC1055	Overhaul Manual
	Barry Controls	Engine Mount	7350721	Maintenance Manual
73) FUEL SYSTEM	Lucas	Gear Pump	025323–101–3	Maintenance Manual
	Wright Components	Fuel Shutoff Solenoid Valve	15410 (U9500)	Component Maintenance Manual

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