CHANGE

MODEL
401 401A AND 401B
402 402A AND 402B
BUSINESSLINER
AND
UTILILINER
1967 THRU 1978
SERVICE MANUAL

CHANGE 21

3 JUNE 2002

D777C21-13

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TEMPORARY REVISIONS.

1. Additional information which becomes available may be provided by temporary revisions. This service is used to provide, without delay, new information which will assist in maintaining safe flight/ground operations. Temporary revisions are designed to replace or add to existing pages in the manual and are numbered to match pages in the manual. Temporary revisions are normally incorporated into this manual at the next scheduled change, revision or reissue.

REVISION/CHANGE BARS.

1. Additions, or revisions to text in an existing section will be identified by a revision bar on the page adjacent to the change.
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CUSTOMER CARE SUPPLIES AND PUBLICATIONS CATALOG.

A Customer Care Supplies and Publications Catalog is available from your Cessna Service Station or directly from Cessna Propeller Aircraft Product Support, P.O. Box 7706, Wichita, KS. 67277, phone (316) 517-7950 or fax (316) 942-9006. This catalog lists all publications and Customer Care Supplies available from Cessna for prior year models as well as new products. To maintain this catalog in a current status, it is revised yearly and issued in paper and Aerofiche format.

SUPPLEMENTAL TYPE CERTIFICATE INSTALLATIONS.

Inspection, maintenance and parts requirements for supplemental type certificate (STC) installations are not included in this manual. When an STC installation is incorporated on the airplane, those portions of the airplane affected by the installation must be inspected in accordance with the inspection program published by the owner of the STC, since STC installations may change systems interface, operating characteristics and component loads or stresses on adjacent structures. Cessna provided inspection criteria may not be valid for airplanes with STC installations.

CUSTOMER COMMENTS ON MANUAL.

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D. TESTING.

New static tests were conducted to verify the mathematical models which were developed. Test results from previously conducted static tests and fatigue cyclic tests were also reviewed to identify the critical areas of the PSE's.

E. INSPECTION OF AIRPLANE.

A high-time airplane was purchased from a customer for disassembly and inspection in 1988. The airplane had over 20,000 flight hours and 60,000 landings. Locations where cracking was discovered during disassembly are included as inspection locations.

3-0. DURABILITY - FATIGUE AND DAMAGE TOLERANCE

3-1. Airplane Usage

Airplane usage data for the SID program is based on the evaluation of the in-service utilization of the airplane. This data was used to develop the representative fatigue loads spectra. Operational data for development of the Supplemental Structural Inspection Program was obtained from two sources:

A. A survey of operators identified by the FAA which encompassed 14 operators and 85 airplanes. The airplanes included in the survey represent a usage of 1777 flights per week. This survey was conducted in January 1996.

B. A survey of three operators representing 4 airplanes was conducted by Cessna in 1974.

The usage data from each survey is shown in Figure 1.

Usage for spectra determination is defined in terms of a mix of different flights representing typical average in-service utilization of the airplane. This usage, in terms of different flights, reflects the typical in-service flight variation of flight length, takeoff gross weight, payload and fuel.

Each flight is defined in detail in terms of a flight profile. The profile identifies the gross weight, payload, fuel, altitude, speed, distance, etc., required to define the pertinent flight and ground parameters needed to develop the fatigue loads. The flight is then divided into operational segments, where each segment represents the average values of the parameters (speed, payload, fuel, etc.) that are used to calculate the loads spectrum.

3-2. Stress Spectrum

A fatigue loads spectrum, in terms of gross area stress, was developed for each PSE to be analyzed based on the usage-flight profiles. The spectrum represents the following loading environments: flight loads (gust and maneuver), landing impact, balancing tail loads, thrust loads, ground loads (taxi, turning, landing, braking, pivoting, etc.), and ground-air-ground cycles. The resulting spectrum is a representative flight-by-flight, cycle-by-cycle random loading sequence that reflects the appropriate and significant airplane response characteristics.

After reviewing the airplane usage data and the way in which the surveyed airplanes were flown, two sets of stress spectra were developed, one for each flight profile, as described in Section 3-3.

SECTION II - LISTING OF SUPPLEMENTAL INSPECTIONS

Supplemental Inspection Number	Title	Effectivity		Inspection Compliance (See Note 1 and Note 2) Initial	Repeat
52-20-01	Emergency Exit Door Lever Mechanism Inspection	401-0001 402-0001	Thru 401B0221 Thru 402B1384	1,000 Hrs or 3 Yrs	1,000 Hrs or 3 Yrs
53-10-02	Fuselage Left and Right Hand Channel Assemblies	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs or 20 Yrs	5,000 Hrs or 10 Yrs
53-10-03	Horizontal Tail Rear Spar Angle Attachment	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
54-10-04	Model 401/402 Engine Beams	401-0001 402-0001	Thru 401B0221 Thru 402B1384	6,500 Hrs 13 Yrs	1,600 Hrs 3 Yrs
54-10-05	Model 401/402 Engine Beam Modification	401-0001 402-0001	Thru 401B0221 Thru 402B1384	Refer to SID details for initial and repeat inspections times.	Refer to SID details for initial and repeat inspections times.
55-10-03	Horizontal Stabilizer Spars and Attachments	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
55-10-04	Horizontal Stabilizer Forward Spar Upper Cap	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
55-10-05	Horizontal Stabilizer Forward Spar Lower Cap	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
55-10-06	Horizontal Stabilizer Forward Spar Attach BL 7.69	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
55-10-07	Horizontal Stabilizer Rear Spar Lower Cap Attach	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
55-10-08	Horizontal Stabilizer Rear Spar Upper Cap, BL 0.0	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
55-10-09	Horizontal Stabilizer Rear Spar Lower Cap, BL 0.0	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs
55-20-01	Outboard Elevator Hinge Bracket and Attachment	401-0001 402-0001	Thru 401B0221 Thru 402B1384	5,000 Hrs 10 Yrs	1,000 Hrs 3 Yrs
55-20-02	Elevator Hinges and Fittings	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	2,500 Hrs 5 Yrs
55-30-01	Vertical Stabilizer Spars and Attachments	401-0001 402-0001	Thru 401B0221 Thru 402B1384	15,000 Hrs 20 Yrs	5,000 Hrs 10 Yrs

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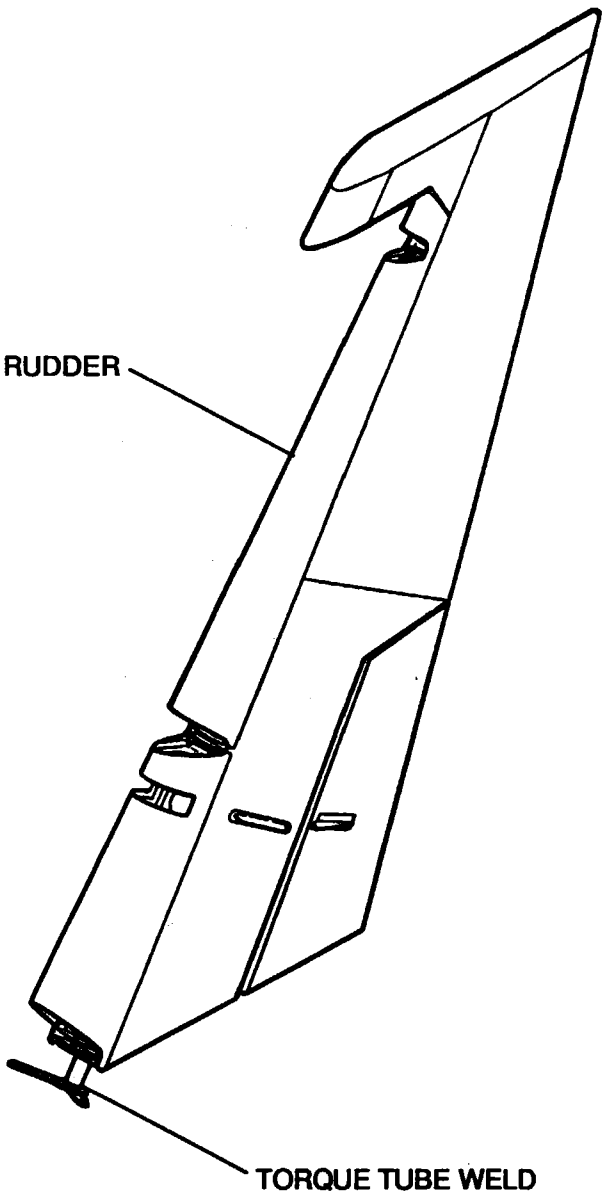

51333002

Rudder Torque Tube
Figure 1 (Sheet 1)

TITLE **SUPPLEMENTAL INSPECTION NUMBER: 32-10-04**

Main Gear Actuator Collar

EFFECTIVITY

	INSPECTION COMPLIANCE				
401-0001 Thru 401B0221 402-0001 Thru 402B1384	**TYPICAL:**	**INITIAL**	12,500 LANDINGS	OR	20 YRS
		REPEAT	2,500 LANDINGS	OR	5 YRS
	SEVERE:	**INITIAL**	12,500 LANDINGS	OR	20 YRS
		REPEAT	2,500 LANDINGS	OR	5 YRS

PURPOSE

Detailed inspection of the main gear actuator collar for cracks due to fatigue, overload, and corrosion.

INSPECTION INSTRUCTIONS

1. Jack airplane off of ground surface according to Service Manual.
2. Remove necessary assemblies to gain access to entire area of Main Gear Actuator Collar per Service Manual.
3. Refer to Section IV (NDI Inspection), Supplemental Inspection Number 32-10-04, for specific instructions.
4. Reinstall any removed assemblies and return aircraft to ground surface. Refer to Service Manual.

ACCESS/LOCATION **DETECTABLE CRACK SIZE**

Main Gear Actuator Collar 0.10 Inch

INSPECTION METHOD

Magnetic Particle

REPAIR/MODIFICATION

Replace the main gear actuator collar if a crack is found. Refer to Service Manual.

COMMENTS

If a crack is detected, contact Propeller Aircraft Product Support of Cessna Aircraft Company.

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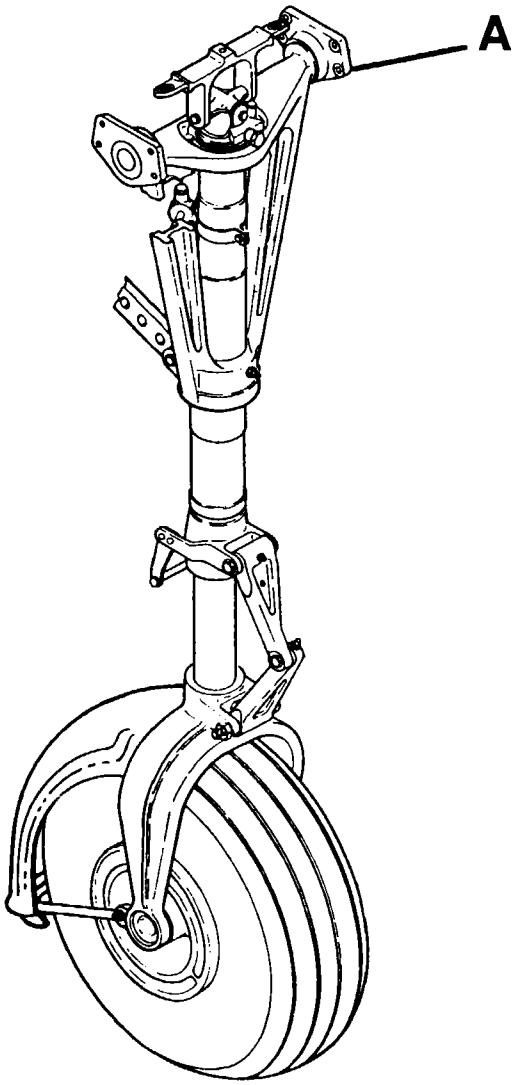
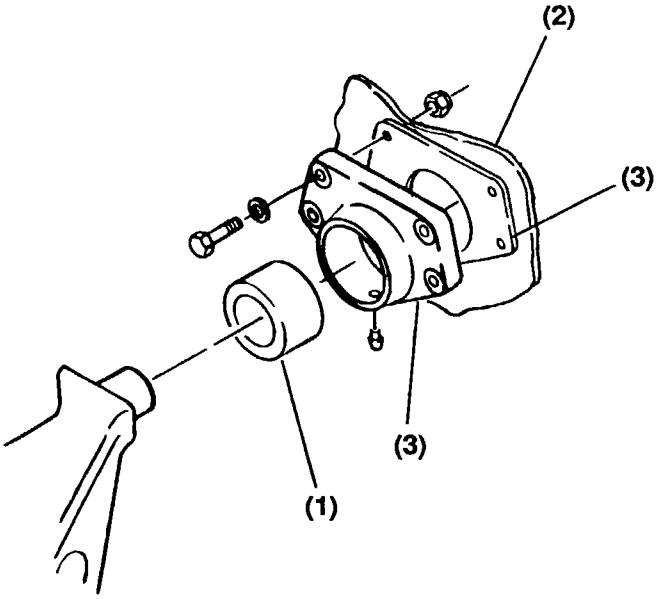
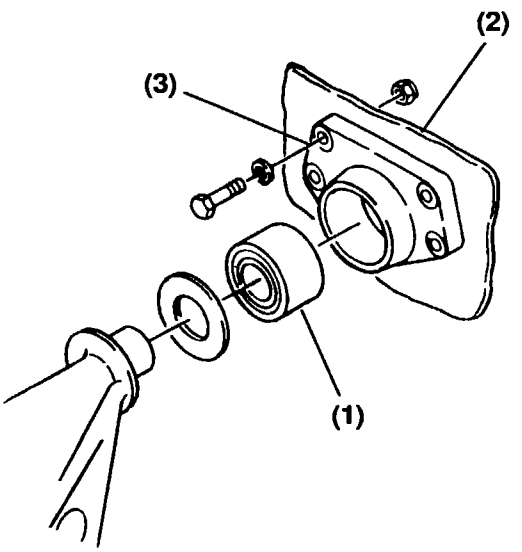




DETAIL A
LARGE LUG (1.31 INCH DIAMETER) TRUNNION



DETAIL A
SMALL LUG (1.19 INCH DIAMETER) TRUNNION

(1) INSPECT FOR FATIGUE CRACKS, CORROSION AND EXCESS WEAR. REPLACE WITH NEW PARTS AS REQUIRED.

(2) INSPECT STRUCTURE FOR CRACKS, CORROSION AND ELONGATED ATTACHMENT HOLES. REPAIR OR REPLACE AS REQUIRED. METHODS OR APPROVED DATA.

(3) INSPECT FOR CRACKS, CORROSION AND ELONGATED ATTACHMENT HOLES. REPLACE WITH PARTS/COMPONENTS AS REQUIRED.

NOTE: REPLACE ALL HARDWARE REMOVED DURING INSPECTION PROCEDURE WITH NEW ATTACHING HARDWARE AT REASSEMBLY/REINSTALLATION.

1442R3004
A1042R1005
A1042R1005

Nose Landing Gear Installation
Figure 2 (Sheet 4)

TITLE **SUPPLEMENTAL INSPECTION NUMBER: 54-10-04**

Engine Beams

EFFECTIVITY

401-0001 Thru 401B0221
402-0001 Thru 402B1384

INSPECTION COMPLIANCE

TYPICAL:	**INITIAL**	6,500 HRS	OR	13 YRS
	REPEAT	1,600 HRS	OR	3 YRS*
SEVERE:	**INITIAL**	6,500 HRS	OR	13 YRS
	REPEAT	1,600 HRS	OR	3 YRS*

PURPOSE

Detailed inspection of the engine beams for the Model 401/402 for cracks due to fatigue, overload, and corrosion.

INSPECTION INSTRUCTIONS

1. Remove engine from the airplane. Refer to Service Manual. Remove four (4) bolts attaching unfeathering accumulator (if installed) and remove to allow access to the engine mount bolts. Do not disconnect hose. Mark all engine mount components for proper orientation. Disconnect the forward and aft mounts from engine and engine beam and remove mounts. Retain bolts and washers.

2. Visually inspect engine support structure for cracks, overload deformations, corrosion, loose fasteners and exhaust leak heat damage.

3. Eddy current inspect the area around and between the fasteners common to the engine beams. Inspect the forward and aft engine mount areas including fastener holes in the unfeathering accumulator attach area. Refer to Section IV, (NDI Inspection), Supplemental Inspection Number 54-10-04.

4. Visually inspect engine support beam for loose or working fasteners. If no cracks/damage is found, restore airplane to appropriate configuration. Refer to Model 401/402 Service Manual.

ACCESS/LOCATION **DETECTABLE CRACK SIZE**

Engine 0.16 Inch

INSPECTION METHOD

Visual and Eddy Current

REPAIR/MODIFICATION

COMMENTS

If a crack is detected, contact Propeller Aircraft Product Support of Cessna Aircraft Company.

*Repeat inspection interval corresponds with every engine overhaul.

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TITLE **SUPPLEMENTAL INSPECTION NUMBER: 55-30-04**

Vertical Stabilizer Rear Spar Cap Attach, WL 108.38

EFFECTIVITY

		INSPECTION COMPLIANCE			
401-0001 Thru 401B0221 402-0001 Thru 402B1384	**TYPICAL:**	**INITIAL**	15,000 HRS	OR	20 YRS
		REPEAT	5,000 HRS	OR	10 YRS
	SEVERE:	**INITIAL**	15,000 HRS	OR	20 YRS
		REPEAT	5,000 HRS	OR	10 YRS

PURPOSE

Detailed inspection of the rear spar attachment at WL 108.38 for cracks due to fatigue, overload, and corrosion.

INSPECTION INSTRUCTIONS

1. Refer to Section IV (NDI Inspection), Supplemental Inspection Number 55-30-04, for specific instructions.

ACCESS/LOCATION **DETECTABLE CRACK SIZE**

Tailcone 0.080 Inch

INSPECTION METHOD

Eddy Current

REPAIR/MODIFICATION

COMMENTS

If a crack is detected, contact Propeller Aircraft Product Support of Cessna Aircraft Company.

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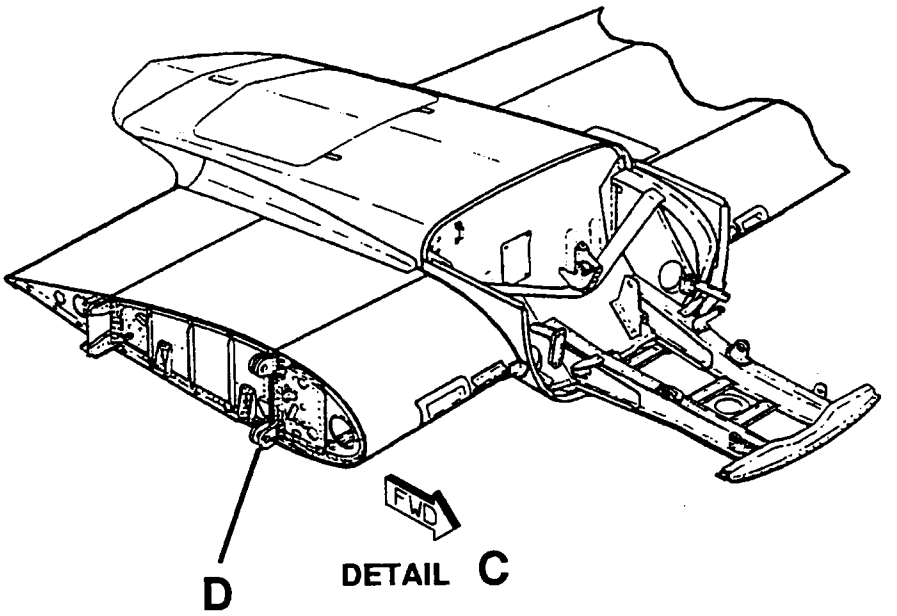

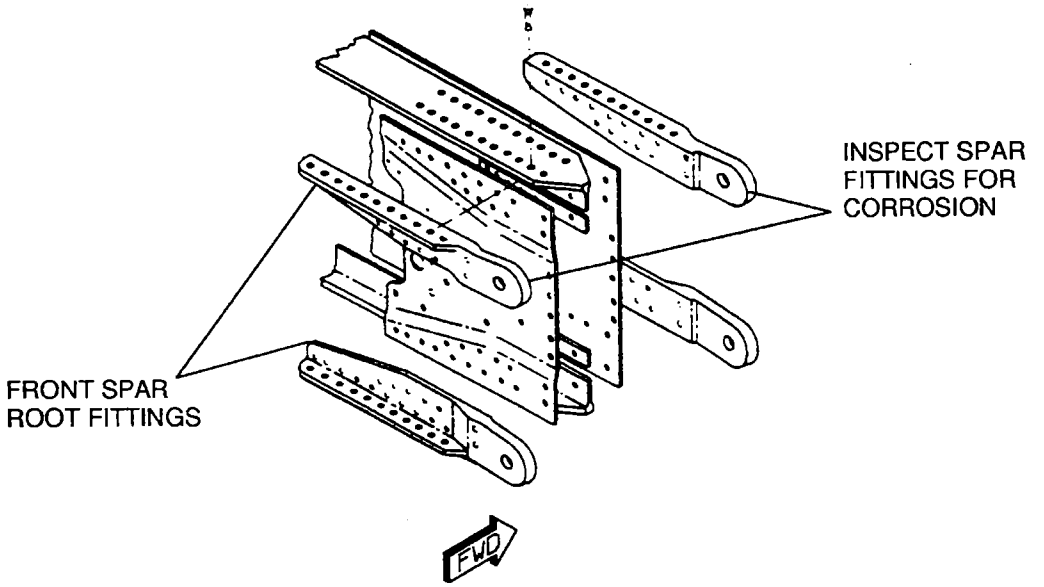

DETAIL D
(FRONT SPAR ROOT FITTINGS)

C52203001

Wing Upper Carry-Thru Front Spar Cap Inspection
Figure 1 (Sheet 2)

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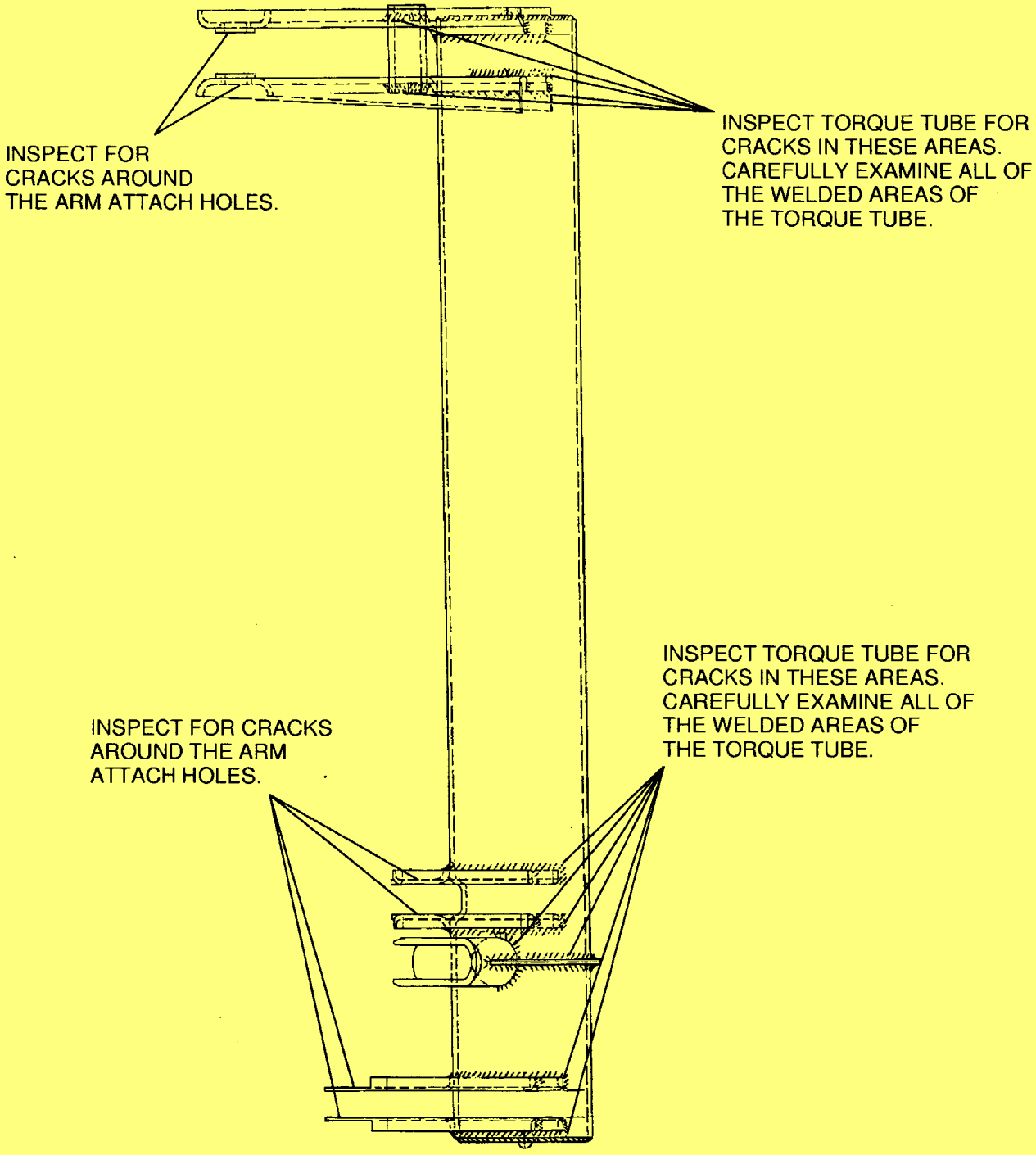

DETAIL A

Main Landing Gear Torque Tube Assembly
Figure 1 (Sheet 2)

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TITLE **SUPPLEMENTAL INSPECTION NUMBER: 53-10-02**

Fuselage Left and Right Hand Channel Assemblies

EFFECTIVITY

401-0001 Thru 401B0221
402-0001 Thru 402B1384

DESCRIPTION

Inspect for fatigue cracks around the fasteners common to the channel assembly and the fuselage skin.

PREPARATION

1. Refer to Figure 1.
2. Clean the inspection area with solvent to remove dirt, grease, oil and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Eddy Current

CRACK SIZE

Minimum detectable crack size: 0.15 Inch

EQUIPMENT

The following equipment was used to develop this procedure; equivalent eddy current test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

PART NUMBER	QUANTITY	DESCRIPTION
Model 19e"	1	EDDY CURRENT INSTRUMENT Staveley Instruments Incorporated 421 North Quay Kennewick, WA 99336
VM202AF-6 shielded absolute coil, 0.10" coil dia, 100-500 KHz.	1	EDDY CURRENT PROBE Surface Pencil Probe VM Products 11208 62 Avenue Puyallup, WA 98373
VM89A	1	REFERENCE STANDARD VM Products 11208 62 Avenue Puyallup, WA 98373

Reference Standard Notch Depths Surface Eddy Current: 0.005", 0.020" and 0.040".

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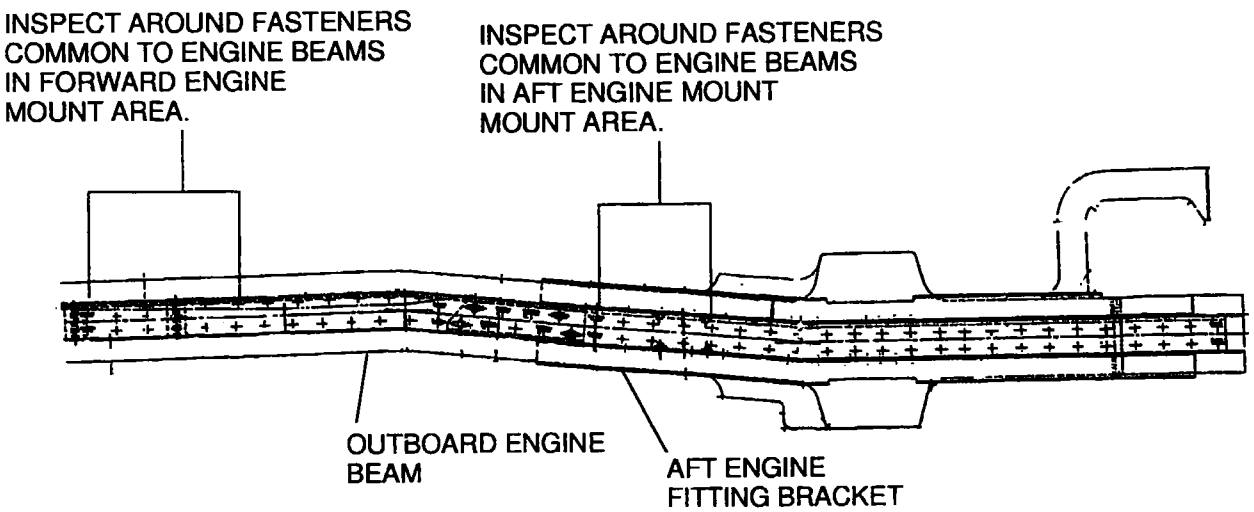


VIEW B-B

LOOKING DOWN AT OUTBOARD ENGINE BEAM
(INBOARD ENGINE BEAM OPPOSITE)

B-B5251R1003

Engine Mount Inspection - Model 401/402
Figure 1 (Sheet 2)

TITLE **SUPPLEMENTAL INSPECTION NUMBER: 55-10-06**

Horizontal Stabilizer Forward Spar Attach, BL 7.69

EFFECTIVITY

401-0001 Thru 401B0221
402-0001 Thru 402B1384

DESCRIPTION

Inspect for fatigue cracks in the horizontal stabilizer forward spar attach points.

PREPARATION

1. Remove the horizontal stabilizer forward spar attach bolts. Refer to Figure 1.

2. Clean the inspection area with solvent to remove dirt, grease, oil and other substances that may interfere with the inspection.

INSPECTION METHOD

Bolt Hole Eddy Current

CRACK SIZE

Minimum detectable crack size: 0.080 Inch

EQUIPMENT

The following equipment was used to develop this procedure; equivalent eddy current test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

PART NUMBER	QUANTITY	DESCRIPTION
Model 19e"	1	EDDY CURRENT INSTRUMENT Staveley Instruments Incorporated 421 North Quay Kennewick, WA 99336
VM101BS 5/8" shielded absolute coil, 0.10" coil dia, 100-500 KHz.	1	EDDY CURRENT PROBE Bolt Hole Probe VM Products 11208 62 Avenue Puyallup, WA 98373
VM89A	1	REFERENCE STANDARD VM Products 11208 62 Avenue Puyallup, WA 98373

Reference Standard Notch Size Bolt Hole Inspection: 45 Degree Corner Notch 0.050" x 0.050", width 0.005".

INSPECTION INSTRUCTIONS

1. Connect the bolt hole probe (0.156" dia.) to the eddy current instrument and adjust the instrument frequency to 200 KHz.

2. Null the probe in the appropriate reference standard hole away from the calibration notch.

3. Adjust liftoff on impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from right to left as the probe is lifted from the part surface.

4. Adjust the instrument gain controls to obtain a signal amplitude response from the calibration notch that is a minimum of 2 major screen divisions.

5. Inspect the inner circumference of the four fastener holes common to the horizontal stabilizer rear spar upper cap around BL 0.00. Inspect the entire depth of each hole. Observe the phase and amplitude changes on the eddy current instrument. Refer to Figure 2.

6. If an indication is noted, carefully repeat the inspection in the opposite direction of probe rotation to verify the indication.

7. If no cracks are detected during this inspection:

A. Install four each MS20470AD4 Rivets in horizontal stabilizer rear spar upper cap.

B. Reinstall the horizontal stabilizer. Refer to the Service Manual.

8. Cracks detected during this inspection shall be reported to Propeller Aircraft Product Support along with the following information: hole diameter; location of hole; hole edge distance, and approximate length and depth of the crack.

TITLE **SUPPLEMENTAL INSPECTION NUMBER: 57-10-03**

Wing Rib Improvement - Main Landing Gear Side Brace

EFFECTIVITY

401-0021 Thru 401B0221
402-0001 Thru 402B1003

DESCRIPTION

Inspect for fatigue cracks in the main landing gear side brace support.

PREPARATION

1. Clean the inspection area with solvent to remove dirt, grease, oil and other substances that may interfere with the inspection.

INSPECTION METHOD

Surface Eddy Current

CRACK SIZE

Minimum detectable crack size: 0.10 Inch

EQUIPMENT

The following equipment was used to develop this procedure; equivalent eddy current test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

PART NUMBER	QUANTITY	DESCRIPTION
Model 19e"	1	EDDY CURRENT INSTRUMENT Staveley Instruments Incorporated 421 North Quay Kennewick, WA 99336
VM202AF-6 shielded absolute coil, 0.10" coil dia, 100-500 KHz.	1	EDDY CURRENT PROBE Surface Pencil Probe VM Products 11208 62 Avenue Puyallup, WA 98373
VM89A	1	REFERENCE STANDARD VM Products 11208 62 Avenue Puyallup, WA 98373

Reference Standard Notch Depths Surface Eddy Current: 0.008", 0.020" and 0.040"

EQUIPMENT

The following equipment was used to develop this procedure; equivalent eddy current test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

PART NUMBER	QUANTITY	DESCRIPTION
Model 19e"	1	EDDY CURRENT INSTRUMENT Staveley Instruments Incorporated 421 North Quay Kennewick, WA 99336
VM101BS 5/16" shielded absolute coil, 0.10" coil dia, 100-500 KHz.	1	EDDY CURRENT PROBE Bolt Hole Probe VM Products 11208 62 Avenue Puyallup, WA 98373
VM89A	1	REFERENCE STANDARD VM Products 11208 62 Avenue Puyallup, WA 98373

Reference Standard Notch Size Bolt Hole Inspection: 45 Degree Corner Notch 0.050" x 0.050", width 0.005".

INSPECTION INSTRUCTIONS

1. Connect the bolt hole probe (0.312" dia.) to the eddy current instrument and adjust the instrument frequency to 200 KHz.
2. Null the probe in the appropriate reference standard hole away from the calibration notch.
3. Adjust liftoff on impedance plane instrumentation so the deflection of the liftoff trace is horizontal and deflects from right to left as the probe is lifted from the part surface.
4. Adjust the instrument gain controls to obtain a signal amplitude response from the calibration notch that is a minimum of 2 major screen divisions.
5. Inspect the inner circumference of each hole common to the wing lower rear carry-thru spar cap and wing attach fitting. Inspect the entire depth of each hole. Observe the phase and amplitude changes on the eddy current instrument. Refer to Figure 2.
6. If an indication is noted, carefully repeat the inspection in the opposite direction of probe rotation to verify the indication.
7. If no crack is found reinstall, AN5-7 Bolts and MS24694 Screws in the wing lower carry-thru rear spar cap and attach fitting.
8. Cracks detected during this inspection shall be reported to Propeller Aircraft Product Support along with the following information: hole diameter; location of hole; hole edge distance, and approximate length and depth of the crack.

TITLE **SUPPLEMENTAL INSPECTION NUMBER: 57-10-11**

Wing Lower Front Spar Cap at Root Fitting Attach, WS 46.70

EFFECTIVITY

401-0001 Thru 401B0221
402-0001 Thru 402B1384

DESCRIPTION

Inspect for fatigue cracks in the inboard vertical flange of the lower front spar root fitting at WS 46.70 (WS 28.96 per Service Manual).

PREPARATION

1. Remove the two bolts through the vertical flange at the inboard end of the lower front spar root attach fitting. Refer to Figure 1.
2. Remove the upper inboard rivet through the vertical flange at the inboard end of the lower front spar root attach fitting. Refer to Figure 1.
3. Clean the inspection area with solvent to remove dirt, grease, oil and other substances that may interfere with the inspection.

INSPECTION METHOD

Eddy Current

CRACK SIZE

Minimum detectable crack size: 0.080 Inch

EQUIPMENT

The following equipment was used to develop this procedure; equivalent eddy current test equipment may be used providing the equipment is capable of achieving the required frequency range and test sensitivity.

PART NUMBER	QUANTITY	DESCRIPTION
Model 19e"	1	EDDY CURRENT INSTRUMENT Staveley Instruments Incorporated 421 North Quay Kennewick, WA 99336
VM101BS 3/16" and VM101BS 1/4" shielded absolute coil, 0.10" coil dia, 100-500 KHz.	1 1	EDDY CURRENT PROBE Bolt Hole Probe VM Products 11208 62 Avenue Puyallup, WA 98373
VM89A	1	REFERENCE STANDARD VM Products 11208 62 Avenue Puyallup, WA 98373

Reference Standard Notch Size Bolt Hole Inspection: 45 Degree Corner Notch 0.050" x 0.050", width 0.005".

NOTE 2: The probe shall have a maximum coil dimension of 1/8" and operate at 200 kHz. The attach fittings have a nominal hole diameter of 5/8" although this dimension may increase due to over sizing of the hole.

NOTE 3: Any NIST (National Institute of Standards and Technology) (or equivalent) traceable bolt hole standard may be used provided it is an aluminum alloy and has 0.050" X 0.050" corner EDM (Electro Discharge Machined) notches.

NOTE 4: The dial calipers shall be used to set the index points on the eddy current probe.

CALIBRATION

1. The instrument shall be calibrated and operated in accordance with this procedure and the manufacturer's instructions.

2. Instrument calibration shall be performed prior to inspection. Calibration shall be checked at intervals necessary to maintain calibration during continuous use and at the conclusion of the inspection. The instrument shall be recalibrated if any part of the system is replaced or if any calibrated control settings are changed.

3. The test system sensitivity shall be established by setting the instrument frequency to 200 kHz and adjusting the instrument controls to achieve a minimum signal deflection of three major divisions when the coil is centered over the EDM notch (Refer to 57-10-22 Figure 1).

INSPECTION INSTRUCTIONS

1. Both the left and right fitting assemblies shall be inspected for 100% of their thickness (all 5 lugs) by indexing the bolt hole probe and scanning a total of 16 times per assembly as indicated in 57-10-22 Figure 2.

2. Calibrate the instrument to establish sensitivity in accordance with above.

3. Establish each index point by measuring the distance from the center of the probe coil to the edge of the probe collar.

4. After setting each index point, position the probe in the hole and balance the instrument if necessary. Rotate the probe through better than 360 degrees.

NOTE: This procedure assumes the eddy current probe has a working length of 2.0" or greater. If necessary, the procedure may be accomplished by indexing the probe through points 1 to 10 from both the forward and aft sides of the fitting assembly.

5. Indications found during the inspection may be confirmed with a right angle surface probe that has a 1/8" or less diameter coil.

6. If no cracks are found, reinstall the wing attach bolt.

7. Cracks detected during this inspection shall be reported to Propeller Aircraft Product Support along with the following information: hole diameter, location of hole, hole edge distance, and length and depth of the crack.

G. Elevator Trim Tab

(1) Elevator Trim Tab Up Travel	5° +1° -0°
(2) Elevator Trim Tab Down Travel	30° +1° -0°
(3) Elevator Trim Tab Cable Tension (At 65° to 95°F)	0 ±3 Pounds

H. Electric Elevator Trim Tab Control

(1) Elevator Trim Tab Cable Tension (At 65°F to 95°F)	18 ±3 Pounds

I. Flaps

(1) Flaps Down Travel	45° +1°, -0°
(2) Flaps Extend Cable Tension (At 65°F to 95°F)	85 ±10 Pounds
(3) Flaps Return Cable Tension (At 65°F to 95°F)	225 ±25 Pounds

LANDING GEAR:

Main Wheels and Brakes (Triple-Piston, Disc-Type)	
Tire Size	6.50 x 10 - 8-Ply Rating
Tire Pressure	62 PSI
Main Strut Pressure (Without Load)	300 PSI
Nose Wheel	
Tire Size	6.00 x 6 - 6-Ply Rating
Tire Pressure	40 PSI
Nose Strut Pressure (Without Load)	165 PSI

WING DIHEDRAL 5.0°

RECOMMENDED NUT TORQUES

THE TORQUE VALUES STATED ARE POUND-INCHES, RELATED ONLY TO STEEL NUTS ON OIL-FREE CADMIUM PLATED THREADS.

