NAVIGATION - GENERAL

1. General

A. **Scope** - This chapter describes the navigation systems that provide navigational information. Flight instruments are operated by ram air, barometric air pressure differences, electric current variations, aircraft electrical power, or references to the earth's magnetic field.

B. **Descriptions**

(1) **Description of Pitot-Static System** - The pitot-static system has two components: ram air from the pitot tube and static air from the static air vent. Tubing in the pitot system is blue in color and tubing in the static system is white in color. Refer to Figure 1, Figure 2 or Figure 3.

(a) The pitot tube is located under the right wing, and the static port is located on the right side of the fuselage, halfway between the copilot's cabin door and the horizontal stabilizer.

(b) The pitot system provides ram air pressure. The amount of ram compression depends on air density and the rate of travel through the air. The ram air, in conjunction with static air operates the airspeed indicator. The static system also provides static (uncompressed) air for the altimeter, vertical speed indicator, autopilot altitude sensor, and a blind encoder that is integrated with the airplane's transponder.

(c) The pitot-static system is extremely sensitive to changes in pressure, therefore, it must be kept free of moisture and obstruction. If moisture, leaks, or obstructions exist within the system, the three diaphragm gauges will give inaccurate readings.

(d) The pitot-static system includes drains located under the belly of the fuselage (center of the wing stub) and another located inside the fuselage near the static air vent.

(2) **Description of Stall Warning System** - The stall warning system consists of an audible buzzer mounted to the speaker in the forward section of the overhead console The system is activated by a vane-type switch located on the leading edge of the left wing. Under normal flight conditions, the airflow over the wing keeps the vane in a neutral position. The vane is connected to an electrical switch that is open under normal flight operations. When the airplane approaches its critical angle of attack, the airflow moves the vane up and closes the switch. The switch is set to activate approximately 5 to 10 knots above the aircraft stall speed in all normal flight configurations.

WARNING: The stall warning system is a critical component of the Cessna 350 rudder limiter system and must be accurately set and calibrated by flight test.

(3) **Description of Systems Using Gyros** - The Cessna 350 uses three gyro systems: attitude gyros, directional gyro, and turn coordinator (roll rate gyro). The directional gyro, attitude gyros, turn coordinator gyro, and the Allied Signal KI 525 horizontal situation indicator (HSI) are electrically driven.

(4) **Description of Marker Beacon/Instrument Landing System**

(a) **Marker Beacon Receiver** - The marker beacon receiver is located on the far left, top side of the audio panel. There are three lights: blue (outer marker), amber (middle marker), and white (inner/airway marker), which give a visual and aural indication when the respective marker is crossed in flight. The MKR button in the audio panel must be depressed for the audio portion to function. There is also a marker beacon repeater indicator located in the flight instrument panel above the artificial horizon. The repeater indicator must be functional for IFR operations using the marker beacon receiver.

(b) **High and Low Sense Switches** - The high and low sense switches control the sensitivity of the marker beacon receiver. In the High position, the outer marker is received about a mile from the receiver. In the Low position, the airplane must be directly above the marker beacon receiver to receive the aural and visual indication. Many pilots set the marker switch to the higher sensitivity for an advance indication of approaching the outer marker and then set the sensitivity to the lower level when the marker signal is received. Doing this will silence the tone and visual indication until the airplane is closer to the marker, giving the pilot a more precise indication.

(c) Holding the three-position switch in the Test position applies voltage to all three marker lamps to indicate they are functioning. The Test position is spring-loaded so that when the toggle switch is released, it returns to the Low position. The photocell in the faceplate automatically measures ambient light conditions and dims the marker lights, the label backlighting, and the lights in the audio selector buttons.

(d) The aural portion of the marker is turned on and off by selecting and deselecting the MKR audio switch on the audio selector panel. The markers do not have an automatic audio reset function, and if the audio for the markers is deselected, it stays off until reselected.

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(e) **Remote Marker Beacon Repeater Indicator** - A remote marker beacon (MB) repeater indicator (Outer, Middle, and Inner Markers) is located above the artificial horizon. The three remote lights are connected to the SL15 or GMA 340 audio panel, which contains the receiver and controls for the remote marker beacons. MB annunciations are also displayed on the SL15 or GMA 340 audio panel and provide a backup source for station passage. The remote marker beacon repeater indicator must be functional for IFR operations.

(5) **Description of GPS, GPS/Comm, Nav Systems**

(a) **Garmin AT Apollo ACU** - The Apollo ACU is a self-contained unit that will provide all necessary functions required for switching HSI/CDI data inputs between a conventional Nav (VOR) receiver and an approach certified GPS receiver. The Apollo ACU will also denote specific modes selected by the front panel switches and various outputs from the GPS receiver. The GARMIN AT ACU is only installed when the GARMIN AT Apollo avionics suite and GPS is installed.

(b) **Garmin AT Apollo GX50 GPS** - The Apollo GX50 is an approach certified GPS receiver with a high definition moving map display on a sunlight viewable screen. The aircraft's position relative to airports, runways, VOR's, NDB's, intersections, and SUA's can be viewed on the large moving map display. Operation is through the use of Smart Keys. Airspaces and navaids can be added or deleted from the map with a single press of a button. The GX50 is TSO-C129a Class A1 approved for IFR non-precision approach operation.

(c) **Garmin GNS 430 GPS System** - The GNS 430 system is a fully integrated, panel-mounted instrument, which contains a VHF communications transceiver a VOR/ILS receiver, and a GPS navigation computer. The aircraft's position relative to airports, runways, VOR's, NDB's, intersections, and SUA's can be viewed on the unit’s display or the EX5000 moving map, if installed.

(d) **Mid-Continent MD41-146A Annunciator Control Unit** - The Mid- Continent ACU is a self-contained unit that will provide all necessary functions required for switching HSI/CDI data inputs between a conventional Nav (VOR) receiver and an approach certified GPS receiver. The ACU will also denote specific modes selected by the front panel switches and various outputs from the GPS receiver. The Mid-Continent ACU is only installed when the Garmin avionics suite and GPS is installed.

(e) **Garmin AT Apollo SL30 Nav/Comm** - The SL30 provides navigation and communications with a 760 channel VHF communications transceiver with 200 channel VOR, Glideslope, and Localizer receivers. Besides traditional Nav/Comm features, the SL30 also incorporates functions such as automatic decoding of the Morse code station identifier for VOR/LOC/ILS, most-used frequency storage in unit memory, and a built-in course deviation indicator. The SL30 has the ability to monitor the standby comm frequency. The SL30 VHF receiver operates from 108 MHz to 117.95 MHz, decoding both the VHF Omni Range and Localizer navigation signals. The built-in glideslope receiver will automatically tune the corresponding glideslope paired frequencies (328 MHz to 335 MHz) when the localizer is tuned.

(f) **Garmin AT Apollo SL15 Audio Panel** - The Apollo SL15 Audio Selector Panel/Intercom System (ASPIS) is frequently referred to as the audio panel. The primary purpose of the panel is to control communication and navigation selections, intercom functions, and the marker beacons. In addition, the unit has provisions for two stereo entertainment inputs. Refer to Chapter 23.

(g) **Garmin GMA 340 Audio Panel** - The Garmin GMA 340 audio panel is the primary control of communications and navigation selections, intercom functions, marker beacons, and entertainment units. Refer to Chapter 23 for more details.

(h) **Garmin AT Apollo SL70 Transponder** - The Apollo SL70 Mode A/C transponder is designed for both VFR and IFR operations. Features include a squawk code display and selection; transponder mode selection for standby, on, and altitude; built-in altitude hold; and ident button and single-button VFR squawk.

(i) **Garmin GTX 327 Transponder** - The Garmin GTX 327 Mode A/C transponder is designed for both VFR and IFR operations. Features include a squawk code display and selection; transponder mode selection for standby, on, and altitude; flight time, countdown, and count up timers; pressure altitude display; and ident button and single-button VFR squawk.

(j) **Garmin GTX 330 Transponder** - The Garmin GTX 330 Mode A/C and S transponder is designed for both VFR and IFR operations. Features include a squawk code display and selection; transponder mode selection for standby, on, and altitude; flight time, countdown, and count up timers; pressure altitude display, altitude monitor; OAT/DALT display, ident button and single-button VFR squawk, verbal alerts, and datalink traffic capability.

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(k) **Bendix-King KCS 55A Compass System** - The KCS 55A system consists of four basic components: the panel mounted KI 525A Horizontal Situation Indicator (HSI), a magnetic slaving transmitter, a directional gyro, and the panel mounted KA 51B Slaving Control and Compensator Unit.

(l) **Bendix-King KI 525A Horizontal Situation Indicator (HSI)** - The Bendix/King KI 525A pictorial navigation indicator displays constantly slaved gyro magnetic heading information along with VOR/LOC course deviation and glideslope deviation indications. It is part of the King KCS 55A compass system, which also includes a KA 51B slaving control and compensator unit, a KMT 112 magnetic slaving transmitter, and a KG 102 directional gyro.

(m) **Attitude Indicator** - The attitude indicator is an electrically-driven basic attitude display.

(6) **Description of Avidyne Entegra Primary Flight Display (PFD)** - The Avidyne Entegra is a panel mounted 10.4-inch diagonal Primary Flight Display (refer to Figure 4) that provides the display capabilities found in Table 1.

Table 1. Avidyne Entegra Primary Flight Display Capabilities

Attitude	Airspeed
Altitude	FMS-Supplied Flight Plan
Horizontal Situation Indication	Glideslope Indication
Outside Air Temperature	Navigation Source Slection
True Airspeed	Slip Indication
VOR/LOC Indication	Vertical Speed
Wind Direction/Velocity	

The PFD is capable of receiving data from and transmitting data to compatible GPS Nav/Comm systems and autopilots.

The PFD operation is controlled through 8 bezel keys, 2 rotary knobs, and a rocker switch. Both the right and left knob's function are specific to the particular page that is currently selected, and are appropriately labeled on the LCD above each knob. The rocker switch located in the top right corner of the bezel controls the display brightness, which is adjusted manually by pushing the top portion of the switch to increase intensity and pushing the bottom portion of the switch to decrease intensity. The functionality of the 8-bezel keys is specific to the currently displayed page, and each key is labeled accordingly on the LCD adjacent to the key.

(7) **Description of Avidyne Entegra Multi-Function Display (MFD)** - The Avidyne Entegra MFD is a panel mounted 10.4-inch diagonal situational awareness display as shown in Figure 5.

The MFD provides the following display capabilities:

1. Moving Map
2. Traffic (optional)
3. Lightning
4. FMS-Supplied Flight Plan
5. Obstacles
6. Datalink Weather (optional)
7. Engine Instruments (optional)
8. Outside Air Temperature

The MFD operation is controlled through 10 bezel keys, 2 rotary knobs, and a rocker switch. The left knob selects the desired page, which is identified on the LCD above the knob. The particular page that is currently selected defines the right knob's function. The knob is appropriately labeled on the LCD above the knob. The rocker switch located in the top right corner of the bezel controls the display brightness, which is adjusted manually by pushing the top portion of the switch to increase intensity and pushing the bottom portion of the switch to decrease intensity. A pop-up window that presents the percentage of full intensity, along with a bar graph representation, indicates the intensity setting of the display. The functionality of the 10 bezel keys is defined by the page that is currently displayed, and are labeled accordingly on the LCD adjacent to the key.

(8) **Description of Avidyne Magnetometer/OAT Sensor Assembly** - The Magnetometer/OAT Sensor Assembly provides magnetic heading and outside air temperature information to the PFD for display and computations. Refer to Figure 6.

(9) **Description of Avidyne Engine Instrument Sensor Interface Unit (SIU)**

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(a) **The SIU interface** - The SIU interfaces with the following engine-mounted sensors: exhaust gas temperature (EGT), cylinder head temperature (CHT), fuel flow, manifold pressure, tachometer, and oil pressure. The SIU interfaces with the following airframe-mounted sensors: battery current and OAT.