FINE THREADS

TAP SIZE	TENSION NUTS TORQUE STD (NOTE 1)	TENSION NUTS TORQUE ALT (NOTE 2)	SHEAR NUTS TORQUE STD (NOTE 3)	SHEAR NUTS TORQUE ALT (NOTE 2)
8-36	12-15		7-9	
10-32	20-25	20-28	12-15	12-19
1/4-28	50-70	50-75	30-40	30-48
5/16-24	100-140	100-150	60-85	60-106
3/8-24	160-190	160-260	95-110	95-170
7/16-20	450-500	450-560	270-300	270-390
1/2-20	480-690	480-730	290-410	290-500
9/16-18	800-1000	800-1070	480-600	480-750
5/8-18	1100-1300	1100-1600	660-780	660-1060
3/4-16	2300-2500	2300-3350	1300-1500	1300-2200
7/8-14	2500-3000	2500-4650	1500-1800	1500-2900
1-14	3700-4500	3700-6650	2200-3300	2200-4400
1-1/8-12	5000-7000	5000-10000	3000-4200	3000-6300
1-1/4-12	9000-11000	9000-16700	5400-6600	5400-10000

COARSE THREAD SERIES

	(NOTE 4)	(NOTE 5)
8-32	12-15	7-9
10-24	20-25	12-15
1/4-20	40-50	25-30
5/16-18	80-90	48-55
3/8-16	160-185	95-100
7/16-14	235-255	140-155
1/2-13	400-480	240-290
9/16-12	500-700	300-420
5/8-11	700-900	420-540
3/4-10	1150-1600	700-950
7/8-9	2200-3000	1300-1800
1-8	3700-5000	2200-3000
1-1/8-8	5500-6500	3300-4000
1-1/4-8	6500-8000	4000-5000

NOTES

1. Covers AN310, AN315, AN345, AN362, AN363, AN366, MS17825, MS20365, MS20500, MS21044, MS21043, MS21044, MS21045, MS21046, MS21047, MS21048, MS21078 and other NAS fine thread tension nuts except NAS 679.
2. When using AN310, AN320, MS17825 or MS17826 castellated nuts where alignment between the bolt and cotter pin hole is not reached using normal torque values, use alternate torque values, use alternate torque values or replace the nut.
3. Covers AN316, AN320, AN7502, MS17826, MS20364, MS21043, MS21083, MS21245 and other NAS fine thread shear nuts except NAS 679.
4. Covers AN340 MS20341, MS20365, MS21042, MS21043, MS21044, MS21045, MS21046 and other coarse thread tension nuts.
5. Covers MS20364 and other coarse thread shear nuts.

CAUTION

During removal and replacement of component parts, all self-locking nuts and castellated self-locking nuts must be replaced with new nuts.

These torque values are recommended for all procedures contained in this manual except where other values are stipulated. They are not to be used for checking tightness of installed parts during service.

Figure 1-8. Torque Values

gusts will not shift aircraft into another. Leave no long chains dangling. Release the parking brake to prevent seizing and chock the wheels.

Indefinite Storage

a. Drain the engine oil and service with corrosive preventive mixture.

NOTE

Corrosion preventive mixture consists of one part compound MIL-C-6529, Type I, (Esso Rust-Ban 628, Cosmoline No. 1223 or equivalent) mixed with three parts new lubricating oil of the grade recommended for service. This mixture should be thoroughly mixed and maintained at a temperature of at least 221° F at the time it is added to the engine. Immediately after servicing with the corrosion preventive mixture, fly the aircraft for a period of time not to exceed a maximum of 30 minutes.

b. Remove the top spark plug from each cylinder and spray with corrosion preventive mixture (221° to 250° F). To thoroughly cover all surfaces of the cylinder interior, move the nozzle of the spray gun from the top to the bottom of the cylinder. If by accident the propeller is rotated following this spraying, respray the cylinders to insure an unbroken coverage of corrosion preventive mixture on all surfaces.

c. Install protex plugs in each of the top spark plug holes, making sure that each plug is blue in color when installed. Protect and support the spark plug leads with AN-4060-1 protectors.

d. Place a bag of desiccant in the exhaust pipes and seal the openings with moisture resistant tape.

e. Seal the cold air inlet to the heater muff with moisture resistant tape to exclude moisture and foreign objects.

f. Seal the engine breather by inserting a protex plug in the breather hose and clamping in place.

g. Attach a red streamer to each place on the engine where bags of desiccant are placed. Either attach red streamers outside of the sealed area with tape or to the inside of the sealed area with safety wire to prevent wicking of moisture into the sealed area.

h. Engines, with propellers installed, that are preserved for storage in accordance with this section should have each propeller tagged in a conspicuous place with the following notation of the tag: "DO NOT TURN PROPELLER - ENGINE PRESERVED."

i. After the engine has been prepared for storage, remove the battery from the aircraft, store in a cool dry place and check regularly for state of charge. Note serial number on battery and reinstall in the same aircraft.

j. If the aircraft is stored outdoors, place control locks on all movable control surfaces and tie the aircraft down snugly, not tightly, with enough clearance so wind gusts will not shift aircraft into another. Leave no long chains dangling. Release the parking brake to prevent seizing and chock the wheels.

k. Cover the aircraft with moisture resistant paper and tape as necessary.

NOTE

The necessity of this requirement can be determined by the condition at the storage area.

l. Install pitot tube cover, seal static source, install ground locks on retractable gear aircraft and attach red streamers at each location. Close all vents and plug cowl openings to prevent bird nests in the engine compartment.

NOTE

As an alternate method of indefinite storage, the aircraft may be serviced in accordance with the procedures under Temporary Storage providing the aircraft is run-up at maximum intervals of ninety (90) days and then reserviced per the temporary storage requirements.

m. Aircraft stored in accordance with the indefinite storage procedures should be inspected per the following instructions:

1. Aircraft prepared for indefinite storage shall have the cylinder protex plugs inspected weekly. The plugs should be changed as soon as their color indicates unsafe conditions of storage. If the dehydrator plugs have changed color in one half or more of the cylinders, all desiccant material on the engine should be replaced.

2. The cylinder bores of all engines prepared for indefinite storage should be resprayed with corrosion preventive mixture every 6 months. Before spraying, the engines shall be inspected for corrosion as follows: Inspect the interior of at least one cylinder on each engine through the spark plug hole. Remove at least one rocker box cover from each engine and inspect the valve mechanism.

Restoring Aircraft to Service.

If the proper procedures were followed for extended storage, the aircraft should require the following service:

a. Airframe:
1. Remove all covers from vents and air inlets.
2. Clean aircraft, if required.
3. Lubricate the aircraft.
4. Check the brake cylinders, struts, and shimmy dampener.
5. Drain a small amount of fuel from all fuel drains and check for water and sediment.
6. Check fuel selector valve for evidence of leaking.

b. Battery:
1. Charge and check battery.
2. Install battery.

c. Engines:
1. Remove all vent and opening covers installed.
2. Remove oil sump drain plugs.

equipment is thoroughly cleaned before using. Apply approximately 1.5-2.0 dry mils film thickness using either conventional air atomization or airless spray equipment. After application at the above film thickness and when cured at 75-80°F with a relative humidity between 50% and 60%, the coating will be dry to tape within 6-8 hours. The coating has good chemical resistance within 16-24 hours after application and ultimate cure is achieved after 10-14 days.

Vinyl Enamel.

Vinyl enamel shall be prepared for spraying by adding 1 part T-5321 lift control to 20 parts paint with T-1866B thinner added to bring viscosity to a range of 19 to 20 seconds in a Number 2 Zahn cup. The vinyl enamel should be applied in full cross coats to ensure a total cured film thickness of 0.002 to 0.003 inch. Vinyl enamel should be force dried for a minimum of 6 hours at 130-140°F prior to handling or masking.

NOTE

Clean spraying equipment immediately with Methyl Ethyl Ketone or Toluene.

Epoxy Enamel.

Epoxy enamel 54E series epoxy shall be catalyzed with T-6221A in a ratio of 4 parts base to 1 part adduct. This mixture may be thinned with MEK to a spray viscosity of 19 to 20 seconds in a Number 2 Zahn cup. Induction time for mixture is 10 to 15 minutes and the pot life is 16 hours. Material shall be applied in a spray coat to yield a dry film thickness of 0.0008 to 0.0012 inch. Epoxy enamel finish should be allowed to cure 8 hours at room temperature or force dried by air for 30 minutes followed by 30 minutes at 250°F or 2 hours at 130°F.

NOTE

Clean spraying equipment immediately with Methyl Ethyl Ketone or Toluene.

Heat-Resistant Enamel (See figure 2-4B.)

Surfaces to receive heat-resistant enamel shall be chemically filmed treated with Iridite 14-2 in accordance with manufacturer's instructions. Fiberglass surface should be lightly sanded and solvent cleaned. Heat-resistant enamel may be reduced by adding 1 to 2 volumes of Xylene or Toluene to 1 volume base. Heat-resistant enamel should be spray coated to yield a film thickness of 0.001 to 0.0015 inch. Heat-resistant enamel may be force dried by 250°F to 300°F for 45 minutes. Allow 30 minutes to air dry at room temperature prior to force drying.

NOTE

Clean spraying equipment immediately with Methyl Ethyl Ketone or Toluene.

Clear Urethane Top Coat (U.S. Paint).

Mix the clear urethane C-21C in a 1 to 1 ratio with catalyst C-22B. Mixture may be sprayed at this viscosity or thinned with MEK to a viscosity of no less than 18 to 20 seconds in a Number 2 Zahn cup. Apply the clear coating in three uniform 50% overlap spray coats to an approximate thickness of 2-1/2 to 3 mils dry film thickness. Allow to air dry 4 to 6 hours or force dry at approximately 135°F for 1 hour.

NOTE

If area is to be recoated, lettered or stenciled in any way, this will have to be applied within a 36-hour period. Longer times will necessitate a light sanding (remove gloss) before recoating or lettering.

NOTE

All equipment should be cleaned immediately after use. T-732A, or Cellosolve Acetate should be used for cleaning equipment.

Clear Urethane Top Coat for Metallic Gold (Sterling).

Mix 1 volume of clear polyurethane enamel 65-U-100S (Sterling) with 1 volume of 65-U-1685 (Sterling) polyurethane catalyst. The polyurethane enamel may be thinned with U-1275 (Sterling) thinner to a spray viscosity of 18 to 22 seconds in a Number 2 Zahn cup. Apply the clear polyurethane enamel over the metallic gold by spraying the coating in uniform coats to yield an approximate dry film thickness of 2 mils. Allow to air dry 6 to 8 hours.

Touch Up Polyurethane.

When it is necessary to touch up or refinish a small area, the edge of the finish adjacent to the defect shall be feathered by sanding with #320 grit sandpaper followed by #400 grit sandpaper. Avoid, if possible, sanding through the primer. If the primer is penetrated over an area of 1-inch square or larger, the surface must be reprimed.

NOTE

Avoid spraying metal primer on the adjacent paint as much as possible.

Fill the feathered areas by spraying on several coats of Sterling 65-U-1761 and U-1762 primer surfacer and allow 1 hour drying time for each coat. Sand the entire area with #400 grit sandpaper and apply original finish.

NOTE

Remove dry overspray from adjacent painted surfaces as soon as possible with Methyl Ethyl Ketone.

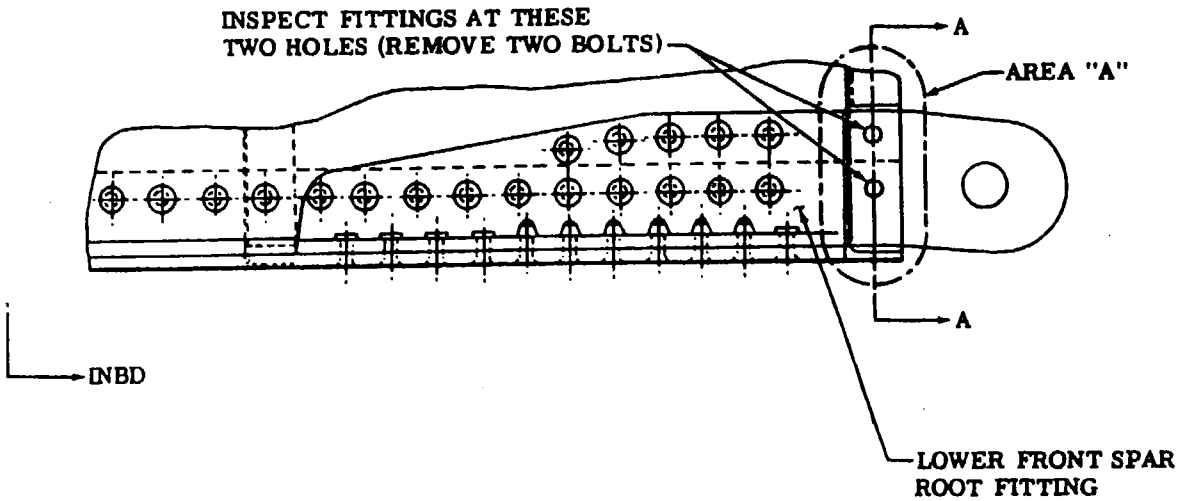
A. Every 500 hours.
B. Every 24 years or 4380 cycles.
C. Every 1400 hours.
D. Refer to latest McCauley Accessory Division Service Bulletin.
E. On condition.
F. Every 2800 hours.
G. Replace the engine compartment flexible fluid-carrying rubber hoses (Cessna-installed only) every five years or at engine overhaul, whichever occurs first (this does not include drain hoses). Rubber hoses which are beyond these limits and are in a serviceable condition must be put on order immediately and then replaced in 120 days or less after receiving the new hose(s) from Cessna Aircraft Company.
H. Every 1000 or 3 years.
I. Not used.
J. Multi-segmented V-band clamps must be replaced every 400 hours total time in service (TTIS) until the initial 500 hour Complete Disassembly Inspection or 500 hour Partial Disassembly Inspection is done. After you complete the Disassembly Inspection and installation of new clamps, the clamps must be replaced every 500 hours.
K. Every 1000 hours of heater operation.
L. Every 2000 hours.
M. Every 5000 hours.
N. Not used.
O. Not used.
P. Not used.
Q. Refer to the Continental Motors Maintenance Manual and Continental Motors Service Bulletins.
R. At engine overhaul or every 4 years.
S. At engine overhaul.
T. Refer to the latest Woodard Governor Service Bulletin.
U. Every 3000 flight hours or 3 years, whichever occurs first.
V. Every ten years.
W. Every 2000 hours or 4000 landings, whichever occurs first.
NOTE: For gearboxes on which the overhaul limits have been exceeded, do the overhaul by no later than the next 300 hours or 400 landings or May 30, 2006, whichever occurs first.

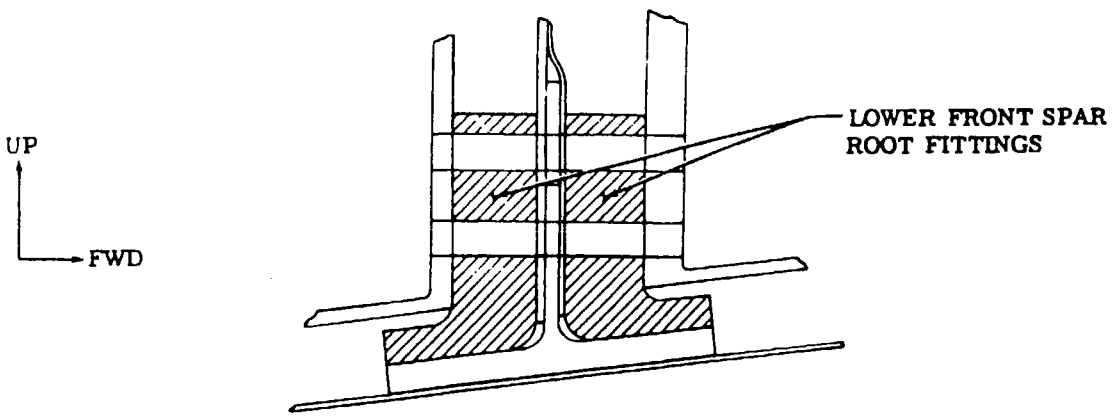
Figure 2-6. Inspection Chart (Sheet 1 of 3)

© Cessna Aircraft Company

INSPECTION AREA	HOLE DIA.	PART TO BE INSPECTED	PROBE DEPTH* IN INCHES	SIDE TO BE INSPECTED FROM
Area "A" - Two Holes Lower Front Spar Root Fittings Wing Station 28.81 46.55	.257 in.	Aft Fitting	Setting 1 .22 Setting 2 .48 Setting 3 .60	Aft
		Fwd Fitting	Setting 4 1.20 Setting 5 1.34 Setting 6 1.60	Aft

*Three probe settings are required to inspect each fitting.



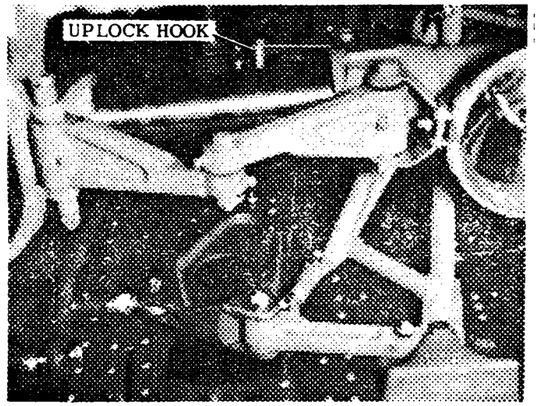


SECTION VIEW A-A

52221002
52221004

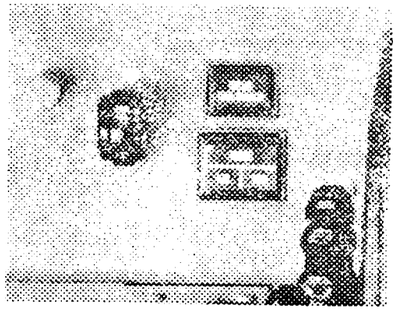
Figure 2-6J. Wing Inspection Area (Sheet 1)

Detail **M** CHECKING Nose Gear Uplock Engagement



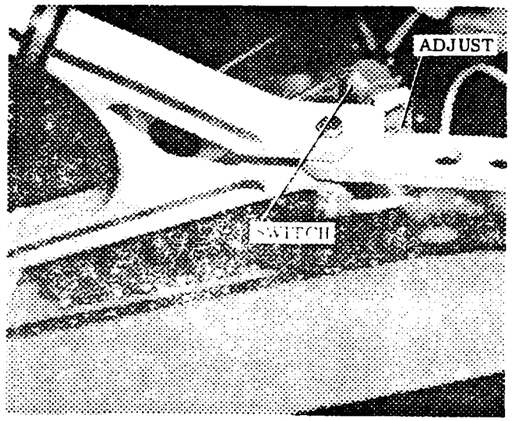
1. **Inspect nose gear uplock for proper engagement.**
2. **Uplock hook must be fully engaged with hook against the nose strut bolt and spacer engages and disengages freely with no binding.**
3. **Uplock hook must be engaged with . 003 to . 060 inches clearance between the spacer on the nose strut and surface of the hook.**

Detail **N** CHECKING Gear Down Indicator Light Switches for Proper Adjustment

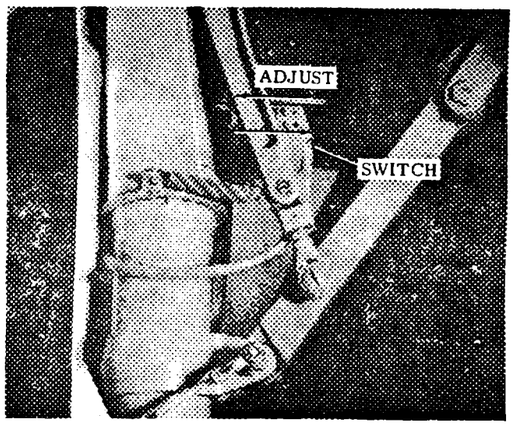


1. Retract landing gear approximately halfway.
2. Engage manual extension handle (see Detail B). Crank toward the down position and stop when green light comes on. Note the angular position of the manual extension handle.
3. Check applicable gear of illuminated light for being down and locked with the overcenter linkage overcenter.
4. Resume cranking toward the down position noting the number of turns required to reach the internal stop in the actuator.
5. **The number of turns required to reach the internal stop should not be less than 8 or more than 14 for the nose gear.** Not less than 4 or more then 8 on the main gear.

Detail **N-1** ADJUSTING Nose Gear Down Indicator Light Switch



Detail **N-2** ADJUSTING Main Landing Gear Down Indicator Light Switch



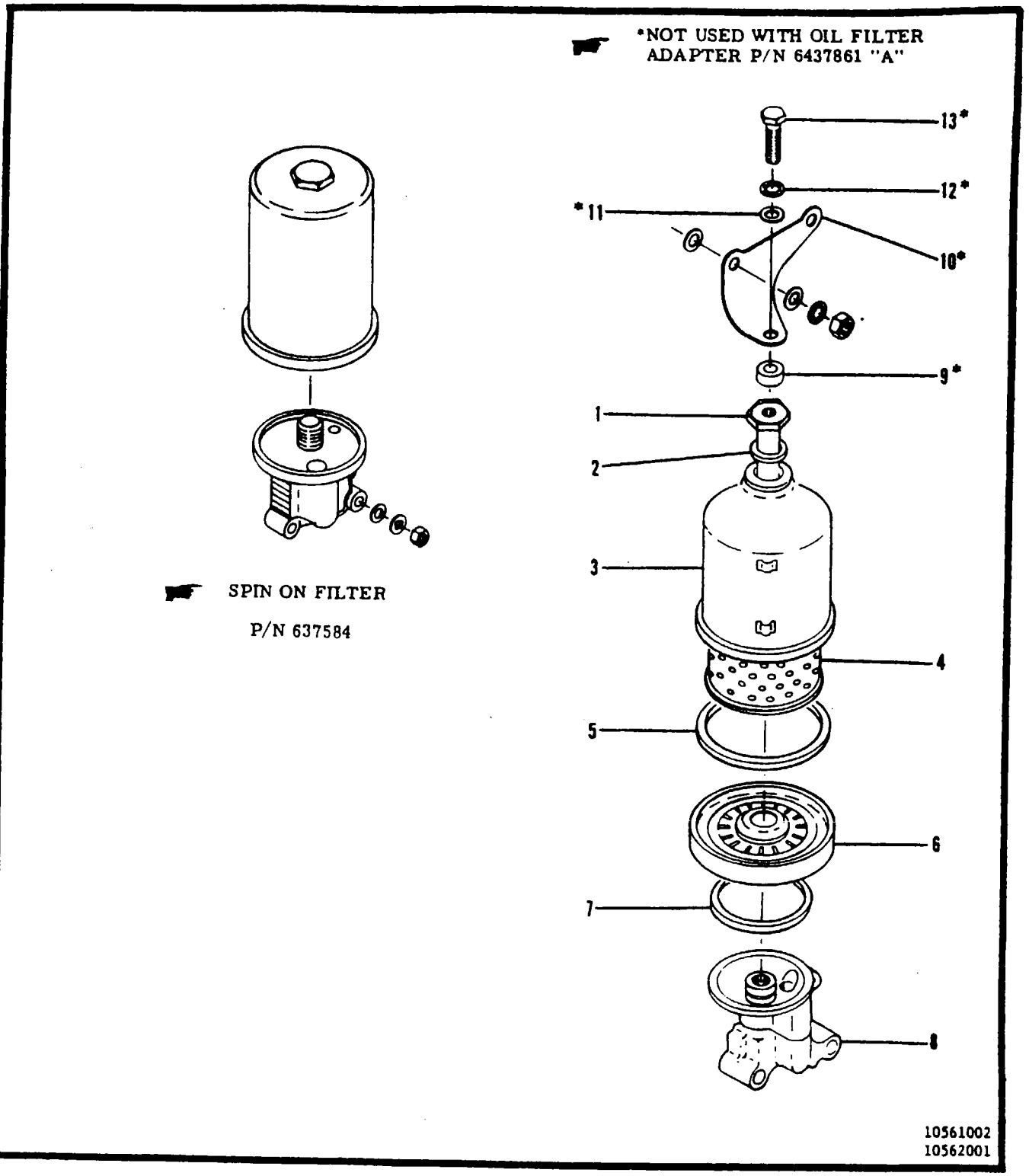
1. If the indicator lights do not illuminate within the required number of turns or the overcenter linkage is not overcenter when the lights illuminate make the following adjustments.
2. Adjust the nose gear down and locked indicator light switch by adjusting the switch actuating bolt. Turn bolt CCW to actuate switch farther from the internal stop inside the actuator. Turn bolt CW to actuate switch closer to the internal stop inside the actuator.
3. Adjust the main landing gear switches by repositioning. Loosen mounting screws and move switch towards the bellcrank to actuate switch farther from the internal stop in the actuator. Move the switch away from the bellcrank to actuate it closer to the internal stop in the actuator.

NOTE

After adjusting indicator switches check to see that gear indicator lights do not illuminate before gear is down and locked by overcenter linkage.

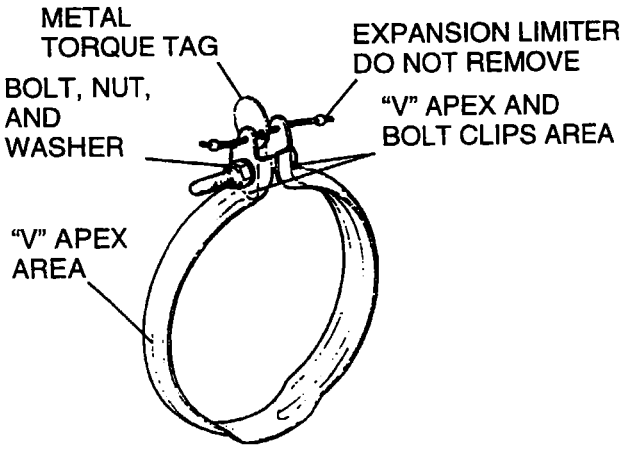
M5440P6025
M5440P6026
M5440P6027
M5440P6028

Figure 2-7. Nose and Main Landing Gear Rigging Inspection (Sheet 6)

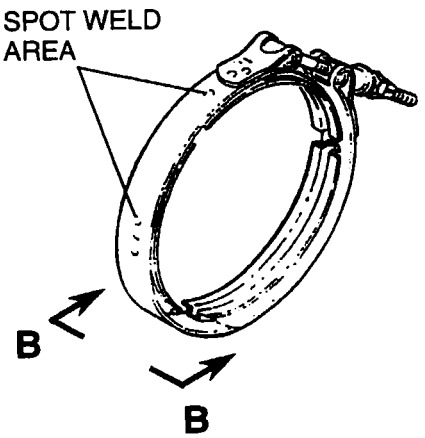


1. Bolt
2. Washer
3. Filter Case
4. Filter Element
5. Gasket (Case to Cover)
6. Cover
7. Gasket (Cover to Adapter)
8. Adapter
9. Spacer
10. Bracket
11. Washer
12. Lockwasher
13. Bolt

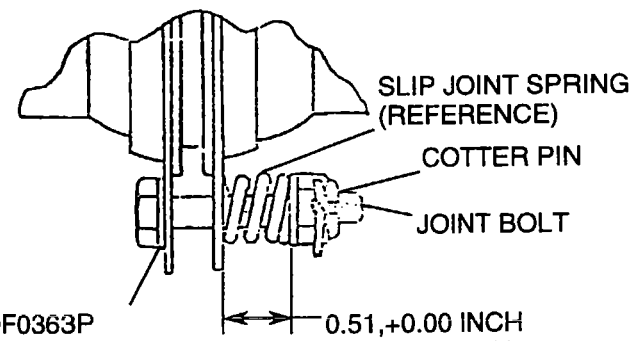
Figure 2-9. Oil Filter Assembly



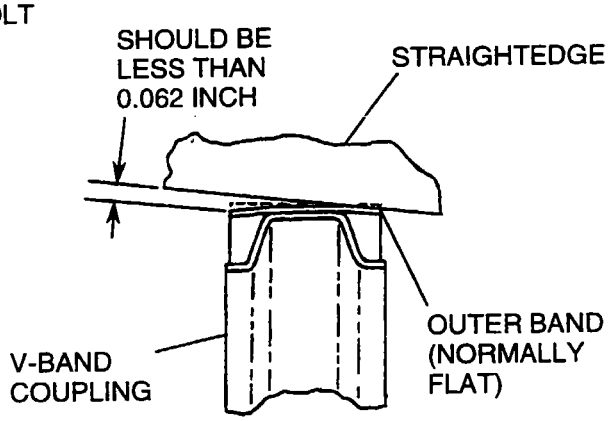
DETAIL F
ONE PIECE "V" BAND CLAMP



DETAIL D
MULTI-SEGMENT "V" BAND CLAMP



DETAIL E
TYPICAL EXHAUST JOINT
SPRING INSTALLATION



VIEW B-B
MULTI-SEGMENT "V" BAND
CLAMP OUTER BAND
DEFORMATION CHECK

C2655X1032
D2655X1031
E2655X1030

Exhaust System Inspection
Figure 2-11A (Sheet 3)

cy on local tower frequency.

(c) Contact local tower personnel and alert them of the proposed test.

(d) Place the ELT function selector switch in the ON position until the emergency tone is heard on the aircraft radio (not over 10 seconds). Immediately place the function selector switch in the ARM position. (Ensure that the transmitter has been deactivated.)

NOTE

After accumulated test or operation time equals 1 hour, battery pack replacement is required.

3. 100 Hours.

(a) Check ELT operation in accordance with procedure outlined under "Periodic".

NOTE

Check calendar date for replacement of battery pack. This date is supplied on a sticker attached to the outside of the ELT case and to each battery.

(b) Verify compliance with Cessna Avionics Service Letter AV74-10 and AV74-14.

4. 12 months.

Have your dealer check your ELT for the following:

(a) Power output (refer to Section 13).
(b) Carrier frequency.
(c) Modulation.
(d) Sweep.
(e) Sweep rate.
(f) Operation of "g" switch.
(g) Replacement of battery pack.
(h) Mounting security.

EXTREME WEATHER MAINTENANCE.

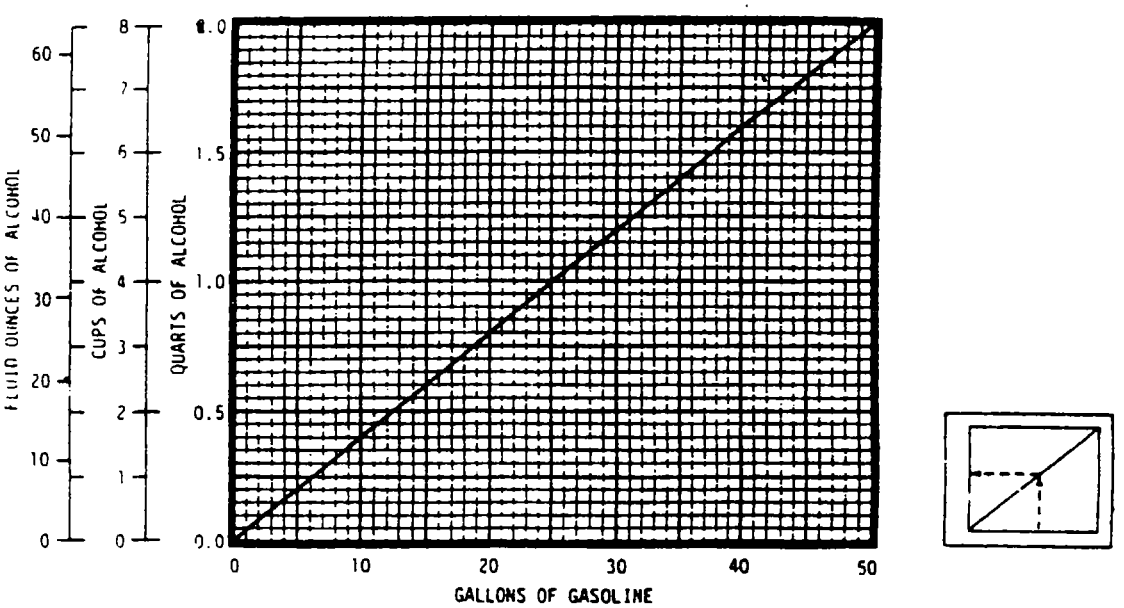
Hot Weather.

In hot weather, with a hot engine, fuel may vaporize at certain points in the fuel system. Vaporized fuel may be purged by setting the mixture control in IDLE CUT-OFF and operating the auxiliary fuel pump on "HI."

Engine mis-starts characterized by weak, intermittent explosions followed by puffs of black smoke from the exhausts are caused by over-priming or flooding. This situation is more apt to develop in hot weather, or when the engine is hot. If it occurs, repeat the starting routine with the throttle approximately one-half OPEN, the mixture control in IDLE CUT-OFF, and the auxiliary fuel pump switch OFF. As the engine fires, move the mixture control to full RICH and decrease the throttle to desired idling speed.

Engine mis-starts characterized by sufficient power to disengage the starter but dying after 3 to 5 revolutions are the result of an excessively lean mixture after the start. This can occur in either warm or cold temperatures. Repeat the starting routine but allow additional priming time with the auxiliary fuel pump switch on "LO" before cranking is started, or place the auxiliary fuel pump switch on "HI" immediately for a richer mixture while cranking.

If prolonged cranking is necessary, allow the starter motor to cool at frequent intervals, since excessive heat may damage the starter.



Alcohol - Fuel Mixing Ratio Chart

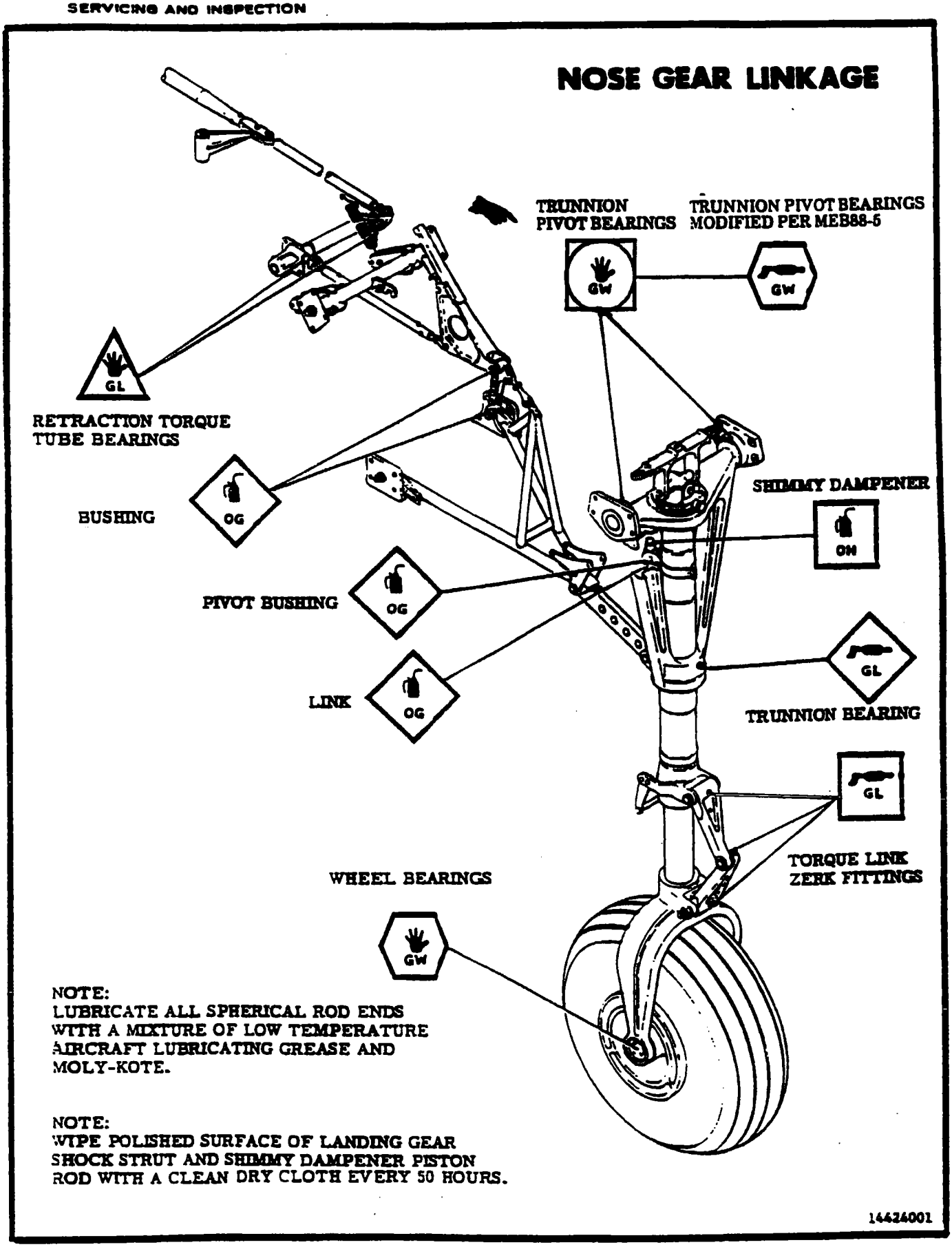


Figure 2-12. Lubrication Diagram (Sheet 6)

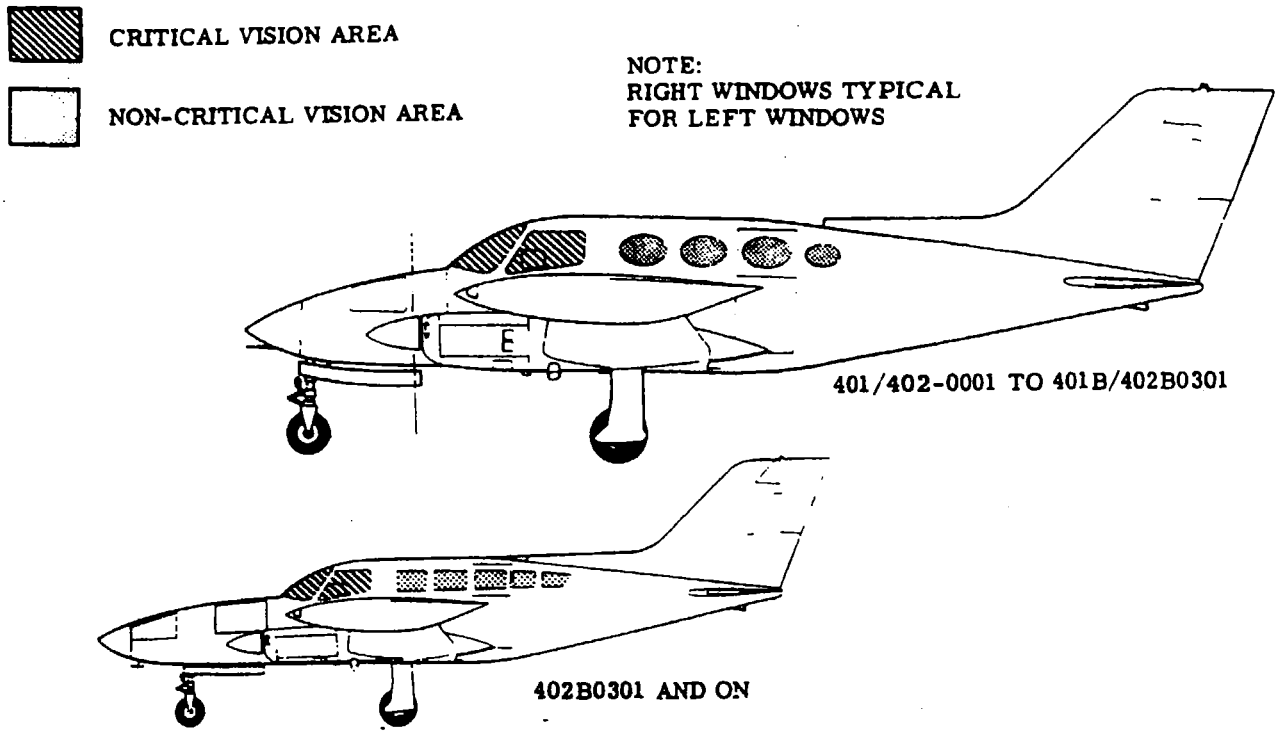


Figure 3-1. Critical Vision Diagram

FUSELAGE

Windshield and Windows.

Removal of Windshield (See figure 3-2).

a. Disconnect electrical wiring to magnetic compass, remove screw, nut and compass.

NOTE NOTE

The following instructions are given for the removal of the LH windshield only, the RH windshield is removed in the same manner.

b. Remove screws attaching center doubler (2) to retainer assembly (8) and remove doubler.
c. Remove upper doubler (5) by removing attaching screws.
d. Remove screws attaching LH lower doubler (14) to windshield retainer (13) and remove lower doubler from aircraft.
e. Remove windshield (7) from aircraft.

Installation of Windshield (See figure 3-2).

a. Apply seal tape (S1078-6, Minnesota Mining and Mfg. Co.) between all glass and metal and apply sealing compound (No. 5120C, Schnee-Morehead Chemicals, Inc.) along edges of outside doublers.
b. Position windshield half (7) in place and temporarily secure LH lower doubler (14) to windshield retainer (13) with several screws.
c. Place upper doubler (5) in position and temporarily secure with several screws.
d. Align slots in windshield half (7) with holes in retainer assembly (8) and secure center doubler (2) to retainer with attaching screws.
e. Finish securing lower doubler (14) and upper doubler (5) with attaching screws.
f. Install trim strip (12) and magnetic compass with attaching screws and nuts.

Removal and Installation of Pilot's Window (See figure 3-3).

a. Remove pilot's window upholstery trim (26) from inside of aircraft.
b. Loosen screws attaching upper doubler (23) to retainer (25).
c. Drill out rivets attaching skin (21) to window retainer (25) and remove skin.
d. Remove window (24) from window retainer (25) repairing and replacing as necessary.
e. On installation, apply sealing compound (No. 5120C, Schnee-Morehead Chemicals Inc., or equivalent) along

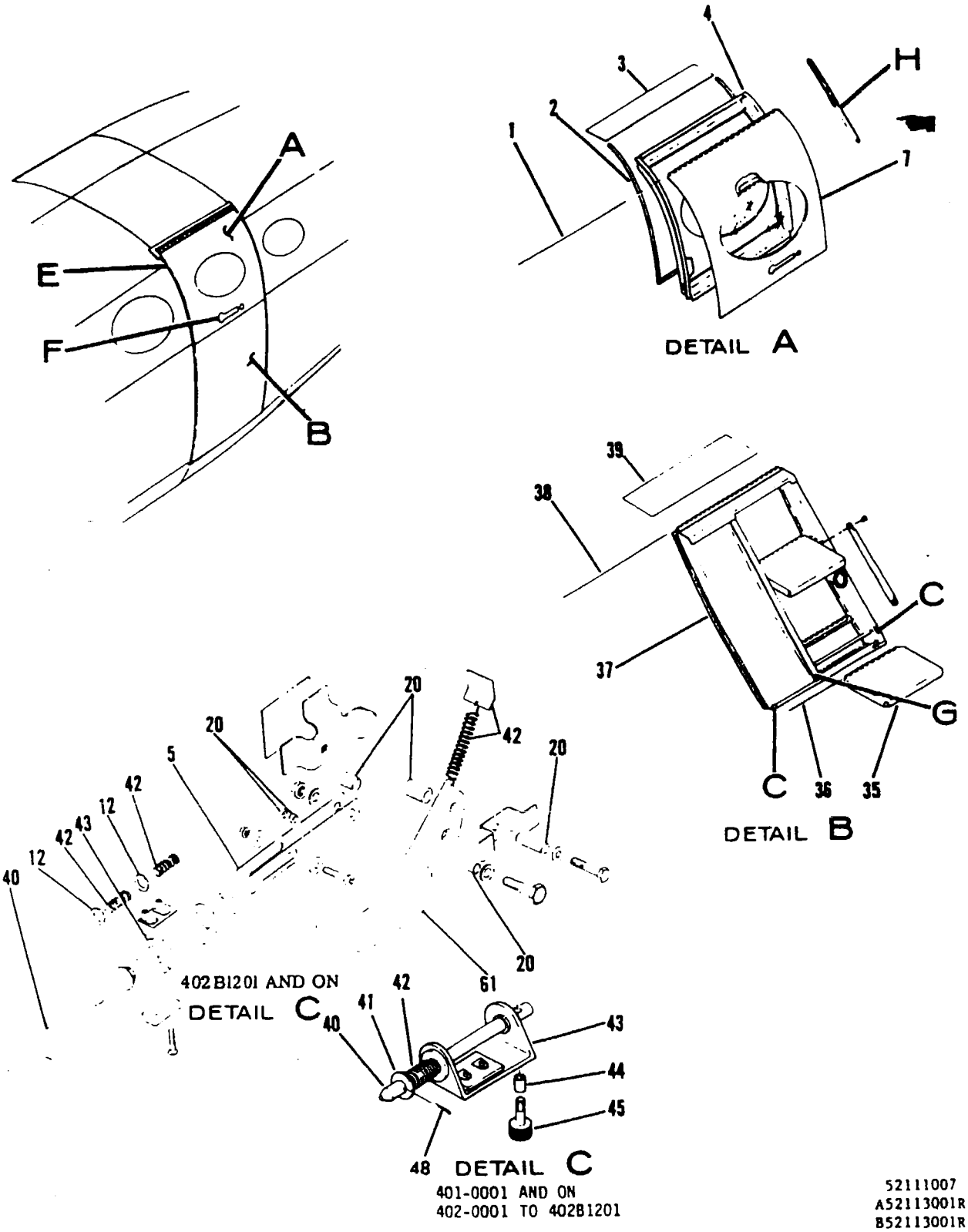
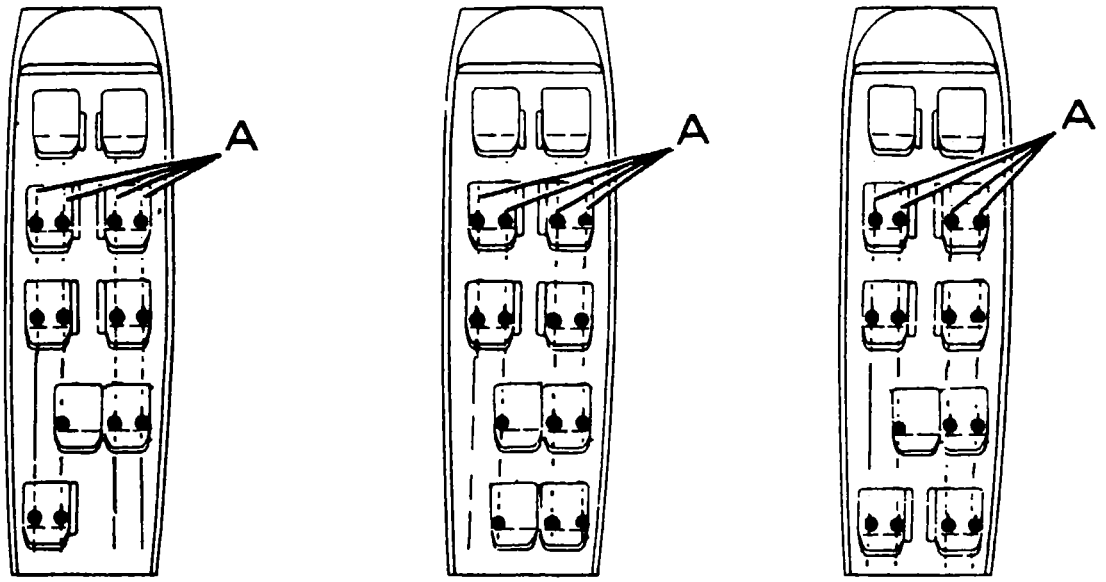


Figure 3-4. Cabin Door Installation (Sheet 1 of 4)



PASSENGER SEATING ARRANGEMENTS

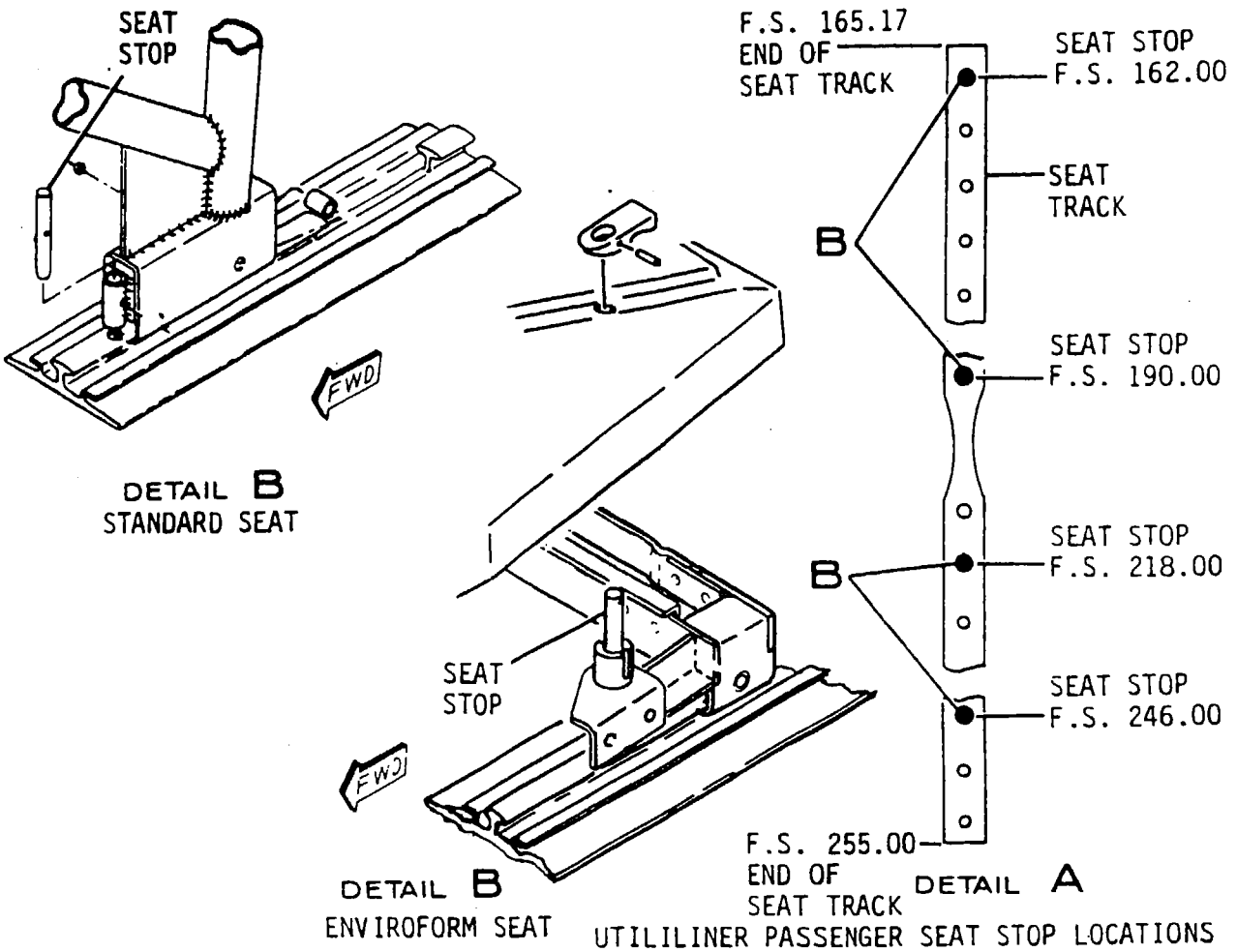


Figure 3-4C. Seating Arrangement Schematic and Seat Stop Locations (Sheet 5)

1. Install headliners by reversing removal procedures and bonding affected areas with cement (EC880, Minnesota Mining and Manufacturing Co., or equivalent).

Removal and Installation of Upholstery and Upholstery Trim (Model 402). (See figure 3-8.)

a. Remove pilot, copilot and passenger seats in accordance with seat removal procedures.
b. Remove headliner as follows:

NOTE

Removal of LH headliner is given, removal of RH headliner is the same.

1. Remove sun visor.

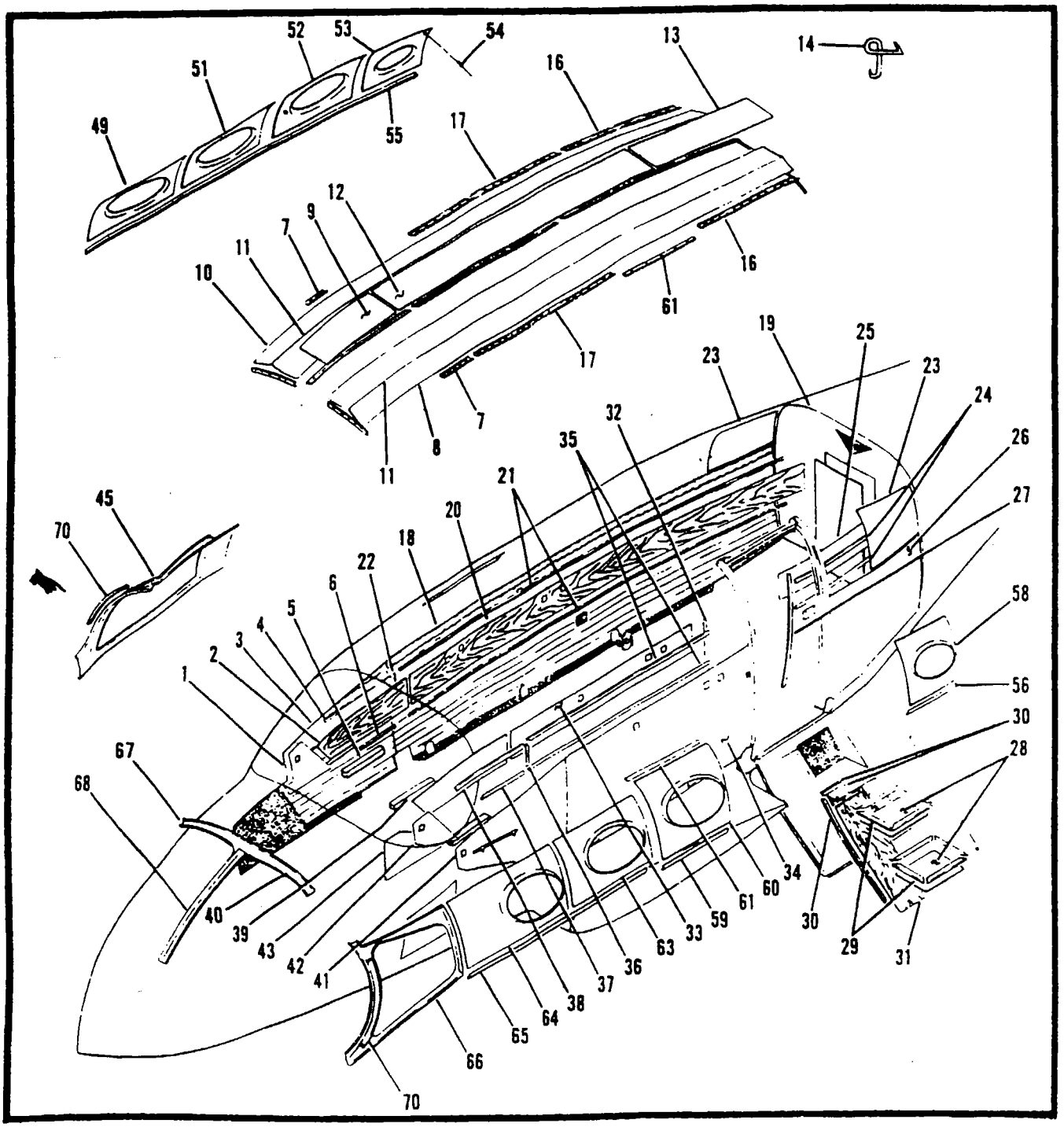


Figure 3-8. Upholstery and Upholstery Trim Installation (Sheet 1 of 3)

m. Refer to Section 2, jack aircraft in accordance with jacking procedures.

CAUTION

If both engines were removed, the aircraft is tail heavy. To balance the aircraft, place weight (such as shot bags or sand bags) in engine nacelles prior to jacking.

n. Remove padded supports from beneath cabin section, tail section and wings.
o. Place tailcone support beneath tailcone bumper.
p. If landing gear was removed, refer to Section 4 and install and rig landing gear.

NOTE

To facilitate rigging of landing gear, use external power source.

q. Refer to Section 9, install engine in accordance with installation procedures and connect flex cables (12, 13 and 14) to engine.

NOTE

To prevent serious binding, check flex cables for proper security during rigging of cables.

r. Remove support from beneath tailcone bumper.
s. Remove aircraft jacks.
t. Refer to Section 14, install batteries in accordance with installation procedures.
u. Check control cables for proper operation and correct directional travel.
v. Service aircraft and check for fuel leaks, especially at the fuel tank and wing connections.
w. Install wing root and fuel tank fairings.
x. Refer to Section 9, perform an engine operation check and observe engine controls and electrical equipment for proper operation.

Checking Wing Twist and Location of Thrust Line.

a. Refer to Section 16 for procedures for checking wing twist and location of thrust line.

Wing Locker.

The wing locker compartments are located just aft of the engine nacelle compartment in each wing. The wing locker door is located on the aft side of the wing and can be opened by a push type, flush mounted snap latch. The door locks are opened by the same key that operates the cabin door. Door stays are provided to support the door when it is open.

Removal of Wing Locker Door. (See figure 3-12.)

a. Open wing locker door.
b. Remove screw (7), washer (8) and spacer (9) from lower end of stop assembly (10).
c. Remove nuts (5), washers (2), stat-o-seals (3) and bolts (1) from door hinge (4) and remove door from nacelle.

Installation of Wing Locker Door. (See figure 3-12.)

Install wing locker door by reversing removal procedures.

Removal and Installation of Wing Locker Door Latch.

Removal and installation of wing locker door latch is not recommended unless replacement or repair is necessary. Use figure 3-12 as a guide for removal and installation.

Sealing.

The wing assembly is sealed in the areas where it is possible for fumes from the engine exhaust to enter the cabin. In the wheel well area, the landing gear door hinges are sealed with a fume boot over each hinge. The corners of the wheel well are sealed with Permagum and between the wheel well structure and upper wing skin, Polyfoam is glued in each opening. All wires and cables have grommets installed in the wing at the wing root area. All lightening holes in the wing root and on the rear spar in the forward flap area have fabric seals glued in place, each push rod in the flap area has a fume boot glued to the rear spar and tied around the push rod. Likewise, the landing gear push-pull tubes have a fume boot installed. The ailerons are sealed by a leather seal in a retainer fastened to the wing rib. The aileron cable seals, and the landing gear door hinge seals have a retainer ring to facilitate removal and installation. The fume boots installed on the push-pull tubes are tied to the push rod and glued to the wing rib or rear spar; should removal be necessary, the fume boots should be reglued in place with CES 1124, Type 4, Class 2.

CAUTION

After removal and installation of fume boots, check operation of push-pull tubes for interference.

STABILIZERS.

Removal of Vertical Stabilizer (See figure 3-13)

a. (See figure 3-14.) Remove fairings and stinger cap.
b. (See figure 1-2.) Remove access covers (12 and 22).
c. Place suitable support under tailcone.
d. Remove rudder in accordance with Section 7.
e. Remove rudder cables from vertical stabilizer pulley bracket.
f. Disconnect elevator cables from elevator bellcrank in accordance with Section 6.
g. Disconnect elevator downspring from elevator bellcrank.

upper bellcrank and slide both tubes outboard so they will not interfere with removal.

j. Disconnect manual extension disengage rod by removing cotter pin and clevis pin.

k. Disconnect the manual extension drive tubes by removing the three clevis pins, washers, and cotter pins; then slide outer shaft aft and inner shaft forward to disconnect.

l. Remove safetywire from bolts to be removed; then remove the four bolts and washers attaching reduction unit and actuator assembly to the aft bulkhead.

NOTE

When removing the motor, disconnect and tag all electrical wires at the quick-disconnects provided.

m. Remove switch brackets from actuator assembly. Do not disturb switch adjustments except to replace switches or brackets.

NOTE

If switches are to be replaced, tag wires before disconnecting.

n. Remove the two bolts, washers, and nuts attaching actuator assembly to forward bracket and bolt, washers and nut attaching actuator to side support.

o. Lift actuator assembly vertically and remove from aircraft.

p. Remove the lower bellcrank and spacer from end of sector shaft.

Installation of Landing Gear Actuator. (See figure 4-1.)

a. Install lower bellcrank on lower end of sector shaft.

NOTE

When installing lower bellcrank, align the index punch mark on the bellcrank with the chamfered spline on the sector shaft.

b. Position actuator assembly in position, aligning manual extension outer and inner shaft so they will mate.

NOTE

To facilitate installation, install all actuator attaching bolts before any bolts are tightened.

c. Install the two bolts attaching the actuator to the forward mounting bracket and bolt, washers and nut attaching actuator to side support.

d. Install the four bolts and washers attaching actuator assembly and reduction unit to bulkhead.

e. If motor was removed, connect the electrical wires at the quick-disconnect provided.

f. Install switch brackets with bolts, washers, and nuts.

CAUTION

Check switches thoroughly for proper operation. A faulty switch may cause damage to the landing gear actuator.

g. Tighten all nuts and bolts which were installed but not tightened.

h. Safetywire bolts attaching actuator to the bulkhead.

i. Connect manual extension drive tubes with clevis pins and washers, and safety with cotter pins.

j. Connect the manual extension disengage rod with clevis pin and safety with cotter pin.

k. Position upper bellcrank above actuator assembly, slide main gear drive tubes inboard and attach to bellcrank with bolts, washers, and nuts. Install bolts with their threaded ends UP.

NOTE

Main gear drive tubes must be installed with half-round side of end fitting upwards.

l. Place spacer and upper bellcrank on sector shaft.

NOTE

When installing upper bellcrank, align the index punch mark on the bellcrank with the chamfered spline on the sector shaft.

m. Insure that lower bellcrank and spacers are correctly in position and install bolt, caps, washer, and nut.

n. Attach nose gear drive tube to lower bellcrank with bolt, washer, and nut.

o. Connect main landing gear drive tubes to idler bellcranks with bolts, spacers, and nuts.

p. Connect forward push-pull tube to fork bolt with bolt and nut.

q. Perform an operational check of landing gear, checking especially that limit switches are correctly adjusted and landing gear is correctly rigged.

r. Install cabin floor panel and access hole cover on underside of fuselage beneath landing gear actuator.

s. Install rear carpet and seats removed for removal of actuator.

t. After making sure landing gear is DOWN and

l. Position brake hose and clamps on bolt (15) and connect upper and lower torque links at hinge point using bolt (15), spacer (19), washers (17 and 21) and nut (20).

CAUTION

Steel washers (17) must be properly located under bolt head and nut.

NOTE

Make sure all new and existing washers at torque link hinge point are in place and free in movement; tighten bolt and nut to insure a side play of .004 to .020 is maintained between torque link braces. It may be necessary to position the washers in different locations to maintain alignment and side play simultaneously.

m. Insure brake line is clear of all structure and secure.
n. Safety bolt (15) and nut (20) with cotter pin (23).
o. Install wheel and brake assembly in accordance with installation procedures.
p. Torque all nuts properly and safety.
q. Lubricate upper and lower torque link fittings.
r. Inflate landing gear strut in accordance with Section 2.
s. Connect landing gear door and cycle landing gear to make sure door fits properly and all lines are clear.
t. Remove aircraft from jacks.

Main Landing Gear Doors.

The main landing gear is equipped with wheel well doors and strut doors. Each strut door, pivoting on a continuous hinge located at its outboard end, is operated by a push-pull rod attached to the main landing gear strut. Each wheel well door, pivoting on forged aluminum hinges located at its inboard end, is operated by a bell-crank and push-pull tube, which is connected to the landing gear retracting linkage. The operating mechanism is so arranged that the wheel well door is closed when the main gear is either fully retracted or fully extended.

Removal of Main Landing Gear Doors (See figure 4-6).

a. Remove strut door as follows:
1. Disconnect door link rod from strut by removing nut and bolt.
2. On aircraft 401/402-0001 to 401/402-0138, remove cotter pins in the ends of hinge pin; remove the hinge pin and outboard door.
3. On aircraft 401/402-0138 and on, remove nine screws (30) securing door to structure; remove door.
b. Remove wheel well door as follows:
1. Jack aircraft and engage manual extension crank. Operate toward the up position until the main landing gear door opens sufficiently to remove link from door.
2. Disconnect the door link tube from actuator arm.
3. Remove wheel well door by removing nuts, washers and bolts attaching hinge arm to door.
c. If desired, remove hinge arm as follows:
1. Remove lower wing root fillet and hinge covers by removing attaching screws.
2. Remove hinge arms from brackets by removing nuts, washers, spacers and bolts.

Installation of Main Landing Gear Door (See figure 4-6).

a. If hinge arms were removed, attach to brackets with bolts, spacers, washers and nuts. Install hinge covers and lower wing root fillet with attaching screws.
b. Install wheel well door as follows:
1. Jack aircraft and engage manual extension to release tension on gear retraction system.

NOTE

The pull required to move door actuator arm from stop must be measured at a right angle to the arm. The tool illustrated in figure 4-11 can be made to facilitate this measurement.

12. Adjust the length of both inboard drive tubes (17 and 21) as necessary to obtain a force of 25 ±10 pounds required to move door actuator arms from stops (5) in the down position.
13. Actuate gear electrically to the UP position and adjust the up limit switch as necessary to obtain a force of 25 ±10 pounds and a maximum difference of 10 pounds from the down position required to move the door actuator arms (6) from stop (5) in the up position.

NOTE

If the up limit switch is readjusted recheck the minimum turn requirement in Step 7.

i. Adjust side brace lock links (9) as follows:
1. With landing gear in the down position, adjust end fitting (8) so that lower side link (11) and upper side link (12) are held firmly in overcenter position when side brace lock link (9) is firmly overcenter.
2. During retraction check for clearance between bellcrank (10) and push-pull tube (2). It may be necessary to add or subtract washers to provide clearance.
3. Manually "Break" the lock link from its overcenter position and move the landing gear to a position five to six inches inboard from the down and locked position, then release. The landing gear must free fall and lock when released from this position.

NOTE

Lengthen side brace lock link end fitting (8) in 1/2 turn increments until the gear will not free fall down and locked.

4. On aircraft 401-0001 to 401A0054 and 402-0001 to 402A0033, shorten lock link end fitting (8) in 1/2 turn increments until gear will free fall to down and locked position.
5. On aircraft 401A0054, 402A0033 and On, shorten adjusting screw (22) in small increments until gear will free fall to down and locked position. Double safety-wire screw in adjusted position.
6. Adjust both main landing gear side brace lock links in this manner.

NOTE

After the preceding steps have been completed the main landing gear retracting system is rigged from the limit switches through the door actuator bellcranks, and both side brace lock links are adjusted at the landing gear. The following procedure is to rig the retraction system from the door actuator bellcranks to the side brace lock links.

j. Make the following preliminary preparations:
1. Remove uplocks (3) and uplock push-pull tubes (13) by removing attaching bolts.
2. Adjust the length of outboard drive tubes (15) to 1/4 inch from as short as possible and install.

NOTE

Bolt (43, figure 4-7) must be installed with head of bolt forward.

3. Disconnect push-pull tube (2) from fork bolt (4).
4. Screw fork bolts (4) into torque tubes (14) as far as possible.
5. Operate landing gear actuator to DOWN position.
6. With landing gear DOWN, adjust length of push-pull tubes (2) so that rod end holes align with the holes in fork bolts (4), then shorten one turn. Operate landing gear actuator toward the UP position far enough to permit installation and install push-pull tubes (2). INSTALL WITH THE COLLAR-END OF THE HI-SHEAR RIVETS POINTING DOWN FOR CLEARANCE.

CAUTION

The bolt installing push-pull tube (2) to the fork bolt (4), must be installed with the bolt head forward.

NOTE

The preceding preparations will result in incomplete retraction, thus eliminating the possibility of damage to the wing structure caused by retracting too far.

k. Operate landing gear to the UP position and observe the highest position reached by the gear during retraction and the amount of drop-off.
l. Adjust the highest position reached by the gear during retraction as follows:
1. Lengthen fork bolt (4) in half-turns to increase the highest position during retraction.

NOTE

Detail A illustrates this adjustment. Lengthening the fork bolt increases dimension "B." the longer dimension "B" is, the higher the gear will retract.

2. The axle dust cover should make light contact with the corrugated reinforcement at the top of wheel well, then drop down a specified distance (drop-off). Refer to step m.
m. Adjust the amount of drop-off as follows:
1. Determine whether drop-off is too little or too much. The minimum and desired amount of drop-off, measured at the center of the wheel, is 1/8 inch. The maximum is 1/4 inch.
2. If drop-off is too much, lengthen outboard drive tube (15) in half-turn increments until drop-off is correct.
3. If drop-off is too little, shorten outboard drive

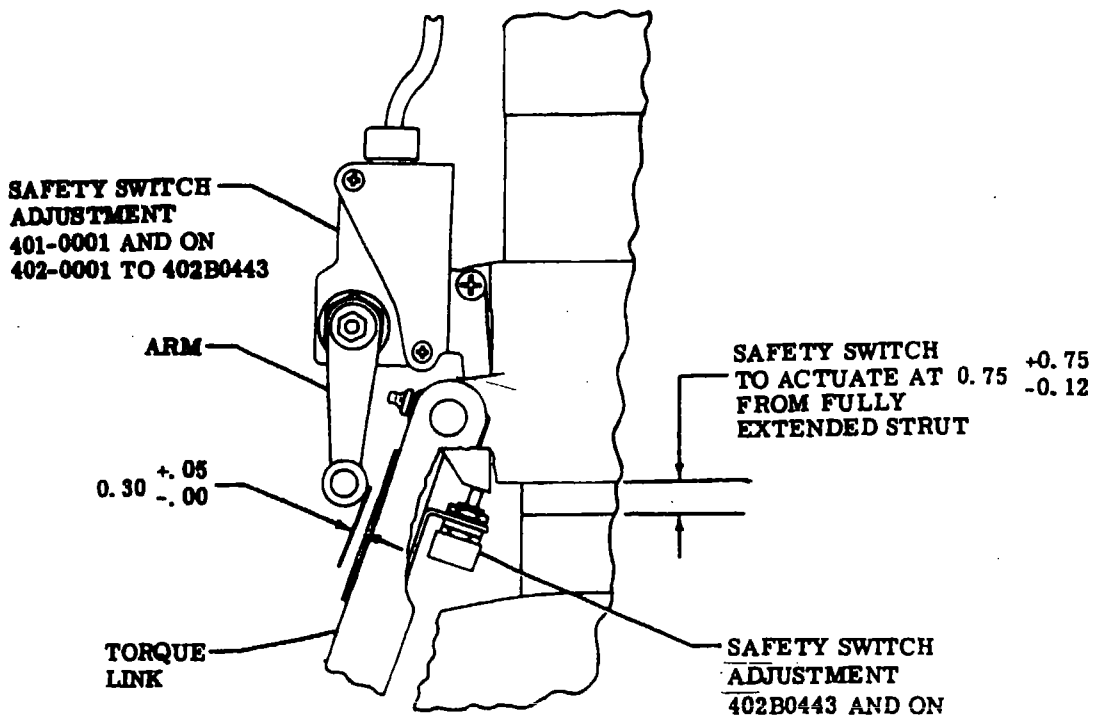
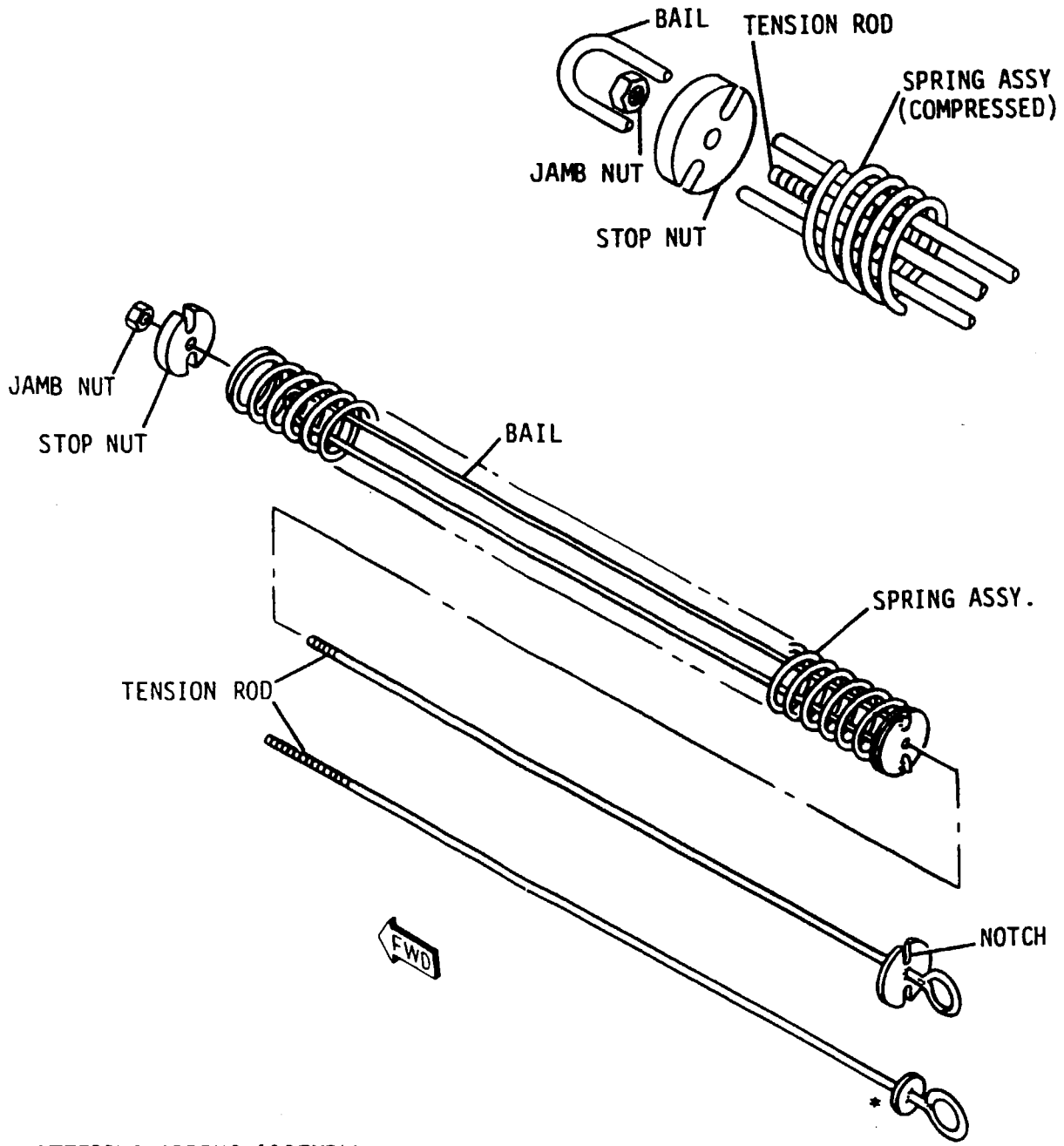


Figure 4-15A. Adjustment of Landing Gear Safety Switch



*STEERING SPRING ASSEMBLY WITH THIS TENSION ROD DOES NOT REQUIRE COMPRESSION TO TURN ROD

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Figure 4-19A. Steering Spring Assembly

Checking the Wearing of Brake Discs (Goodyear Optional).

Brake disc wear can be determined while brake is on aircraft as follows:

a. Apply and release brakes twice.
b. Apply 600 psi pressure to the brake and hold.
c. Measure distance between inside face of brake housing and pressure plate. If this distance measures 0.234 inch or greater, remove brake for inspection.

Inspection of Brake Assembly Parts (Goodyear Optional). (See figure 4-21A.)

Inspect all parts for cracks, nicks, scratches, damaged threads, or other damage. Discard all cracked parts and all parts having damaged threads.

a. Inspect rotating discs (33) as follows:
1. Inspect relief slot terminal holes and drive tang radii for cracks. Discard cracked discs.
2. Inspect disc for wear. Discard disc when minimum thickness across unchipped mix becomes 0.143 inch or less, or if mix is worn to 0.015 inch thick at any location on either face of disc.

NOTE

As much as 25 percent of mix may be lost through chipping, provided mix loss is scattered.

3. Inspect discs for dishing. Discs can be dished a maximum of 0.015 inch if they are to be used with other dished discs.
4. Discard discs that are 0.125 inch or more out of round.
5. Inspect disc tangs for battering. Discard disc when one or more tangs becomes less than 0.615 inch in width.
6. Inspect disc for shrinkage. Discard discs that have shrunk to an inside diameter of 7.750 inches or less.

b. Inspect stationary disc (34) as follows:
1. Inspect relief slot terminal holes and key slot corners for cracks. Discard cracked discs.
2. Inspect disc for wear. Discard disc when worn to a thickness of 0.146 inch or less on aircraft 401B0001 and On and 402A0001 to 402B0501 or when worn to a thickness of 0.170 inch or less on aircraft 402B0501 and On.
3. Inspect for dishing. Disc can be dished a maximum of 0.015 inch if it will be used with other dished discs.
4. Check inside diameter of disc for size by ascertaining that it can be installed on the torque tube without interference.
5. Check key slots for wear and battering. Discard disc with keyslots less than 0.590 inch or greater than 0.640 inch in width.
6. Inspect disc for shrinkage. Discard disc if shrunk to an inside diameter of 7.10 inches or less measured between ends of keyslot tangs.

c. Inspect pressure plate subassembly (32) as follows:
1. Inspect pressure plate subassembly for cracks in corners of keyslots. Discard if cracked.
2. Inspect subassembly for thickness. Replace if worn to 0.225 inch or less in thickness measured between wear pad rivets.
3. Inspect wear pads (41) for thickness. Replace all pads if any rivet is sheared, any pad is cracked, or if one or more pads are worn to a width of 2.75 inches or less, as shown in figure 4-22B. Install new pads in accordance with applicable instructions.
4. Discard if keyslot width is less than 0.615 or greater than 0.640 inch.

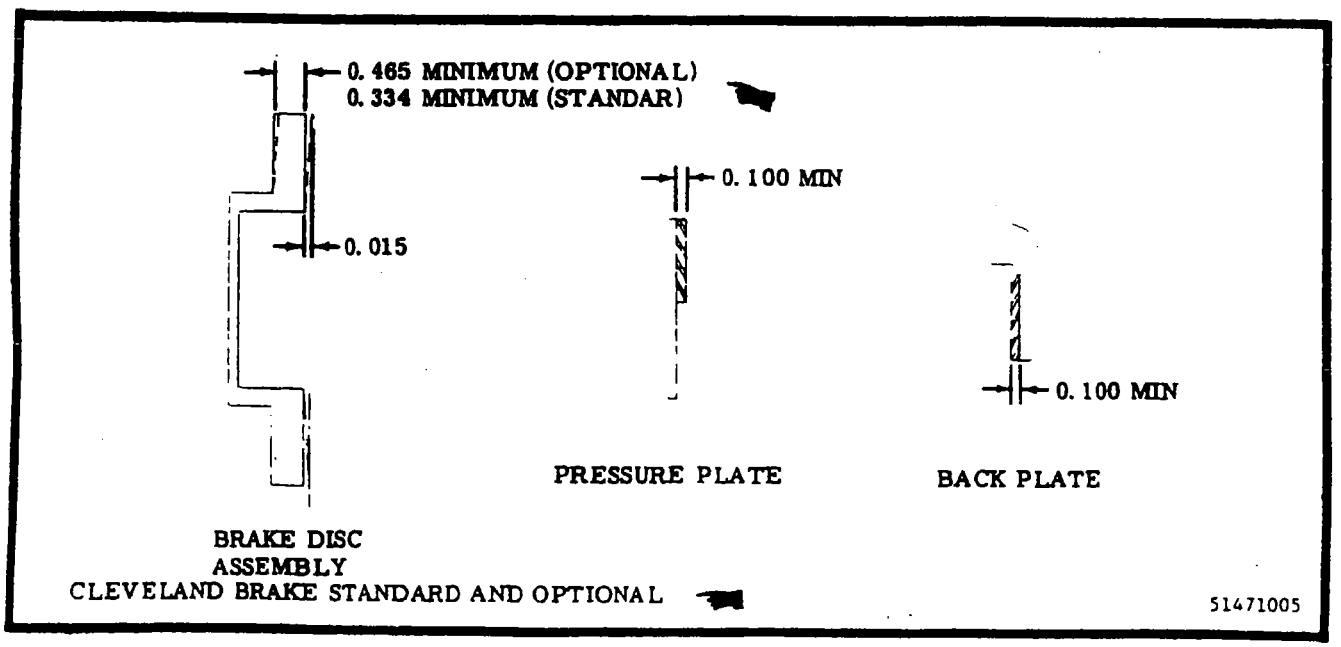


Figure 4-22A. Brake Wear Limits

Removal of Aileron. (See figure 5-4.)

a. When the left aileron assembly is being removed, move aileron trim tab to extreme up position and remove cotter pin, nut, bolt and spacer from aileron trim tab push-pull tube (6).
b. Lower flaps and disconnect aileron push-pull rod assembly (10) by removing bolt and washer.
c. Disconnect bonding strap from aileron by removing attaching screw.
d. Remove bolts attaching aileron to hinge assembly.
e. To remove aileron, move aileron to the rear, clearing hinge assembly, and slightly inboard until outboard hinge pin (9) is free of bearing.

Installation of Aileron.

NOTE

If rigging was correct prior to aileron removal, and the aileron push-pull rod end adjustment was not disturbed, it should not be necessary to rerig the aileron control system. If the push-pull rod end adjustment was disturbed, adjust rod ends to provide 9.65 inches span measuring from center to center of rod end attach holes.

a. Installation of ailerons is the reversal of the removal procedure.

NOTE

After installation of aileron, check travel, alignment, and rerig, if necessary, in accordance with rigging procedures.

AILERON TRIM TAB.

The all metal aileron trim tab attaches to the left aileron and operates by a push-pull tube connected to the aileron trim tab actuator. The aileron trim tab must be installed when static balancing the left aileron.

Removal of Aileron Trim Tab. (See figure 5-4.)

a. Position aileron trim tab to full UP and disconnect bonding strap by removing screw.
b. Remove cotter pin (12), nut (5) and bolt (7) attaching push-pull rod (6) to aileron trim tab.
c. Remove cotter pins in hinge half and remove hinge pin.
d. Remove aileron trim tab from aileron.