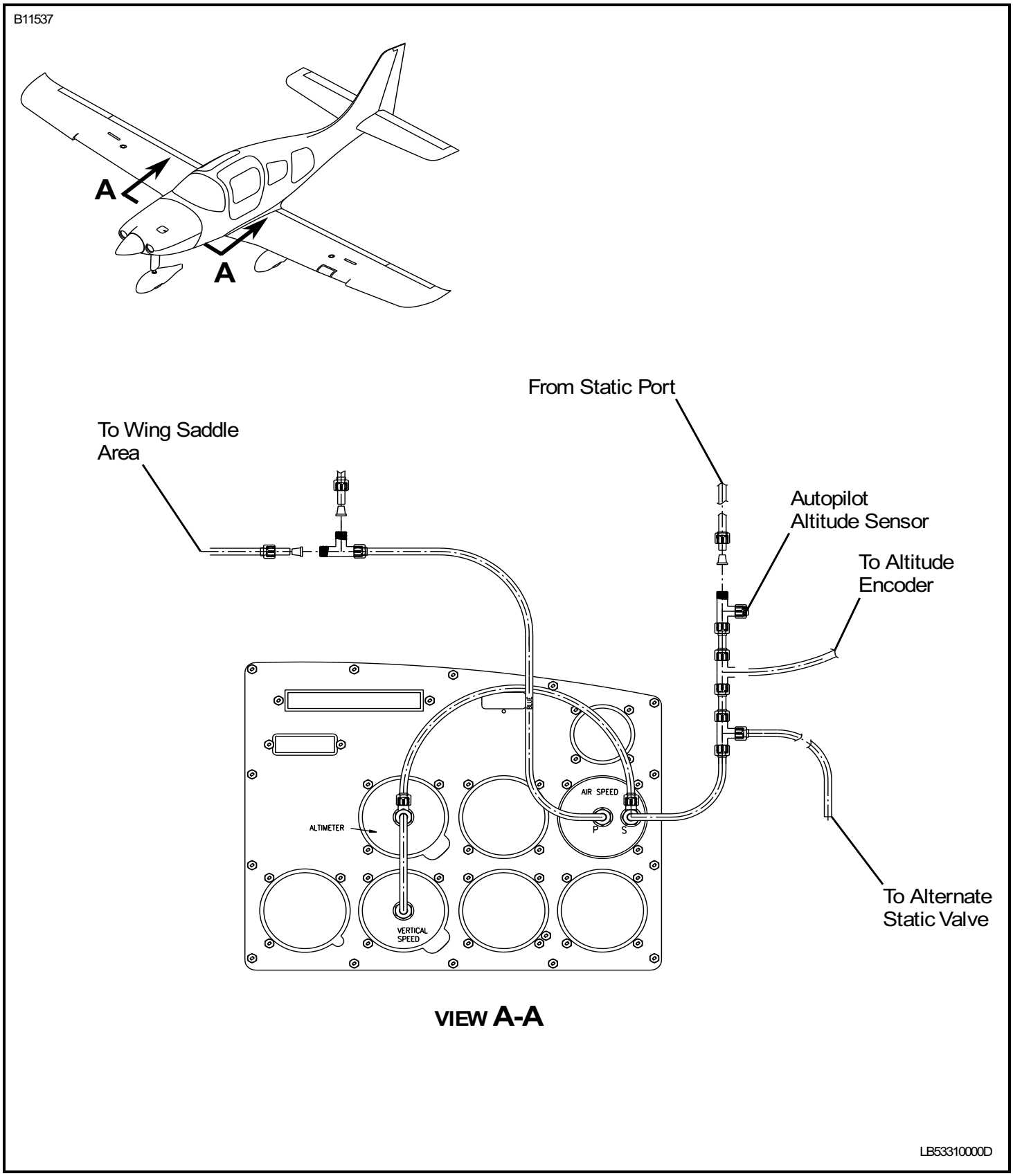
(b) **Wiring Harness** - A wiring harness collects signals from six cylinder head temperature (CHT) sensors and six exhaust gas temperature (EGT) sensors and routes them through a firewall feed-through. The feed-through is sealed with a heat resistant sealant, such as RTV736. A wire harness aft of the firewall routes the signals from the firewall bulkhead and from the existing engine indicators, mounted in the instruments panel, to the SIU.

(c) **The SIU data stream** - The SIU data stream transmits on an RS-232 data bus, and includes diagnostic information. Connect the SIU to an Avidyne MFD. The RS-232 serial data are transmitted at 9600-baud, no parity bit, 8 data bits, 1 stop bit, at a rate of 5 Hz. One header message is transmitted after each group of 15 data records.

CAUTION: Before any maintenance, removal, or installation of equipment is performed, make sure both the master and avionics switches are in the off position. Failure to do so could result in damage to equipment.

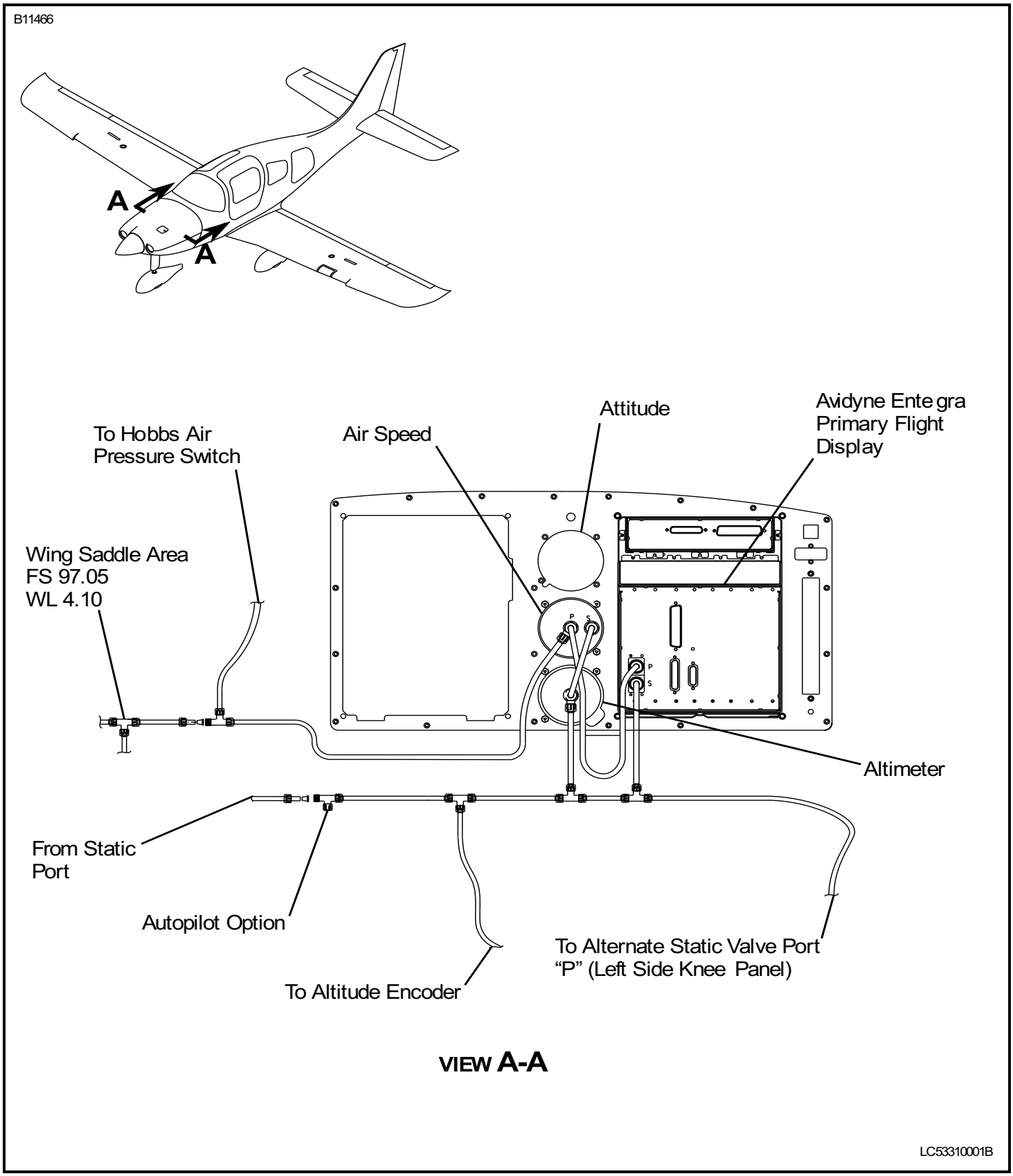
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Figure 1 : Sheet 1 : Pitot-Static System Diagram



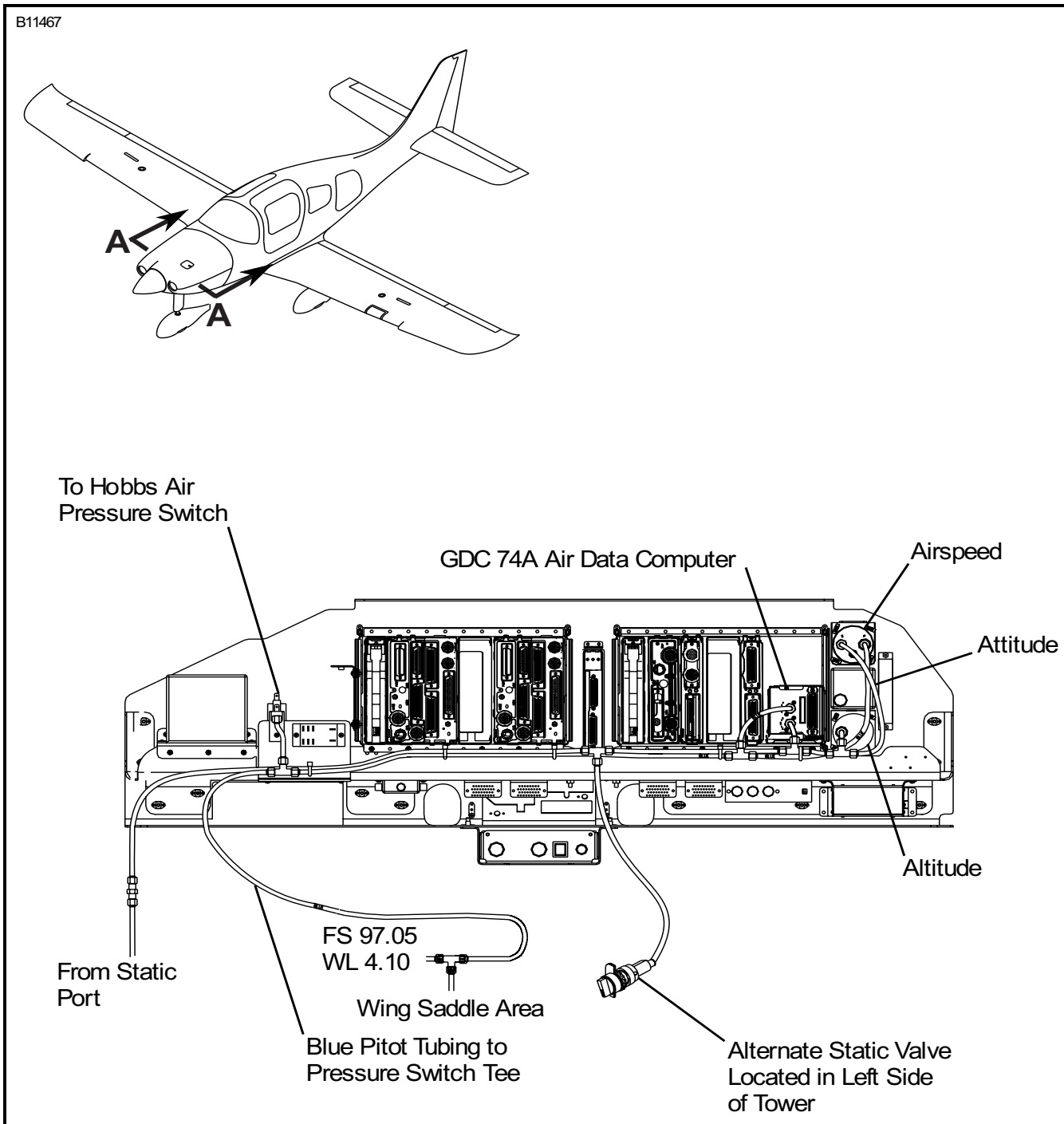
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Figure 2 : Sheet 1 : Pitot-Static System Diagram with Avidyne Entegra Option



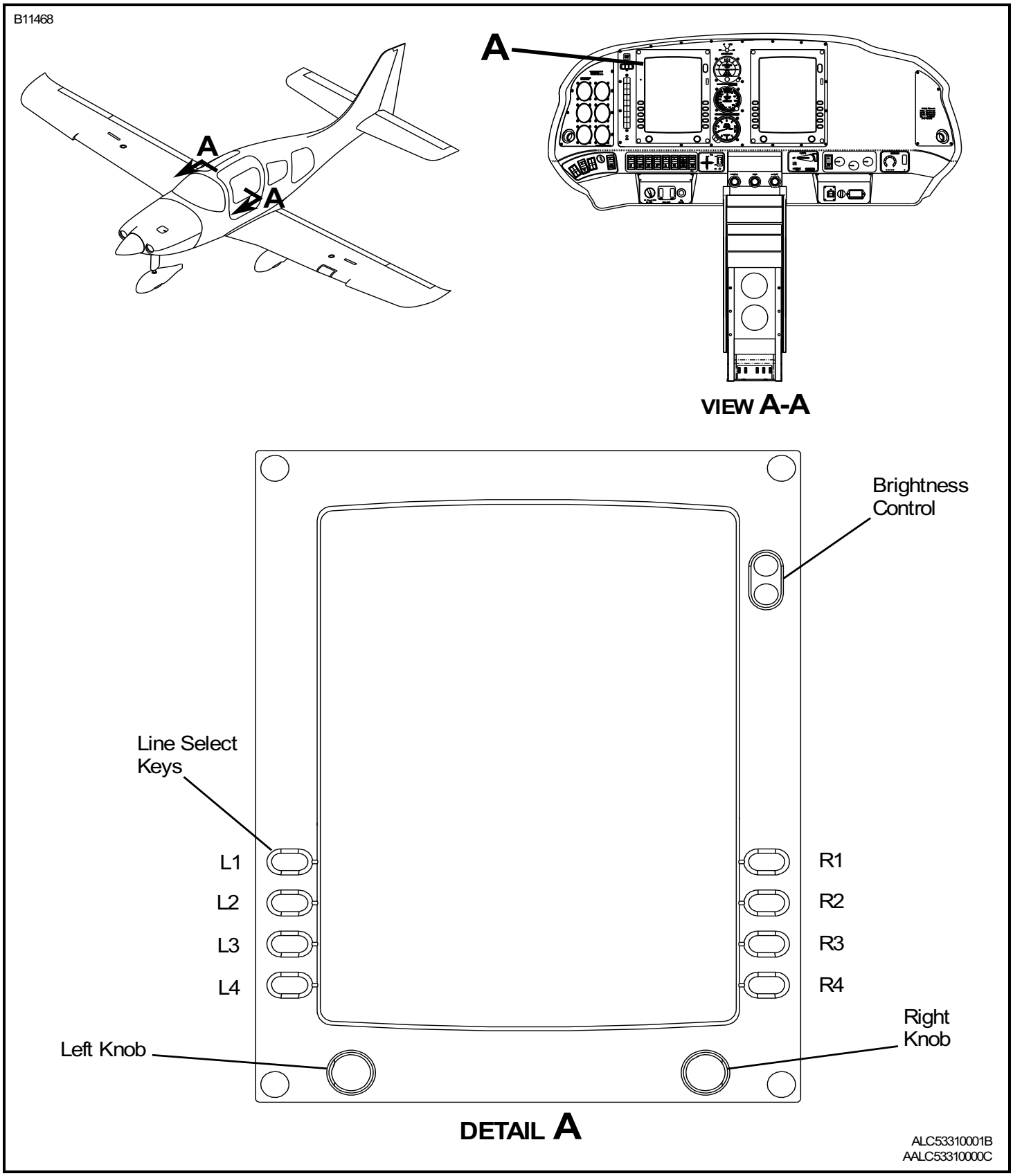
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Figure 3 : Sheet 1 : Pitot-Static System Diagram with Garmin G1000 Option



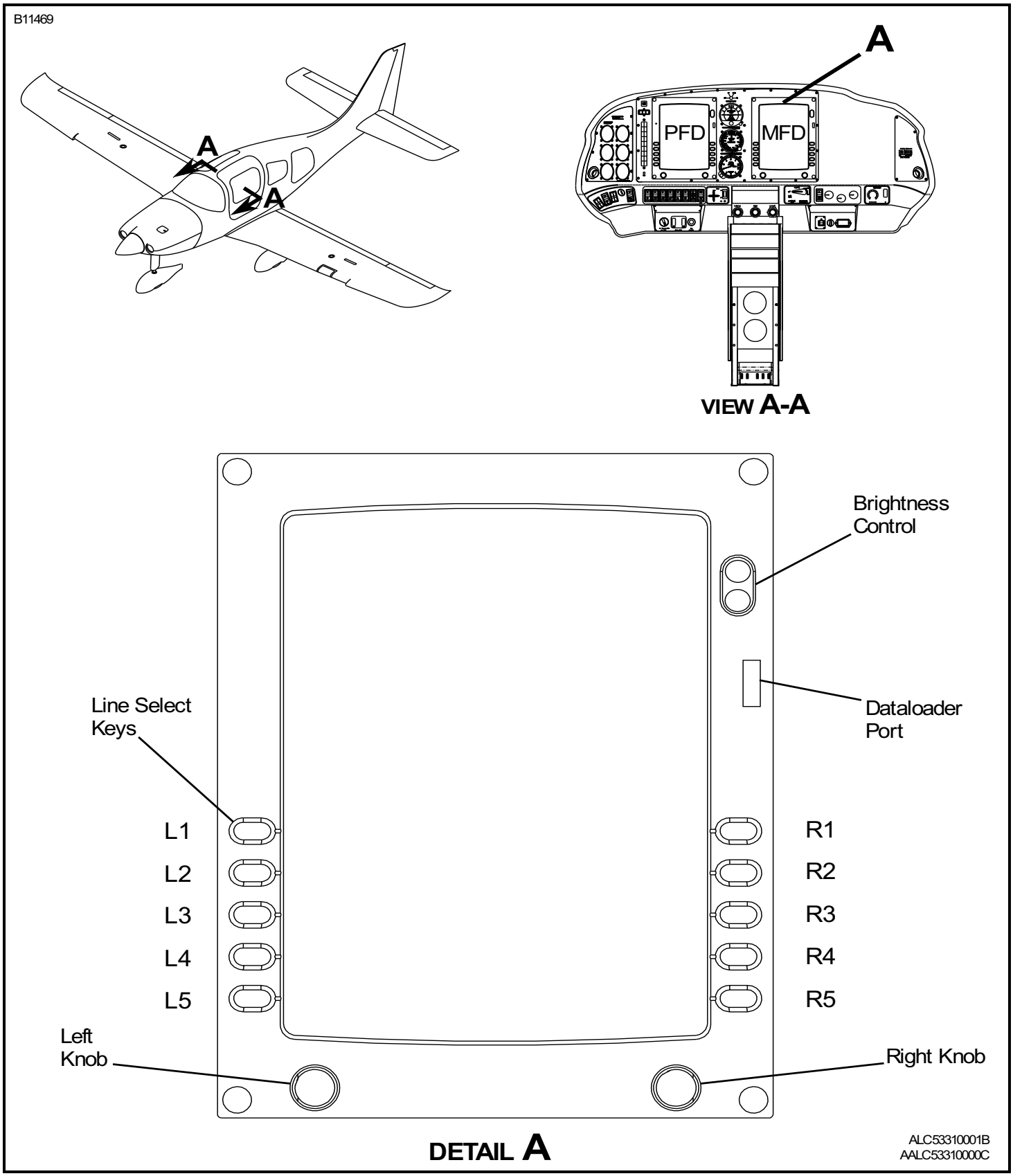
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Figure 4 : Sheet 1 : Avidyne Entegra Primary Flight Display (PFD)



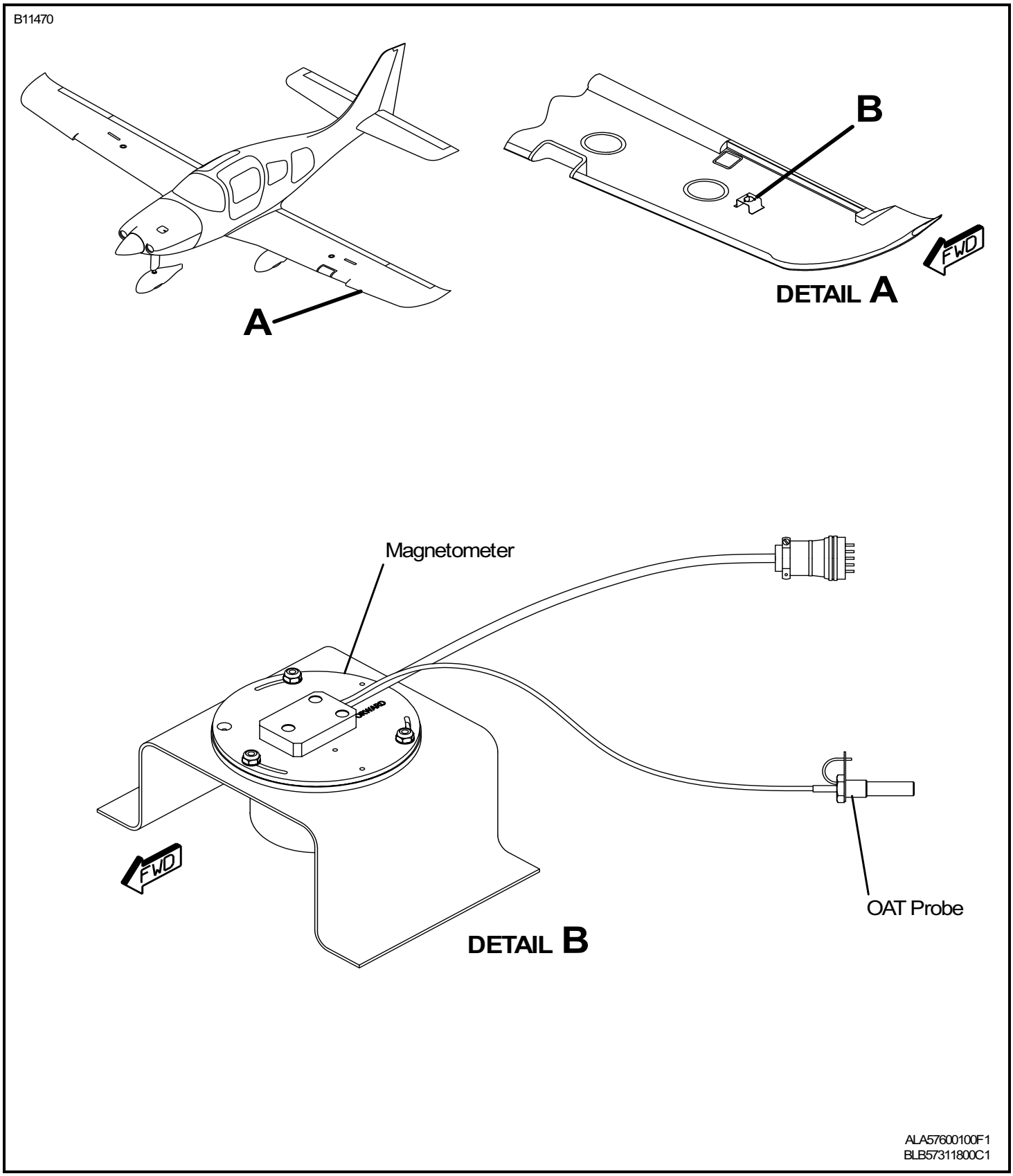
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Figure 5 : Sheet 1 : Avidyne Entegra Multi-Function Display (MFD)



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Figure 6 : Sheet 1 : Avidyne Magnetometer/OAT Sensor Assembly



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PITOT-STATIC SYSTEM - MAINTENANCE PRACTICES

1. Flight Environment Data/Pitot-Static System

A. The pitot heat may be protected by a thermal device between P16 and the pitot heater limiting the temperature to approximately 175° F. When the pitot heat is turned on, this temperature is achieved in 3 to 4 minutes. Once the pitot has reached this temperature and the pitot heat is interrupted, the pitot can take up to 10 minutes to cool before the thermal device resets.

(1) **Pitot System - Removing Pitot Assembly**

CAUTION: To avoid injury, turn off the master switches, and pull the circuit breaker for this unit prior to touching it.

(a) The pitot assembly is located under the right wing.

(b) Remove the access panel next to the pitot assembly.

(c) Remove the screws securing the pitot assembly to the wing. Refer to Figure 201.

(d) Disconnect the ram air tube connector at the B-nut.

(e) Disconnect the electrical plug.

(f) Remove the pitot assembly from the airplane.

(2) **Pitot System - Installing Pitot Assembly**

(a) Secure the pitot assembly to the wing structure with screws. Dry torque 10 to 12 in.-lbs. Refer to Figure 201.

(b) Connect the electrical wire plug for the pitot heater.

(c) Connect the ram air tube connector (B-nut) to the female connector. Tighten the nut finger-tight, and turn a quarter-turn with a wrench.

(d) Install the access panel next to the pitot assembly.

(3) **Pitot System - Pitot Heat Testing**

(a) Turn the master switch (battery side) on.

(b) Turn the pitot heat rocker switch on.

(c) Touch the pitot assembly to determine if the temperature is increasing.

CAUTION: To avoid injury, do not grip the pitot assembly if the heat has been turned on for more than one minute.

(d) If the assembly does not become warm to the touch within 60 seconds, the heating component is malfunctioning and should be replaced.

(4) **Pitot System - Clean Out and Drain**

(a) The pitot system moisture collection tube is located at the low point of the pitot system in the wing (underside of the fuselage). Refer to Figure 202.

(b) Remove the fuel module access panel to gain access to the moisture collection tube.

(c) Remove the plastic nut from the union fitting. Refer to Figure 203.

(d) If moisture is present, remove the entire moisture collection tube for cleaning or replacement.

(e) Reassemble the moisture collection tube (tighten the nut finger-tight and turn a quarter-turn with a wrench).

(f) Reinstall the fuel module access panel.

(5) **Static Pressure System Inspection and Leakage Test** - The following procedure outlines inspection and testing of the static pressure system, assuming the altimeter has been tested and inspected in accordance with current Federal Aviation Regulations.

(a) Ensure the static system is free from moisture and restrictions.

(b) Ensure no alterations or deformations of the airframe surface have been made which would affect the relationship between air pressure in the static pressure system and true ambient air pressure for any flight configuration.

(c) Ensure the static pressure alternate source valve, located on the pilot-side knee bolster, is turned to NORM. Refer to Figure 204 or Figure 205.

(d) Attach a source of suction to the static system moisture collection tube using sealant tape at the connection

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points to ensure a tight seal, and tape over the normal static pressure port (located on the right side of the fuselage, halfway from the door to the tail).

(e) Slowly apply suction until the altimeter indicates a 1000 ft. increase in altitude.

WARNING: When applying or releasing suction, do not exceed the range of the vertical speed indicator or airspeed indicator.

(f) Cut off the suction source to maintain a closed system for one minute. Leakage shall not exceed 100 ft. of altitude loss as indicated on the altimeter.

(g) If the leakage rate is within tolerance, slowly release the suction source, and remove the tape from the static port. If the leakage rate exceeds the maximum allowable, tighten all connections, and repeat the leakage test. If the leakage rate still exceeds the maximum allowable, proceed to the next step, otherwise, the test is complete.

(h) Disconnect the static pressure lines from the airspeed indicator, vertical speed indicator, and altitude encoder. Use suitable fittings to connect the lines together so the altimeter is the only instrument still connected into the static pressure system.

(i) Repeat the leakage test to check whether the static pressure system or the bypassed instruments are the cause of leakage. If instruments are at fault, they must be repaired by an appropriately rated repair station or replaced. If the static pressure system is at fault, use the following procedure to locate leakage.

(j) Attach a source of positive pressure to the normal static port.

WARNING: Do not apply positive pressure with the airspeed indicator connected to the static pressure system.

CAUTION: To avoid damage to equipment, always apply pressure or suction slowly when testing.

(k) SLOWLY apply positive pressure until the altimeter indicates a 500 ft. decrease in altitude, and maintain this altimeter indication while checking for leaks. Coat the line connections and static source flange with LEAK-TEC or a solution of mild soap and water. Watch for bubbles to locate leaks.

(l) Tighten leaking connections, and repair or replace parts found to be defective.

(m) Reconnect the airspeed and vertical speed indicators into the static pressure system, and repeat the leakage test beginning by ensuring the static pressure alternate source valve is turned to NORM.

(6) **Static Pressure System - Clean Out and Drain**

(a) The static system sump/water trap is located behind the interior panel on the right side of the aircraft's baggage area. Refer to Figure 206.

(b) Remove the right-side interior curved side panel to gain access to the static sump.

(c) Remove the plastic nut from the union fitting.

(d) If moisture is present, remove the entire moisture collection tube by unscrewing the B-nut at the T-connection. When the sump assembly is no longer connected to the pitot system, slide it out of its bracket.