Installation of Aileron Trim Tab. (See figure 5-4.)

a. Install aileron trim tab by reversing removal procedures.

NOTE

Check aileron trim tab travel, alignment and rerig, if necessary, in accordance with rigging procedures.

AILERON CONTROL SYSTEM.

The ailerons are actuated by the rotational movement of either control wheel. The actuation of ailerons is accomplished by cable assemblies which are attached to the control column interconnect assembly and routed through the fuselage to the fuselage bellcrank located just forward of the rear spar. From the fuselage bellcrank, cable assemblies are then routed through each wing to a wing bellcrank assembly where a push-pull tube is connected to the aileron. An aileron trim system is provided and is routed directly through the fuselage and left hand wing to the aileron trim tab actuator. An aileron trim control knob is provided on the control pedestal.

Trouble Shooting Aileron Control System.

TROUBLE	PROBABLE CAUSE	CORRECTION
LOST MOTION BETWEEN CONTROL WHEEL AND AILERON	Cable tension too low.	Adjust cable tension in accordance with rigging procedures.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys or wing bellcranks.	Install cables correctly. Check cable guards.
RESISTANCE TO CONTROL WHEEL ROTATION	Cable tension too high.	Adjust cable tension in accordance with rigging procedures.
	Wing bellcrank bolts over-torqued.	Loosen bolts.

Removal of Aileron Trim Control Knob and Pointer Assembly (See figure 5-7).

a. Remove pedestal access covers.
b. Remove gear assembly (6) and support bracket (3) from pedestal assembly.
c. Remove pointer assembly (1) by removing four attaching screws (13).
d. Relieve tension on aileron trim control system by loosening turnbuckles.
e. Disengage roller chain (10) from sprocket (9) and slide sprocket from shaft (7).
f. Remove roll pin (12) from gear (11) and slide shaft (7) from pedestal assembly.

Installation of Aileron Trim Control Knob and Pointer Assembly (See figure 5-7).

a. Installation of aileron trim control knob and pointer assembly is the reversal of the removal procedure.
b. Check rigging and rig in accordance with Rigging of Aileron Trim Control procedure if necessary.

Rigging of Aileron Trim Control System (See figure 5-5).

NOTE

See figure 1-2 and remove cable access panel (18), floorboard access panels (73 and 77) and wing access panel (27).

a. Remove safety clips if installed from turnbuckles (45 and 49) and adjust cable tension to 10 ±3 pounds.

NOTE

Cable tension should be adjusted when ambient temperature is 60°F to 90°F. Allow aircraft temperature to stabilize for a period of 4 hours.

b. Move aileron trim control system to neutral by rotating control knob.
c. Check to make sure ends of chain (30) are equal in length from sprocket on actuator (27) and ends of chain (52) are equal in length from sprocket (53).
d. (See figure 5-7.) Adjust aileron trim indicator to neutral by removing screw and raising bracket (3) to allow pointer (1) to move to the center.
e. (See figure 5-5.) Remove cotter pin (22), nut (23) and bolt (24) attaching push-pull tube (25) to aileron trim tab.
f. Align aileron with trailing edge of wing and place trim tab in neutral.
g. (See figure 5-5.) Adjust trim tab actuator (27) so that hole in push-pull tube (25) is in line with hole in trim tab horns.
h. (See figure 5-5.) Install bolt (24), nut (23) and cotter pin (22).
i. Loosen stop blocks (42), move trim to full down travel. Refer to Section 1 for travel. Slide outboard stop block against outboard rib assembly at station 129.24 and tighten stop block.

NOTE

Stop blocks (42) should be installed so that bottom cable passes through bushing and top cable is clamped by stop blocks.

j. Move trim tab to full up travel. Refer to Section 1 for travel. Slide inboard stop block against inboard rib assembly at station 118.24 and tighten stop block.
k. For deflection and friction forces check, refer to Section 2.

WARNING

Insure that aileron tab moves in the proper direction when operated by trim knob.

l. Install all access panels.

(19) by removing cotter pin, nut, washer and bolt.
b. Remove cotter pin (17) from each end of hinge pin (18).
c. Remove trim tab by removing hinge pin (18).

Disassembly and Assembly of Elevator Trim Tab.

Disassembly and assembly of the trim tab is limited to the removal and installation of the trim tab horn. To remove the elevator trim tab horn, proceed as follows:
a. Remove nuts, washers and screws securing trim tab horn to trim tab.
b. Install elevator trim tab horn by installing screws through elevator trim tab horn and secure with washers and nuts.

Installation of Elevator Trim Tab. (See figure 6-2.)

a. Reverse the elevator trim tab removal procedures.
b. Check the elevator trim tab for proper operation and travel in accordance with the rigging of elevator trim tab control system.

ELEVATOR CONTROL SYSTEM.

The elevators are operated by the fore and aft movement of the control column. The quadrant assembly is attached to the control column by links which are attached to a swivel bearing on the control column. The elevator cable assemblies and adjustable turnbuckles are attached to the quadrant assembly and then routed down under the floor panel and through the fuselage by pulleys to the elevator bellcrank in the tail section. A push-pull tube connects the bellcrank to the elevator arm which is attached to the elevator torque tube. The elevator travel stops, consisting of bolts mounted in a bracket just aft of the bellcrank are provided for the recommended adjustment. The elevator bob weights are attached to the quadrant assembly torque tube and are provided for elevator counterbalance weight. The elevator down spring is attached to the bellcrank and is provided for better elevator balance during flight.

Troubleshooting Elevator Control.

TROUBLE	PROBABLE CAUSE	CORRECTION
LOST MOTION BETWEEN CONTROL WHEEL AND ELEVATORS	Cable tension too low.	Adjust cable tension in accordance with elevator rigging instructions.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
	Aft push-pull tube disconnected.	Connect push-pull tube.
	Cable disconnected.	Connect cables.
RESISTANCE TO ELEVATOR CONTROL MOVEMENT	Cable tension too high.	Adjust cable tension in accordance with elevator rigging instructions.
	Pulley binding or rubbing.	Replace binding pulleys. Provide clearance if rubbing pulley brackets or cable guards.
	Cable not in place on pulleys.	Install cable correctly.
	Bent elevator or hinge.	Repair or replace elevator or hinge.
	Defective bearing or bushing in control column or interconnect assembly.	Replace defective bearing or bushing.
INCORRECT ELEVATOR TRAVEL	Elevator bellcrank stops incorrectly adjusted.	Adjust in accordance with elevator rigging instructions.
CORRECT ELEVATOR TRAVEL CANNOT BE OBTAINED BY ADJUSTING BELLCRANK STOPS	Elevator cables incorrectly rigged.	Rig cables in accordance with elevator rigging instructions.
	Control wheel is not rigged in the neutral position.	Rig cables in accordance with elevator rigging instructions.
	Bob weights contacting or rubbing.	Provide clearance if rubbing or contacting other equipment.

Removal of Elevator Trim Actuator Assembly. (See figure 6-3.) (401A0001 and On, 402A0001 To 402B1001.)

a. Remove seats and center carpets in accordance with Section 3.
b. (See figure 1-2.) Remove floorboard center access panels, as necessary, to gain access to the elevator trim control cable turnbuckles.
c. Remove locking clips (6) from turnbuckles.
d. Loosen and disconnect turnbuckles (7 and 10) from the control cable.
e. Remove screws attaching actuator assembly (8) to support (9) and disconnect electrical wiring (27).
f. Remove actuator assembly (8) and control cable (5).

Disassembly and Assembly of Elevator Trim Actuator Assembly. (401A0001 and On, 402A0001 To 402B1001.)

a. Loosen screws (12, 17 and 19) and remove top and side covers (16 and 20).
b. Loosen screws (22 and 24) and remove cover assembly (21), cover assembly (23) and cable guard (26).
c. Slide clutch assembly (25) from housing (18) and remove chain assembly (15) from clutch assembly sprocket and sprocket (14).
d. Remove control cable (5) from clutch assembly (25).
e. Remove motor assembly (11) from housing (18).
f. Clean component parts by wiping with a clean cloth saturated with a suitable solvent.
g. Check clutch assembly sprocket and sprocket (14) for broken, chipped and worn teeth.
h. Do not attempt to repair damaged or worn parts of the actuator assembly. Discard all defective parts and install new parts during reassembly.
i. Adjust clutch assembly (25) to slip at 25 ±3 inch-pounds.
j. (See figure 2-10.) During reassembly, lubricate clutch assembly (25).
k. Install motor assembly (11) in housing (18).
l. Install control cable (5) on clutch assembly (25).

NOTE

Clutch assembly cable drum and control cable must be free of grease and oil and control cable must make 3 full wraps around the cable drum.

m. Install chain assembly (15) on sprocket (14) and clutch assembly sprocket and slide clutch assembly into housing (18).
n. Install cover assembly (23), cover assembly (21) and secure with screws (22 and 24).
o. Install cable guard (26) in housing (18).
p. Install top and side covers (16 and 20) and secure with screws (12, 17 and 19).

Installation of Elevator Trim Actuator Assembly. (See figure 6-3.) (401A0001 and On, 402A0001 To 402B1001.)

a. Install actuator assembly (8) on support (9) and secure with screws.
b. Connect electrical wiring (27).
c. Connect turnbuckles (7 and 10) to elevator trim control cable.
d. Install locking clips (6) in turnbuckles.
e. Rig elevator trim control system in accordance with Section 6, except cable tension should be 18 ±3 pounds.

NOTE

Cable tension should be adjusted when ambient temperature is 60°F to 90°F. Allow aircraft temperature to stabilize for a period of 4 hours.

f. (See figure 1-2.) Install removed floorboard access panels.
g. Install carpets and seats in accordance with Section 3.

Removal of Elevator Trim Actuator Assembly. (See figure 6-3.) (402B1001 and On.)

a. Place a suitable support under tailcone.
b. Remove tailcone access.
c. Remove center floor panels as necessary to gain access to the elevator trim control cable turnbuckles.
d. Disconnect electrical connector from actuator (32).

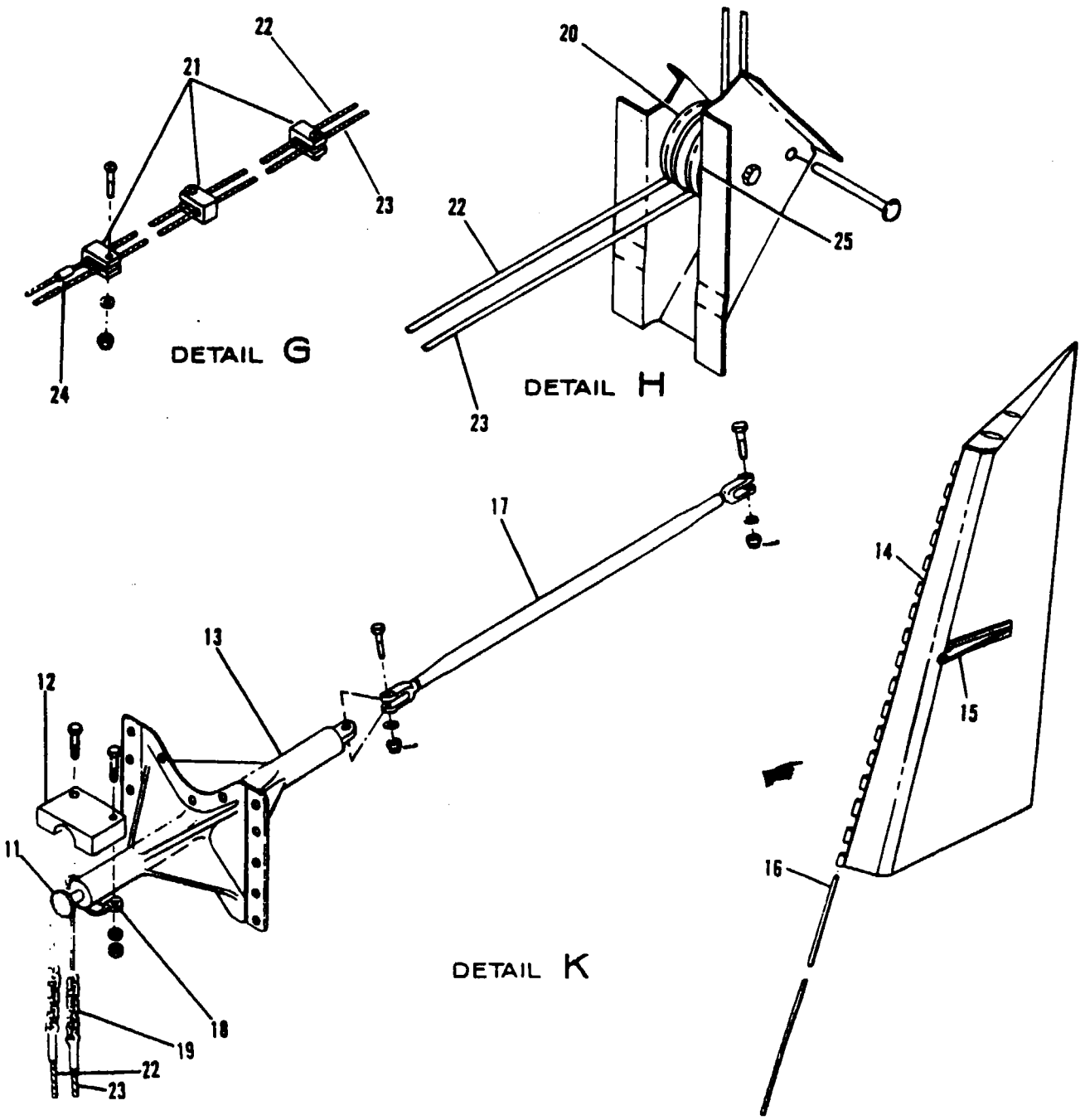
NOTE

The electric trim actuator may be removed from the mount without disturbing cable tension if only the actuator is being replaced.

e. Remove actuator (32) from mount (28) by removing bolts (11).
f. If actuator mount is being removed proceed as follows:
1. Remove cable guard pins from actuator mount.
2. Disconnect turnbuckle (7) and remove cable from actuator capstan.
3. Remove mount (28) from support (30) by removing screw, washer and lockwasher.

Installation of Elevator Trim Actuator Assembly. (See figure 6-3.) (402B1001 and On.)

a. Position actuator mount (28) to support (30) and secure with screws, washers and lockwashers.
b. Install elevator trim cable (31) as follows:
1. Position and hold elevator trim tab in normal full nose down position.



G52613001
H54613004
K51334001
K54613004

11. Sprocket
12. Chain Guard
13. Trim Actuator
14. Trim Tab
15. Trim Tab Horn
16. Hinge Pin
17. Push-Pull Rod
18. Clamp
19. Chain
20. Pulley
21. Stop Block
22. Right Cable
23. Left Cable
24. Bushing
25. Pulley

Figure 7-3. Rudder Trim Control System (Sheet 2 of 2)

Installation of Flap Actuator Assembly. (See figure 8-2.)

a. Place flap actuator assembly near floorboard access hole and attach wires from wire bundle to respective limit switch and remove tags.

NOTE

Insure that terminals of the limit switches are covered with insulated sleeving and wiring is secure and clear of all control cables and moving parts.

b. Insert actuator assembly through floorboard access hole and install with screws and nuts.
c. On aircraft 401/402-0001 to 401A/402A-0001 connect the electrical connector to the position transmitter (25).
d. Engage chains with sprockets (12) and rig flap control system in accordance with rigging procedure.
e. Install fuselage and floorboard access panels.
f. Install carpets and cabin seats in accordance with Section 3.

Rigging of Flap Control System. (See figure 8-1.)

The flap control system should be rigged using a 24 volt external power source, with a two-position momentary switch, attached to the flap motor wires so the flaps may be observed while being operated during rigging.

CAUTION

Use caution while operating flaps with an external power source, as the limit switches are inoperative.

a. Remove center seats and center carpet in accordance with Section 3.
b. Remove access hole covers from aft cabin floor, aft side of wheel wells, aft underside of wings and underside of fuselage beneath flap motor.
c. Check length of inboard push-pull rod on each inboard flap and adjust to 10.45 inches.

NOTE

Length of push-pull rods is measured between the centerline of rod and bolt holes.

d. Check length of interconnecting push-pull tubes and adjust, if necessary, to the following dimensions:

Outboard interconnecting push-pull tubes - 33.35 in.
Center interconnecting push-pull tubes - 23.80 in.
Inboard interconnecting push-pull tubes - 23.80 in.

e. Adjust remaining push-pull rods from bellcranks to scissors so that flaps fit evenly when in the UP position.
f. (See figure 8-2.) Adjust limit switches as follows:
1. Loosen limit switches (21 and 26) and position them where they will be activated by the cam (18).
2. With the flaps in the UP position on 401/402-0001 to 401A/402A-0001 aircraft, flex flap switch bracket (3) aft to disengage cam (18) from pinion gear (16). Rotate cam to a point where it actuates the UP limit switch (26). (upper switch) and re-engage cam with pinion gear by releasing tension on bracket (3).
3. With the flaps in the UP position on 401A/402A-0001 and ON aircraft, loosen setscrew (6), rotate cam (18) to a point where it actuates the UP limit switch (26), and then tighten setscrew (6).

g. With flaps in the UP position, check for correct positioning of the flap chains on sprockets and, if necessary, adjust as follows:
1. (See figure 8-2.) Engage right chain with forward sprocket so that approximately 3-1/2 links extend around bottom side of sprocket.
2. Engage left chain with aft sprocket so that 3 links extend around the top side of sprocket.
3. (See figure 8-1.) Connect RH extend cable (25) to the long end of chain (top side of forward sprocket) and route over pulleys (32 and 29) to bellcrank (17).
4. Connect RH return cable (27) to the short end of chain (bottom side of forward sprocket) and route over pulleys (31 and 30) around outboard pulley (28) and back to bellcrank (17).
5. Connect LH extend cable (34) to the long end of chain (bottom side of aft sprocket) and route over pulleys (39 and 37) to bellcrank (42).
6. Return cable (35) to the short end of chain (top side of aft sprocket) and route over pulleys (40 and 38) around outboard pulley (43) and back to bellcrank (42).
7. With flaps secured in UP position, tighten turnbuckles until the slack is out of cables. Temporarily rig both return cables (35 and 27) with 185 lbs. tension. Final tension on return cables should be 190 to 210 pounds on aircraft 401/402-0001 to 401/402-0250 and 200 to 250 pounds on aircraft 401/402-0250 and ON with flaps in the UP position.

NOTE

At this point, recheck flap position as to being in proper position with respect to wing trailing edge. Should further adjustment of push rods be necessary, this must be accomplished before final setting of cable tension.

8. Remove tape or other means of securing flaps in the UP position and lower flaps a few degrees at a time, observing for any unusual tension build-up in the cables or binding of the push-pull tubes and push rods, until flaps reach their bottom position as measured with a protractor. Tolerance for flap travel is plus one degree, minus zero degrees.
9. Rig both extend cables (25 and 34) with 75 lbs. of tension. (Final tension of extend cables, with flaps down, should be 75 to 95 lbs. tension with the differential tension between left and right cables not exceeding 25 lbs.).

NOTE

Cable tension should be adjusted when ambient temperature is 60°F to 90°F. Allow aircraft temperature to stabilize for a period of 4 hours.

TROUBLE	PROBABLE CAUSE	CORRECTION
CLIMBING TO ALTITUDES ABOVE 12,000 FEET, ENGINE QUITS WHEN POWER REDUCED (CONT)	Fuel vaporization (cont.).	Engines may quit when throttles are brought back for power adjustment after leveling off. If this occurs, lean mixture and engines will pick up again.
MANIFOLD PRESSURE BETWEEN ENGINES VARIES WITH ALTITUDE	1000 feet critical altitude tolerance between engines.	Advance one throttle in front of the other to keep manifold pressure the same.
MANIFOLD PRESSURE VARIES WITH AIRSPEED. INCREASE WHEN AIRSPEED INCREASED, DECREASE WHEN AIRSPEED DECREASED.	Waste-gate stuck or closed.	Adjust throttles after airspeed has increased from climb speed.
NOTE The left engine manifold pressure will not normally increase or decrease more than right engine because of location of left engine air intake duct.		
MANIFOLD PRESSURE VARIES WITH ENGINE RPM	Waste-gate OPEN, RPM increase manifold pressure slight decrease. RPM decrease manifold pressure slight increase.	Normal turbocharged engines characteristic.
MANIFOLD PRESSURE VARIES WITH INCREASING OR DECREASING FUEL FLOW	Waste-gates CLOSED.	Fuel flow increase the mass flow of exhaust increased, turbine turns faster more induction air furnished engine resulting in manifold pressure increase.

ENGINE COWLS.

Removal and Installation of Engine Cowls. (See figure 9-1.)

NOTE

This removal procedure is for the LH engine cowling. RH engine cowling is removed in a similar manner.

a. Release fasteners retaining cowl doors (2 and 6) to upper cowl assembly (1).
b. Release fasteners retaining upper cowl assembly (1) to nacelle structure and remove upper cowl assembly (1) from aircraft.
c. Remove screws attaching lower portion of cowls (2 and 6) to engine beam assembly.
d. Remove screws retaining nose cap assemblies (4 and 5) to forward engine beam assembly.
e. Remove screws securing nose cap assemblies (4 and 5) together and remove nose cap assemblies.
f. Install engine cowling by reversing removal procedures.

COWL FLAPS.

Removal and Installation of Cowl Flaps. (See figure 9-2.)

a. Position cowl flaps to OPEN.
b. Disconnect lower link (8) from bracket (9) on door assembly (10) by removing attaching nut and bolt.
c. Remove hinge pin (11) from hinge halves and remove door assembly (10) from engine nacelle.
d. Install door assembly (10) by reversing removal procedures.

NOTE

Secure hinge pin (11) with cotter pins upon installation.

Removal and Installation of Cowl Flap Torque Tube Assembly. (See figure 9-2.)

a. Position cowl flap to OPEN.
b. Disconnect turnbuckle (7) from inboard bellcrank assembly (2) by removing nut, bushing, washers and screws attaching upper link (10) to bellcrank assembly (2).

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NOTE

Align propeller spinner with cowling nose cap contours to assist in determining correct engine position.

CAUTION

For each 0.125 inch of spacer thickness added, it will be necessary to add the next size longer length engine mount bolt.

c. Secure engine mount in place and safety in accordance with engine mount installation procedures.
d. Recheck all clearances, especially in the exhaust stack area.

Installation of Engine Mount. (See figure 9-4.)

a. Install engine mounts by reversing removal procedures.

NOTE

Aircraft 401B/402B0200 and On and aircraft equipped with J9613-58 engine mounts do not require heat shields (9). When replacing an engine mount with J9613-58 engine mount, replace all existing mounts on same engine.

b. Torque bolt (8) to 300, +50, -0 inch-pounds.

NOTE

Lock washers (7) with a break tool. This break tool is available from Cessna Dealers' Organization (Tool No. SE608).

Engine Exhaust System.

The exhaust system consists of a riser pipe attached to the exhaust port of each cylinder. The three risers at each bank of cylinders are joined together into a common pipe to form a stack assembly. The inboard and outboard stack assemblies are routed aft through flexible bellows to a manifold header which is mounted to the turbine side of the turbocharger. On aircraft 401/402-0201 and On, the risers are joined together by slip joints and are routed aft through slip joints to the manifold header. From the exhaust port of the turbine, the tailpipe routes the exhaust out of the aircraft through the lower nacelle. The exhaust port of the waste-gate assembly is routed overboard from the system when not required at the turbine. The waste-gate is automatically controlled by the turbo control system.

Removal of Exhaust System. (See figure 9-4A.)

a. Remove engine cowling in accordance with removal procedures.
b. If exhaust temperature system (optional equipment) is installed, disconnect probes.
c. Remove shields (2 and 15) by removing attaching screws.
d. Disconnect exhaust assembly (1) from bellows assembly (3) on aircraft 401/402-0001 thru 401B/402B0199 or from slip joint (18) on aircraft 401B/402B0201 and On by removing cotter pins, nuts, bolts, washers and springs. Route exhaust assembly from engine.
e. Disconnect exhaust assembly (1) from bellows assembly (3) on aircraft 401/402-0001 thru 401B/402B0199 or from slip joint (18) on aircraft 401B/402B0201 and On by removing cotter pins, nuts, bolts, washers and springs. Remove exhaust assembly (1) as a unit by removing four nuts, and washers from each of the three exhaust risers. Do not attempt to separate exhaust assembly to remove.
f. Disconnect tube assemblies (11) from manifold header (10) by loosening clamp. Route tube assembly (11) through canted bulkhead and from engine nacelle.
g. (See figure 9-14.) Remove hose (11) from air box (5) and adapter.
h. Remove shield (7) from tube assembly (6) by loosening two clamps.
i. Disconnect tube assembly (6) from manifold header (10) by loosening clamp. Route tube assembly (6) through canted bulkhead and from engine nacelle.
j. Remove overboard exhaust tube (9) by removing clamp (22), support (21) and v-band coupling (23). Route overboard exhaust tube (9) from engine nacelle.
k. Remove manifold header (10) as follows:
1. (See figure 9-12.) Disconnect line assemblies (25, 26 and 27) from actuator (28).
2. (See figure 9-14.) Remove turbocharger in accordance with removal procedures.
3. Remove turbocharger support brace by removing attaching bolts.
4. Route manifold header from engine nacelle.

Removal and Installation of Exhaust Slip Joint Seals and Expansion Ring. (See figure 9-4A.)

a. Remove exhaust system in accordance with removal procedures.
b. Remove seal and expansion rings.

NOTE

Prior to installing slip joint ensure it is to proper dimensions as outlined in SK421-40.

c. Install new expansion ring and seal in accordance with SK421-40.

CAUTION

During replacement or when a new seal has been installed, ensure the slip joint has been expanded properly, the attach spring properly compressed, and the slip joint forward and aft ends are fitted properly. An improperly fitted slip joint will allow hot gases to escape causing structural damage to the aircraft.

d. Install three guide rods (7) on left mounting plate (4) with three attaching screws (1).
e. Assemble control levers, spacers, guides, etc., on hub and guide rods as illustrated.

NOTE

Do not lubricate the parts to be assembled on the hub. These parts must remain dry for proper operation of the control friction lock.

f. Install right mounting plate (14) with three screws (1), attaching mounting plate to guide rods (7).
g. Install the two lugs (36) with the spacers (37), bolts (40) and nuts (41).
h. Install screws (37) in lugs (36) and terminals (30) on screws (37), then with terminals (30) located on spacer lugs, install pins (39), washers (31), and cotter pins (32).
i. Adjust forward screw (33) on the gear warning switches (35) so that when the throttle levers (27) are CLOSED, the switches (35) are closed.

Installation of Control Quadrant. (See figure 9-6.)

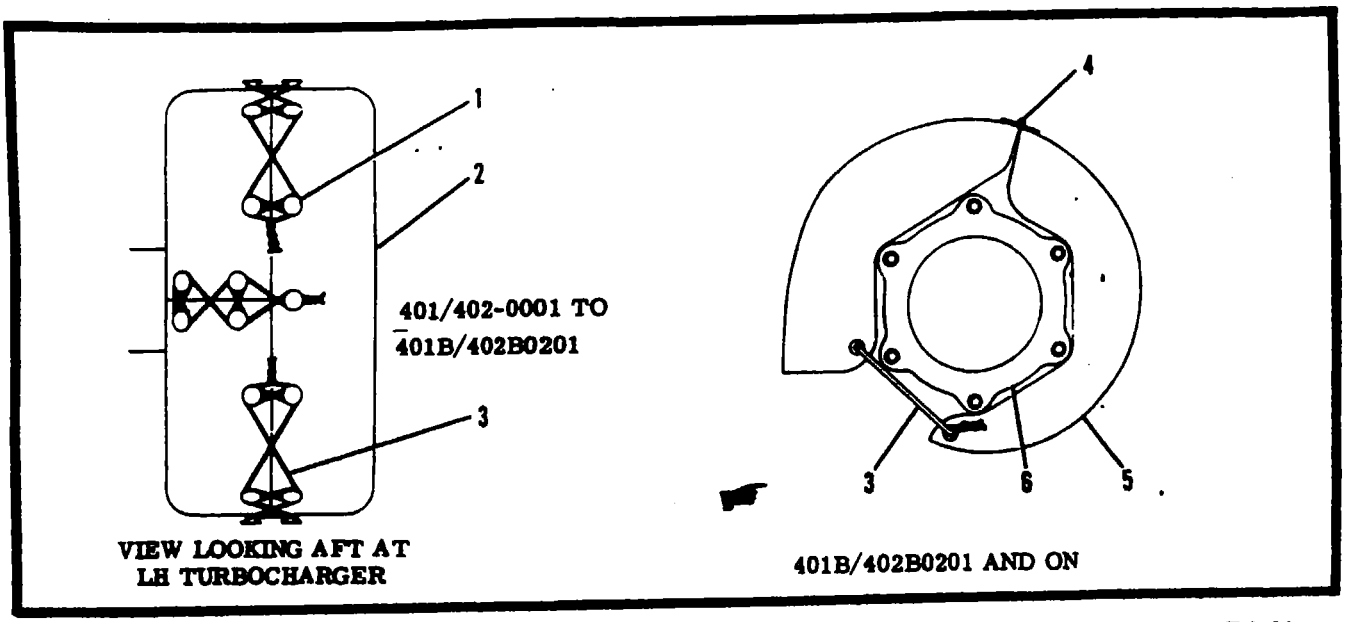
Reverse the removal of control quadrant procedure.

FUEL INJECTION SYSTEM.

The fuel injection is a simple, low pressure system of injecting fuel into the intake valve port in the cylinder head. It is a multi-nozzle, continuous flow type which controls fuel flow to match engine airflow. Any change in throttle position, engine speed or a combination of both, causes changes in fuel flow in the correct relation to engine airflow. A manual mixture control and a flow gage, indicating metered fuel pressure, are provided for precise leaning at any combination of altitude and power setting. The continuous flow system uses a typical rotary vane fuel pump. There are no running parts in this system except for the engine-driven fuel injection pump.

Trouble Shooting the Fuel Injection System.

TROUBLE	PROBABLE CAUSE	CORRECTION
ENGINE WILL NOT START AND NO FUEL FLOW GAGE INDICATION	No fuel to engine.	Check tank fuel level.
	Mixture control improperly rigged.	Check mixture control for proper rigging.
	Engine not primed.	Auxiliary pump switch in PRIME position.
	Selector valve in wrong position.	Position selector valve to MAIN TANK position.
ENGINE WILL NOT START WITH FUEL FLOW GAGE INDICATION	Engine flooded.	Re-set throttle, clear engine of excess fuel, try another start.
	No fuel to engine.	Loosen one line at nozzle. If no fuel shows, with fuel flow on gage, replace fuel manifold valve.
ROUGH IDLE	Nozzle restricted.	Remove nozzles and clean.
	Improper idle mixture.	Adjust fuel-air control unit in accordance with adjustment procedures.
POOR ACCELERATION	Idle mixture incorrect.	Adjust fuel-air control unit in accordance with adjustment procedures.
	Unmetered fuel pressure too high.	Lower unmetered fuel pressure.
	Worn linkage.	Replace worn elements of linkage.
ENGINE RUNS ROUGH	Restricted nozzle.	Remove and clean all nozzles.
	Improper mixture.	Improper pump pressure, replace.
LOW FUEL FLOW GAGE INDICATION	Restricted flow to metering valve.	Check mixture control for full travel. Check for clogged fuel filters.



1. Wrap Button
2. Insulation Blanket
3. 0.032 Monel Wire
4. Hinge
5. Shield
6. Turbine

Figure 9-13. Turbocharger Insulation Installation

g. Install oil pressure lines by reversing removal procedures.

Removal and Installation of Waste-gate Actuator Oil Control System Lines and Hoses. (See figure 9-12.)

a. Disconnect and remove hose assembly (12) from ratio controller (7) and scavenger pump.
b. Disconnect and remove line assemblies (21 and 23) from ratio controller (7) and absolute controller (22).
c. Disconnect and remove line assembly (26) from absolute controller (22) and waste-gate actuator (28).
d. Disconnect and remove line assembly (25) from waste-gate actuator (28) and elbow.
e. Disconnect and remove hose assembly (9) from oil cooler and elbow.
f. Install lines and hoses by reversing removal procedures.

Removal and Installation of Turbocharger Return Oil Lines and Hoses. (See figure 9-12.)

a. Disconnect and remove hose assembly (16) from crankcase fitting (15) and drain line (1) by removing two bolts attaching drain line to turbocharger. Remove clamp and route from engine nacelle.
b. Install drain line (1) and hose assembly (16) by reversing removal procedures.

OIL PRESSURE ADJUSTMENT.

The oil pressure relief valve, located on the oil pump is adjustable to increase or decrease oil pressure. Refer to figure 9-12A and adjust pressure to 30 to 60 psi.

TURBOCHARGER INSULATION.

On aircraft 401/402-0001 to 401B/402B0201, the turbocharger insulation installation consists of three individual blankets which are wrapped around the turbocharger turbine and held in place with Monel wire. Each blanket is made from material which consists of high temperature insulation sandwiched between two thin jackets of quilted stainless steel. Small holes are provided on the inside of each blanket to provide the necessary breathing. These blankets, when installed, will contain the high temperatures which are emitted by the turbocharger turbine. On aircraft 401B/402B0201 and On, the turbocharger is insulated by a shield. The shield consists of two shield halves hinged together which are fitted around the turbocharger turbine and held in place with Monel wire.

Removal and Installation of Turbocharger Insulation. (See figure 9-13.)

CAUTION

Extreme care should be taken when removing, installing, or working near the turbocharger blankets to prevent puncturing the stainless steel jackets.

a. Cut and unlace the Monel wire securing the turbocharger insulation blankets in place.
b. Carefully remove the three insulation blankets from the turbocharger turbine.

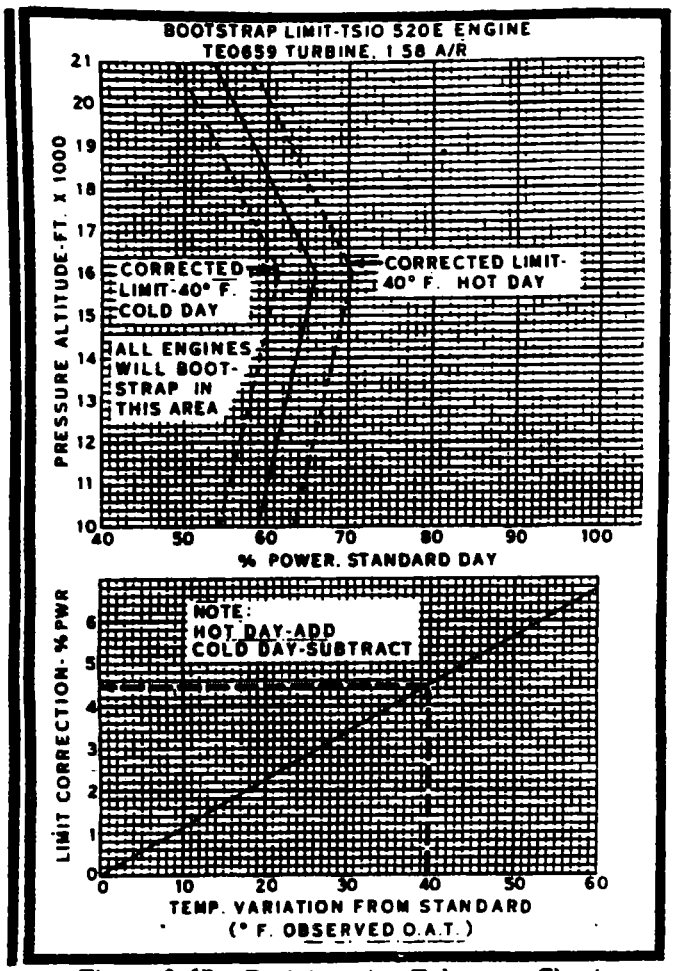


Figure 9-17. Bootstrapping Tolerance Chart

The absolute pressure controller controls the maximum turbocharger compressor discharge pressure (34.5 ±0.5 in. Hg. to critical altitude, approximately 16,000 feet). On 401/402-0001 to 401A/402A-0001 aircraft, the rate-of-change controller controls the rate at which the turbocharger compressor discharge pressure will increase. The pressure ratio controller begins to function at critical altitude (approximately 16,000 feet) and controls the maximum turbocharger compressor discharge pressure at a ratio of 2.2 times nacelle ambient pressure. On 401A/402A-0001 and ON aircraft, a relief valve is incorporated in the turbocharger compressor and will be actuated if the intake manifold pressure exceeds 36.5 in. Hg.

b. Operation. The waste-gate actuator is spring-loaded to position the waste-gate to the normally open position when there is not adequate oil pressure in the waste-gate actuator power cylinder during engine shutdown. When the engine is started, oil pressure is fed into the waste-gate actuator power cylinder through the capillary tube. This automatically fills the waste-gate actuator power cylinder and lines leading to the controllers, which are blocked by normally closed metering and poppet valves. As the oil pressure builds up in the waste-gate actuator power cylinder, it overcomes the force of the waste-gate open spring, closing the waste-gate. When the waste-gate begins to close, the exhaust gases are routed through the turbocharger turbine. As the engine increases its power and speed, the increase of temperature and pressure of the exhaust gases causes the turbocharger to rotate faster, raising the turbocharger compressor outlet pressure. As the compressor outlet pressure rises, the aneroid bellows in the absolute pressure controller senses the increase in pressure. When at high engine speed and load and the proper absolute pressure is reached, the force on the aneroid bellows opens the normally closed metering valve. When the oil pressure in the waste-gate actuator power cylinder is lowered sufficiently, the waste-gate actuator open spring forces the mechanical linkage to open the waste-gate. A portion of the exhaust gases then bypasses the turbocharger turbine, thus preventing further increase of turbocharger speed and holding the compressor discharge absolute pressure to the desired value. At engine idle, the turbocharger runs slowly with low compressor pressure output and low pressure applied to the aneroid bellows is not sufficient to affect the unseating of the normally closed metering valve. Engine oil pressure keeps the waste-gate closed.

On 401/402-0001 to 401A/402A-0001 aircraft, the rate-of-change controller senses the compressor outlet pressure in the upper chamber and through an internal capillary restrictor tube in the lower chamber. When the compressor discharge pressure increases more rapidly than approximately 6.5 inches of mercury per second, a pressure differential exists between the upper and lower chambers of the diaphragm. As the pressure in the upper chamber becomes greater than that of the lower chamber, the diaphragm between the upper and lower chamber is forced downward, causing the poppet valve to open and lowering the oil pressure in the waste-gate actuator power cylinder and causing the waste-gate to open. This prevents the turbocharger from increasing at too fast a rate and prevents overboosting the engine.

After critical altitude is reached (approximately 16,000 feet), the absolute pressure controller is unable to maintain 34.5 ±0.5 inches of manifold pressure. The pressure ratio controller, which senses both nacelle ambient pressure and discharge pressure, begins the function of controlling turbocharger discharge pressure at a ratio of 2.2 times nacelle ambient pressure. When the proper discharge pressure is reached inside the pressure ratio controller, the force on the aneroid bellows opens the poppet valve and lowers the oil pressure in the waste-gate actuator power cylinder.

CAUTION

The absolute pressure controller, rate of change controller and the pressure ratio controller are adjusted at the factory. The controller adjustments should not be tampered with except for the prescribed minor adjustments to obtain the rated manifold pressure.

b. Turn rotor in direction of rotation until painted chamfered tooth of distributor gear is just becoming visible in timing window. Continue turning rotor of magneto until line on end of cam is aligned with neutral mark in housing.

c. Fabricate a timing pointer from a piece of wire (approximately 2-1/2 inches) as follows:

1. Remove the cam screw and flat washer from cam.
2. Bend one end of the wire around the threads of the screw, loose enough to allow wire to be rotated.
3. At the outside diameter of the large washer, bend wire straight up parallel to the screw.
4. Install assembly on cam shaft and tighten screw just enough to hold wire in position and yet allowing it to be rotated.
5. At a height sufficient to clear breaker housing, make a right angle bend in the wire.

NOTE

Pointer should be over timing marks, but not touching the housing.

d. On retard breaker magnetos, it is necessary to set the retard breaker to open a predetermined number of degrees after the main breaker opens within +2°, -0°. The number of degrees retard for a particular magneto is stamped at the bottom of the breaker compartment. After main breaker has been set to open a "E" gap (10° ± 4°) move pointer back until it is over the zero mark without moving rotor from its position where main breaker just opened. Turn rotor until pointer is over correct retard mark. Using a timing light, adjust retard breaker contacts to open at this point. A tolerance of 1.16 inch past the point can be used to get proper contact clearance. Continue rotating rotor until cam follower is on the high point of the lobe. Measure contact clearance. It should be 0.018 ± 0.006. If not, readjust breaker and recheck to be sure that contacts will open within retard degree tolerance. Replace breaker assembly if retard degree tolerance and contact clearance cannot be obtained.