(e) Clean or replace the drain.

(7) **Static Pressure System - Install Moisture Collection Tube**

(a) Before reconnecting the moisture collection tube assembly to the T-connector, be sure that both are clean and free of debris.

(b) Reassemble the moisture collection tube assembly to the static pressure line (tighten the nut finger-tight and turn it a quarter-turn with a wrench).

(c) Assure the collection tube is fastened to the tie-wrap block.

(d) Reinstall the interior panel.

(e) Retest the static system as per the Static Pressure System Inspection and Leakage Test paragraph above.

(8) **Static Pressure System - Removal and Install Normal Static Port**

(a) The static port is bonded into the fuselage skin. It must be removed and replaced by qualified composite technicians.

(b) Refer to Chapter 51 for composite repairs.

(9) **Alternate Static Source Valve Removal**

(a) Disconnect the left knee bolster from the instrument panel. Refer to Chapter 25-11-00 for instructions on the

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removal of the knee bolster.

(b) Disconnect the static line from the alternate static source valve. Refer to Figure 207 or Figure 208.

(c) Insert a small, flat bladed tool into the slot on the backside of the valve to spread the valve from the operator control knob, and remove the valve.

(10) **Alternate Static Source Valve Installation**

(a) Snap the static source valve to the operator. Refer to Figure 207 or Figure 208.

(b) Operate the valve to check for freedom of operation.

(c) The back of the static source valve has two connectors labeled P and V, respectively. Reconnect the static line to the P connector of the valve and finger-tighten the nut plus a quarter-turn.

(d) Reinstall the left knee bolster.

(11) **Alternate Static Source Testing**

(a) Note the altimeter elevation setting.

(b) Turn the alternate static source valve operator knob to NORM.

(c) Attach a source of suction to the normal static port.

(d) SLOWLY apply suction until the altimeter indicates a 100 ft. increase in altitude.

CAUTION: To avoid damage to equipment, always apply pressure or suction slowly when testing.

(e) Turn the alternate static source valve operator knob to ALT. The altimeter should drop back to its original setting.

(f) If the altimeter does not drop back to its original setting, replace the alternate static source valve assembly.

2. Static Air Instruments

A. The altimeter, vertical speed indicator, blind altitude encoder (not present if Garmin G1000 installed), and airspeed indicator all utilize static air as a major component of their functionality. Altitude, vertical speed, and airspeed information are displayed on the PFD if the Avidyne or Garmin G1000 options are installed. Removal and installation of redundant altimeter and airspeed indicator instruments is as follows:

(1) **With Avidyne Entegra Primary Flight Display (PFD) –**

(a) Altimeter Vertical Speed Indicator, and Airspeed Indicator Removal

1 Remove the Avidyne Multi-Function Display. Refer to the Avidyne Entegra EX5000 Multi-Function Display (MFD) section.

2 Unfasten the instrument from the instrument panel.

3 Disconnect the wires and the static lines going into the back of the instrument.

(b) Altimeter, Vertical Speed Indicator, and Airspeed Indicator Installation

1 Slide the instrument into its slot in the instrument panel.

2 Connect the proper wires and static lines into the back of the instrument.

3 Fasten the instrument to the instrument panel.

4 Reinstall the Avidyne Multi-Function Display. Refer to the Avidyne Entegra EX5000 Multi-Function Display (MFD) section.

(2) **Without Avidyne Entegra Primary Flight Display (PFD)**

(a) Altimeter, Vertical Speed Indicator, and Airspeed Indicator Removal

1 Unfasten the main plate of the instrument panel. Refer to Figure 209.

2 Carefully slide the plate out of the instrument panel.

3 Unfasten the instrument from the plate.

4 Disconnect the wires and the static lines going into the back of the instrument.

(b) Altimeter, Vertical Speed Indicator, and Airspeed Indicator Installation

1 Slide the instrument into its slot in the main plate.

2 Fasten the instrument to the main plate.

3 Connect the proper wires and static lines into the back of the instrument.

4 Gently slide the main plate into the instrument panel.

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5 Refasten the main plate to the instrument panel. Refer to Figure 209.

(3) **With Garmin G1000 Option**

(a) Altimeter, Vertical Speed Indicator, and Airspeed Indicator Removal

1 Reach under the instrument panel and unfasten the instrument.

2 Disconnect the wires and the static lines going into the back of the instrument.

(b) Altimeter, Vertical Speed Indicator, and Airspeed Indicator Installation

1 Slide the instrument into its slot in instrument panel.

2 Fasten the instrument to the panel.

3 Connect the proper wires and static lines into the back of the instrument.

(4) **Blind Altitude Encoder - Removal**

(a) The blind altitude encoder is attached to a metal plate connected to the firewall on the passenger's side. To access the blind encoder, remove the instrument panel glare shield. The blind encoder is on the bottom left corner of the metal plate. Refer to Figure 210.

(b) Unhook the wires and static lines connected to the blind encoder.

(c) Remove the knurled bolt above the blind encoder, and remove it from its mounting.

(5) **Blind Altitude Encoder Installation**

(a) Place the blind encoder in its mounting.

(b) Finger-tighten the knurled bolt above the blind encoder.

(c) Reconnect the wires and static lines to the blind encoder.

(6) **Blind Altitude Encoder Testing**

(a) Set the system master and avionics switch on.

(b) Wait for the blind encoder to warm up (must be warmed internally to within a few degrees of 140° F (60° C). The instrument is encased within a heated thermostatically controlled manifold that keeps the temperature of the unit within a few degrees of 140°F (60°C). The encoder will not provide altitude information to the transponder until the manifold assembly reaches its operating temperature. The time required for the heat assembly to reach operating temperature is a function of cockpit ambient air temperature. Table 201 shows the approximate warm-up times for a range of different ambient temperatures.

(c) Set the barometric scale on the altimeter to 29.92 inches of Hg.

(d) View the blind encoder altitude readout on the Garmin AT Apollo GPS or the Garmin AT Apollo Transponder. The difference between the blind encoder readout and the altimeter cannot be more than 125 ft.

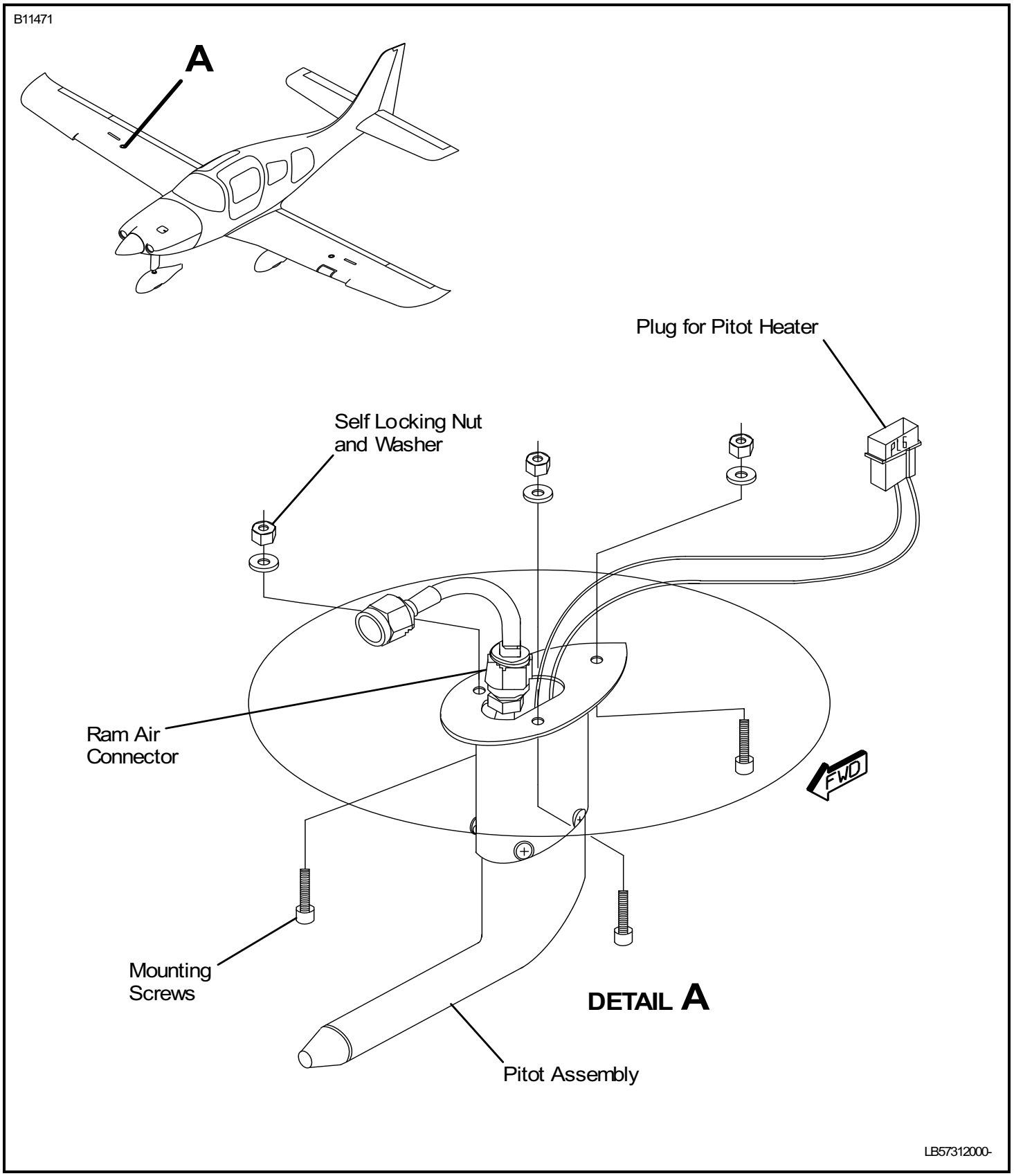
(e) If the altitude encoder readout is not within 125 ft., have a qualified avionics shop check out the system.

Table 201. Blind Altitude Encoder Heating Requirements

Cockpit Ambient Temperature	Required Operating Temperature	Warm-Up Time (minutes)
70° F (21° C)	140° F (60° C)	4 to 6
0° F (-18° C)	140° F (60° C)	Up to 10
-55° F (-48° C)	140° F (60° C)	30 to 45

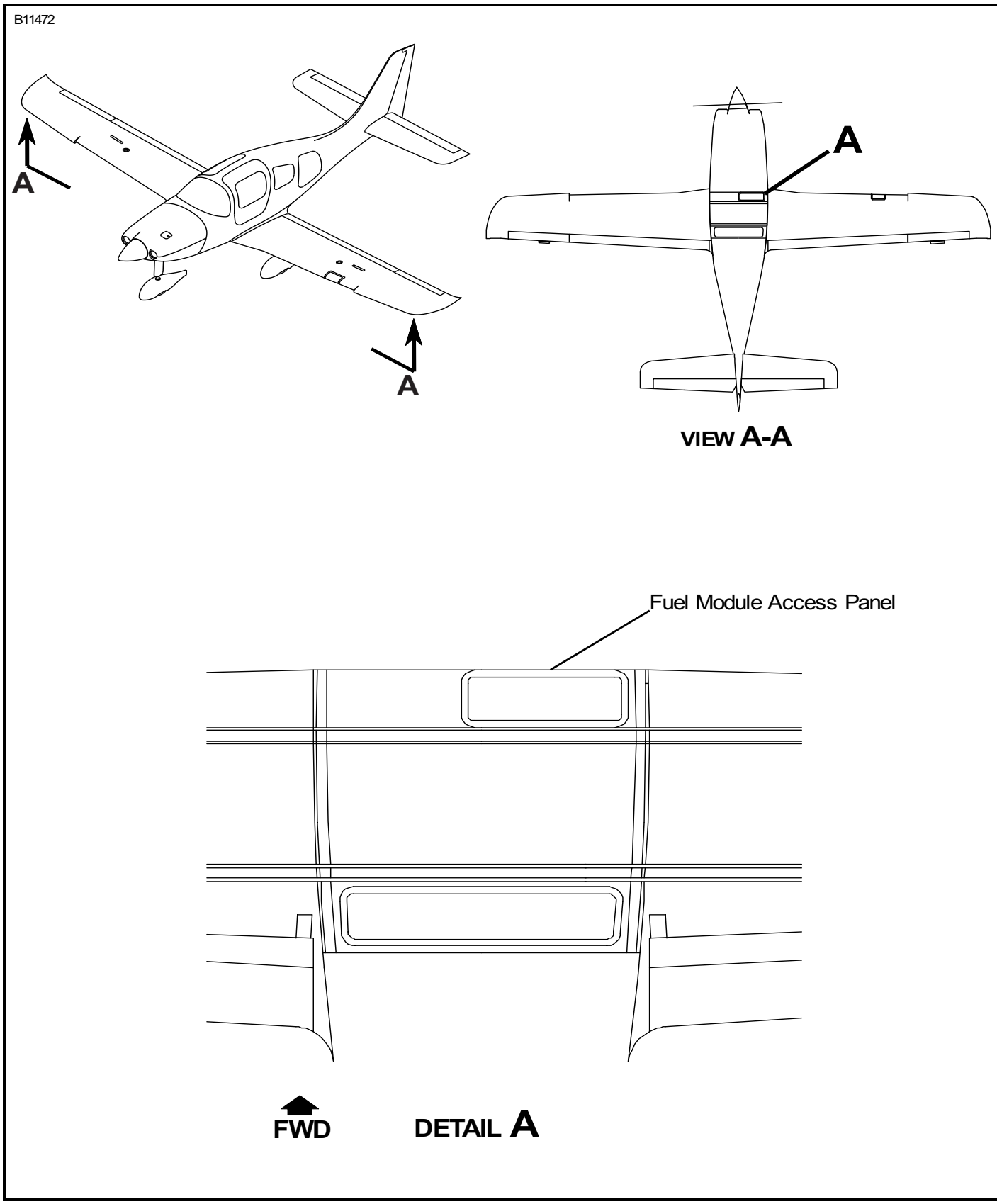
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Figure 201 : Sheet 1 : Pitot Installation



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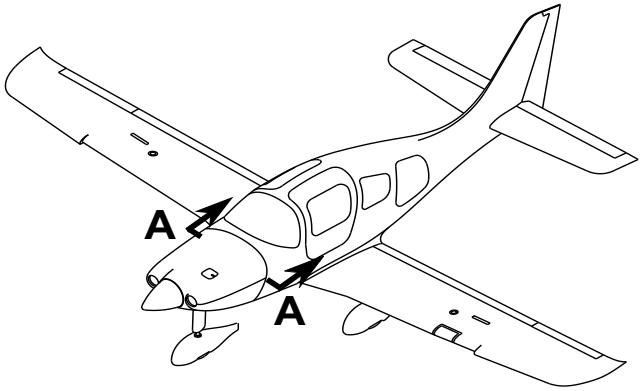
Figure 202 : Sheet 1 : Fuel Module Access Panel

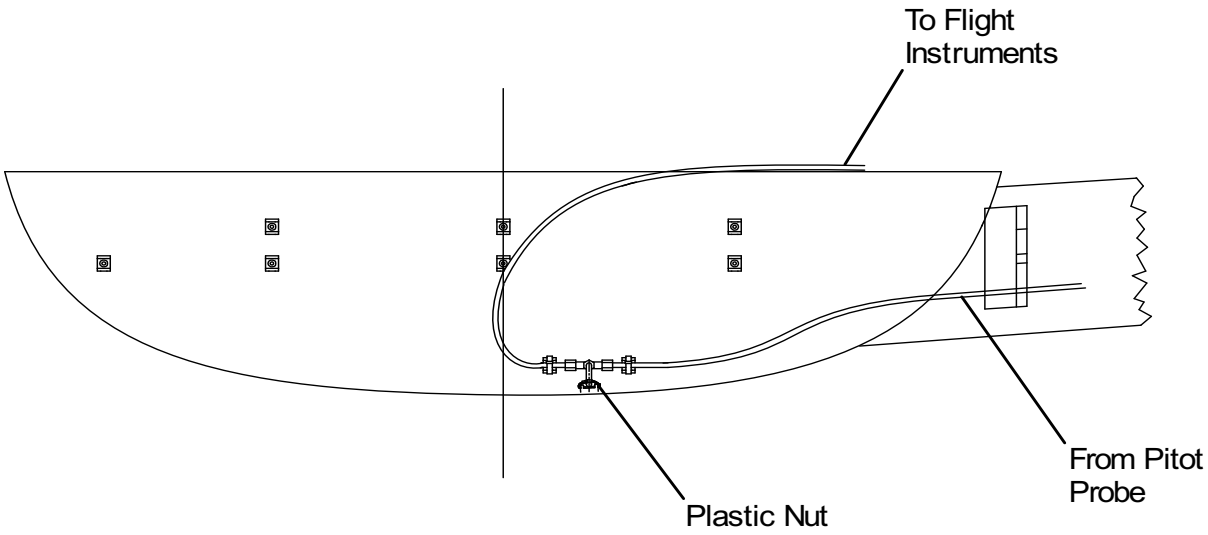


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Figure 203 : Sheet 1 : Pitot Moisture Collection Tube

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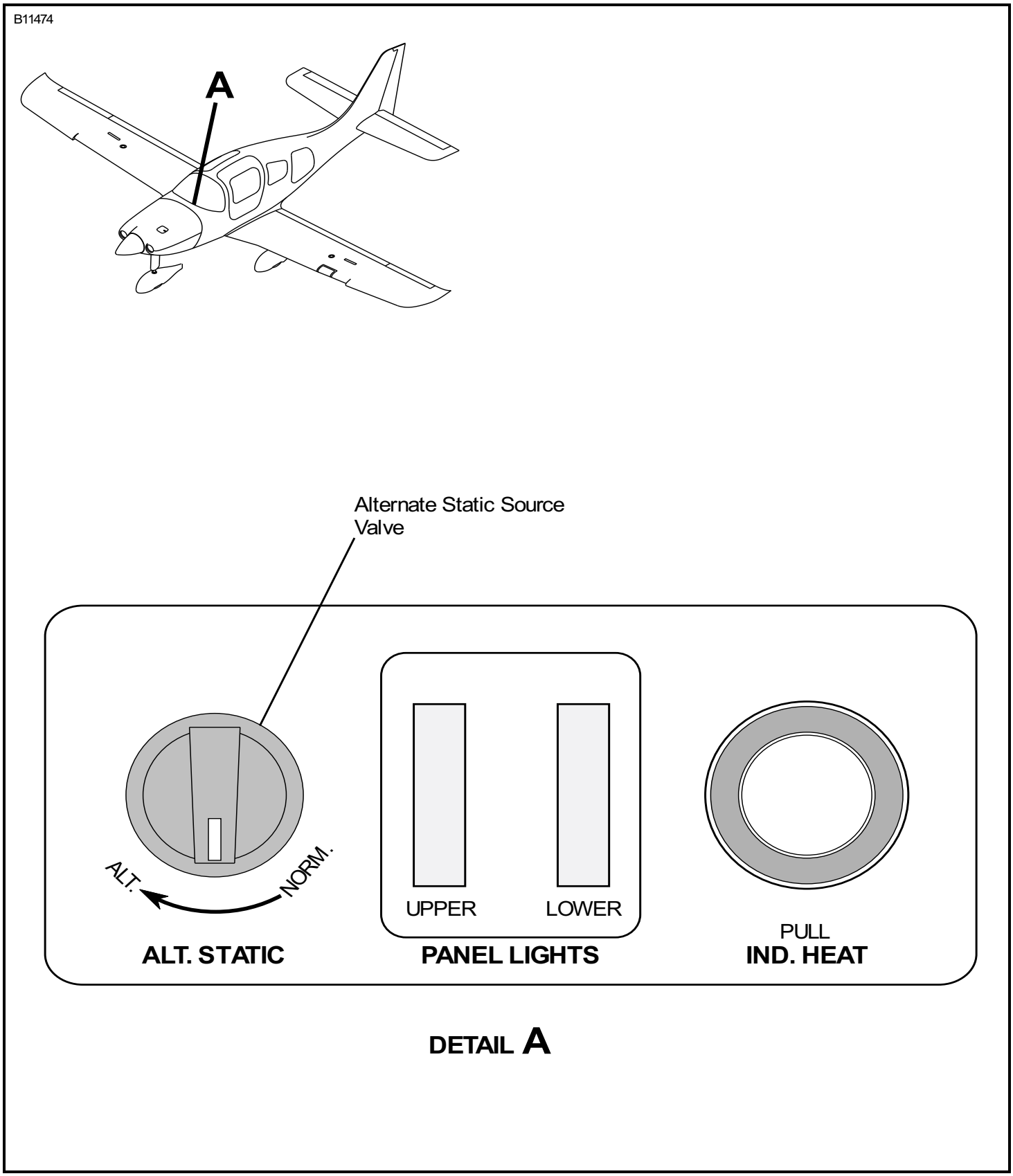


VIEW A-A

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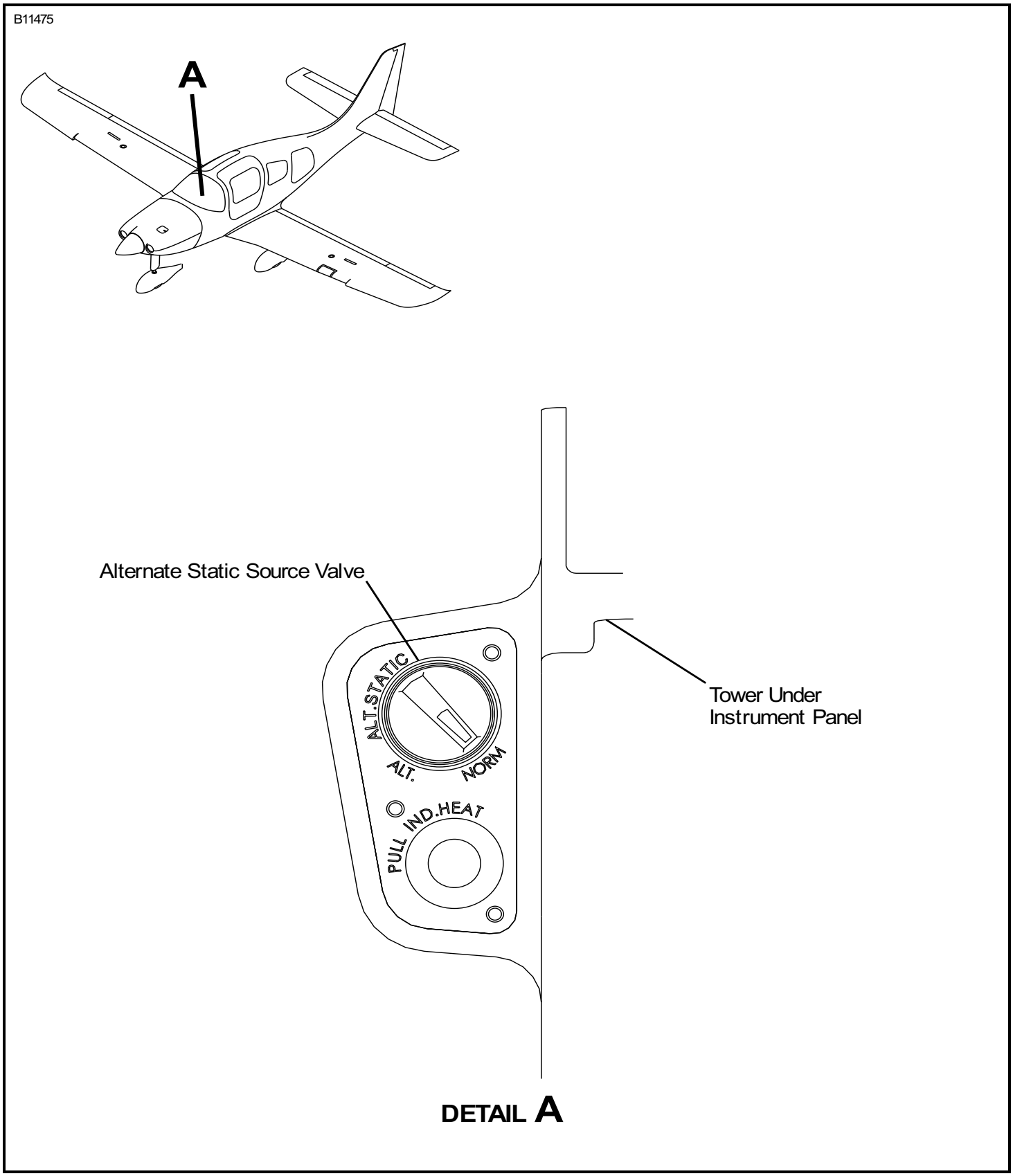
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Figure 204 : Sheet 1 : Alternate Static Source Valve



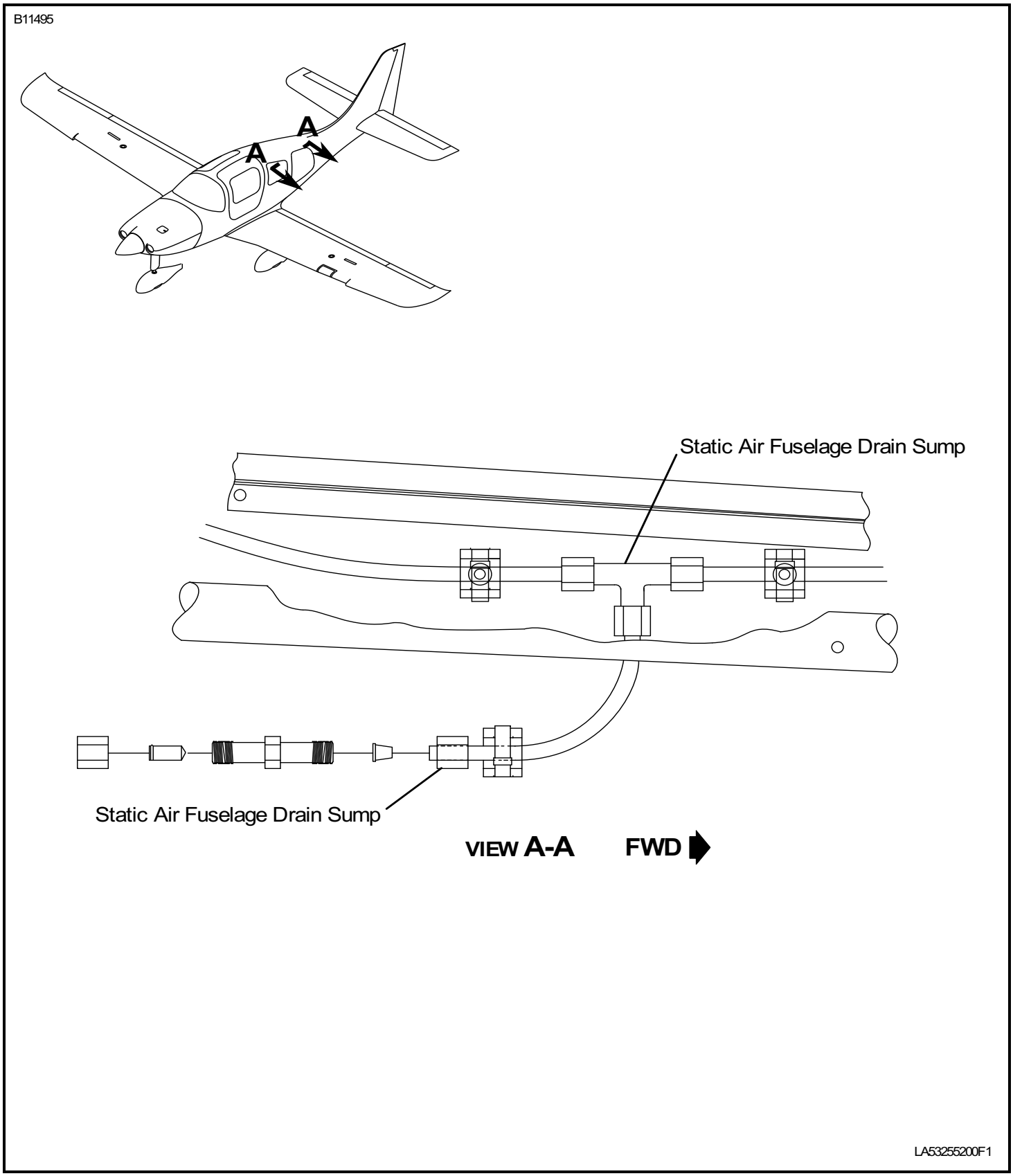
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Figure 205 : Sheet 1 : Alternate Static Source Valve (Garmin G1000 Option)