NOTE

Extreme care must be taken not to move the rotor from the main breaker opening position when returning the pointer back to the zero mark.

Installation of Magnetos and Ignition Timing.

a. (See figure 9-21.) Remove the magneto timing inspection hole plug. Rotate the magneto shaft until the timing pointer inside the magneto case is aligned with marked gear tooth.

b. Remove either the upper or lower spark plug from No. 1 cylinder.

c. (See figure 9-21.) Remove timing inspection plug located on the side of the crankcase and forward of No. 6 cylinder.

d. Rotate propeller to locate timing mark scribed on ring gear through timing inspection hole, when No. 1 cylinder is on the compression stroke.

NOTE

The No. 1 cylinder firing position is 20° BTC. This can be accomplished by rotating the propeller shaft and centering the timing mark on ring gear with the centerline of the crankcase timing inspection hole (see figure 9-21).

e. Check magneto to see that it is internally timed for right-hand drive rotation.

NOTE

The magneto installation procedure is identical for installing either magneto on either engine.

f. Set the magneto in place on the crankcase accessory mounting pad.

g. Attach a timing light to the magneto in accordance with the timing light manufacturer's instructions.

h. If timing light is extinguished, rotate magneto housing in direction of its magneto rotation a few degrees beyond point where light illuminates. Slowly rotate magneto in opposite direction until light is extinguished.

i. Tighten magneto clamp nuts to prevent any further movement of the magneto.

CAUTION

Do not adjust breaker points to compensate for ignition timing. Breaker point adjustment is for internal magneto timing only, The adjustment of magneto breaker points to compensate for ignition timing will ultimately produce a weak ignition spark and reduce engine performance.

j. Rotate propeller in opposite direction of normal rotation for a few degrees after light illuminates.

k. Rotate propeller in direction of normal rotation until light extinguishes. If timing mark is visible, through the crankcase timing inspection hole, when light extinguishes, the magneto is correctly timed to the engine.

l. Repeat timing procedure for other magneto, if applicable.

m. When ignition timing has been checked (breaker points open on both magnetos at 20° BTC) retighten magneto clamp nuts to prevent movement of the magnetos.

n. Remove timing light from the magneto and engine.

o. Replace magneto timing inspection hole plug and attach high tension outlet plate to magneto.

p. Attach the magneto ground wire to magneto.

q. Replace the removed spark plug.

r. Install the engine cowling.

IGNITION CABLES.

Removal of Ignition Cables.

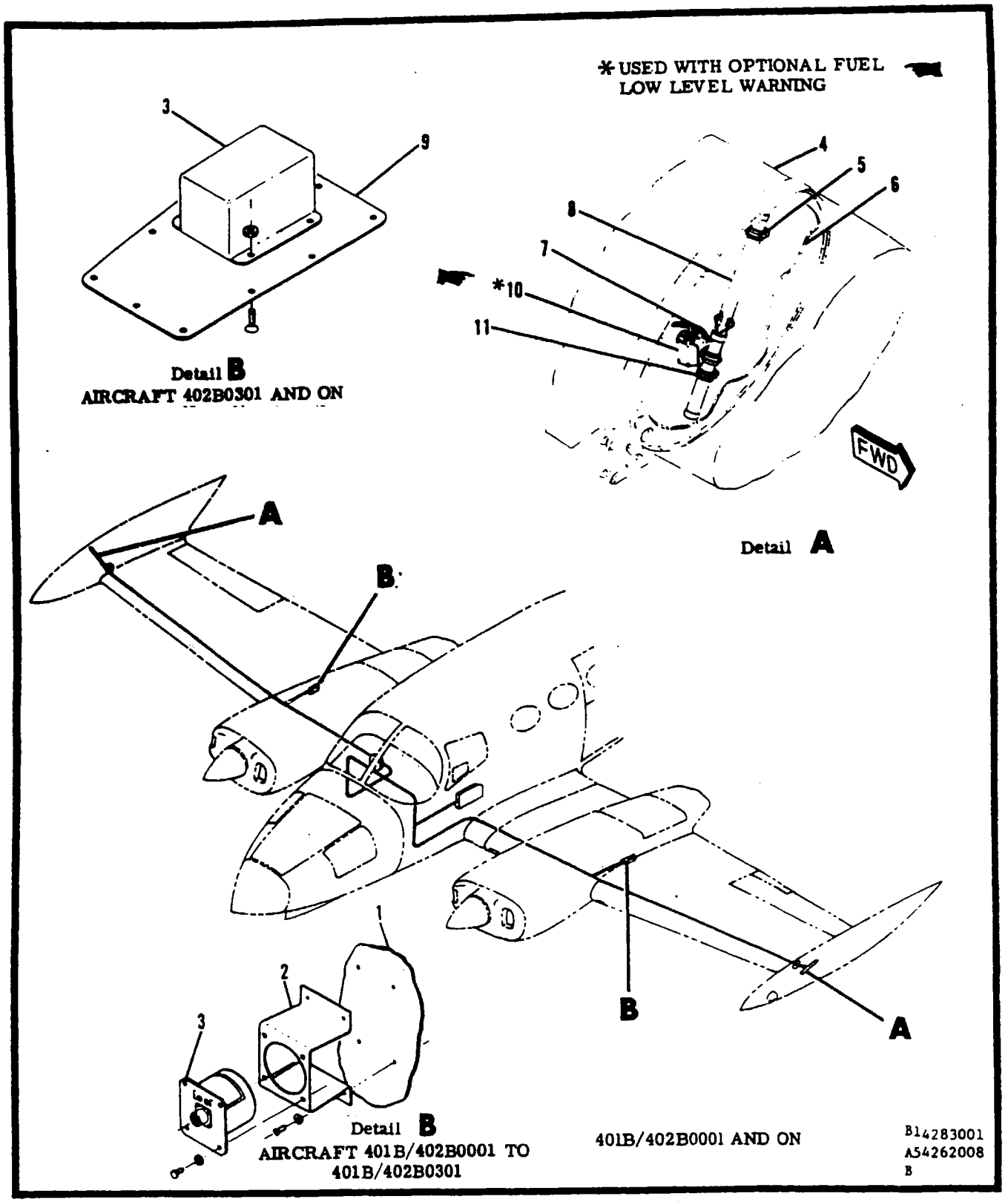
a. Remove engine cowling in accordance with removal procedure.

SECTION 11

FUEL SYSTEM

Table Of Contents

	Page	Fiche/ Frame
FUEL SYSTEM	11-2	2 I4
Troubleshooting	11-2A	2 I5
MAIN FUEL TANKS	11-3	2 I7
Removal	11-3	2 I7
Disassembly	11-6	2 I12
Inspection and Repair	11-6	2 I12
Assembly	11-6	2 I12
Installation	11-6	2 I12
SNIFFLE VALVE	11-6	2 I12
Removal	11-6	2 I12
Installation	11-6	2 I12
MAIN TANK FUEL TRANSFER PUMP	11-6	2 I12
Removal	11-6A	2 I13
Disassembly	11-6A	2 I13
Cleaning and Inspection	11-6A	2 I13
Assembly	11-6A	2 I13
Installation	11-6A	2 I13
AUXILIARY FUEL PUMP	11-6B	2 I14
Removal	11-10	2 I20
Installation	11-10	2 I20
Low Adjustment	11-10	2 I20
AUXILIARY FUEL CELLS	11-10	2 I20
Removal of Auxiliary Fuel Cells	11-10	2 I20
Inspection and Repair of Auxiliary Fuel Cells	11-10	2 I20
Inspection of Fuel Cell Cavity Following Fuel Cell Removal	11-10	2 I20
Installation of Auxiliary Fuel Cells	11-10A	2 I21
FUEL INDICATOR SYSTEM	11-10A	2 I21
Troubleshooting	11-10A	2 I21
Removal of Main Tank Unit (401/402-0001 To 401B/402B0001)	11-11	2 I23
Installation of Main Tank Unit (401/402-0001 To 401B/402B0001)	11-11	2 I23
Removal of Auxiliary Fuel Quantity Sending Unit (401/402-0001 To 401B/402B0001)	11-11	2 I23
Installation of Auxiliary Fuel Quantity Sending Unit (401/402-0001 To 401B/402B0001)	11-11	2 I23
Fuel Quantity Sending Unit Operational Check (401/402-0001 To 401B/402B0001)	11-11	2 I23
Removal of Main Fuel Tank Unit	11-12B	2 J2
Installation of Main Fuel Tank Unit	11-12B	2 J2
Removal and Installation of Signal Conditioner	11-12B	2 J2
Removal of Auxiliary Fuel Quantity Tank Units	11-12B	2 J2
Installation of Auxiliary Fuel Quantity Tank Unit	11-12B	2 J2
Fuel Quantity System Calibration	11-12B	2 J2
Dry Calibration	11-12E	2 J5
Alternate Method	11-12F	2 J6
FUEL LINES AND VENTS	11-13	2 J8
Removal	11-13	2 J8
Installation	11-13	2 J8
SELECTOR VALVES	11-13	2 J8
Removal	11-13	2 J8
Installation	11-13	2 J8
FUEL SELECTOR CONTROL SYSTEM	11-14	2 J9
Removal	11-14	2 J9
Installation	11-14	2 J9
Rigging	11-15	2 J10
WING LOCKER FUEL TANKS	11-17	2 J12
Removal	11-17	2 J12
Installation	11-17	2 J12
Removal	11-17	2 J12
Installation	11-17	2 J12
WING LOCKER FUEL PLUMBING	11-20	2 J15
Removal	11-20	2 J15
Installation	11-20	2 J15
FUEL VENT HEATERS	11-20	2 J15



1. Rib
2. Bracket
3. Signal Conditioner
4. Tank
5. Clip
6. Baffle
7. Electrical Bundle
8. Tank Unit
9. Cover Plate
10. Fuel Low Level Warning Switch
11. Collar

Figure 11-4A. Main Fuel Tank Unit

Dry Calibration is the preferred method, being the most accurate. Dry Calibration uses the actual tank units for Zero Calibration, this compensates for the tank unit and wiring harness tolerances. A fixed capacitance is added in parallel with the actual dry tank unit to represent the fuel in the tank to provide for calibration in the full condition.

To calibrate the fuel system with the Dry method, it will be necessary to use the Field Calibration Unit Test Box. The capacitance test box may be purchased through your Cessna Dealers' Organization. The Field Calibration Unit Test Box is designed for complete maintenance, troubleshooting and calibration, enabling the technician to measure capacitance of system, individual components, harness insulation breakdown, resistance of system components and substitute capacitance to simulate the tank sensors or actual fuel in the tank.

The Alternate method of calibrating the fuel quantity system may be accomplished without the capacitance test box. This method checks fuel temperature and density versus total capacitance of fuel. The Alternate method of calibration has a greater chance for error than the Dry Fuel System Calibration.

Dry Calibration.

a. Preparation for Calibration.

(1) Defuel airplane. Refer to Chapter 2. Complete defueling from each tank by removing drain valves from the bottom side of each fuel sump located on the underside of each wing outboard of main gear.

(2) Remove access panels to signal conditioner.

NOTE

The fuel quantity indicator visually displays fuel quantity for each wing separately, therefore calibration procedure is repeated for the opposite wing.

(3) Set up capacitance fuel system test box and harness assembly for insulation resistance test. See Figure 11-6A. Check insulation breakdown resistance on left and right main and auxiliary tanks. Circuit resistance should be tested for the following circuits:

MEGOHMS	CIRCUIT
0	= Shield to Structure Ground
1500	= Hi Z to Lo Z
100	= Hi Z to Shield
100	= Lo Z to Shield
100	= Hi Z to Structure Ground
100	= Lo Z to Structure Ground

(4) Interconnect test box and harness assembly into circuit as shown on Figure 6, Fuel Quantity Capacitance Test.

(5) Make capacitance measurements on LH and RH main and auxiliary circuits. Net gain capacitance on the circuit and probe combinations should not exceed 2.00 pf of that of the probes noted on Table II. The 2.00 pf allowance is for additional capacitance induced by the system wiring.

NOTE

Any discrepancies found in Steps (3) and (5) should be checked out completely and corrected before proceeding further with the calibration.

(6) Restore the circuits to original configuration.

b. Calibration.

(1) Apply airplane power and assure battery is adequately charged; voltage should never be less than 22 volts when calibration.

(2) Place fuel selector valve to "Main" position and adjust the signal conditioner "Aux Empty" potentiometer to read exact pounds on the indicator.

NOTE

Slight tapping on the indicator may be required to overcome friction when meter is in static condition.

(4) Place fuel selector valve to "Main" position and interconnect test box and harness assembly and adjust variable capacitance for added capacitance value. See Table I.

(5) Adjust signal conditioner "Main Full" potentiometer to read 50 gallons or 310 lbs. (Ref. Table I) on the indicator. Tap indicator slightly to insure that pointer has stabilized in final position.

(6) Disconnect test box and harness assembly and restore circuit to original configuration.

INSTRUMENTS.

Trouble Shooting the Instruments.

TROUBLE	PROBABLE CAUSE	CORRECTION
	VACUUM SYSTEM INSTRUMENTS	
BOTH VACUUM INSTRUMENTS MALFUNCTIONING	Dirty filter element. Restricted airflow or improper adjustment.	Clean and replace filter. Adjust vacuum relief valve.
ONE VACUUM INSTRUMENT MALFUNCTIONING, OTHER VACUUM INSTRUMENT OPERATING NORMALLY	Defective instrument.	Replace instrument.
DIRECTIONAL GYRO PRECESSES AND/OR SPINS	Insufficient suction.	Repair or replace vacuum pump and/or check system.
	Filter element dirty.	Replace filter element.
	Excessive vibration of instrument panel.	Replace instrument panel shock mounts.
	Operation limits exceeded.	Replace or overhaul directional gyro.
GYRO HORIZON WILL NOT ERECT, TUMBLES, AND IS SLUGGISH IN OPERATION	Insufficient suction.	Repair or replace vacuum pump and/or check system.
	Filter element dirty.	Replace filter element.
	Excessive vibration of instrument panel.	Replace instrument panel shock mounts.
	Operational limits exceeded.	Replace or overhaul gyro horizon.
	SENSITIVE ALTIMETER	
INDICATING POINTERS FAIL TO RESPOND	Static line obstructed.	Disconnect static line from all instruments and blow out line with dry compressed air.
EXCESSIVE POINTER VIBRATION	Excessive vibration of static line or hose.	Secure to aircraft structure or components.
ERRONEOUS INDICATIONS	Water or foreign matter in static line.	Disconnect static line from all instruments and blow out line.
	Loose static line connection.	Test and repair in accordance with testing of static pressure line.
	Defective instrument.	Replace instrument.
	AIRSPEED INDICATOR	
POINTER FAILS TO RESPOND	Clogged pitot line.	Disconnect tube from instrument and blow out line with dry compressed air.

d. The Engine Combination Gages, (Cylinder Head Temperature, Oil Temperature and Oil Pressure), one for each engine, are remote electrical indicators. Each gage is connected electrically to a cylinder head temperature bulb located in the number three or one cylinder of the respective engine. As the temperature of the bulb changes, the combination gage measures the change and the pointer indicates the temperature in degrees Fahrenheit. The oil temperature is electrically received from the oil temperature bulb, located in the engine oil passage, and calibrated in degrees Fahrenheit. Oil pressure is taken directly from the pressurized engine oil passage. It is routed through small lines and hoses to the combination gage which calibrates the pressure to pounds per square inch.

e. The Dual Tachometer is a remote electrical instrument that is connected by electrical leads to a tachometer generator on each engine. The tachometer calibrates electrical current from the tachometer generator to revolutions per minute. The pointers, one marked for each engine, are concentrically mounted so that the engines may be synchronized visually by over-lapping the pointers.

f. Economy Mixture Indicator (Optional Equipment) is an exhaust gas temperature (EGT) sensing device which is used to aid the pilot in selecting the most desirable fuel-air mixture for cruising flight of less than 75% power. EGT varies with the ratio of fuel-to-air mixture entering the engine cylinders.

Miscellaneous Instruments.

a. The Turn-and-Bank Indicator is a combination instrument. The turn indicator is an electrically-driven gyro mounted in a horizontal gimbal that is attached to a pointer which indicates the rate of turn. The slip indicator consists of a curved, liquid-filled glass tube in which an inclinometer ball, moving with dampened motion, changes positions according to the gravitational and centrifugal force acting upon the aircraft.

b. The Voltammeter, located on the left console, measures the current received from each alternator, individually or simultaneously, the battery drain or the battery bus voltage, whichever is desired, using the Voltammeter Selector Switch.

c. The Magnetic Compass is located on the windshield centerstrip. It consists of a pair of parallel magnetic bars surrounded by a circular calibrated compass card visible through a window in the compass case. The compass case is a metal bowl filled with liquid to dampen dial oscillation. Lighting is integral and controlled by a rheostat on the switch panel. The compass has two adjusting set screws, one for N-S headings and one for E-W headings. These set screws are located on the lower face of the compass behind the metal disc.

d. The Stall Warning Unit is mounted in the left console. A stall warning transmitter, mounted on the leading edge of the left wing, energizes the electrical circuit when the aircraft approaches a stall and causes the horn to operate. The stall warning transmitter incorporates a heater element, operated by the pitot heater switch, to prevent ice from hampering its operation.

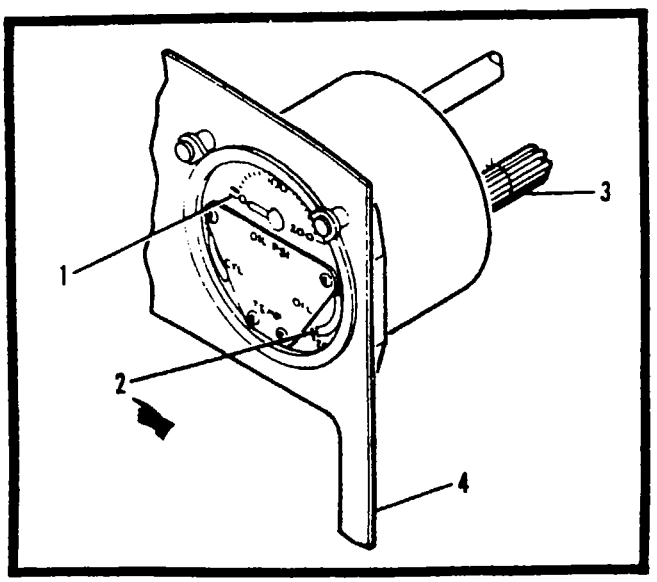
e. The Outside Air Temperature Gage is calibrated in degrees Fahrenheit and operated electrically from a free air temperature bulb located in the fresh air duct in the nose of the fuselage.

f. (401/402-0001 to 401A/402A0001) The flap position indicator is electrically operated and indicates the position of the flaps from zero to 45 degrees. The indicator operates in conjunction with a flap position transmitter located just aft of the fuselage rear spar. Refer to Section 8 for flap position transmitter removal and installation procedures.

g. (401A/402A0001 and On) The flap position is indicated by the flap preselect system pointer on the stationary instrument panel.

h. The clock is a standard eight-day aircraft clock with a sweep second hand. A winding stem is provided in the lower left-hand portion of the case.

i. Davtron digital clock removal/installation illustrated is typical. The clock may be located at various locations including different panels. The electrical cable assembly routing will vary according to the clock location. Battery replacement is accomplished by splicing electrical leads.



1. Oil Pressure
2. Oil Temperature
3. Cylinder Head Temp Wiring Bundle
4. Instrument Panel

Figure 12-3 . Engine Combination Gage

d. Removal of line (4) is not recommended except as required for replacement. The replacement of the line with a hose to facilitate installation is permissible.
e. Remove pilot's seat, copilot's seat, cabin divider (optional), kick plates and carpet in accordance with Section 3.
f. Remove line (5) by detaching at the fuselage skin union and wing splice rib union. Remove by pulling out through the fuselage floorboards.
g. Remove aft cabin vacuum line (6) by disconnecting from vacuum relief valve (7) and fuselage skin union.
h. Remove relief valve (7) by disconnecting clamps and hoses from each side of relief valve (7).
i. Remove left upholstery side panel in accordance with Section 3.
j. Remove cabin vacuum line (8) by disconnecting from union at support bracket, and from relief valve (7) by removing hose clamps.
k. Remove cabin vacuum line (9) by disconnecting from union at support bracket and from union at manifold (18).
l. Remove manifold (18) by disconnecting all hoses from manifold (18) and removing two screws.
m. Remove hose from manifold (18) to instrument, to filter by removing clamps at each end of hoses and then remove hose.

Installation of Vacuum System Plumbing. (See Figure 12-7.)

NOTE

Slight bending of the vacuum line may be required to facilitate installation; however,

	Page	Fiche/ Frame
Removal of Condensers	13-86	3 F21
Installation of Condensers	13-86	3 F21
Removal of Receiver-Dryer	13-86	3 F21
Installation of Receiver-Dryer	13-86	3 F21
Removal of Low Pressure Switch	13-86	3 F21
Installation of Low Pressure Switch	13-87	3 F22
Removal of High Temperature Switch	13-87	3 F22
Installation of High Temperature Switch	13-87	3 F22
Removal of Evaporator Blower Motor	13-87	3 F22
Installation of Evaporator Blower Motor	13-87	3 F22
Removal of Expansion Valve	13-87	3 F22
Installation of Expansion Valve	13-87	3 F22
Removal of Evaporators	13-88	3 F23
Installation of Evaporators	13-88	3 F23
Removal of Air Conditioning Plumbing	13-88	3 F23
Installation of Air Conditioning Plumbing	13-88	3 F23
AIR CONDITIONING SYSTEM	13-89	3 F24
Troubleshooting Air Conditioning Hydraulic System	13-90	3 G1
Hydraulic System Operational Test	13-90	3 G1
Removal of Hydraulic Pump	13-92	3 G3
Installation of Hydraulic Pump	13-92	3 G3
Removal of Hydraulic Reservoir	13-92	3 G3
Installation of Hydraulic Reservoir	13-92	3 G3
Removal of Hydraulic Motor	13-93	3 G4
Installation of Hydraulic Motor	13-93	3 G4
Removal of Manifold and Valve Assembly	13-93	3 G4
Installation of Manifold and Valve Assembly	13-93	3 G4
Removal of Compressor	13-93	3 G4
Installation of Compressor	13-93	3 G4
Removal of Condenser Blower Motor	13-96	3 G7
Installation of Condenser Blower Motor	13-96	3 G7
Removal of Condenser	13-96	3 G7
Installation of Condenser	13-96	3 G7
Removal of Receiver-Dryer	13-96	3 G7
Installation of Receiver-Dryer	13-96	3 G7
Removal of Low Pressure Switch	13-96	3 G7
Installation of Low Pressure Switch	13-96	3 G7
Removal of High Temperature Switch	13-96	3 G7
Installation of High Temperature Switch	13-97	3 G8
Removal of Thermostat Switch Assembly	13-97	3 G8
Installation of Thermostat Switch Assembly	13-97	3 G8
Removal of Evaporator Blower Motor	13-97	3 G8
Installation of Evaporator Blower Motor	13-97	3 G8
Removal of Expansion Valve	13-97	3 G8
Installation of Expansion Valve	13-97	3 G8
Removal of Evaporator	13-102	3 G13
Installation of Evaporator	13-102	3 G13
Removal of Air Conditioning Plumbing	13-102	3 G13
Installation of Air Conditioning Plumbing	13-102	3 G13
Removal/Installation of Inlet Scoop Door	13-102	3 G13
Removal/Installation of Bellcrank	13-102	3 G13
Removal/Installation of Actuating Cylinder	13-102	3 G13
Adjustment of Inlet Scoop Door	13-102A	3 G14
ANGLE OF ATTACK SYSTEM	13-103	3 G16
Removal of Angle of Attack Indicator	13-103	3 G16
Installation of Angle of Attack Indicator	13-103	3 G16
Removal of Angle of Attack Transducer	13-103	3 G16
Installation of Angle of Attack Transducer	13-103	3 G16
Operational Check of Angle of Attack System	13-103	3 G16

Maintenance of Oxygen System.

Before and during maintenance on oxygen systems, the following general rules must be followed:

a. Clean hands, tools, and working area.

b. Keep grease, oil, water, and all foreign matter from system.

c. Keep all lines dry and capped until installed.

d. All compounds used on fittings must conform to MIL-T-5542. No compound shall be used on aluminum alloy flared fittings. Compounds are used only on the first three threads of male fittings. No compounds is used on coupling sleeves, or outside of tube flares.

CAUTION

Whenever a component of the oxygen system (lines, gages, cylinders, or regulators, etc.) has been removed, reinstalled, replaced, or system has been disassembled in any way, the oxygen system must be leak checked and purged as per procedures outlined in this section.

e. Inspection of the cylinders is required before charging. Do not attempt to charge the cylinder if any of the following conditions exist:

1. Contamination or corroded fitting on cascade cylinder or filler valve.

2. Cylinder out of hydrostatic test date.

3. Cylinder bears no I. C. C. or D. O. T. designation.

4. Cylinder completely empty after shutoff valve has been turned off for a length of time. Must be completely disassembled and inspected in an FAA approved facility before charging.

f. Fabrication of pressure lines is not recommended. Lines should be replaced from factory by part number.

g. Lines and fittings shall be clean and dry. One of the following methods may be used to clean lines:

CAUTION

Most air compressors are oil lubricated, and a minute amount of oil may be carried by the air stream. A water lubricated compressor should be used to blow tubing clean.

1. Flush with naphtha type III conforming to Specification TT-N-95, then blow clean and dry with clean, dry filtered air. Flush with anti-icing fluid conforming to MIL-F-5566 or anhydrous ethyl alcohol. Rinse thoroughly with fresh water and dry with clean, nitrogen gas (BB-N411) Type 1, Class 1, Grade A or technical Argon (MIL-A-18455).

2. Flush with hot inhibited alkaline cleaner until free from oil and grease. Rinse with fresh water and dry with clean dry, nitrogen gas (BB-N411) Type 1, Class 1, Grade A or technical Argon (MIL-A-18455).

NOTE

Cap lines at both ends immediately after drying to prevent moisture from entering.

Bleeding the Oxygen System. (Refer to Figure 13-5.)

a. Pull oxygen control knob (15) to ON position.

b. Remove tubing from the mask end of one of the passengers oxygen masks and insert into oxygen outlet and route the hose outside the cabin area through the pilots foul weather window.

WARNING

The bleeding procedure should be accomplished outdoors. If the bleeding is done indoors, extreme care must be exercised to prevent oxygen flow from oils, grease, contaminations and electrical sparks. The area should be roped off and no smoking or open flame in or near the area.

c. Bleeding the oxygen system into the cabin area is not recommended.

Removal of Oxygen Cylinder-Regulator Assembly. (Refer to Figure 13-5.)

a. Push control knob (15) to OFF position and disconnect lines (16, 17, and 21) from oxygen cylinder-regulator assembly (8).

b. Disconnect control cable (10) from oxygen cylinder-regulator assembly (8). Safety in the OFF position.

CAUTION

Do not rotate control lever to the "ON" position with outlet ports (low pressure) open to atmosphere. Damage to regulator metering poppet will occur.

c. Remove control mounting clamp (20) from oxygen cylinder-regulator assembly (8).

d. Loosen mounting straps (7) and lift oxygen cylinder from airplane.

Installation of Oxygen Cylinder-Regulator Assembly. (Refer to Figure 13-5.)

WARNING

Extreme care must be taken when installing or working near the cylinder-regulator to prevent damaging the cylinder. The slightest scratch, nick or dent is cause for immediate condemnation of the cylinder.

a. Install oxygen cylinder-regulator assembly (8) in the nose compartment position in mounting straps (7). Do not tighten at this time.

b. Align oxygen cylinder-regulator assembly (8) with control cable (10) and three line assemblies (16, 17, and 21). Tighten nuts on mounting straps (7).

3. Connect a source of clean air with an inline hand-operated valve and a pressure gauge to the disconnected deice hose.
4. Disconnect tube from overboard port of the control valve and plug the port with an AN933-4 plug.
5. Disconnect the electrical power lead to each of the three flow valves.

CAUTION

Do not attempt air leakage test with deice timer module connected in circuit.

6. Connect a 28VDC electrical source to each flow valve to energize the valves.

NOTE

For test or troubleshooting, one valve may be actuated at a time to test and isolate each system.

7. Apply 12 PSI pressure to the deice system and by means of the inline hand-operated valve, trap the pressure in the deice system.
(a) Observe the deice system for leaks. The leakage rate should not exceed a pressure drop of 4.0 PSI per minute.
8. Insure that all deice boots inflate and that there are no leaks, remove test equipment and reconnect hose to pneumatic pump, remove plug from control valve and connect tube.
9. Remove 28VDC electrical source and reconnect electrical leaks to aircraft system.

Cold Patch Repair of Deicer Boots.

There are four types or areas of damage that are most common to the deicer boots. An outline of the cold patch repair procedure for each follows:

NOTE

When repairing the deicer boots and replacement layers are being installed, exercise care to prevent trapping air beneath the replacement layers. If air blisters appear after material is applied, remove them with a hypodermic needle. Should air blisters appear after boots have been installed for a length of time, it is permissible to cut a slit in the deicer boot, apply adhesive and repair in accordance with the following cold patch repair procedures. An alternate method of repair is to peel the deicer boot back using Ketone solvent, and reapply using normal adhesives.

a. Scuff or Surface Damage. This type of damage is the most commonly encountered and is usually caused by scuffing the outer surface of the deicer boots while using scaffolds, refueling hoses, ladders, etc. Repair is generally not necessary because the thick outer veneer provides protection to the natural rubber underneath. If the scuff is severe and has caused removal of the entire thickness of veneer (exposing the brown natural rubber underneath) the damage should be repaired using Cold Patch Repair Kit, Part Number 74-451-C, available from Cessna Aircraft Company. Repair damage as follows:
1. Select a patch (Part Number 74-451-16, 74-451-17 or 74-451-18 of ample size to cover damaged area.
2. Clean the area to be repaired with a cloth dampened slightly with solvent.
3. Buff the area around the damage with steel wool so that the area is moderately but completely roughened.
4. Wipe the buffed area clean with a cloth slightly dampened in solvent to remove all loose particles.
5. Apply one even, thorough coat of cement (Part Number 74-451-20) to the patch and to the corresponding damaged area of the deicer boot. Allow cement to set a couple of minutes until it becomes tacky.
6. Apply patch to the deicer with an edge or the center adhering first, and work the remainder of the patch down, being careful to avoid trapping air pockets.
7. Roll the patch thoroughly with a stitcher-roller (Part Number 74-451-73) and allow to set for ten to fifteen minutes.
8. Wipe the patch and surrounding area from the center of the patch outward, with a cloth slightly dampeded with solvent.
9. Apply one light coat of A-56-B conductive cement (Part Number 74-451-11) to the patched area.

NOTE

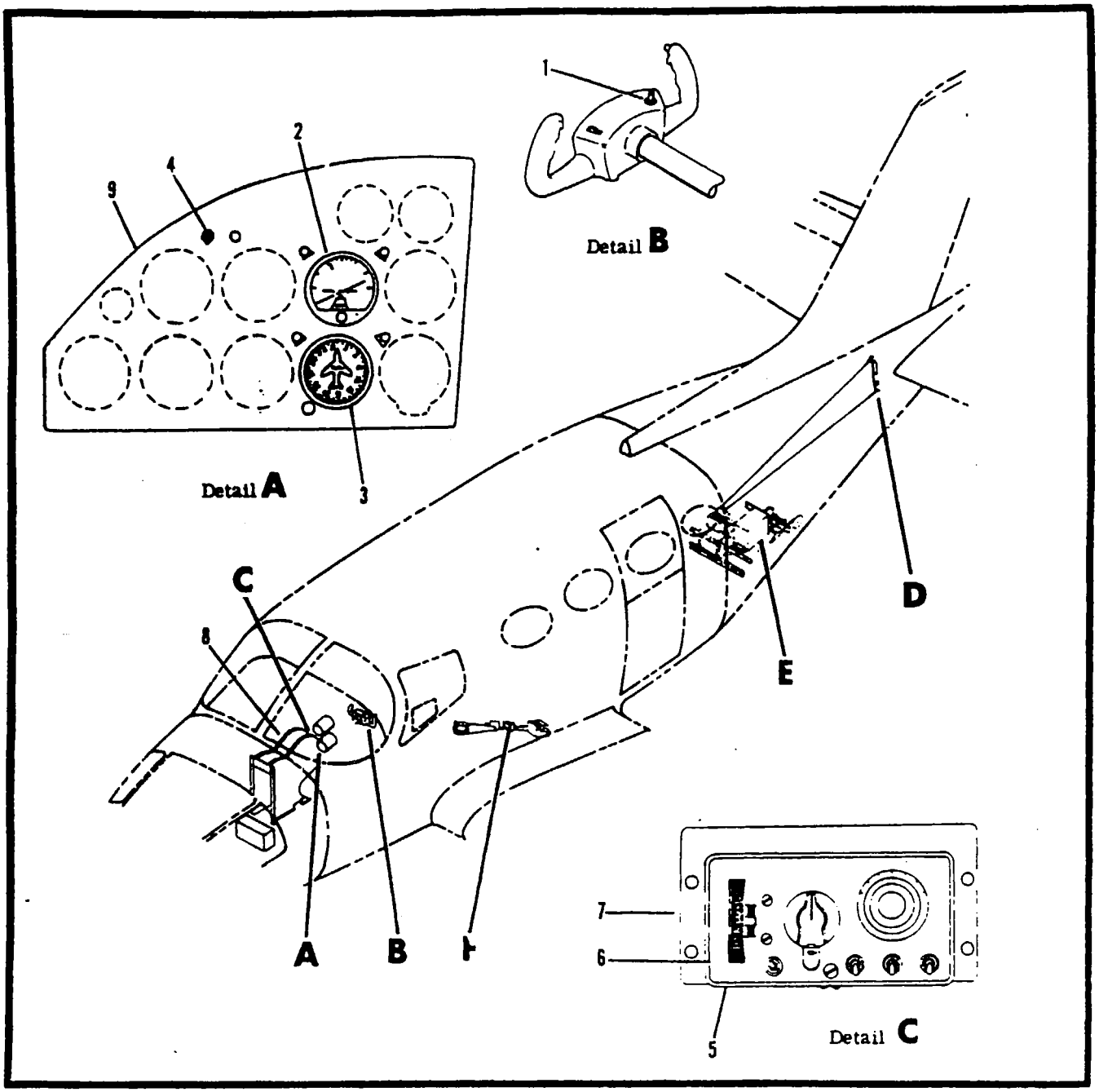
Satisfactory adhesion should be obtained in four hours; however, if the patch is allowed to cure for a minimum of 20 minutes, the deicer boots may be inflated to check the repair.

b. Damage to Tube Area. This type of damage consists of cuts, tears, or ruptures to the inflatable tube area and a fabric reinforced patch must be used for this repair. Damage to the tube area should be repaired as outlined below:
1. Select a patch (Part Number 74-451-16, 74-451-17, 74-451-18 or 74-451-19) of ample size to extend at least 5/8" beyond the damaged area.

NOTE

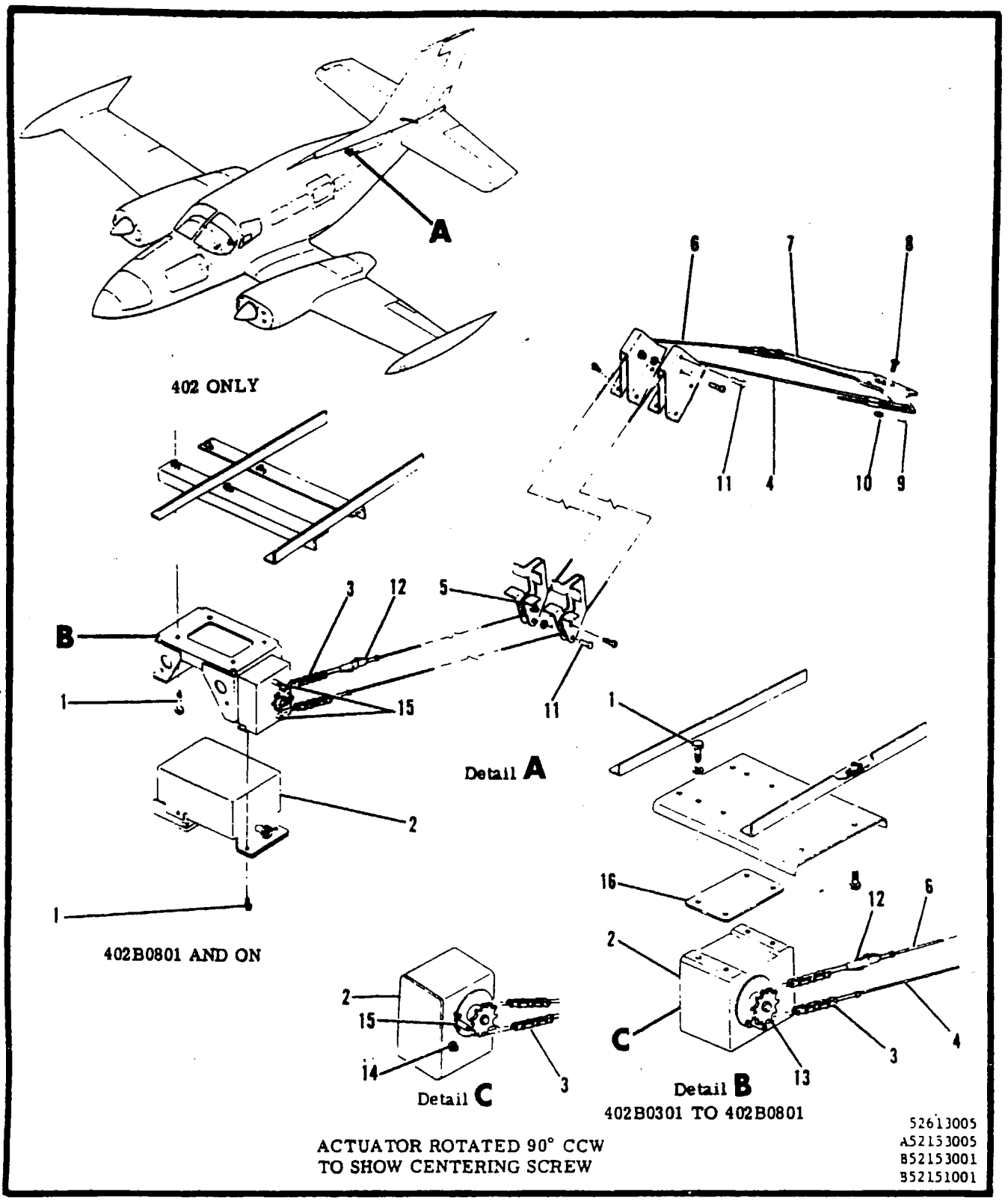
If none of these patches are of the proper size, one may be cut to the size desired from one of the larger patches. If this is done, the edge should be beveled by cutting with the shears at an angle. These patches are manufactured so they will stretch in one direction only. Be sure to cut the patch selected so that the stretch is in the widthwise direction of the inflatable tubes.

2. Clean the area to be repaired with a cloth dampened slightly with solvent.
3. Buff the area around the damage with steel wool so that the area is moderately but completely rough-



1. Disengage Switch
2. Horizontal Gyro
3. Directional Gyro
4. Selector Switch
5. Bezel
6. Controller
7. Retainer
8. Control Quadrant
9. Shock Mounted Instrument Panel
10. Elevator Bellcrank
11. Link
12. Elevator Cable (Upper, servo to bellcrank)
13. Elevator Cable (Lower, servo to bellcrank)
14. Elevator Servo
15. Line Assembly (Computer to static line)
16. Computer Amplifier
17. Cable Assembly (Computer to controller)
18. Cable Assembly (Computer to elevator servo)
19. Cable Assembly (Computer to aileron servo)
20. Sprocket
21. Aileron Cable (Servo to bellcrank)
22. Aileron Servo
23. Bracket
24. Aileron Cable
25. Aileron Bellcrank Assembly
26. Turnbuckle
27. Chain Assembly
28. Aileron Cable Pulley
29. Aileron Cable Pulley
30. Chain Assembly.

Figure 13-9. Nav-O-Matic 400 Autopilot Installation (Sheet 1 of 2)



1. Bolt
2. Actuator
3. Chain
4. Cable
5. Pulley
6. Cable
7. Bellcrank
8. Screw
9. Cotter Pin
10. Nut
11. Guard Pin
12. Turnbuckle
13. Sprocket
14. Centering Screw
15. Chain Guard
16. Plate

Figure 13-10B. Yaw Damper System (Sheet 2)

Dual Pitot and Static System. (See figure 13-16.)

On aircraft 401B/402B0001 and On, an optional pitot system is provided for the copilot's airspeed indicator. It is independent of the pilot's airspeed indicator but uses a common static air source.