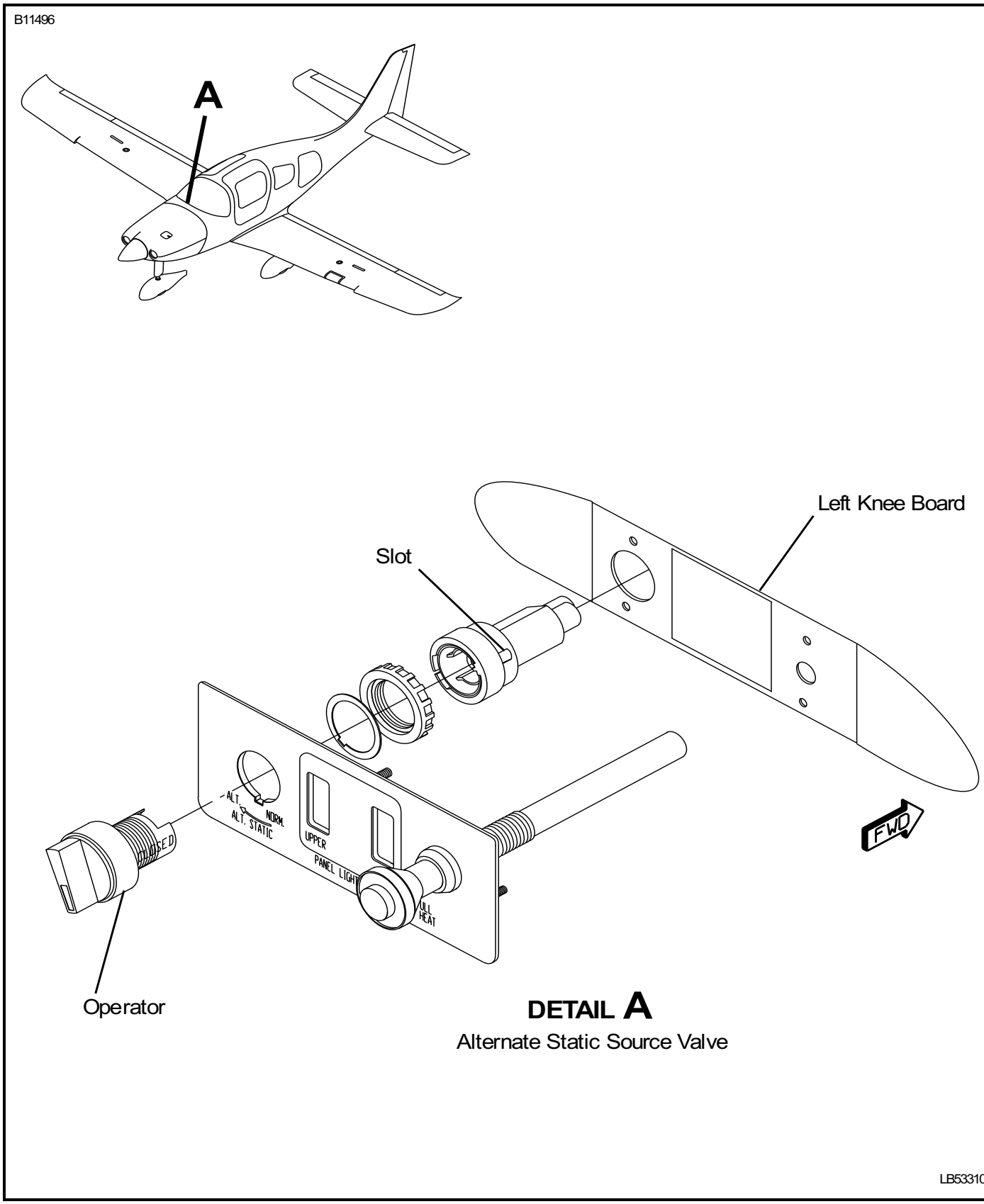
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Figure 206 : Sheet 1 : Baggage Area Static Air Sump Drain



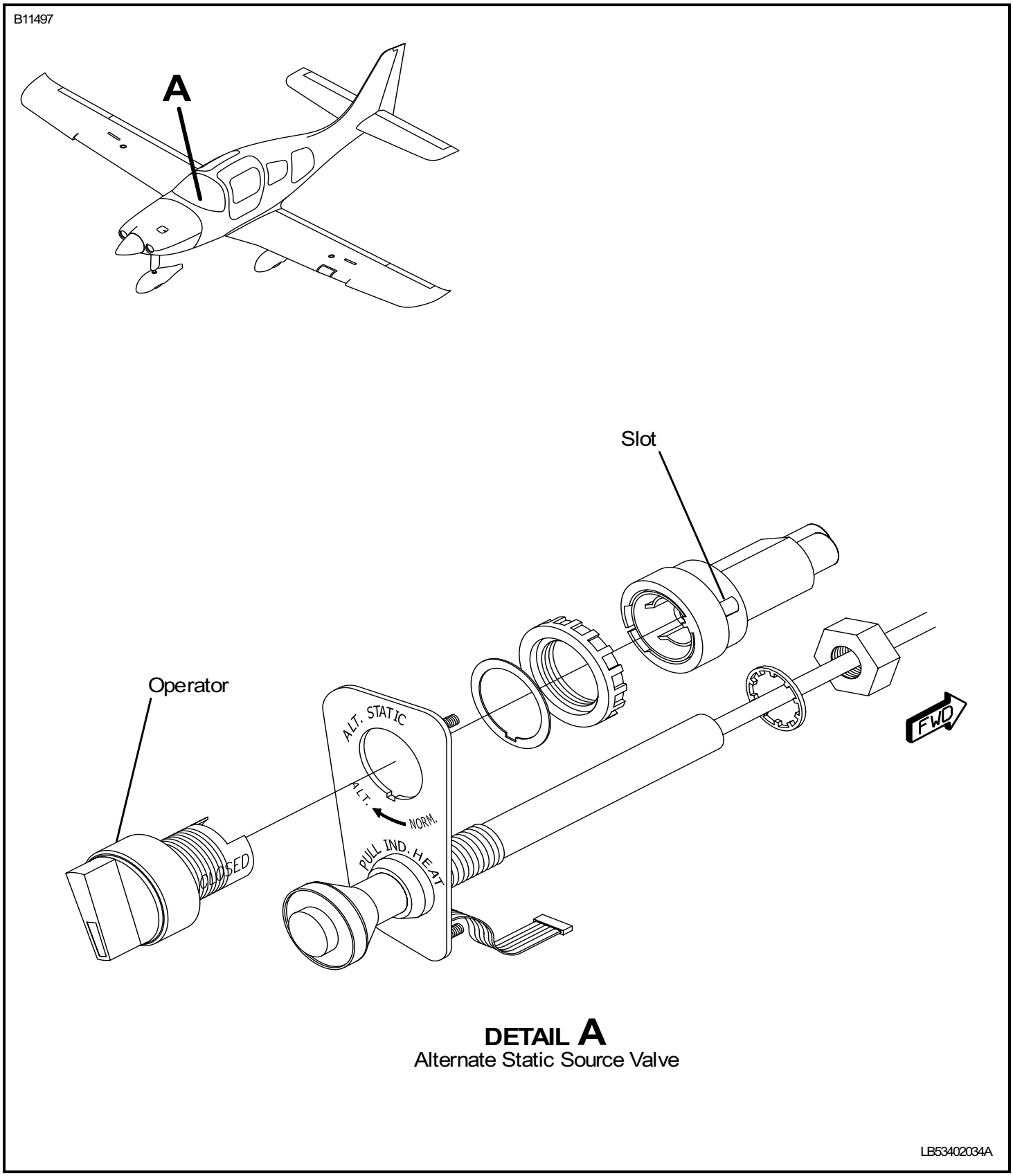
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Figure 207 : Sheet 1 : Alternate Static Source Valve



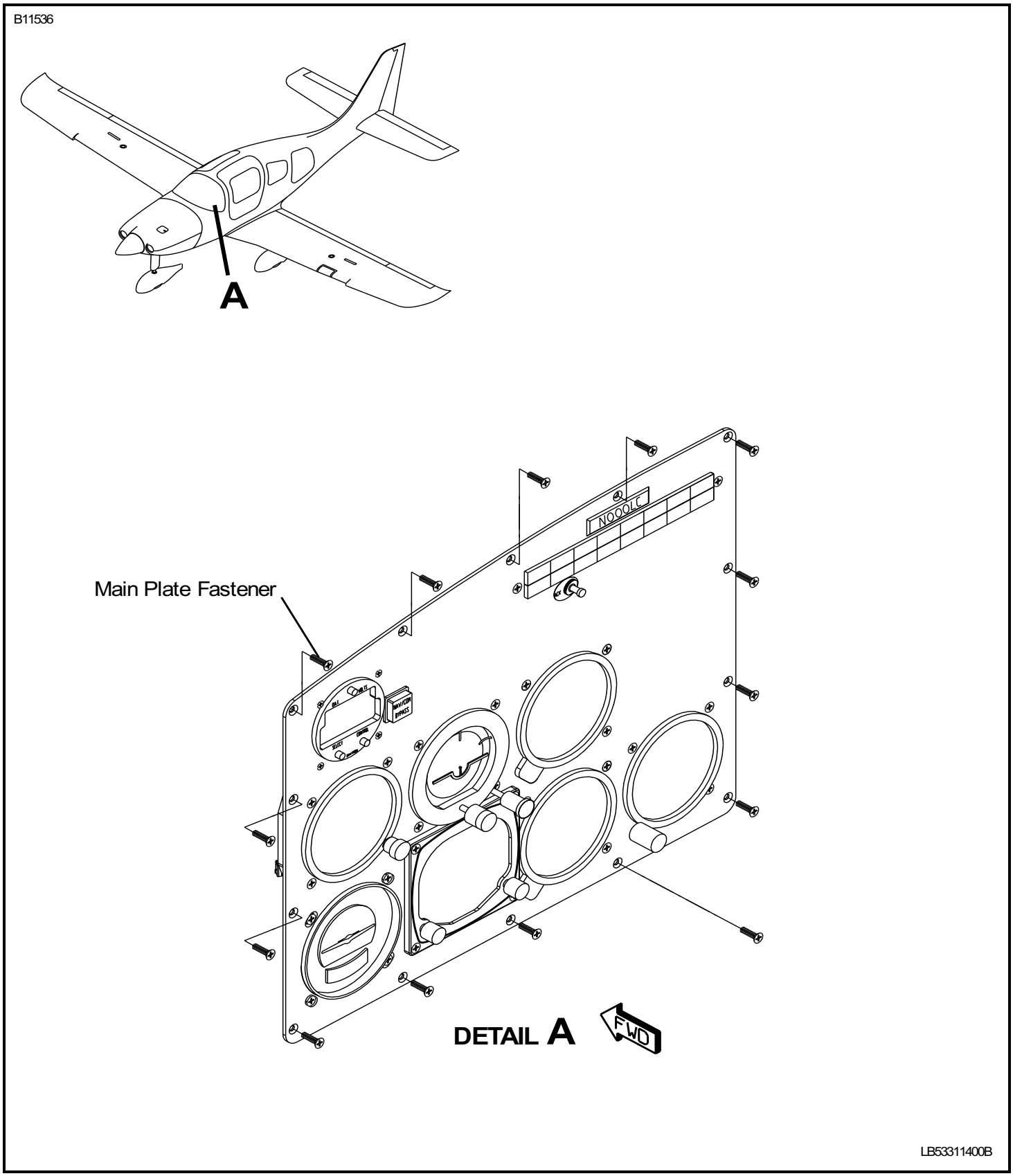
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Figure 208 : Sheet 1 : Alternate Static Source Valve (Garmin G1000 Option)



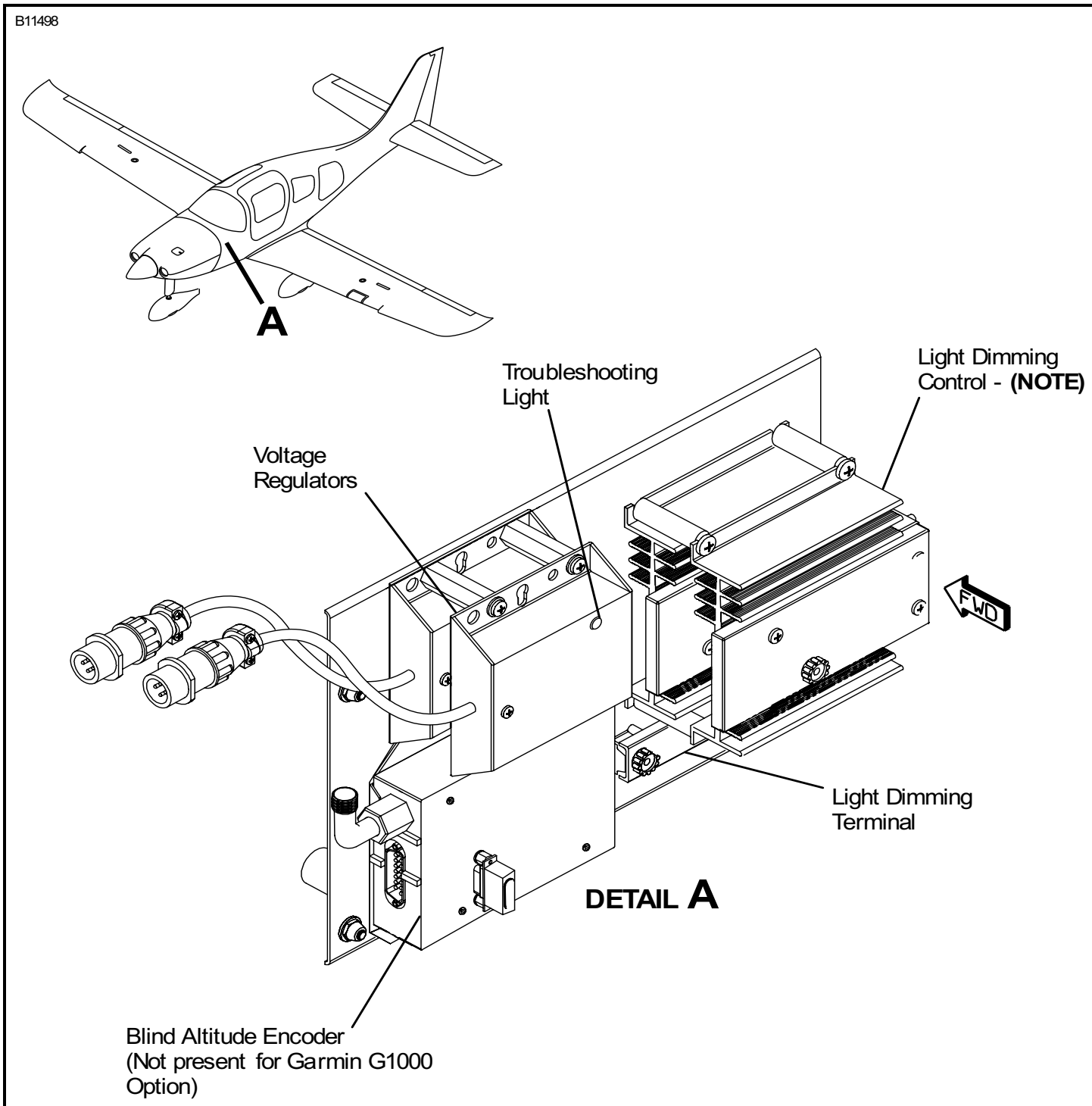
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Figure 209 : Sheet 1 : Instrument Panel Main Plate



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Figure 210 : Sheet 1 : Accessories Unit Mounting Plate



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STALL WARNING SYSTEM - MAINTENANCE PRACTICES

1. Stall Warning System

A. **Stall Warning Detector Assembly Location** - The stall warning detector assembly is located on the left wing on the leading edge about halfway between the fuselage and the wing tip. Refer to Figure 201.

B. **Stall Warning Detector Assembly Removal**

(1) With an erasable marker, mark the outline (position) of the faceplate onto the wing.

(2) Remove the two screws connecting the faceplate to the wing.

(3) Disconnect the two wires after labeling the wires with their corresponding terminal positions.

C. **Stall Warning Detector Assembly Installation**

(1) Connect the two wires to the proper terminals.

(2) To be sure the wires are connected correctly, turn the master switch on and push the stall warning vane up. If the stall warning buzzer sounds, the unit is installed correctly.

(3) Align the faceplate with markings.

(4) Fasten screws in the top right and bottom left corner.

(5) Proceed with the test of the stall warning detector assembly.

D. **Stall Warning Detector Assembly Testing**

(1) Test fly the airplane to determine if the stall warning detector actuates at the correct airspeed. Commence the test at 20 KIAS above the stall speed, and bleed off speed at a rate of 1 knot per second until 5-10 KIAS above stall speed. The stall warning buzzer should sound between 5-10 KIAS above the stall speed.

WARNING: This test should only be performed by qualified pilots at a sufficient altitude to safely recover from a stall and in an area where it is safe to perform this maneuver.

(2) If the stall warning buzzer sounds at speeds in excess of 10 KIAS above the stall speed, loosen the mounting screws on the stall warning detector assembly, and move the assembly slightly down.

WARNING: Do not bend the metal tab (vane) to adjust the stall warning detector.

(3) If the stall warning buzzer does not sound before reaching a speed of 5 KIAS above stall speed, loosen the mounting screws on the stall warning detector assembly, and move the assembly slightly up.

E. **Stall Warning Buzzer Removal**

(1) Remove the overhead console. Refer to Chapter 25-30-00 for overhead console removal. Also refer to Figure 202, Sheet 1 and Sheet 2, for the location of the buzzer in the overhead console.

(2) Disconnect the connector attached to the buzzer. Refer to Figure 203.

(3) Remove the attachment screw.

F. **Stall Warning Buzzer Installation**

(1) Using the screw, fasten the stall warning buzzer to the overhead console. Refer to Figure 202.

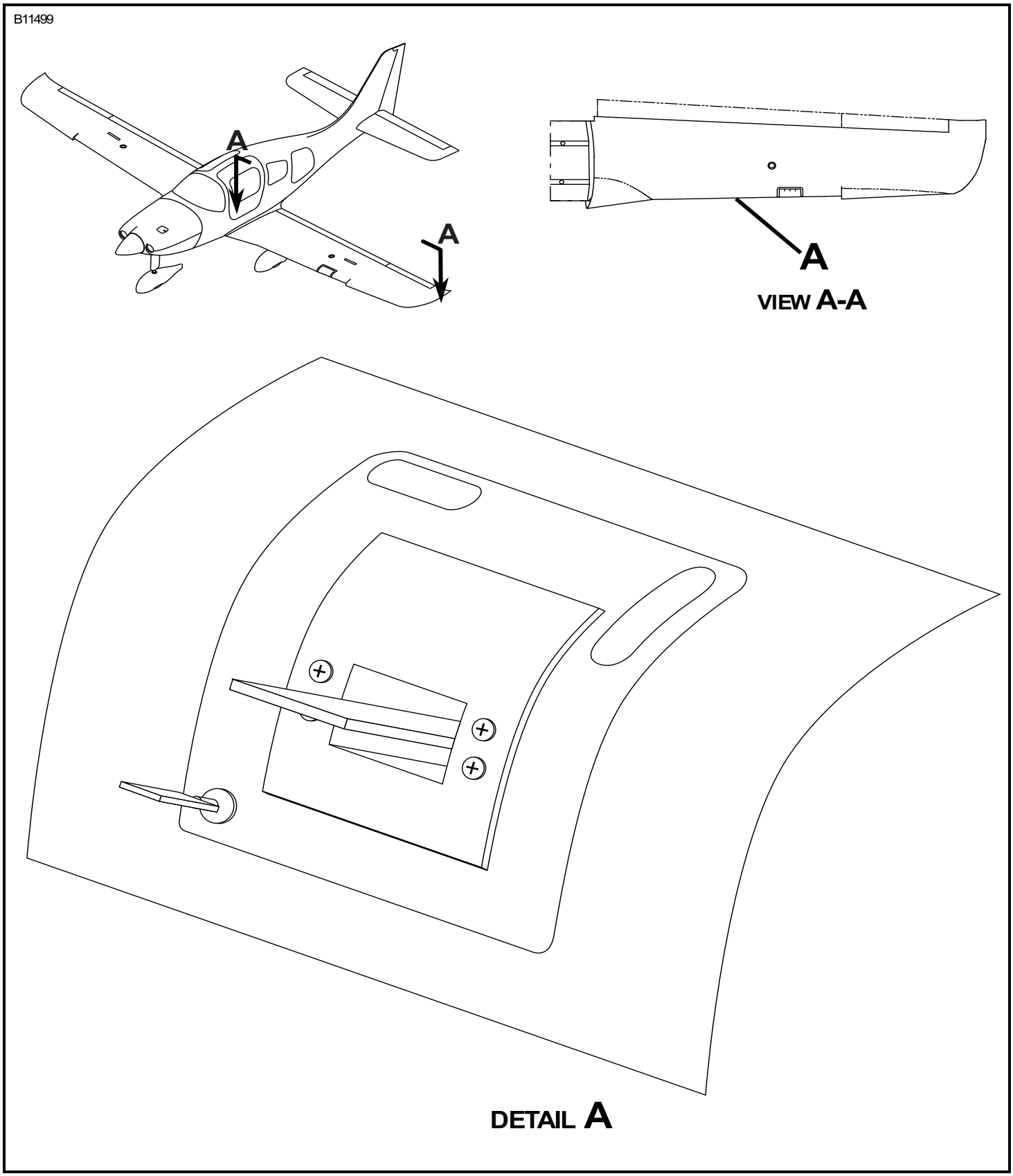
(2) Connect the wires to the buzzer using the connector shown in Figure 203.

(3) Test the operation of the horn by turning the master switch on, and push the stall warning vane up. If the stall warning buzzer can be heard, it is installed correctly.

(4) Reinstall the overhead console as shown in Chapter 25-30-00.

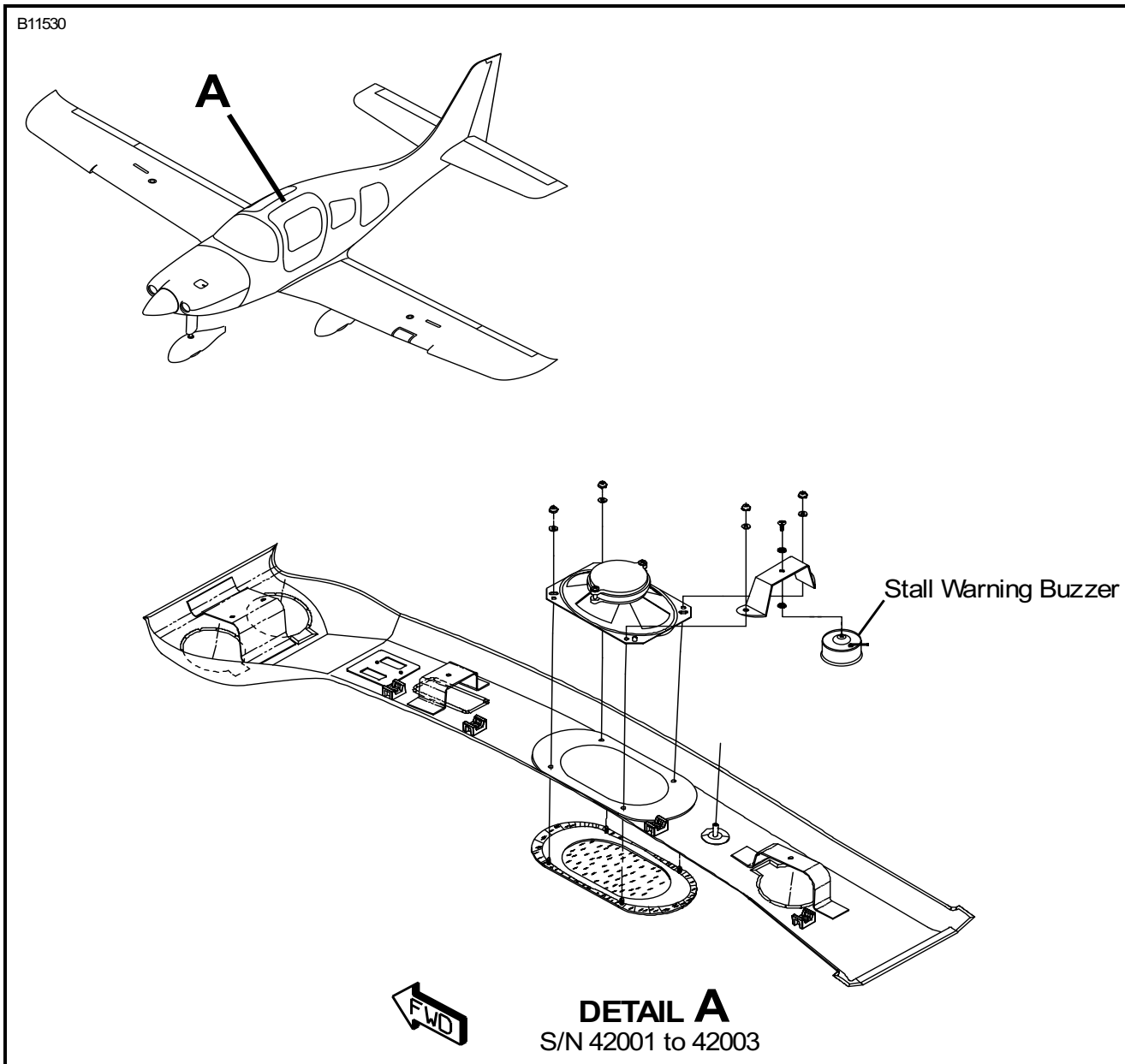
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Figure 201 : Sheet 1 : Stall Warning Detector Assembly