On aircraft 402B0801 and On, an optional static air source is provided for the copilot's flight instruments. It is identical to but independent of the pilot's flight instruments static system.

For maintenance and servicing of the dual pitot and static systems, refer to procedures for removal, installation, testing and purging of the standard systems, Section 12.

FLIGHT HOUR RECORDER.

The flight hour recorder instrument is mounted on the stationary instrument panel. On aircraft 401-0001 to 401B0055 and 402-0001 to 402B0040 the recorder is actuated by a vane switch located on the bottom of the forward fuselage and is actuated by a 40 MPH flow of air. On aircraft 401B0055 and On and 402B0040 and On the recorder is actuated by the landing gear safety switch. The switch is actuated when the strut is extended and will allow the recorder to operate.

NOTE

To prevent the flight hour recorder from recording while the aircraft is on jacks and battery switch is in the ON position, remove fuse located in the left console.

Removal and Installation of Flight Hour Recorder Switch.

a. Remove copilot's seat in accordance with Section 3.
b. Remove carpets and floorboards in accordance with Section 3, as necessary, to gain access to flight hour recorder switch.
c. Disconnect wires from switch, remove screws attaching switch to fuselage skin. Retain spacers for reinstallation.
d. Install switch by reversing removal procedure.

TRUE AIRSPEED INDICATOR. (See figure 13-17.)

The true airspeed indicator is composed of three elements: airspeed, altitude, and temperature. The altitude and temperature mechanism are correlated to operate a rotating dial over which the indicated airspeed indicates the true airspeed. The true airspeed indicator senses changes in pressure and temperature. This combination of altitude and temperature change results in a sum total of airspeed corrections to indicate true airspeed.

Removal and Installation of True Airspeed Indicator. (See figure 13-17.)

a. On aircraft 401B/402B0001 to 401B/402B0201, remove and install the true airspeed indicator in accordance with Typical Instrument Removal and Installation procedures.
b. On aircraft 401B/402B0201 and On, remove and install the true airspeed indicator as follows:
1. Disconnect clamp securing temperature probe on bottom skin.
2. Loosen feed thru nut on temperature tube and work tube up and into the cockpit area.

CAUTION

Avoid sharp bends, nicks or dents in temperature tube. Incorrect readings could result.

3. Remove indicator in accordance with Typical Instrument procedures Section 12.
4. Install true airspeed indicator by reversing removal procedures.

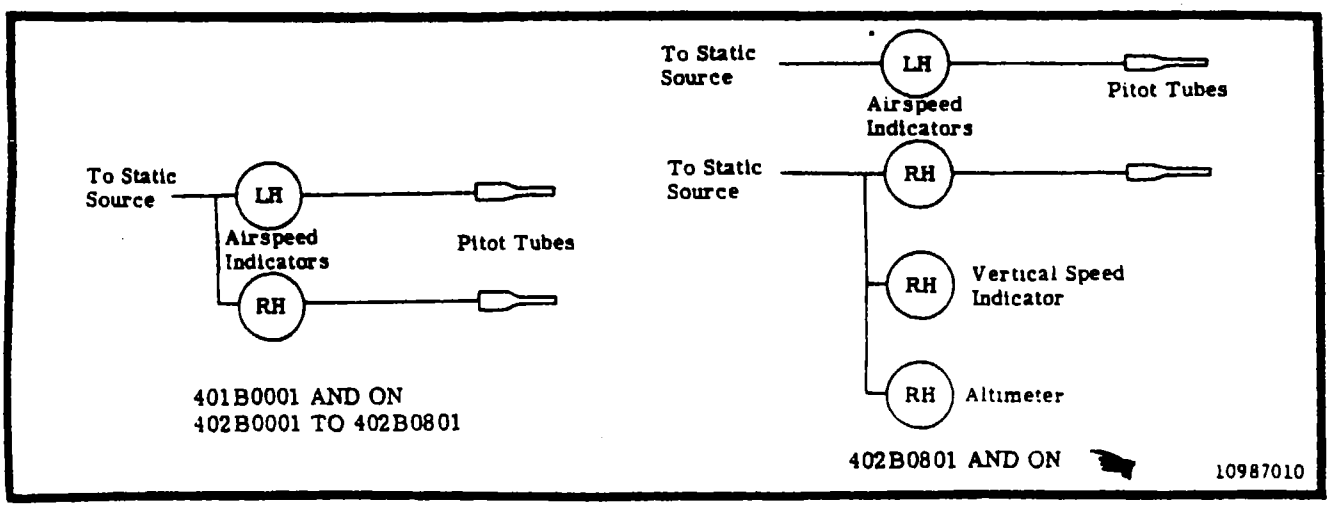
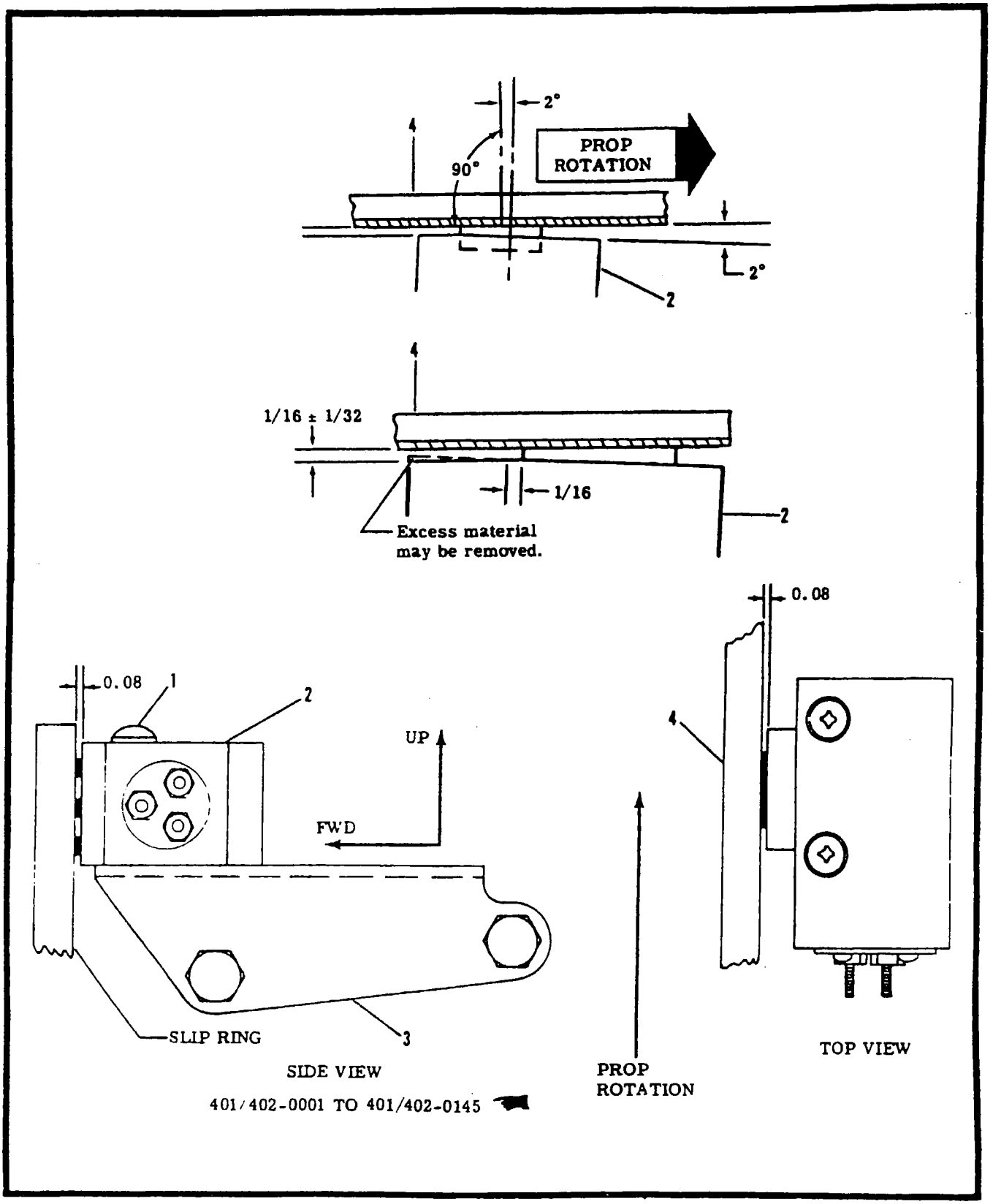
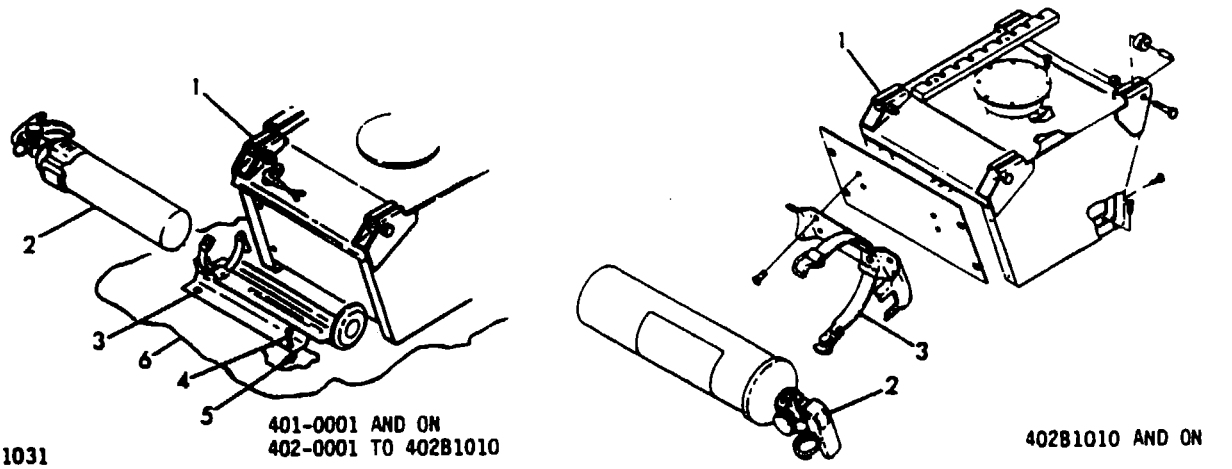


Figure 13-16. Dual Pitot and Static System Schematic



1. Screw
2. Brush Holder
3. Bracket Assembly
4. Slip Ring
5. Washer

Figure 13-21. Deice Brush Holder Installation (Sheet 1 of 2)



401-0001 AND ON
402-0001 TO 402B1010

402B1010 AND ON

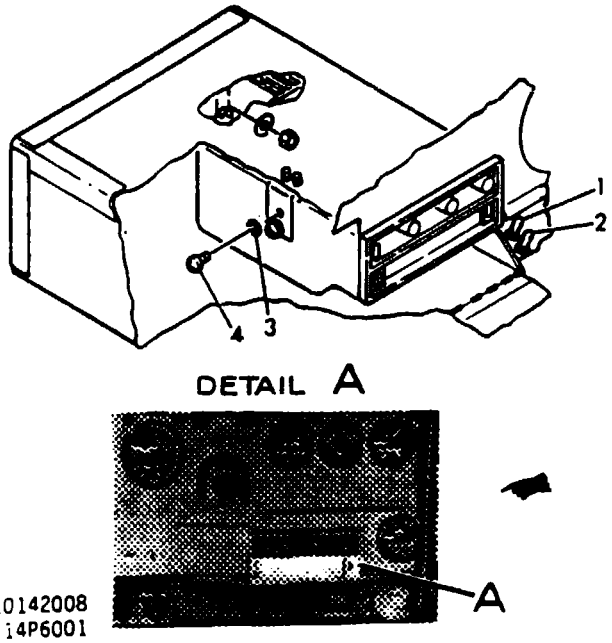
14141031
51143090
51142054

1. Seat Support
2. Fire Extinguisher
3. Bracket
4. Screw
5. Nut
6. Floorboard

Figure 13-28. Fire Extinguisher Installation

FIRE EXTINGUISHER (See figure 13-28).

The fire extinguisher mounted on the floorboard in front of the copilot's seat contains a pressurized dry chemical charge. The fire extinguisher may be used on combustible material and liquid or electrical fires. The charge is nontoxic and noncorrosive. The fire extinguisher may be recharged by most fire equipment dealers.



510142008
5214P6001

1. Headset Speaker Selector Switch
2. "On" "Off" Switch
3. Washer
4. Screw

Figure 13-29. Mini-8 Stereo

Removal of Mini-8 Stereo (See figure 13-29).

a. Turn electrical power OFF.
b. Remove screw (4) and washer (3).
c. Remove stereo unit out far enough to gain access to wiring.
d. Tag and disconnect electrical wiring.
e. Remove stereo.

Installation of Mini-8 Stereo (See figure 13-29).

a. Position stereo in glove box and connect electrical wiring.
b. Remove tags.
c. Secure stereo in place with washers (3) and screws (4).
d. Turn on electrical power and check operation of stereo.

ENGINE COMPARTMENT FIRE EXTINGUISHER.

Description.

a. The engine compartment fire extinguisher (optional) consists of three major components: the annunciator panel, thermal detectors and the extinguisher unit. The annunciator panel is mounted in a standard three-inch instrument hole. Three thermal detectors are mounted in the high seat areas of each engine nacelle and the extinguisher unit is mounted in the aft section of the nacelle. An exit line is routed from the extinguisher into the forward section of the engine where the freon gas is expelled. A smaller discharge hose which routes off the large discharge hose expells freon gas from the center of the nacelle area towards the inboard side of the nacelle.

PILOT AND COPILOT MANUAL AND ELECTRICAL ADJUSTABLE SEAT.

Removal of Pilot and Copilot Manual and Electrical Adjustable Seat. (See figure 13-32.)

The precedures given pertain to either the pilot's or copilot's manual or electrical adjustable seat. The difference between the manual and electrical is the electrical seat utilizes an electric motor in lieu of the manual crank for the up and down and tilting movements.

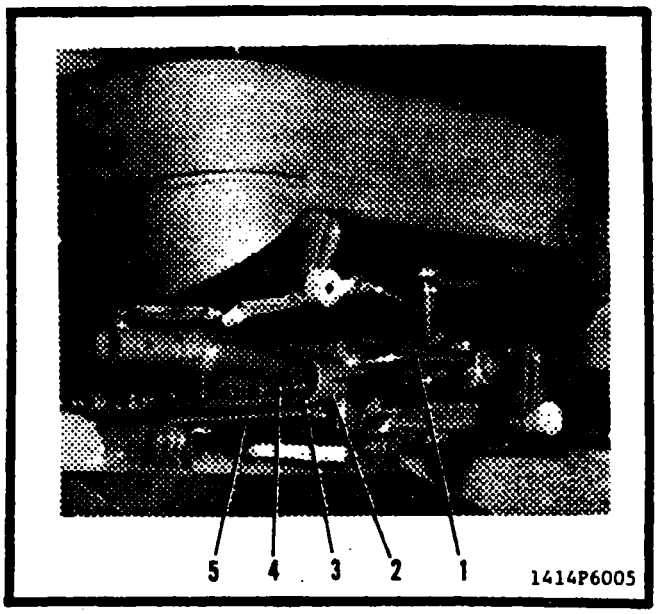
a. Raise seat to the highest position.
b. Remove seat stops on each side of the rail by removing nuts and screws.
c. Tag and disconnect wiring.
d. Pull up on seat stop handle and slide seat aft and remove seat from its mounting.

Disassembly of Pilot and Copilot Manual and Electrical Adjustable Seat. (See figure 13-33.)

a. Disassemble pilot's and copilot's manual and electrical adjustable seat in accordance with Figure 13-33.

Installation of Pilot's and Copilot's Manual and Electrical Adjustable Seat. (See figure 13-32.)

a. Insert seat on seat rails and slide forward.
b. Pull up on seat stop handle and allow seat to move forward far enough to install seat stops.
c. Connect wiring and remove tags.
d. Install seat stops with screws and nuts.
e. Turn on electrical power and check operation.
f. If seat is not electrical, check operation using crank provided on the front of the seats.



1. Adjust Handle
2. Seat Stop
3. Nut
4. Screw
5. Seat Rail

Figure 13-32. Pilot and Copilot Manual and Electrical Seat Installation

System operation in the air conditioning mode, requires that the right-hand engine be operating at a minimum tachometer reading of 950 RPM in order to drive the freon compressor at its design ground capacity. Whenever the system mode switch is selected to the air conditioning mode, with the right-hand engine operating, the magnetic clutch is energized and the compressor starts delivering a high pressure and temperature freon gas to the condensers. This gas is routed through the condensors where cool air from the fan assembly removes heat from the gas and condenses it into a high pressure liquid.

The liquid then enters the receiver-dryer where it is filtered, dried and gas is separated from the liquid. From the receiver-dryer, the liquid flows to the freon distribution manifold which directs the liquid to each evaporator module expansion valve. The expansion valve throttles the liquid into a sub-cooled gas before it enters the evaporator coil. Air from the cabin is pulled across the evaporator coil and is cooled by the sub-cooled freon gas. The refrigerated air is then routed into the air distribution ducts and distributed through directional louvered outlets into the cabin and cockpit.

NOTE

For troubleshooting the Air Conditioning System refer to Air Conditioning System Service/Parts manual.

Removal of Compressor (See figure 13-37.)

a. Verify system power switch is in OFF position.
b. Remove RH cowling.
c. Discharge refrigerant system. Refer to Air Conditioning System Service/Parts Manual.
d. Remove compressor drive belt in accordance with the Removal of Compressor Drive Belt Procedures.
e. Remove suction hose and discharge hose. Cap open ports and hoses.
f. Disconnect electrical wires at splices.
g. Remove bolt and nut securing rod end (125) to bracket (124).
h. Remove bolt, nut and washers securing support assembly (123) to support fitting.
i. Remove compressor, torque link and support assembly as a unit.
j. Disassemble compressor assembly components in accordance with figure 13-37.

Installation of Compressor (See figure 13-37.)

a. Drain all oil from the compressor (53) in accordance with Air Conditioning System Service/Parts Manual.
b. Refill the compressor with 4 ounces of Suniso No. 5 or Texaco Capella E Grade, 500 viscosity oil or equivalent.

NOTE

Special care shall be taken to assure that the system is not over-charged with oil, since an over-charge of oil in the system will decrease system performance.

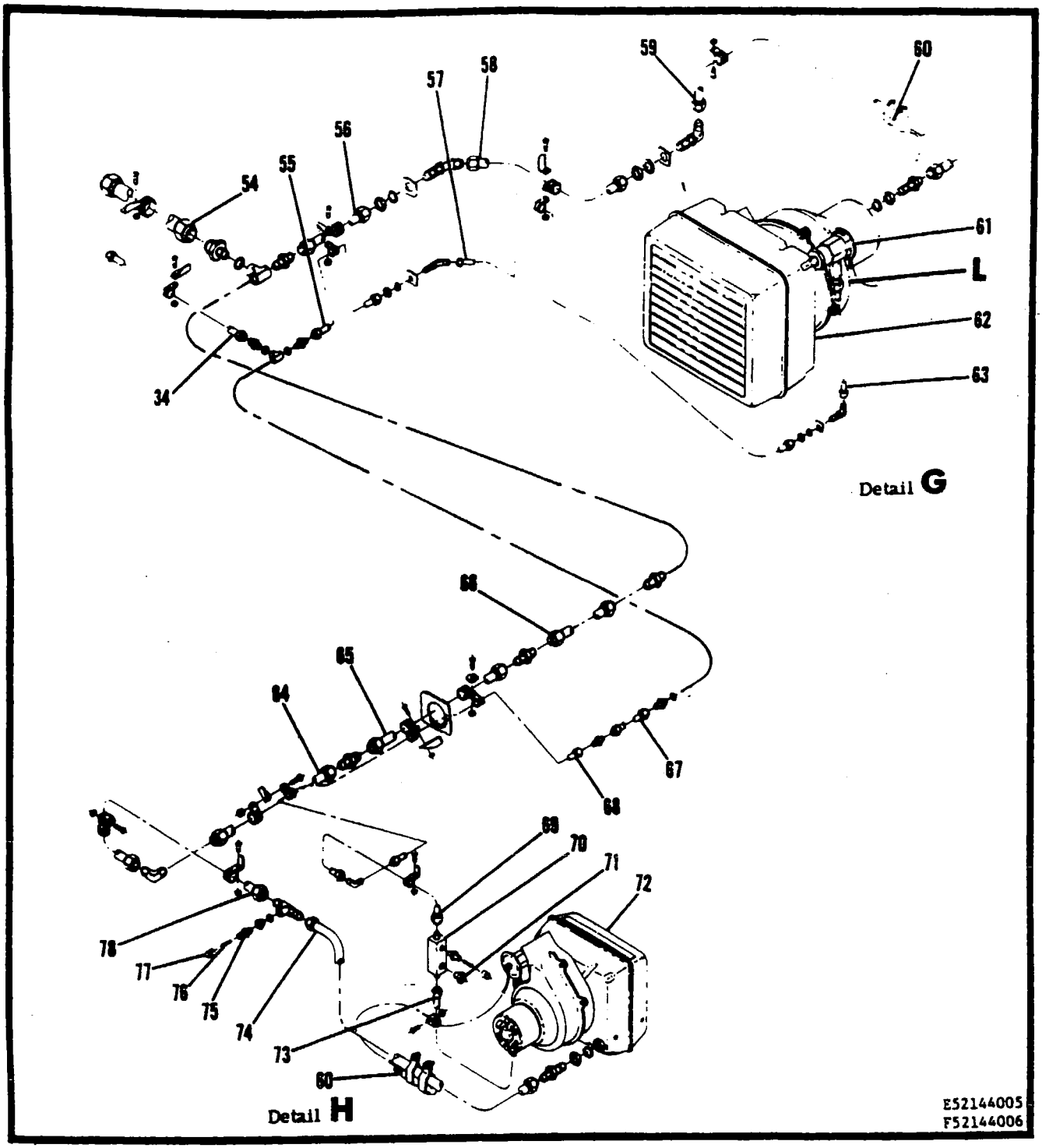
NOTE

All openings in compressor shall be capped off to assure that oil in the compressor does not drain out during installation.

c. Assemble torque link (121) and support assembly (123) to compressor (39) as shown for proper placement of components.

NOTE

Bushings (122) may be lubricated with Dow-Corning DC4 (MIL-G-3278) silicone grease to facilitate installation. Bolts and nuts securing bushings must be snug but not so tight as to deform bushings. Secure nuts with cotter pins.



54. Line
55. Line
56. Line
57. Line
58. Line
59. Line
60. Temperature Sense Bulb
61. Expansion Valve
62. Aft Evaporator
63. Line
64. Line
65. Line
66. Line
67. Line
68. Line
69. Line
70. Service Body
71. Sight Glass
72. Forward Evaporator
73. Line
74. Line
75. Service Valve
76. Valve Core
77. Cap
78. Line

Figure 13-39. Air Conditioning System (Sheet 4)

Removal of Hydraulic Motor. (See figure 13-41.)

a. Remove RH wing nacelle air conditioning access cover.
b. Place container under reservoir drain.
c. Cut safety wire; open drain valve (7) and drain fluid from the reservoir.
d. Disconnect line (19) from manifold and valve assembly (8) and firewall fitting. Cap all openings.
e. Disconnect line (16) and hose (18) from hydraulic motor (17). Cap all openings.
f. Remove nuts, washers and bolts, securing hydraulic motor (17) to retaining ring (28) and adapter (26).
g. Pull hydraulic motor (17) outboard and remove motor from compartment.
h. If further disassembly of hydraulic motor (17) is required, refer to Air Conditioning System Service/Parts Manual.

Installation of Hydraulic Motor. (See figure 13-41.)

NOTE

Before installing hydraulic motor, lubricate spline of motor with Dow Corning No. 4 Compound.

For ease of installation, assemble coupling on motor spline before installing motor in adapter.

a. Install retaining ring (28) and adapter (26) on shaft of motor (17).
b. Engage hydraulic motor (17) with the compressor shaft. Ensure that the adapter and retaining ring are properly engaged.
c. Secure hydraulic motor to adapter and retaining ring with washers and bolts. Torque bolts to 125 inch-pounds.
d. Connection of lines and hose is the reversal of the removal procedures.
e. Close drain valve (7) and safety wire.
f. Fill reservoir (3) with hydraulic fluid.
g. Check system for operation and leakage.
h. Install RH nacelle air conditioning access cover.

Removal of Manifold and Valve Assembly. (See figure 13-41.)

a. Verify electrical power to the air conditioning system is turned OFF.
b. Remove RH nacelle air conditioning access cover.
c. Place container under reservoir drain.
d. Cut safety wire; open drain valve (7) and drain hydraulic fluid from reservoir (3).
e. Identify electrical wiring and disconnect.
f. Disconnect lines (16, 19 and 25) and tee (10) from manifold and valve assembly (8). Cap all openings.
g. Remove nuts, washers and bolts; remove manifold and valve assembly (8).

Installation of Manifold and Valve Assembly. (See figure 13-41.)

a. Installation of manifold and valve assembly (8) is the reversal of the removal procedures.
b. Close drain valve (7) and safety wire.
c. Fill reservoir with hydraulic fluid.
d. Check system operation.

Removal of Compressor. (See figure 13-42.)

a. Discharge refrigerant system. Refer to Air Conditioning System Service/Parts Manual.
b. Remove RH wing nacelle air conditioning access cover.
c. Disconnect lines (11 and 8) from compressor (9). Cap all openings.
d. Disconnect and remove lines as required to gain access for removal. Cap all openings.
e. Remove hydraulic motor. Refer to Hydraulic Motor Removal procedures.
f. Remove bolts and washers securing compressor (9) to mount (10). Slide compressor inboard and lift it out of the compartment.
g. Remove screws, securing baffle to compressor (9).
h. If further disassembly of compressor is required refer to Air Conditioning System Service/Parts Manual.

Installation of Compressor. (See figure 13-42.)

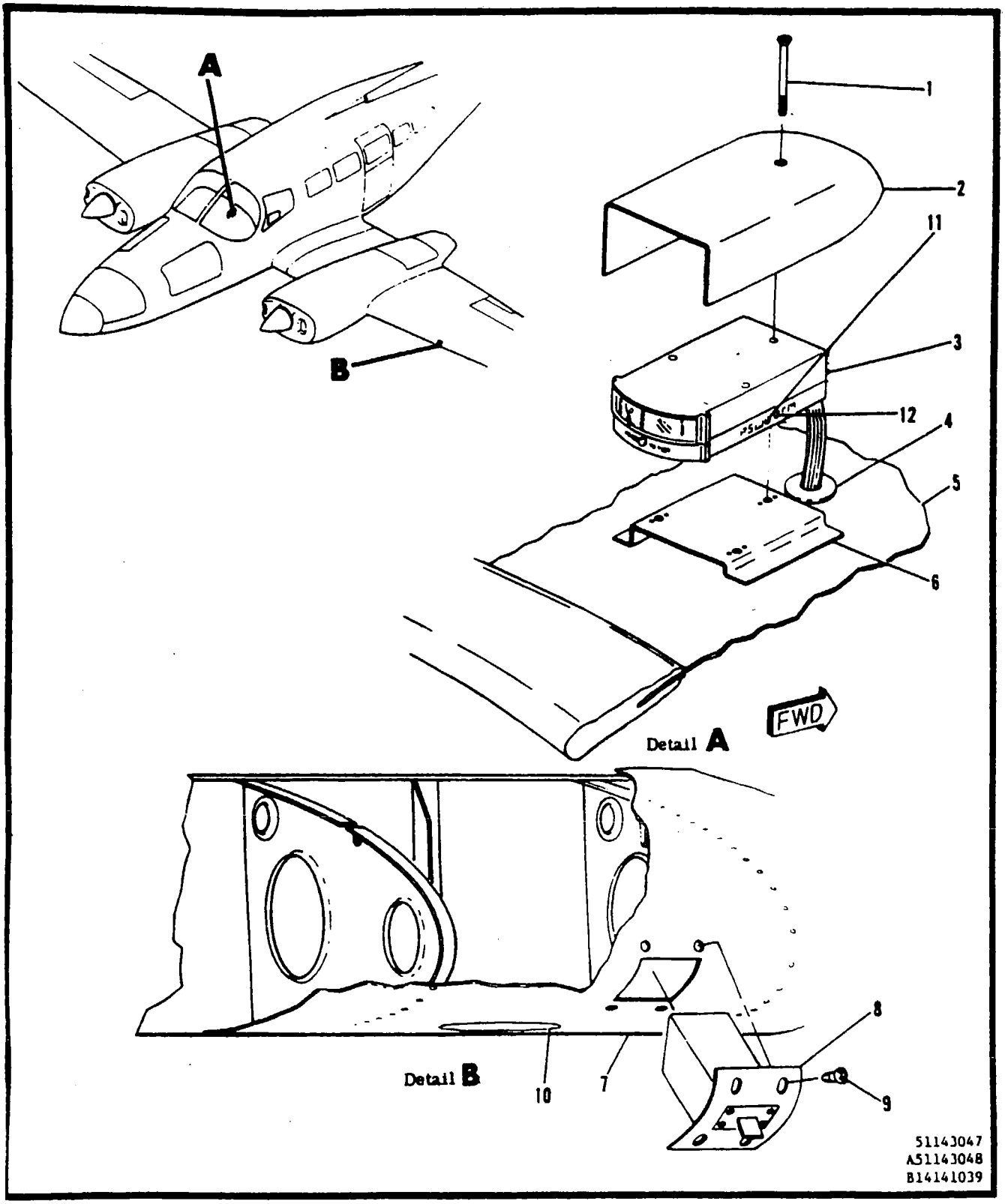
a. Drain all oil from the compressor (9) in accordance with Air Conditioning System Service/Parts Manual.
b. Refill the compressor with 4 ounces of Suniso No. 5 or Texaco Capella E Grade, 500 viscosity oil or equivalent.

NOTE

Special care shall be taken to assure that the system is not over-charged with oil, since an over-charge of oil in the system will decrease system performance.

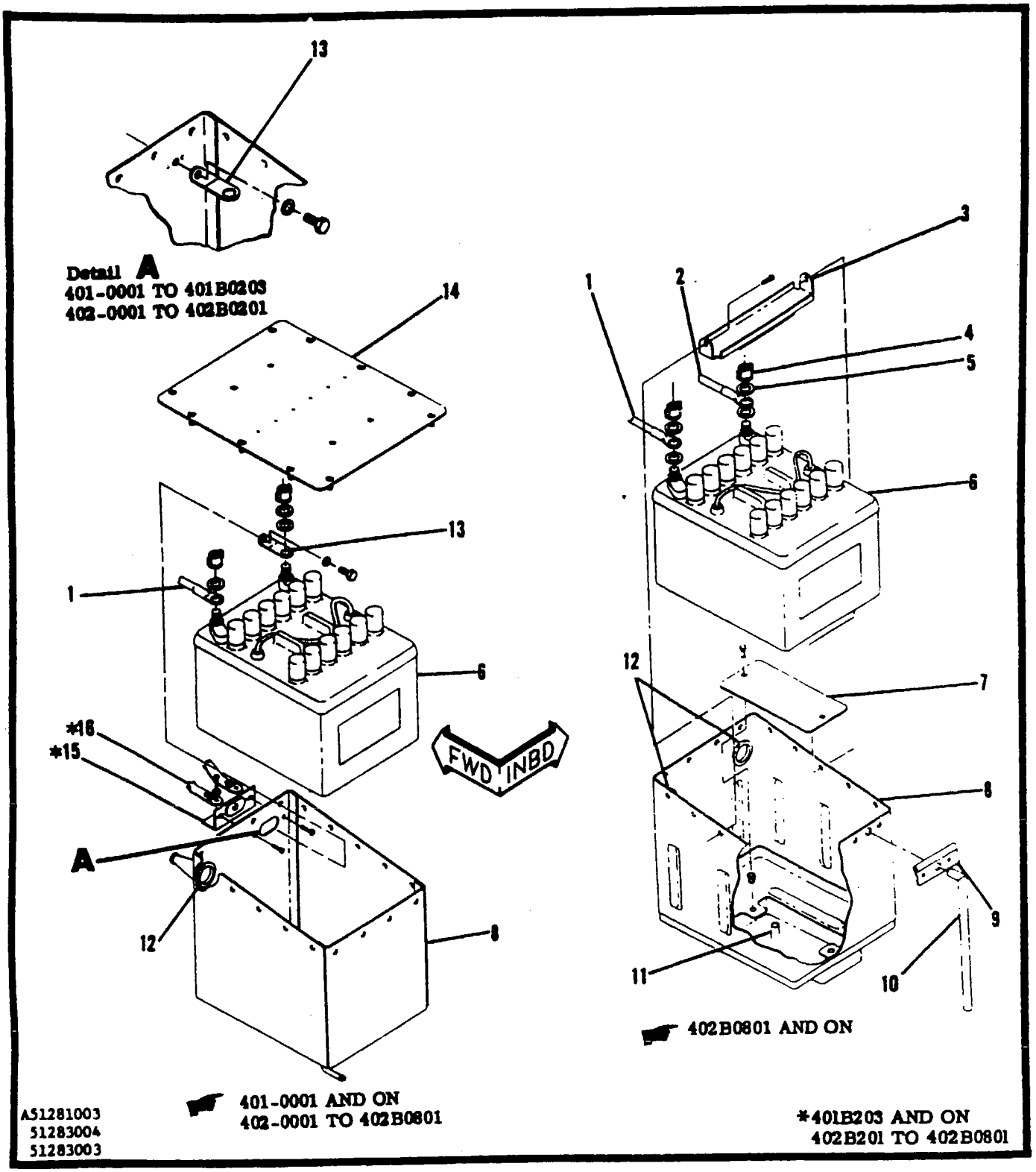
All openings in compressor shall be capped off to assure that oil in the compressor does not drain out during installation.

c. Secure baffle to compressor (9) with screws, before installation.
d. Position compressor (9) in the compartment.
e. Secure compressor (9) to mount (10) with bolts and washers.
f. Install hydraulic motor. Refer to Hydraulic Motor Installation procedures.
g. Connect lines (11 and 8) to the compressor (9).
h. If lines were removed to gain access, reinstall lines.
i. Evacuate and charge system. Refer to Air Conditioning System Service/Parts Manual.
j. Leak check system.
k. Install RH nacelle air conditioning access cover.



1. Screw
2. Cover
3. Indicator
4. Grommet
5. Deck Cover
6. Mounting Bracket
7. Wing Leading Edge
8. Transducer
9. Screw
10. Access Cover
11. PSW Screw
12. CM Screw

Figure 13-43. Angle of Attack System



1. Power Cable
2. Ground Cable
3. Tie Down Strap
4. Wing Nut
5. Lockwasher
6. Battery
7. Sump Cover
8. Battery Box
9. Seal
10. Vent Tube
11. Drain Tube
12. Grommet
13. Bus Bar
14. Battery Access Cover
15. Angle Assembly
16. Bonding Jumper

Figure 14-3. Battery and Battery Box Installation

Removal and Installation of Side Console Circuit Breaker Panel.

NOTE

When optional electroluminescent panels are installed, these panels must be removed first, in order to gain access to circuit breaker panel attaching screws.

a. Remove the four attaching screws, one at each corner, and lift panel up and out.

NOTE

Panel is still attached to console by a wire bundle assembly. Components may be removed, replaced or installed with panel in this position.

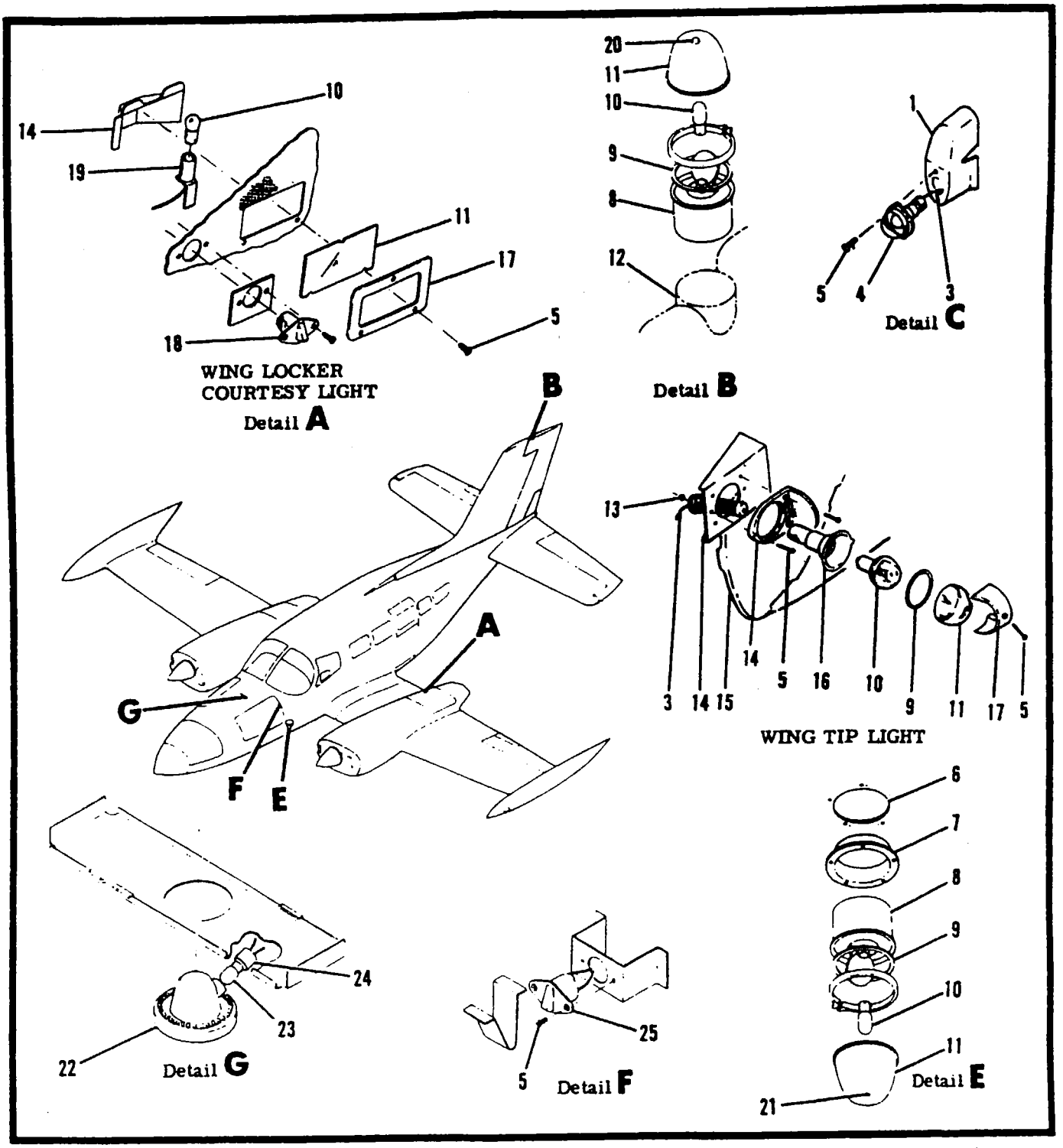
b. Remove and replace components as may be required.
c. If panel is to be removed, tag and disconnect electrical leads.
d. Remove panel from aircraft.
e. Install panel using reverse order.

LIGHTING SYSTEM.

The standard exterior lighting system consists of the navigation lights, tail light, top and bottom rotating beacons and a left landing light. Optional exterior lights consist of the right landing light, deice light and taxi light. The standard interior lights consist of a cabin door light, a dome light, four cabin reading lights, overhead console lights, a map light, three left-hand console panel lights located in the pilot's armrest and fuel selector lights. On aircraft 402B1301 and on, the cabin courtesy light incorporates a delay timer located in the side console to turn OFF the cabin courtesy light after 15 minutes of operation. The instrument panel lights consist of the warning lights, compass light, a call letter light and the instrument mounted post lights. If desired, an optional lighted seat belt sign and post lights for the optional instruments may be installed.

On aircraft 402B0001 and on, a dimmer control and a transistor heat sink assembly controls lighting intensity. On aircraft 402B0001 and on, the transistor heat sink is located on the forward right side at the Station 100.00 bulkhead. On aircraft 402B0001 to 402B0301, the dimmer control is mounted on the stationary switch panel. On aircraft 402B0301 and on, the dimmer control is mounted on the center pedestal.

On aircraft 401B0001 and on and 402B0001 to 402B0301, a dimming control for auxiliary fuel quantity lights is mounted on the forward side of the stationary switch panel at the inboard side of the glove compartment. On aircraft 402B0301 and on, it is mounted above the center pedestal.

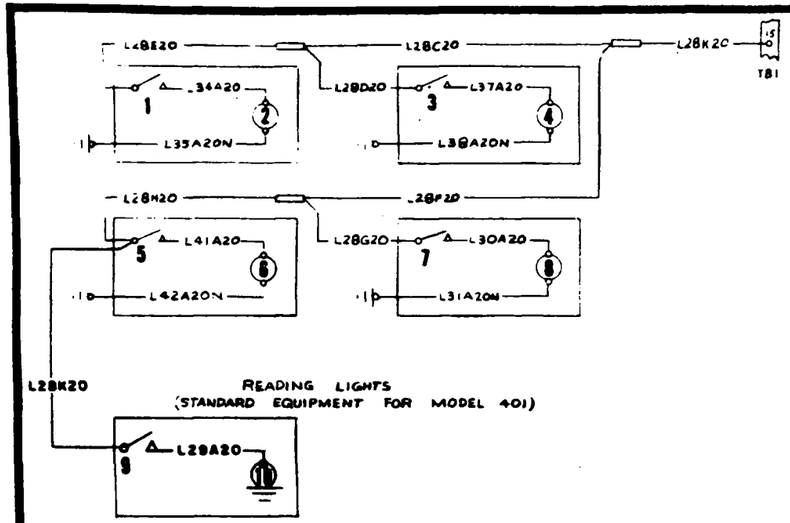


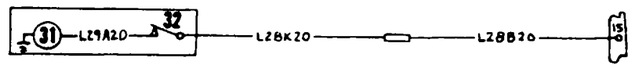
1. Stinger Cap
2. Deleted
3. Wire
4. Tail Light Assembly
5. Screw
6. Skin (Lower)
7. Mounting Ring
8. Beacon
9. Gasket
10. Bulb
11. Lens
12. Fin Mount
13. Nut
14. Bracket
15. Tip Tank
16. Socket
17. Lens Retainer
18. Switch
19. Socket
20. Plug
21. Drain Hole
22. Light - Nose Baggage
23. Bulb
24. Light Socket
25. Switch

Figure 14-9. Exterior Lighting

NONCONTINUOUS LOAD

	Amps Required
Landing Gear Motor	25.00
Flap Motor	13.00
Cabin Heater Overheat Indicator	0.04
Cigar Lighter (Load Required for One Lighter)	5.5
Landing Gear Relay	0.45
Fuel Boost Relay (.09 Amps Each)	0.18
Landing Gear Warning System	0.60
Stall Warning	0.20
Door Warning Indicator	0.04
Door Courtesy Light	0.30
In-Transit Landing Gear Indicator	0.10
Landing Gear Down Indicator	0.20
Landing Light - L.H. (Lamp)	9.0
Landing Light Extension and Retraction Motor	1.25
Shaver Inverter	0.60
Propeller Synchronous Indicator Lights	0.08
Map Light (Copilot)	[illegible]
Electric Seat (10 Amps Each)	[illegible]
Deice Light Wing (.70 Amps Each)	[illegible]
Intercom Call Light	[illegible]
Surface Deice System	[illegible]
Electric Trim	[illegible]
Taxi Light	[illegible]
Landing Light - R.H. (Lamp)	[illegible]
Landing Light Extension and Retraction Motor - R.H.	[illegible]
Cabin Ventilation Blower	[illegible]
Low Fuel Warning (2)	0.20





	Part Number	Nomenclature
1.	S-1158-2-1	Sw - Reading Lt - LH
2.	2225	Reading Lt - L
3.	S-1158-2-1	Sw - Reading Lt - LH
4.	2225	Reading Lt - L
5.	S-1158-2-1	Sw - Reading Lt - RH
6.	2225	Reading Lt - R
7.	S-1158-2-1	Sw - Reading Lt - RH
8.	2225	Reading Lt - R
9.	0713029	Sw - Dome Lt
10.	2109	Dome Lt
11.	OR-55-2	Resistor - Cigar Ltr - RH Rear
12.	051362-6	Cigar Ltr - RH Rear
13.	OR-55-2	Resistor - Cigar Ltr - LH Rear
14.	0513062-6	Cigar Ltr - LH Rear
15.	S-1158-2-1	Sw - Reading Lt Aft - LH
16.	2225	Reading Lt - LH Aft
17.	S-1158-2-1	Sw - Reading Lt - LH
18.	2225	Reading Lt - LH
19.	S-1158-2-1	Sw - Reading Lt - LH
20.	2225	Reading Lt - LH
21.	S-1158-2-1	Sw - Reading Lt Fwd - LH
22.	2225	Reading Lt Fwd - LH
23.	S-1158-2-1	Sw - Reading Lt Aft - RH
24.	2225	Reading Lt Aft - RH
25.	S-1158-2-1	Sw - Reading Lt - RH
26.	2225	Reading Lt - RH
27.	S-1158-2-1	Sw - Reading Lt - RH
28.	2225	Reading Lt - RH
29.	S-1158-2-1	Sw - Reading Lt Fwd - RH
30.	2225	Reading Lt Fwd - RH
31.	2109	Dome Lt
32.	0713029	Sw - Dome Lt

EFF: 401/402-0001 TO 401A/402A-0001

Cessna. P. O. BOX 1977 WALLACE-PROSPECT PLANT WICHITA, KANSAS

TITLE: WIRING DIAGRAM - READING LIGHTS & CIGAR LIGHTERS

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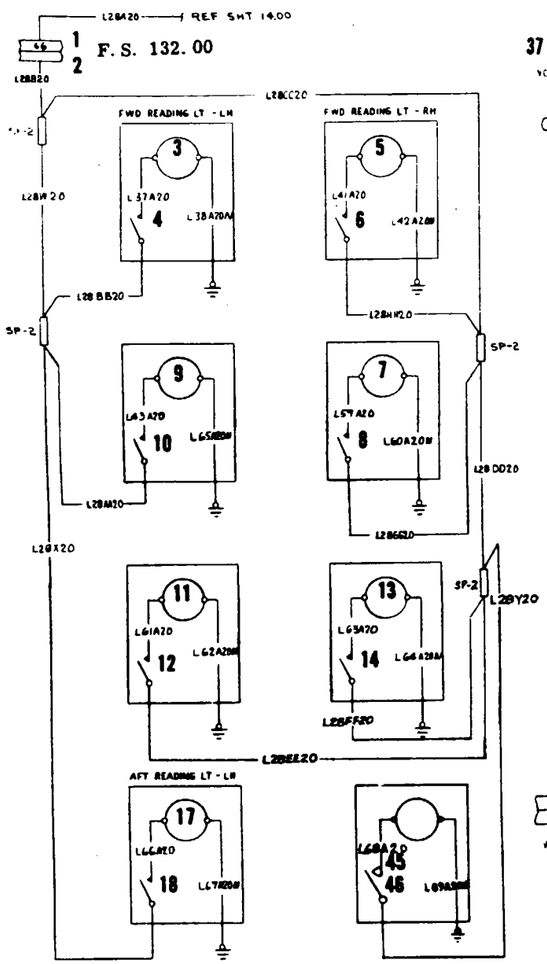
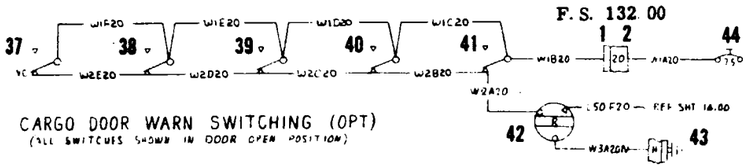
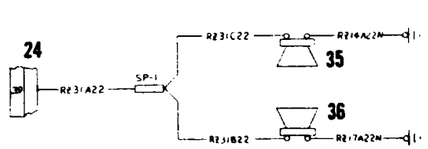
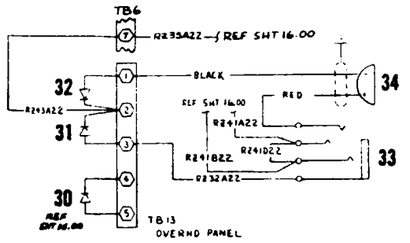
SCALE NONE REV B SHEET 9.00 OF 9.00

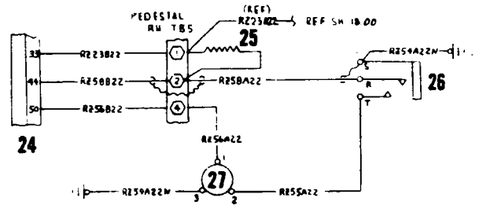


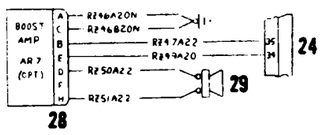
OPTIONAL READING LIGHTS - MODEL 402

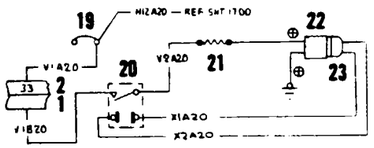












INTERCOMM (OPT)

	Part Number	Nomenclature
1.	201310-3	Recp - Lwr Side Console
2.	201311-3	Plug - Lwr Side Console
3.	2375-25	Reading Lt - LH
4.	S-1158-2-1	Switch - Reading Lts
5.	2375-24	Reading Lt - RH
6.	S-1158-2-1	Switch - Reading Lts
7.	2375-24	Reading Lt - RH
8.	S-1158-2-1	Switch - Reading Lts
9.	2375-25	Reading Lt - LH
10.	S-1158-2-1	Switch - Reading Lts
11.	2375-25	Reading Lt - LH
12.	S-1158-2-1	Switch - Reading Lts
13.	2375-24	Reading Lt - RH
14.	S-1158-2-1	Switch - Reading Lts
15.		
16.		
17.	2375-25	Reading Lt - LH
18.	S-1158-2-1	Switch - Reading Lts
19.	S-1232-507	Ckt Brk - Shaver Inv
20.	7777K1	Recp - 100 V AC
21.	OR-40-5	Resistor - Shaver Inv
22.	12SP2A	Inverter - Shaver
23.	865	Plug - Shaver
24.	7777K1	Recp - 100V AC
25.	470Ω 1 Watt	Resistor - Intercomm
26.	S-1102-1	Mike Jack - Intercomm
27.	VM911M-3	Intercomm Call Lt
28.	126-222	Plug - Boost Ampl
29.	F8WP	Spkr - Fwd P.A.
30.	IN2482	Diode - Ovhd Flood Lt
31.	IN2482	Diode - Audio
32.	IN2482	Diode - Audio
33.	S-13-B	Mike Jack Boom Mike
34.	C596503-0101	Microphone - Boom Mike
35.	55-4	Spkr - Aft P.A. & Stereo
36.	55-4	Spkr - Aft P.A. & Stereo
37.		
38.		
39.	CM2931-1	Switch - Door Warn
40.	CM2931-1	Switch - Door Warn
41.	MS25253-1	Switch - Door Warn
42.	VM911M-2	Door Warn Ind Lt
43.	203540-1	Plug - LH Grd
44.	S-1232-507	Ckt Bkr - Radio Lt
45.	2375-25	Reading Lt RH
46.	S-1158-2	Switch - Reading Lts

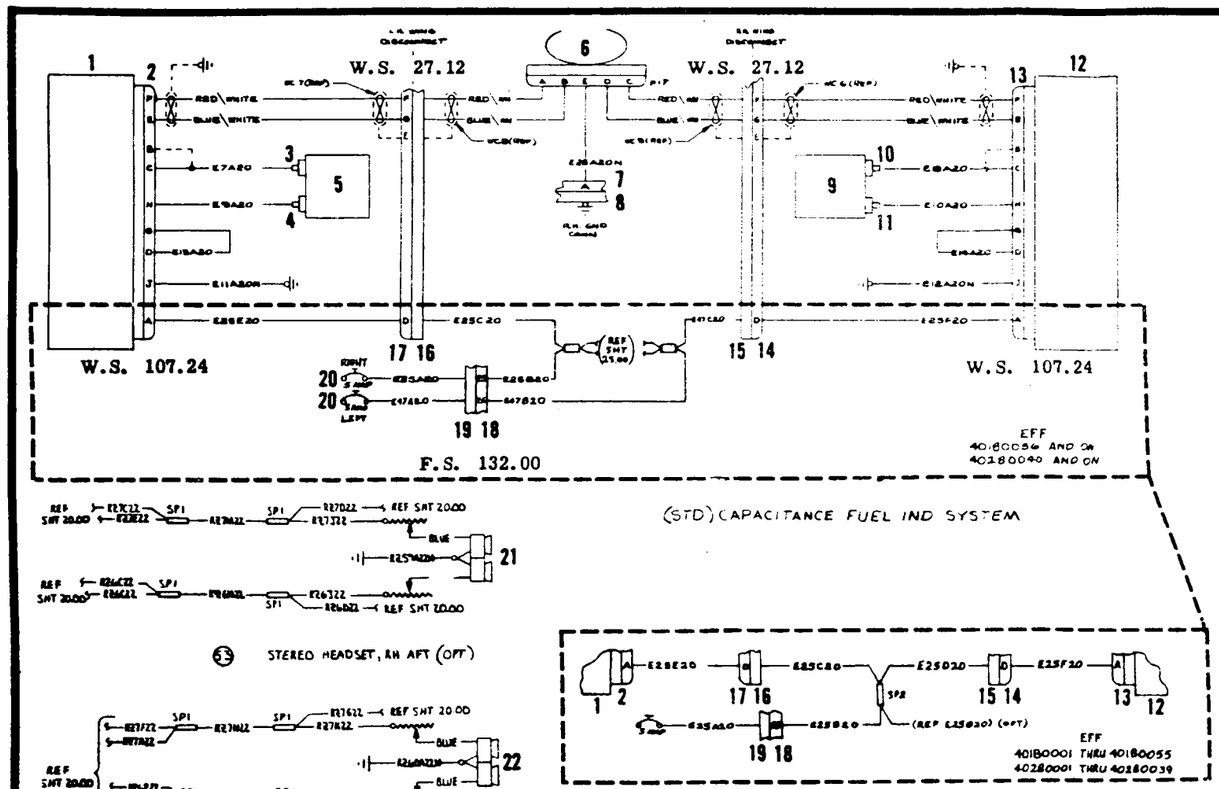
401A/402A0001 to 401B/402B0001

Cessna. P. O. BOX 1977 WALLACE-PROSPECT PLANT WICHITA, KANSAS

OPTIONAL EQUIPMENT - READING LTS, AFT P.A., FWD P.A., BOOM MIKE, INTERCOMM, CARGO DR WARN SWITCHING, SHAVER INV - WIRING DIAGRAM

SIZE D | CAGE IDENT. NO. 71379 | DRAWING NO. 5218008

SCALE NONE | REV A. | SHEET 19.00 of 21.00



	Part Number	Nomenclature
1.		LH Signal Cond
2.	PT06-12-105	Plug - Sig Cond
3.	1-817-1	Plug Cap - Fuel
4.	1-906-1	Plug Cap - Fuel
5.	391053-001	Tank Unit - LH Tip
6.	393011-002	Dual Fuel Qty Ind
7.	201298-1	Recp - Inst Pnl RH Gnd
8.	203540-1	Plug - RH Gnd
9.	391053-001	Tank Unit - RH Tip
10.	1-817-1	Plug Cap - Fuel
11.	1-906-1	Plug Cap - Fuel
12.		RH Signal Cond
13.	PT06-12-105	Plug - RH Sig Cond
14.	MS3106A28-21P	Plug - RH Wing
15.	MS3100A28-21S	Recp - RH Wing
16.	MS3100A28-21S	Recp - LH Wing
17.	MS3106A28-21P	Plug - LH Wing
18.	1-203739-5	Plug - Fwd Ckt Bkr Pnl
19.	1-203739-6	Recp - Fwd Ckt Bkr Pnl
20.	S-1232-505	Ckt Bkr - Fuel Qty, Aux
21.	060880-01TWA	Speaker - Dual Transducer
22.	060880-01TWA	Speaker - Dual Transducer
23.	060880-01TWA	Speaker - Dual Transducer

401B0001 AND ON
402B0001 TO 402B0301

Cessna
CAP FUEL IND SYSTEM, OPT STEREO HEADSETS
D 71378 5208001 REV 8

402B0301 AND ON

WIRING DIAGRAMS

REFERENCE DESIGNATOR CODE:

Reference designators are used in these diagrams to more quickly identify and locate a part or assembly used in an electrical system. A part is identified by a reference number on a diagram. The reference number is called out in page 14-77 and identified by part number, description and diagram sheet location.

EXAMPLE -

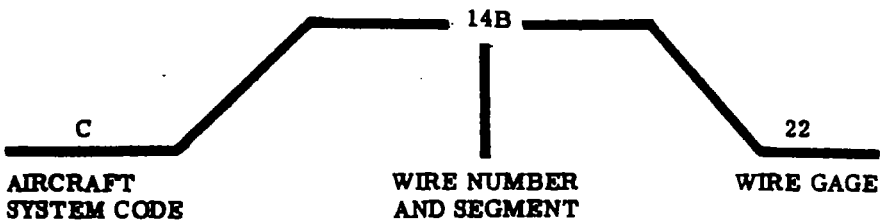
REF. DESIGNATOR	DIAGRAM SHT. NO.	PART NO.	DESCRIPTION
BB20	6	0820001-1	BATTERY BUS

NOTE

Part Numbers referred to on Wiring Diagrams are for reference only. When ordering spare or replacement parts, refer to applicable Parts Catalog.

The Wiring Diagrams contained in this section clearly show the complete wiring on each item of electrical component listed in the Wiring Diagram Index. The first portion of wire number indicates the Aircraft System, the center portion indicates Wire Number, and the last portion indicates Wire Gage Size. Each Wiring Diagram contains part number and nomenclature for each component.

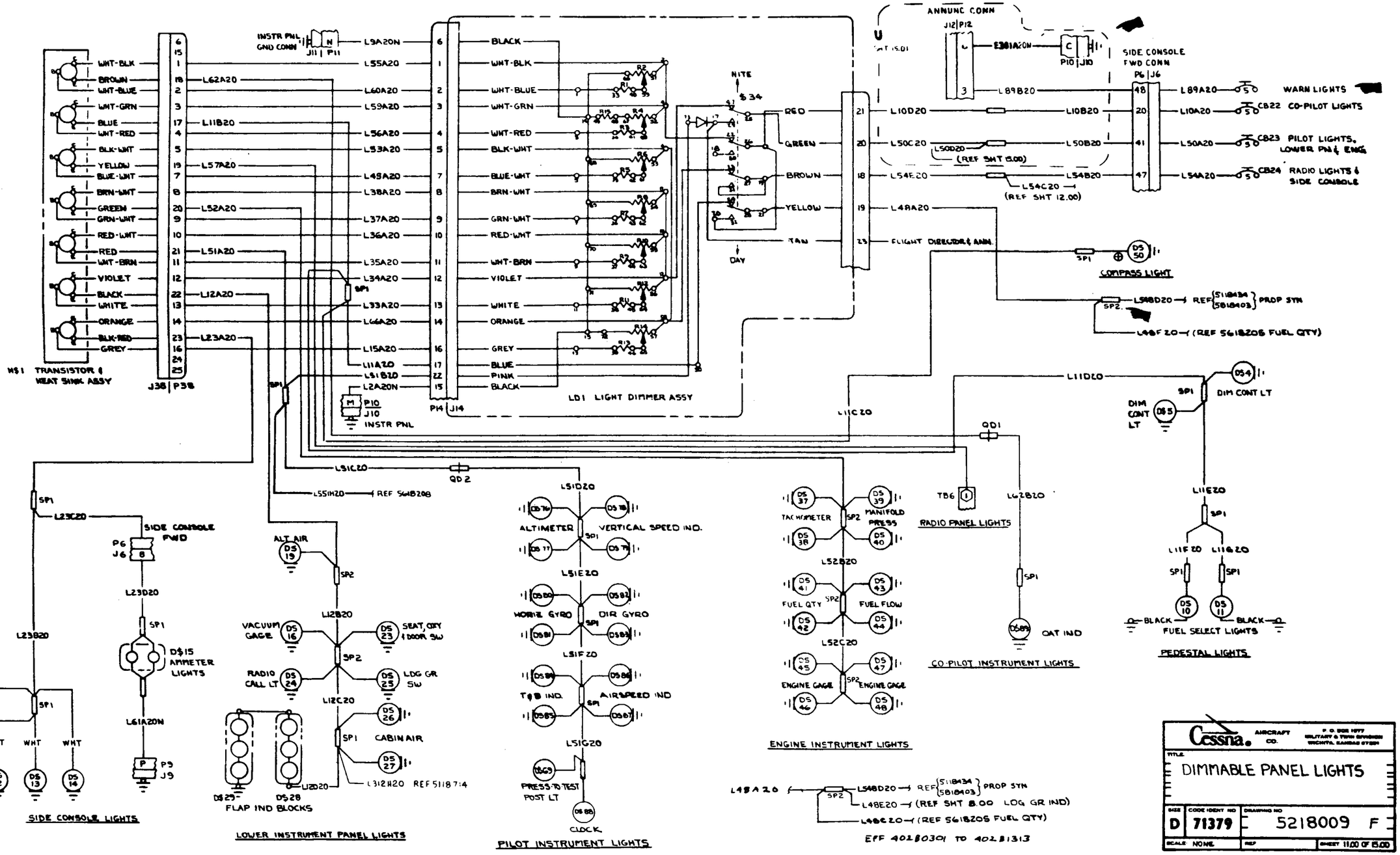
EXAMPLE: C14B22



Three types of wire numbers are shown for optional equipment. An example of the various wire numbers is as follows:

E27B18 E327B18 E527B18

The only difference in the wire is in the installation schedule. The E27B wire numbering system was used on aircraft 402-0001 thru 402B0301. At 402B0301 and On, the same wire would be indicated as E327B18 or E527B18.

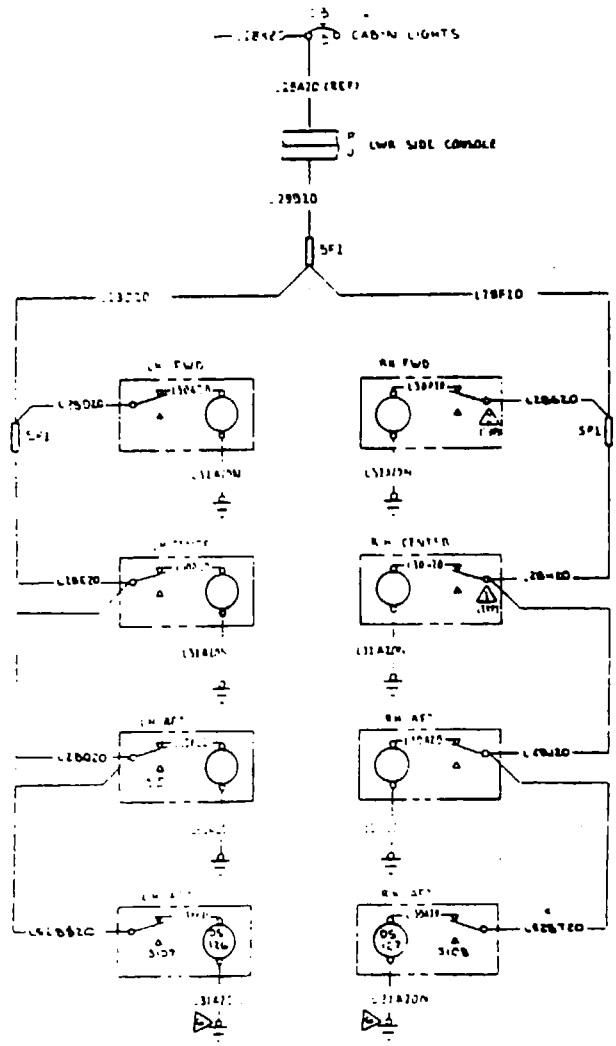


Cessna AIRCRAFT CO.

TITLE: DIMMABLE PANEL LIGHTS

SIZE	CODE IDENT NO	DRAWING NO
D	71379	5218009 F

SCALE: NONE SHEET 11.00 OF 15.00

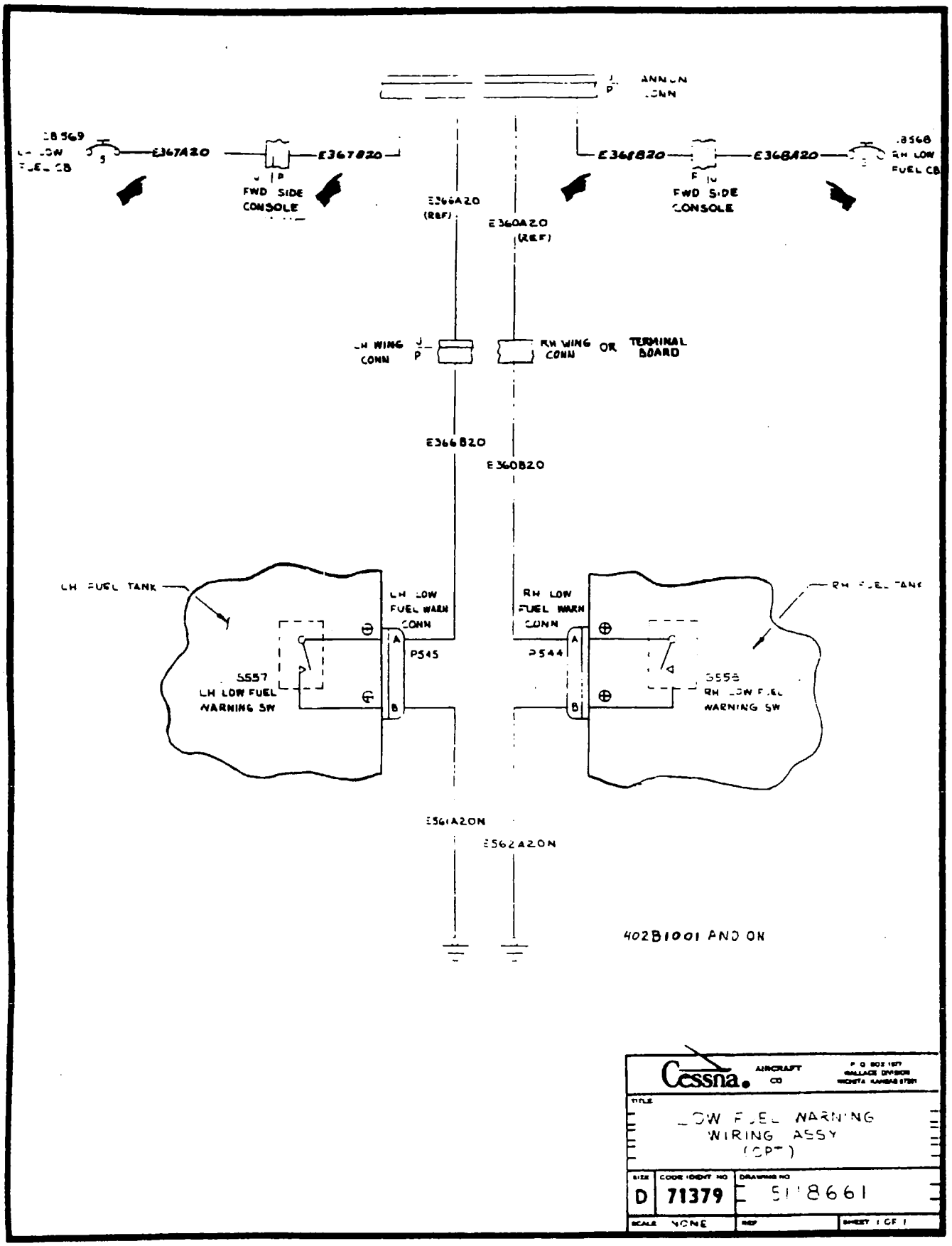


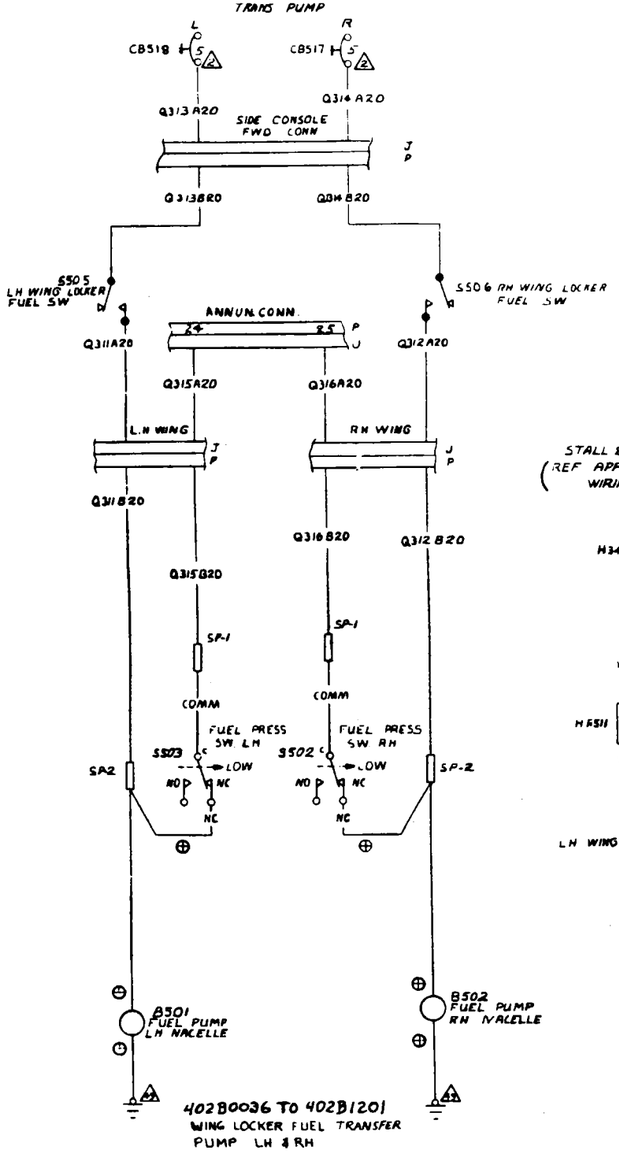
EQUIPMENT LIST		
SYMBOL	PART No.	PART NAME
DS126	[illegible]	READING LIGHT LH
DS127	[illegible]	READING LIGHT RH
S1C1	[illegible]	SWITCH
S1C8	[illegible]	SWITCH
	[illegible]	TERMINAL
	[illegible]	TERMINAL
SP1	[illegible]	SPLICE
	[illegible]	TERMINAL

Cessna. AIRCRAFT CO

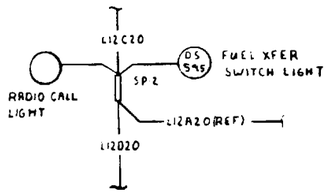
[illegible] WIRING ASSEMBLY

SIZE	CAGE IDENT NO	SPECIFICATION NO
D	71379	5218118

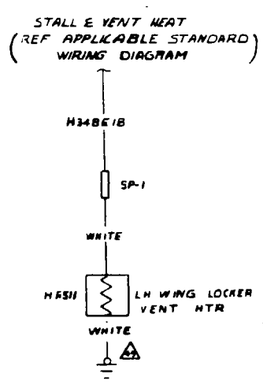

ANNUN CONN
J
P
E368A20
E367A20
E367B20
E366B20
E368B20
FWD SIDE CONSOLE
FWD SIDE CONSOLE
E366A20 (REF)
E360A20 (REF)
LH WING CONN
RH WING CONN
OR TERMINAL BOARD
E366B20
E360B20
LH FUEL TANK
RH FUEL TANK
LH LOW FUEL WARN CONN
RH LOW FUEL WARN CONN
P545
P544
5557 LH LOW FUEL WARNING SW
5558 RH LOW FUEL WARNING SW
E561A20N
E562A20N
402B1001 AND ON
Cessna AIRCRAFT CO
WALLACE DIVISION
WICHITA KANSAS 67201
TITLE
LOW FUEL WARNING WIRING ASSY (CPT)
SIZE D
CODE IDENT NO 71379
DRAWING NO 5118661
SCALE NONE
SHEET 1 OF 1



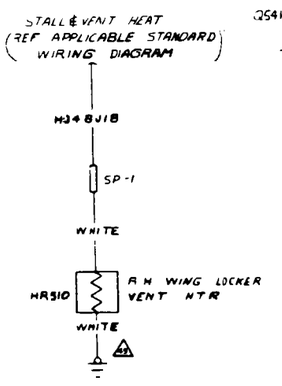
402B0036 TO 402B1201
WING LOCKER FUEL TRANSFER PUMP LH & RH



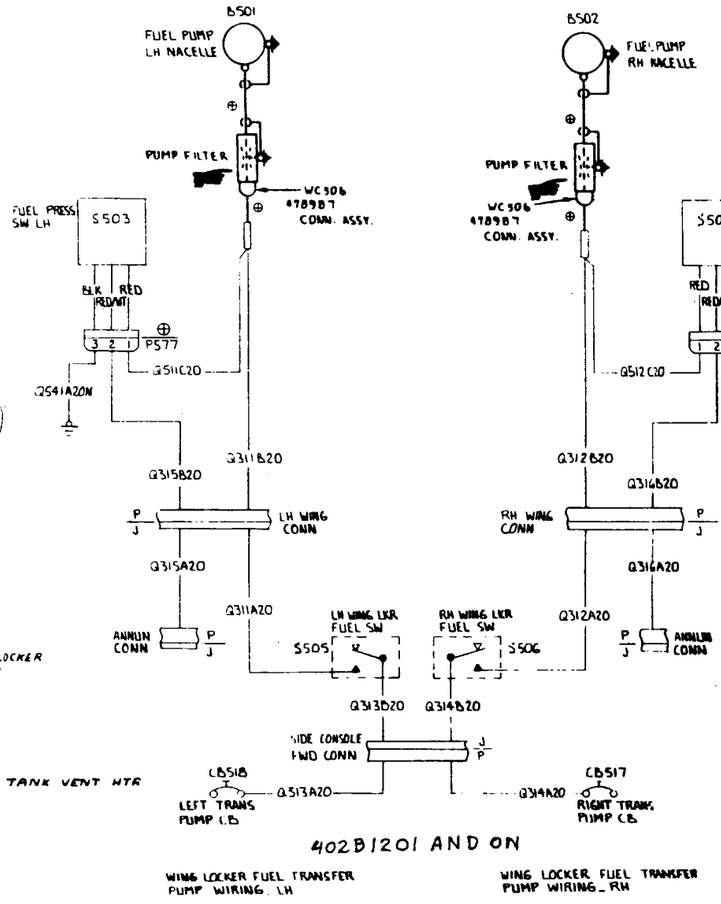
POST LIGHT WIRING (FUEL TRANS SW.)



LH WING LOCKER AUX TANK VENT HTR



RH WING LOCKER AUX TANK VENT HTR



402B1201 AND ON

WING LOCKER FUEL TRANSFER PUMP WIRING LH

WING LOCKER FUEL TRANSFER PUMP WIRING RH

EQUIPMENT LIST

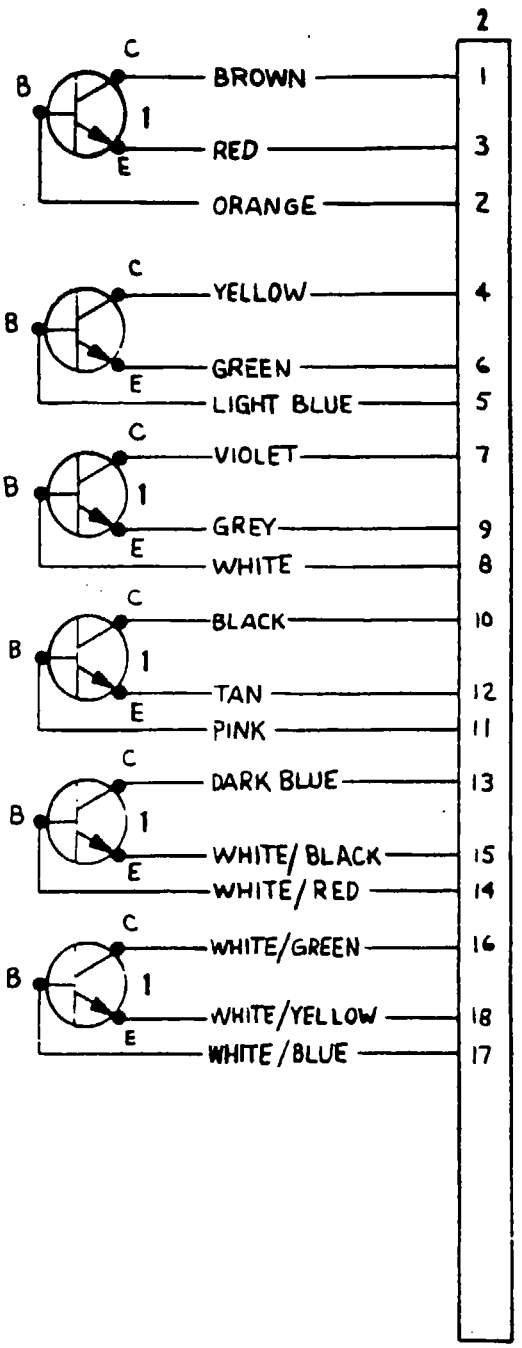
SYMBOL	PART NO.	PART NAME
B501	4140-00-153	FUEL PUMP LH NACELLE
B502	4140-00-153	FUEL PUMP RH NACELLE
CB	7277-2-5	TRANSFER PUMP C.B.
CB	7277-2-5	TRANSFER PUMP C.B.
HR510	105880	VENT HTR WING LOCKER R.H
HR511	105880	VENT HTR WING LOCKER L.H.
S502	9910058-1	SWITCH NAC FUEL PRESS
S503	9910058-1	SWITCH-NAC FUEL PRESS
S506	MST-205N	RH NAC AUX FUEL
S505	MST-205N	L.H. NAC AUX FUEL
SP1	320559	SPLICE
SP2	320562	SPLICE
1	MS25036-149	TERMINAL
2	MS25036-102	TERMINAL
DS595	C2990-2	POST LIGHT

Cessna AIRCRAFT CO. P O BOX 1677 MILITARY & TWIN DIVISION WICHITA KANSAS 67201

TITLE: WING LOCKER FUEL TRANSFER PUMP & VENT HTR RH & LH (OPT) WIRING DIAGRAM

SIZE	CODE IDENT NO	DRAWING NO
D	71379	5118441

SCALE: NONE REF SHEET 1 OF 2



Part Number	Nomenclature
1. MJE3055	Transistor
2. DB-25S	Receptacle

REFER TO PAGE 14-54
FOR SYSTEM WIRING

401B/402B0001 AND ON

Cessna.

TRANSISTOR & HEAT SINK ASSEMBLY

D | 71379 | 5618008

	Page	Fiche/ Frame
Cessna 800 Glide Slope	15-20C	4 J19
Cessna 800 RMI	15-20D	4 J21
Cessna 800 Transponder	15-20E	4 J23
Cessna 800 Antenna Interconnects	15-20G	4 K1
Cessna Transceiver 500	15-20H	4 K2
Cessna Nav/Com 500	15-21	4 K3
Cessna ADF 500 (Rotatable Loop)	15-22	4 K4
Cessna Dual ADF 500	15-23	4 K5
Cessna ADF 500 (Fixed Loop)	15-24	4 K6
Cessna Nav/Omni 500	15-25	4 K7
Cessna 500 Series Antenna Interconnects	15-26	4 K8
Cessna Nav/Com 400	15-26A	4 K9
Cessna Transceiver 400	15-26B	4 K11
Cessna Nav/Omni 400	15-26C	4 K13
Cessna ADF 400	15-26D	4 K15
Cessna Glide Slope 400	15-26E	4 K17
Cessna Transponder 400	15-26F	4 K18
Cessna 400 ADF	15-26G	4 K19
Cessna Marker Beacon 300	15-27	4 K20
Cessna Nav/Com 300	15-28A	4 K22
Cessna Nav/Omni 300	15-29	4 K23
Cessna Transceiver 300	15-30	4 K24
T-10-R Transceiver	15-31	4 L1
Cessna ADF 300	15-32	4 L2
Cessna 300 Transponder	15-32A	4 L3
Cessna 300 DME	15-32B	4 L4
External Power Amplifier and Cabin Interphone System	15-33	4 L5
AVQ-55 Weather Radar	15-34	4 L6
Audio Amplifier	15-34A	4 L7
Radio Junction Box Connections	15-36	4 L16
Cessna Nav-O-Matic 400 Autopilot	15-38	4 L19
Cessna 400A Nav-O-Matic Autopilot	15-38A	4 L21
T-22-RA Transceiver	15-39	5 A3
AVQ-45 and AVQ-46 Weather Radar	15-40	5 A5
PN101 Nav System Coupled with Nav Omni 800 and Nav-O-Matic 800 Autopilot	15-41	5 A7
Cessna Course Director 800	15-42	5 A9
Radio Cooling Blower	15-43	5 A11
Audio Code Filter	15-44	5 A12
Table of Electronic Equipment Loading	15-45	5 A13
Electronic Continuous Load Chart	15-62	5 E7

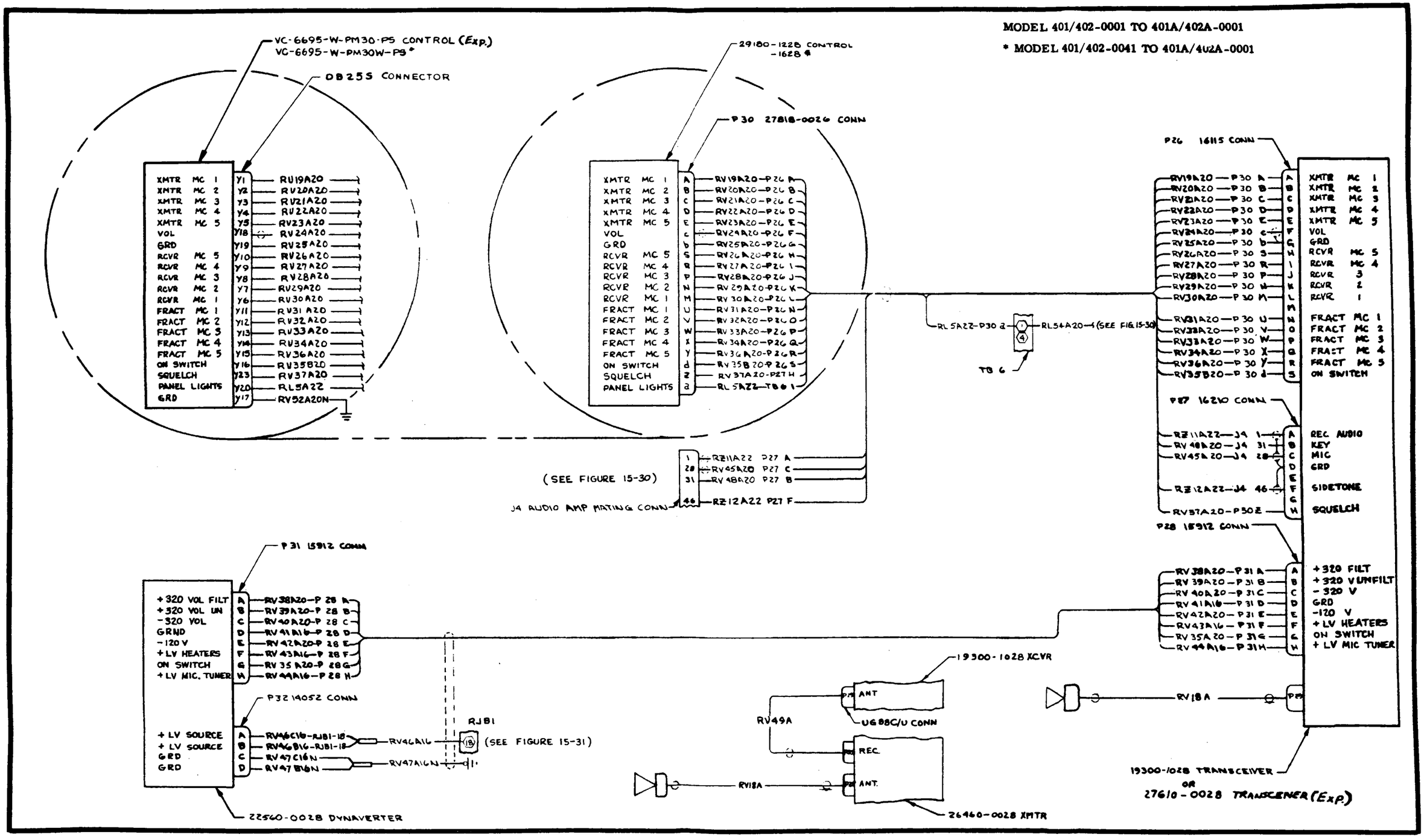
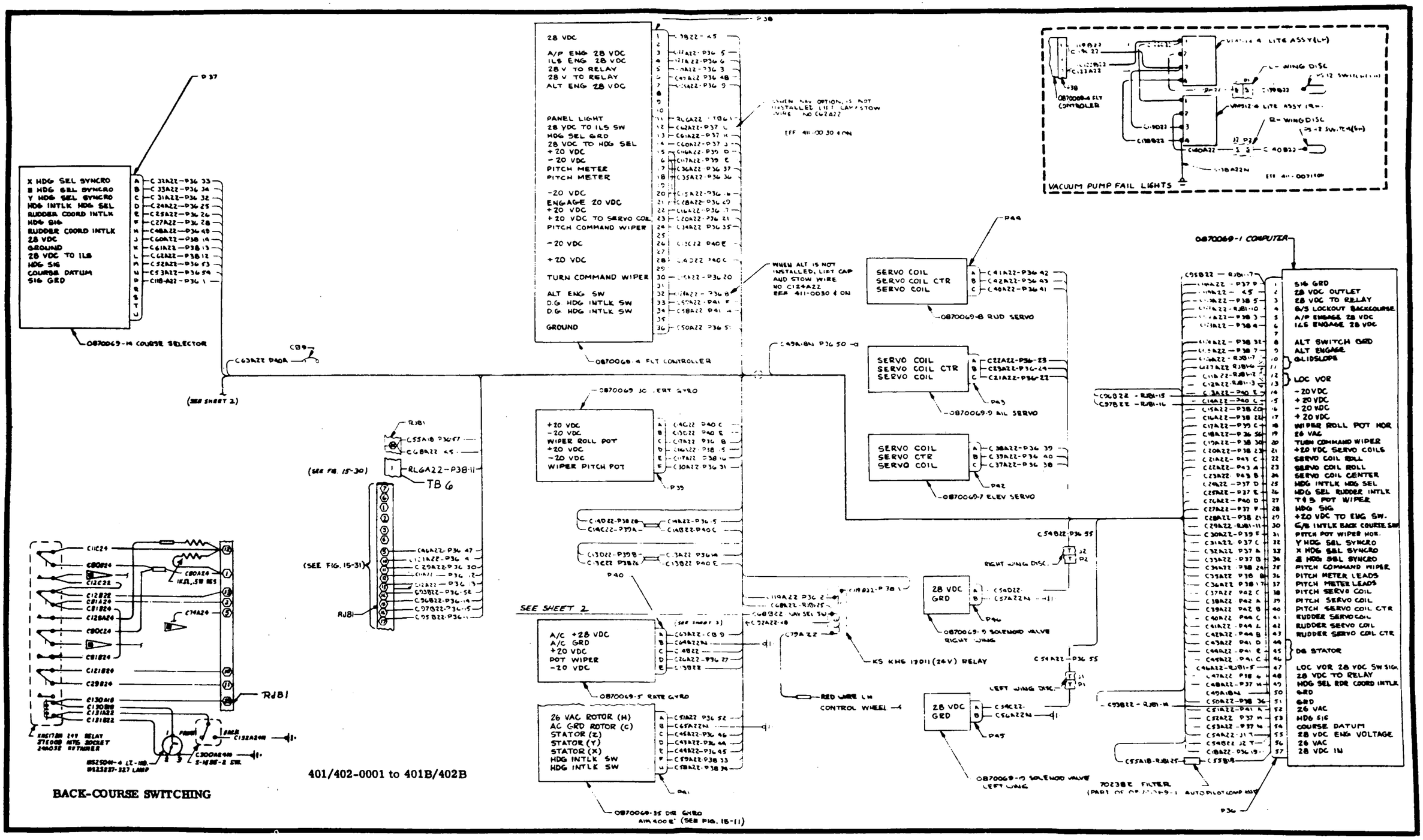


Figure 15-1D. Cessna Transceiver 800



BACK-COURSE SWITCHING

401/402-0001 to 401B/402B

Figure 15-12. Cessna Nav-O-Matic 800 Autopilot (Sheet 1 of 4)

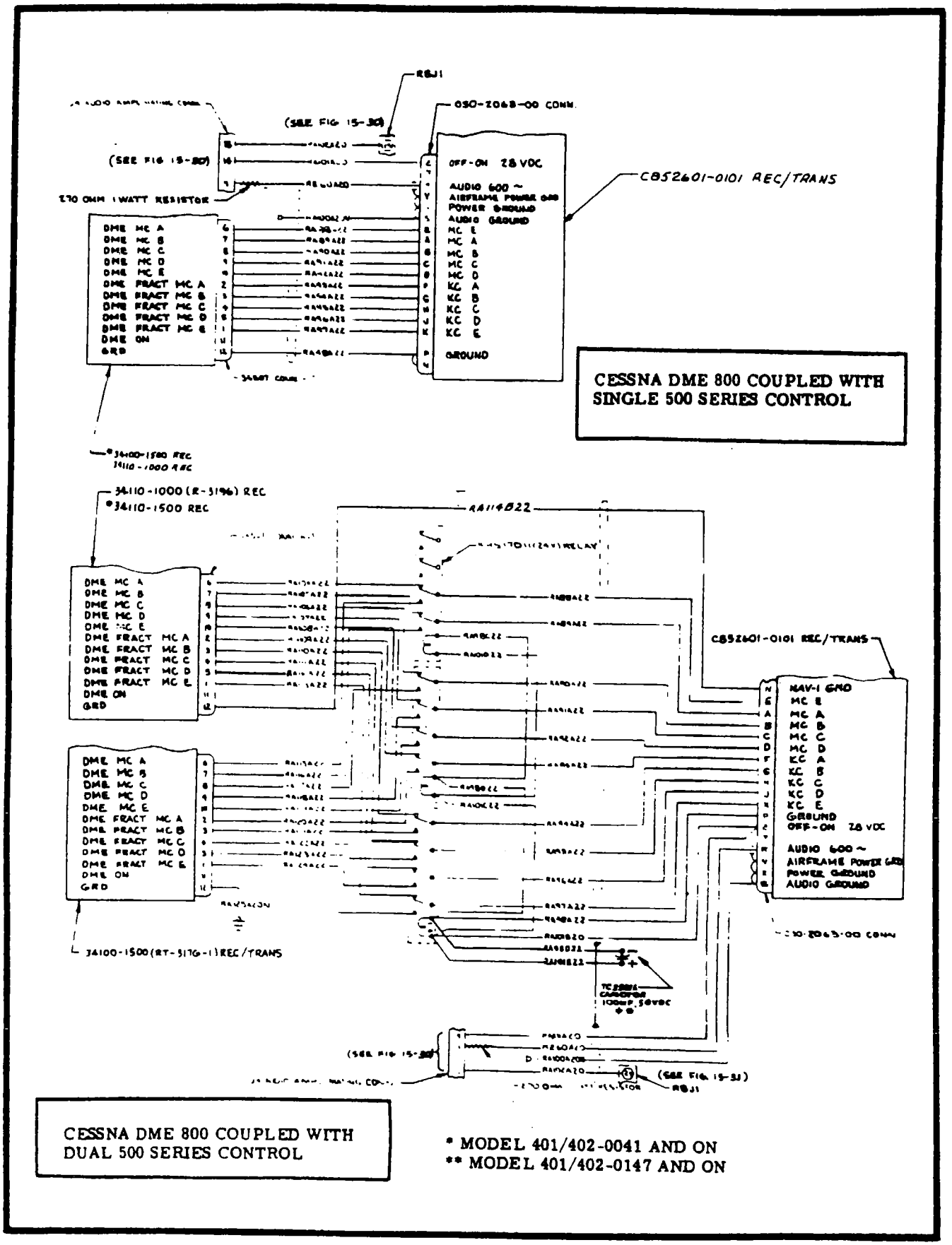


Figure 15-14. Cessna DME 800 (Sheet 2 of 3)

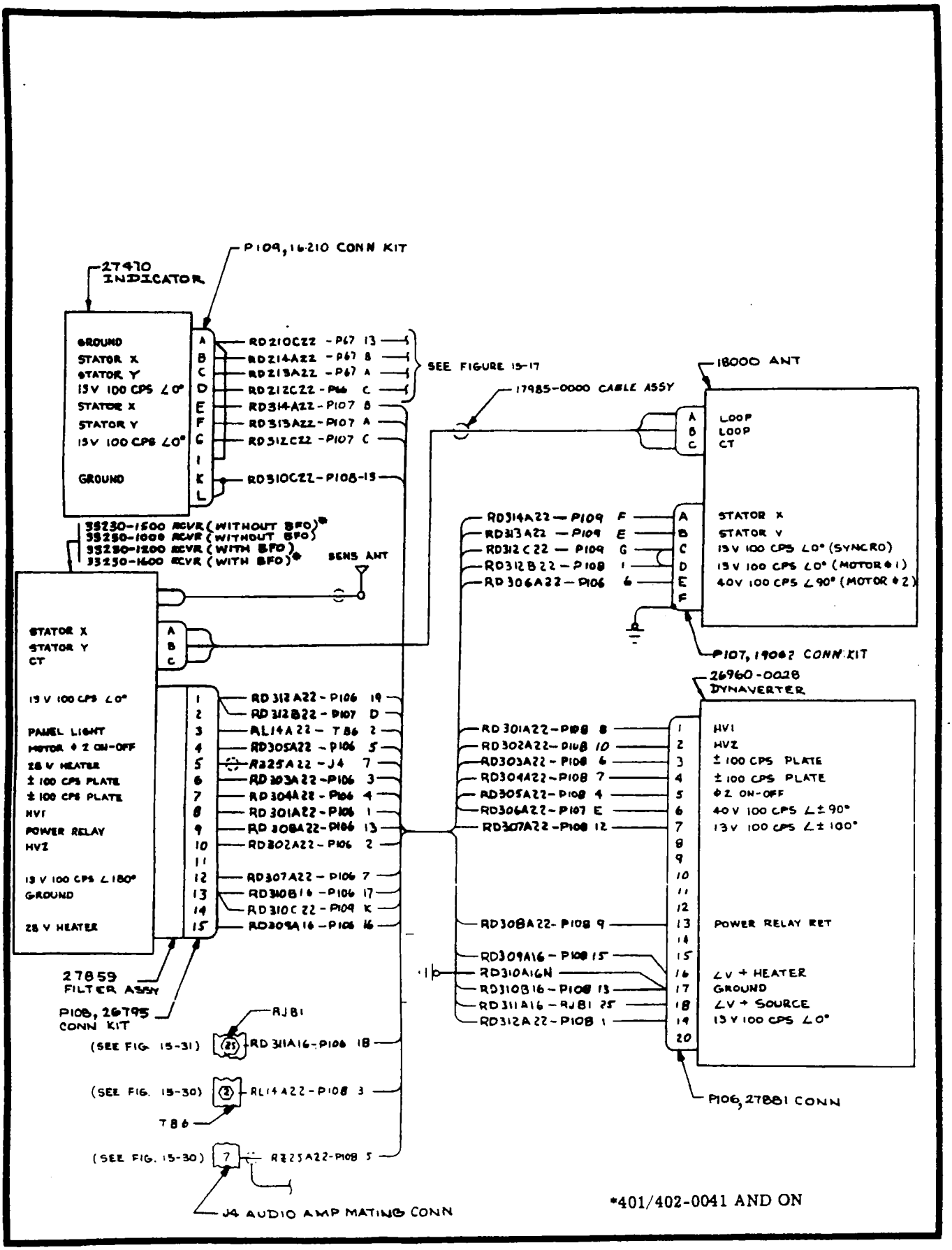


Figure 15-18. Cessna Dual ADF 500

Changed May 1967

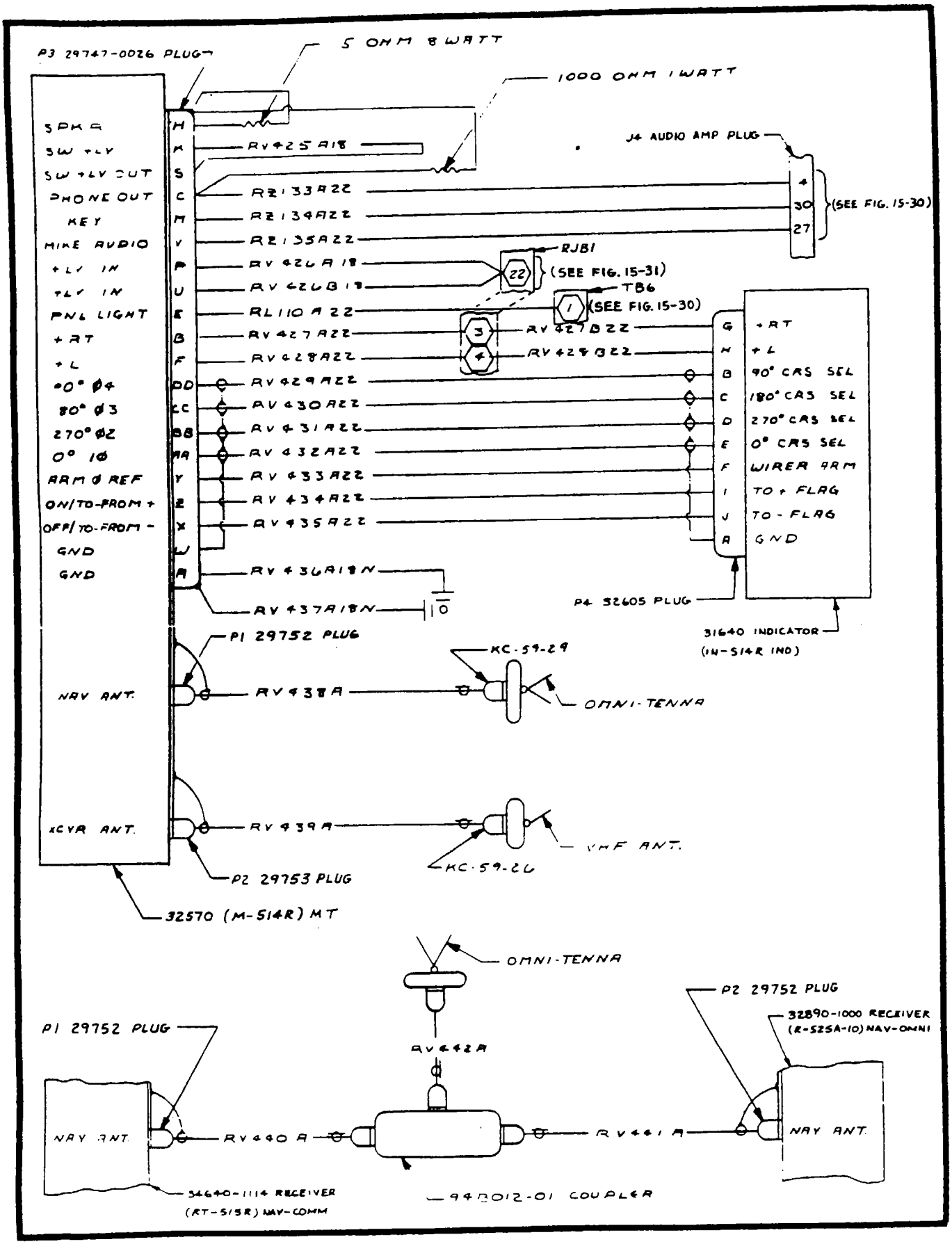


Figure 15-23. Cessna Nav/Comm 300

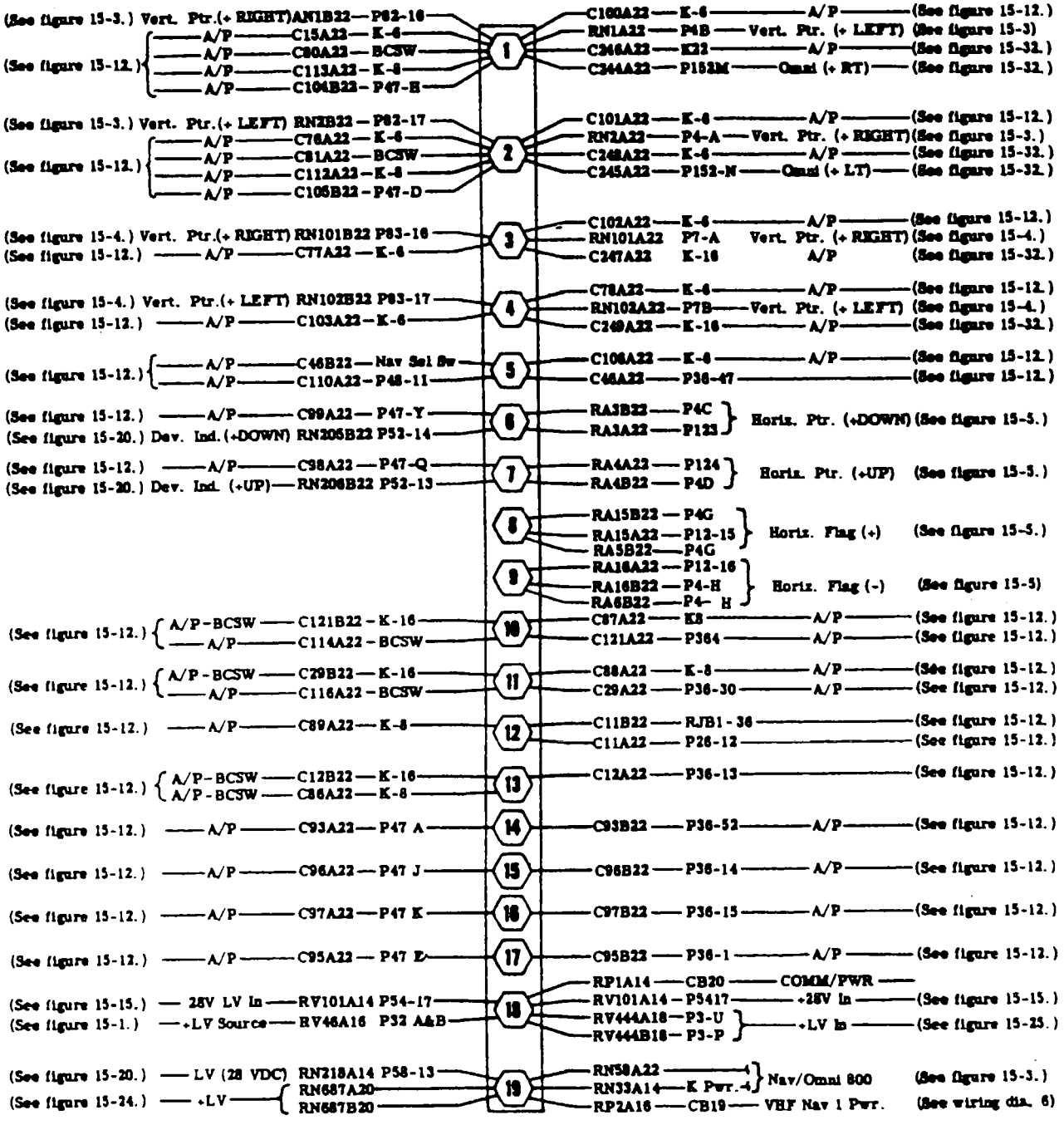
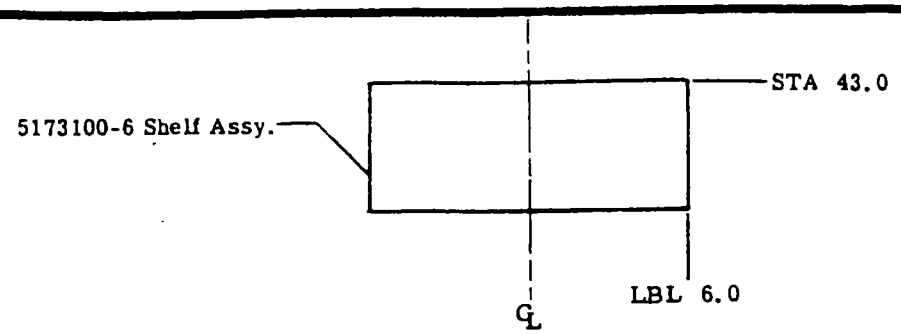
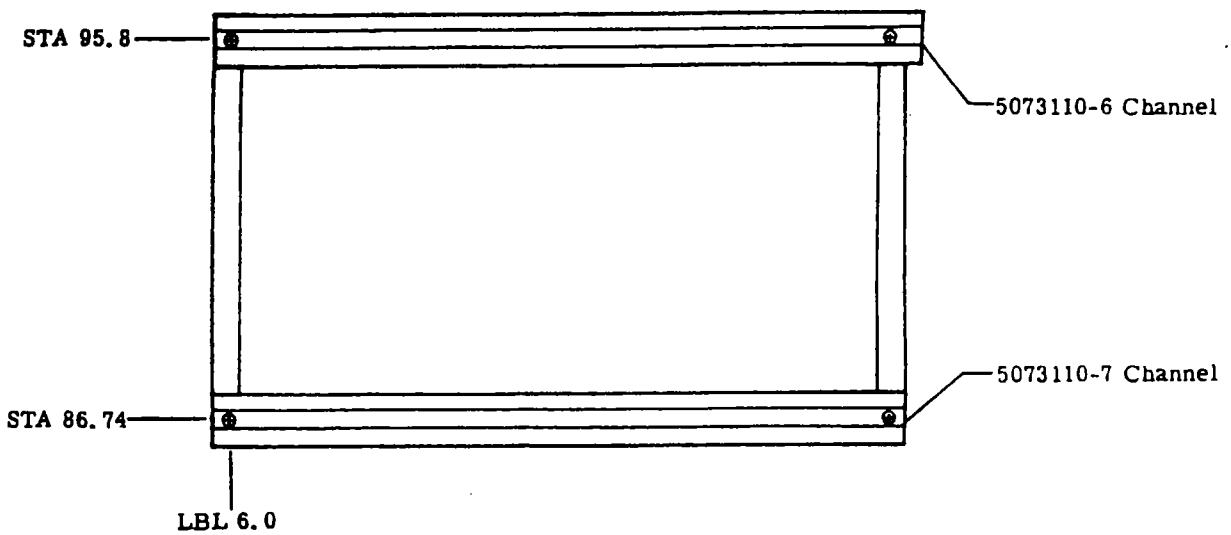


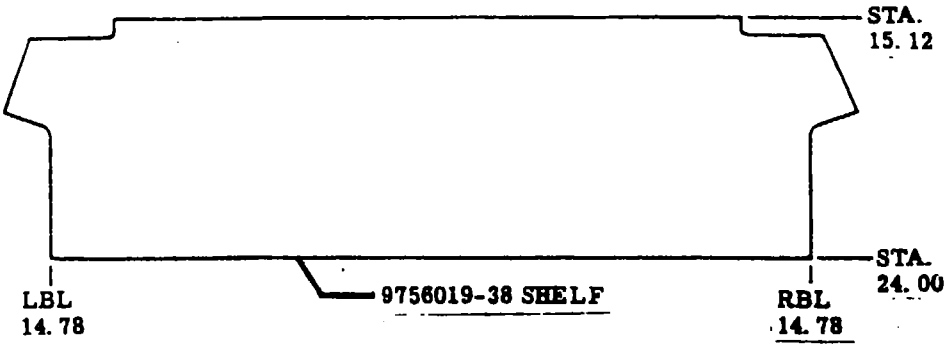
Figure 15-31. Radio Junction Box Connections (Sheet 1 of 2)



EQUIPMENT	COMPONENT WEIGHT	REMARKS
Transceiver 800 22560 Dynaverter 21455 Mounting	6.8	
Nav/Omni 500 24660 Converter	3.7	



EQUIPMENT	COMPONENT WEIGHT	REMARKS
Nav-O-Matic 800 Autopilot 0870069-1 Computer	9.9	W/ Mount
Glide Slope 800 21420-0028 Receiver	6.0	



402A-0001 AND ON

EQUIPMENT	COMPONENT WEIGHT	REMARKS
AVQ-45 Weather Radar		
MI-592021 Transceiver	14.0	
MI-592032 Mount	1.0	
ADF Receiver 800		
34850-0028 Receiver	7.4	Includes Mount
21660-0000 Mount		

TABLE OF ELECTRONIC EQUIPMENT LOADING

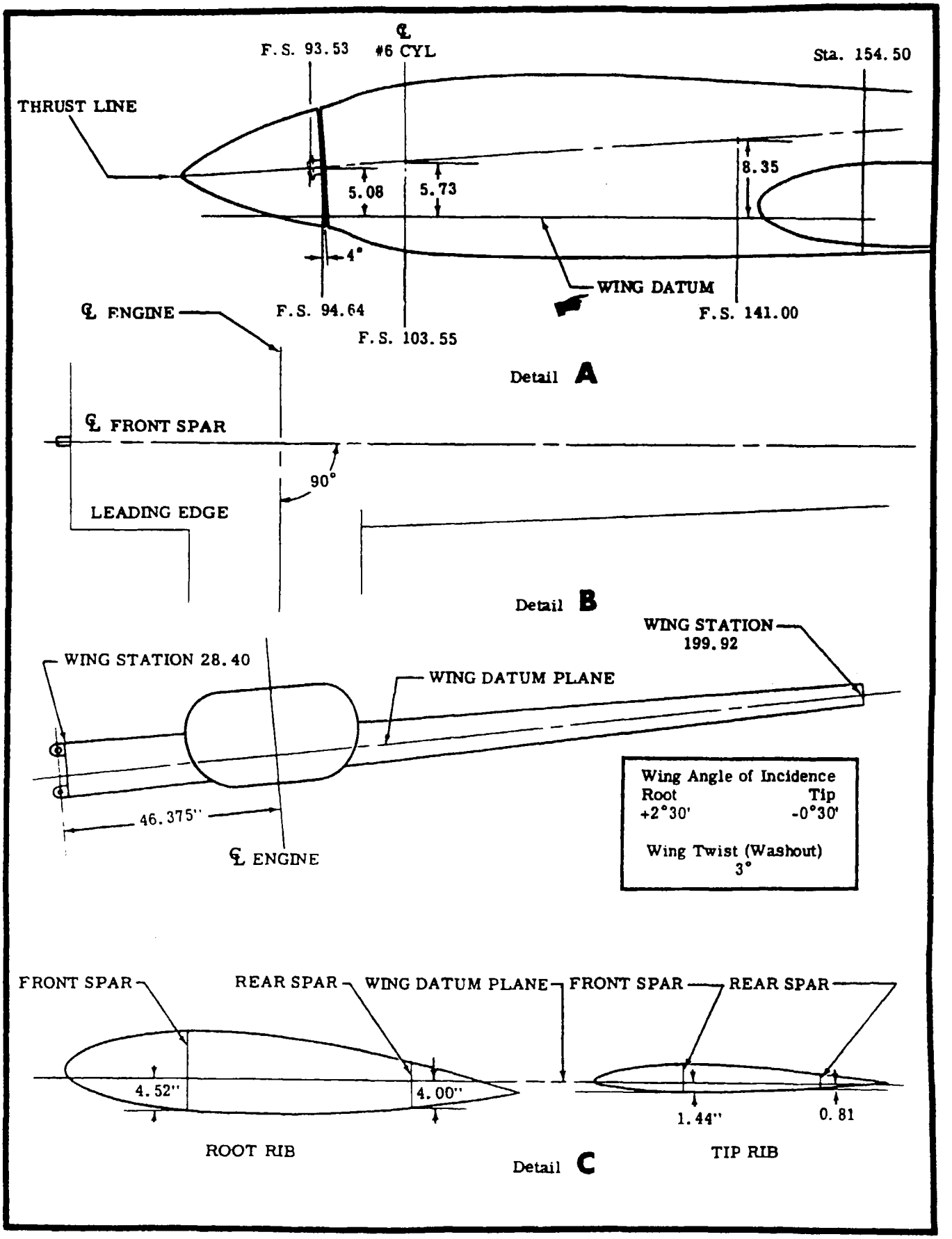


Figure 16-1. Wing Twist and Thrust Line Data

Method I (Stepped Joint Method).

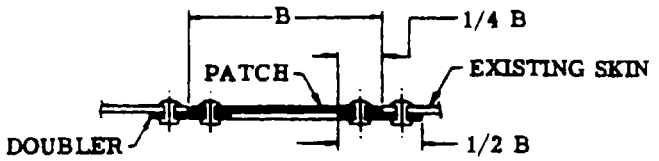
(a) The preferred method of removing damaged plies in accomplishing a Class II Repair is by the stepped joint method. For small damages the scarf method of repair may be used.

(b) Ascertain the extent of the damaged area by visual inspection, using a strong light source, prior to beginning repair. With aid of a straight edge or compass, outline the damaged area by scribing a rectangle with rounded corners or circle that will necessitate removal of a minimum of sound material. Extend the sides of the circle a distance in inches equal to the number of plies to be removed less one inch. (Five inches if six plies are to be removed, four inches if five plies are to be removed, etc.) Overlap should be at least one inch per ply of glass cloth.

(c) With the aid of a straight edge, use a sharp knife or other specially prepared cutter and cut along the lines scribed in the outermost ply. Use extreme care not to cut or score the underlying ply. A suggested method is to cut through the overlaying ply in a series of cuts rather than attempt to cut through the ply in one cut.

(d) Remove the cut outermost ply by inserting the knife blade under the corner and prying loose carefully. When this outermost ply is removed, scribe on the next exposed ply a similar outline except reducing the dimensions one inch in all directions. (Overlaps shall be one inch each ply.) Repeat this procedure until all the damaged plies have been removed. (See Figure 16-2.)

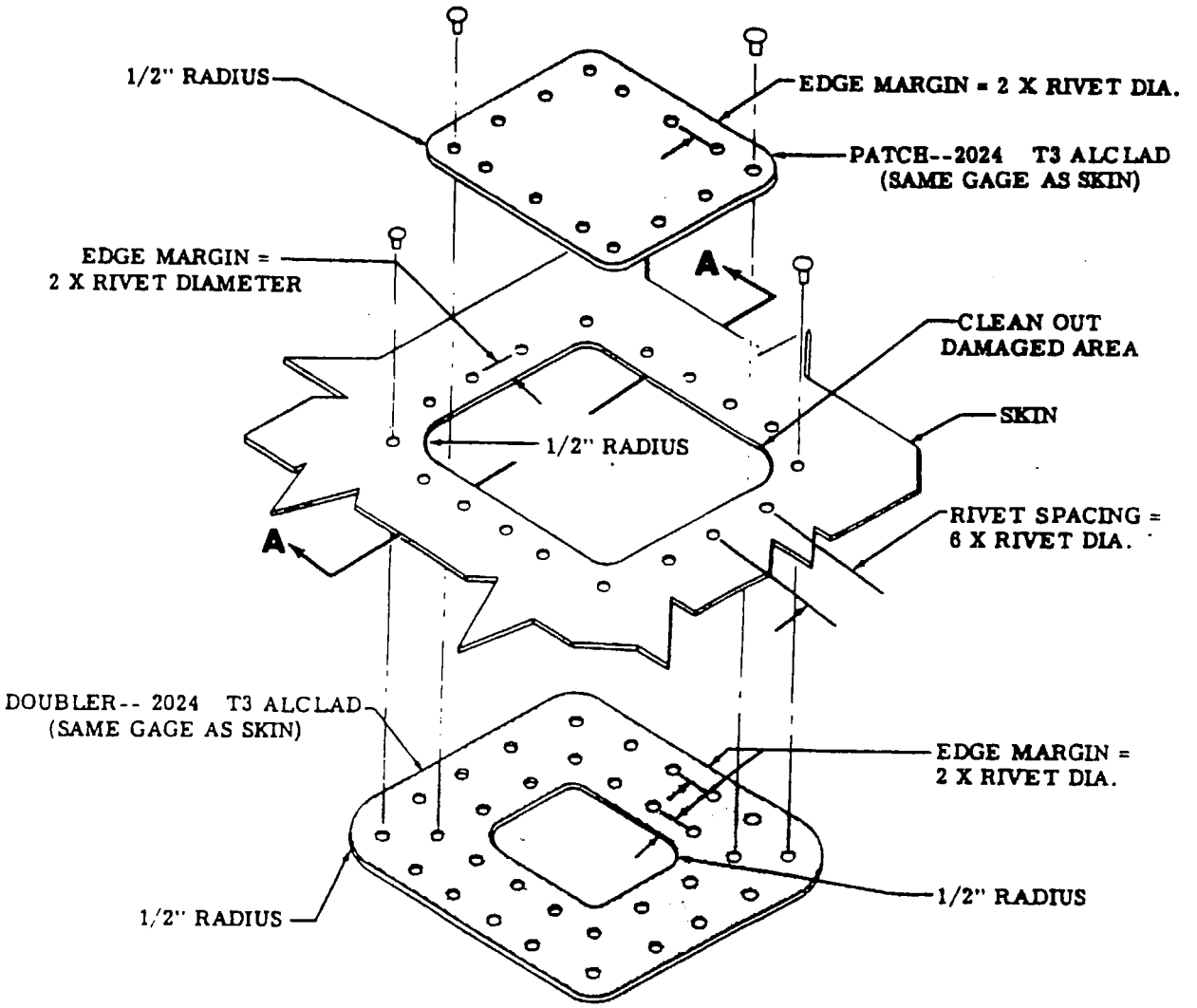
(e) Lightly sand exposed plies and clean surfaces using methyl-ethyl-ketone, Specification TT-M-261, and allow to dry thoroughly preparatory to completing repair buildup.



SECTION THRU ASSEMBLED PATCH

A–A

RIVET TABLE	
SKIN GAGE	RIVET DIA.
.020	3/32
.025	1/8
.032	1/8
.040	1/8
.051	5/32



ORIGINAL PARTS

REPAIR PARTS

REPAIR PARTS IN CROSS SECTION

Figure 16-16. Typical Insert Patch

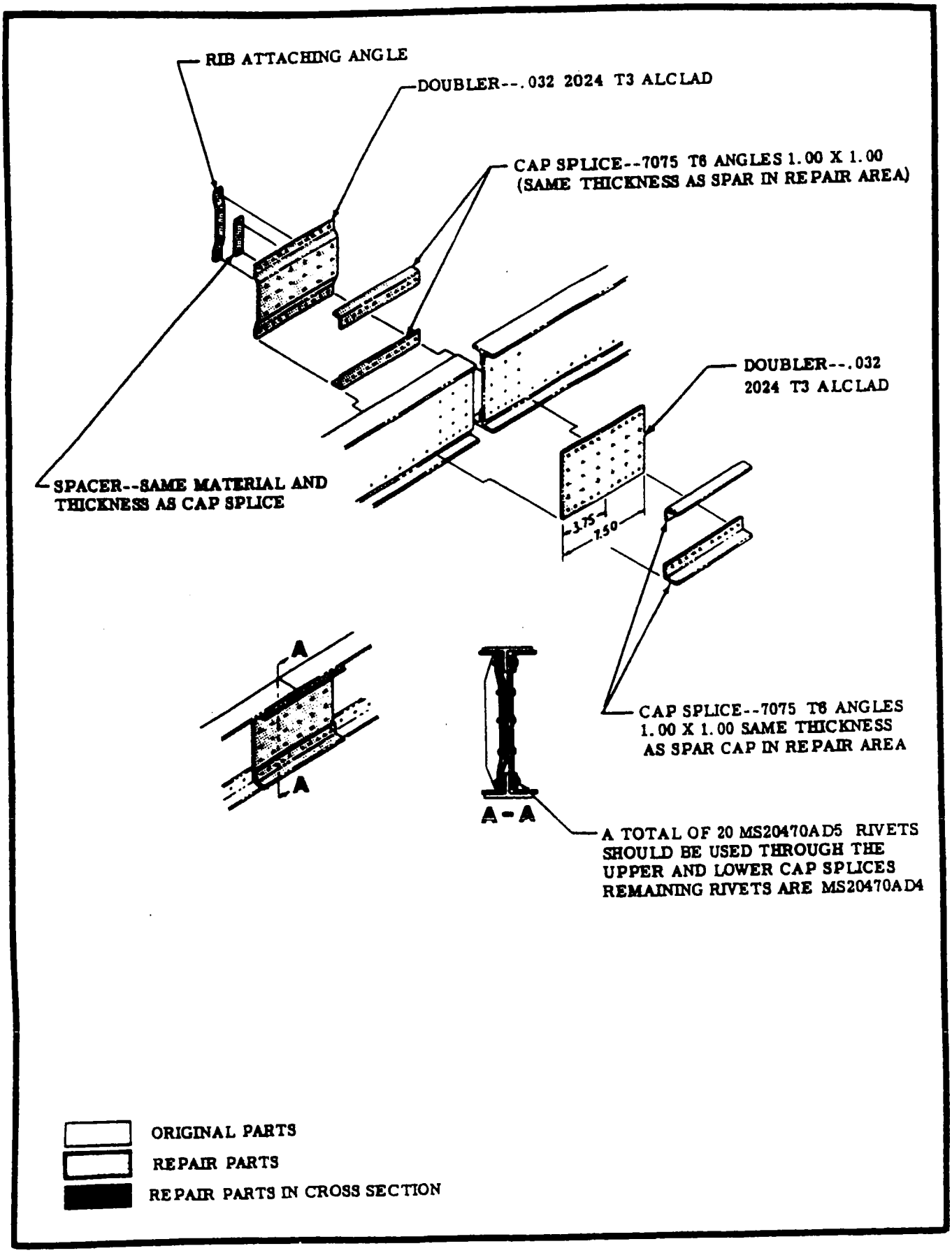


Figure 16-29. Typical Front Spar Repair (Station 168.75 and Outboard)

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