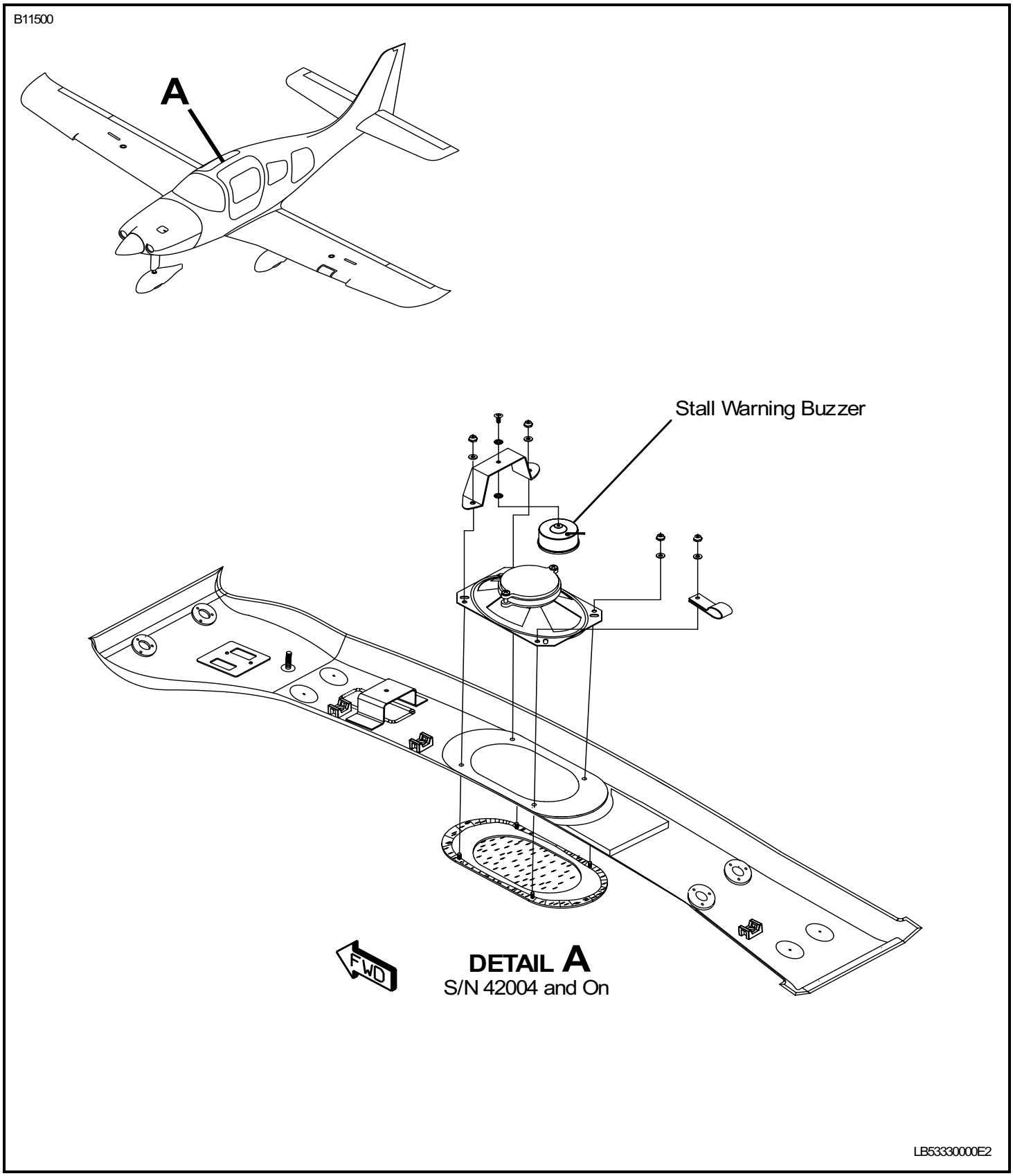
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Figure 202 : Sheet 1 : Stall Warning Buzzer Installation



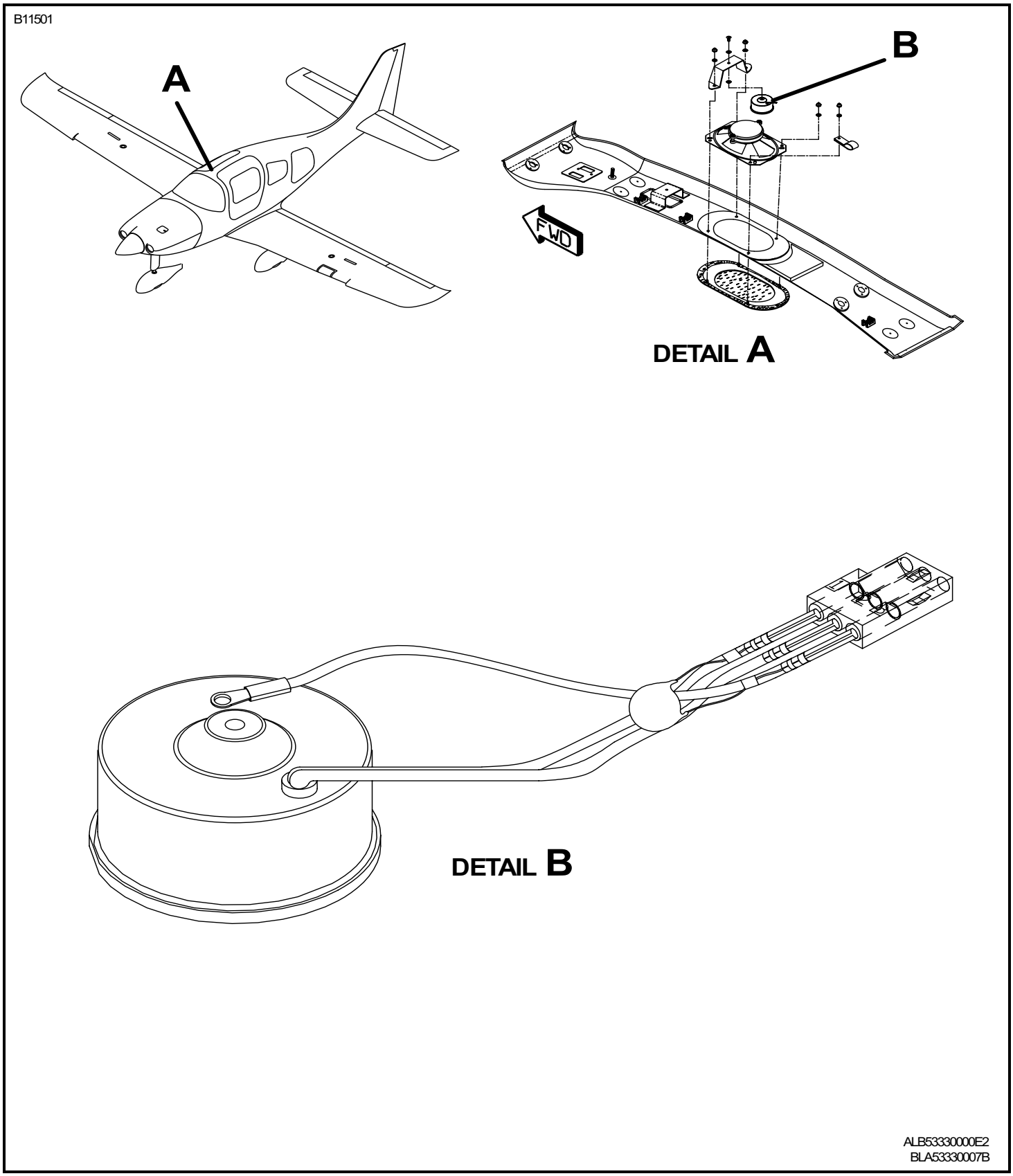
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Figure 202 : Sheet 2 : Stall Warning Buzzer Installation



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Figure 203 : Sheet 1 : Stall Warning Horn



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ATTITUDE AND DIRECTIONAL SYSTEM - MAINTENANCE PRACTICES

1. Attitude and Directional Gyro Indicators

A. **With Avidyne Entegra Primary Flight Display or Garmin G1000 Option** - Attitude and Directional Gyro information are displayed on the PFD. Removal and installation of a backup attitude instrument located to the right of the PFD (basic or Avidyne option) or to the left of the PFD (Garmin G1000 option) is as follows.

(1) **Attitude Indicator Removal**

(a) Turn the master power switches off.

(b) For the Avidyne option, remove the Avidyne Multi-Function Display (refer to the Avidyne Entegra EX5000 Multi-Function Display (MFD) section). For the Garmin G1000 option, access the indicator from under the instrument panel.

(c) Unfasten the instrument from the instrument panel.

(d) Disconnect the wires going into the back of the instrument.

(2) **Attitude Indicator Installation**

(a) Slide the instrument into its slot in the instrument panel.

(b) Connect the proper wires into the back of the instrument.

(c) Fasten the instrument to the instrument panel.

(d) Reinstall the Avidyne Multi-Function Display (Refer to the Avidyne Entegra EX5000 Multi-Function Display (MFD) section), as applicable.

B. **Without Avidyne Entegra Primary Flight Display (Basic Option)**

(1) Attitude and Directional Gyro Indicators Removal

(a) Unfasten the main plate of the instrument panel. Refer to Figure 209.

(b) Carefully slide the plate out of the instrument panel.

(c) Unfasten the instrument from the plate.

(d) Disconnect the wires going into the back of the instrument.

(2) Attitude and Directional Gyro Indicators Installation

(a) Slide the instrument into its slot in the main plate.

(b) Fasten the instrument to the main plate.

(c) Connect the proper wires into the back of the instrument.

(d) Gently slide the main plate into the instrument panel.

(e) Refasten the main plate to the instrument panel. Refer to Figure 209.

C. **Attitude and Directional Gyros Testing**

(1) Turn the master switches on. A red flag on the upper right side of the attitude indicator should disappear, indicating power to the unit.

(2) Listen to determine if the gyro is operating properly.

(3) If the gyro is not heard, remove the indicator, and take it to a qualified technician for replacement or repair.

2. Turn Coordinator (Basic or Avidyne Option Only)

A. **With Avidyne Entegra Primary Flight Display** - The turn coordinator is located in the avionics shelf under the baggage compartment floor refer to Figure 203). No turn coordinator is installed with the Garmin G1000 option.

(1) Turn Coordinator Removal

(a) Remove the carpet and access panel from the baggage compartment floor.

(b) Loosen the mounting screws that hold the turn coordinator into its mounting bracket attached to the avionics shelf.

(c) Disconnect all connectors to the unit.

(d) Remove the unit.

(2) Turn Coordinator Installation

(a) Perform the Turn Coordinator Removal procedure above in reverse order.

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(b) Level the aircraft (Refer to Chapter 8).

(c) If the ball is not in the center, loosen the mounting screws and adjust the turn coordinator until the ball is centered.

B. **Without Avidyne Entegra Primary Flight Display**

(1) Turn Coordinator Removal

(a) Unfasten the main plate of the instrument panel. Refer to Figure 209.

(b) Carefully slide the plate out of the instrument panel.

(c) Unfasten the turn coordinator from the plate.

(d) Remove the connectors attached to the indicator.

(e) Remove the turn coordinator by pulling it out of the instrument panel.

(2) Attitude and Directional Gyro Indicators Installation

(a) Perform the Turn Coordinator Removal steps above, in reverse order.

(b) Level the aircraft. Refer to Chapter 08-00-00.

(c) If the ball is not in the center, loosen the mounting screws and adjust the turn coordinator until the ball is centered.

C. **Turn Coordinator Testing**

(1) Turn the master switches on. A red flag should disappear, indicating the power is on. Refer to Figure 201.

(2) Listen to the indicator to determine if the gyro is properly turning.

(3) If the gyro is not heard, remove the indicator, and take it to a qualified technician for replacement or repair.

3. Magnetic Compass

A. **Magnetic Compass Removal and Installation** - The compass is mounted on the windshield, as shown in Figure 202. The base cannot be removed as it is adhered to the windshield. If the compass needs to be replaced, a new compass unit should be acquired, the pin attaching the compass to the base should be removed, the new compass installed, the pin replaced, and the connector reconnected.

B. **Magnetic Compass Base Re-installation** - Use the following procedure should the compass base become detached from the windshield.

(1) Ensure the windshield is warmer than 60°F.

(2) Disassemble the base from the compass. Remove all adhesive from the compass base and windshield using isopropyl alcohol.

(3) Apply 3M™ pressure sensitive adhesive P/N VHB 4947F to the base.

(4) Center the base on the windshield as shown in Figure 202 and adhere. Secure in place with 20±3 pounds force for at least 30 minutes.

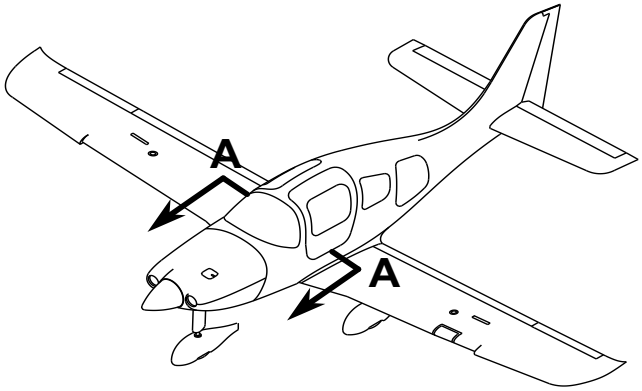
(5) Reinstall the compass onto the base.

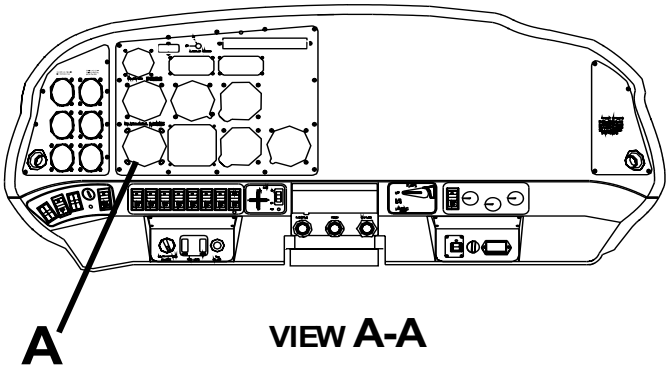
NOTE: The magnetic compass should be checked for calibration each time avionics or instruments are installed in the instrument panel. A qualified technician must calibrate the magnetic compass under these conditions.

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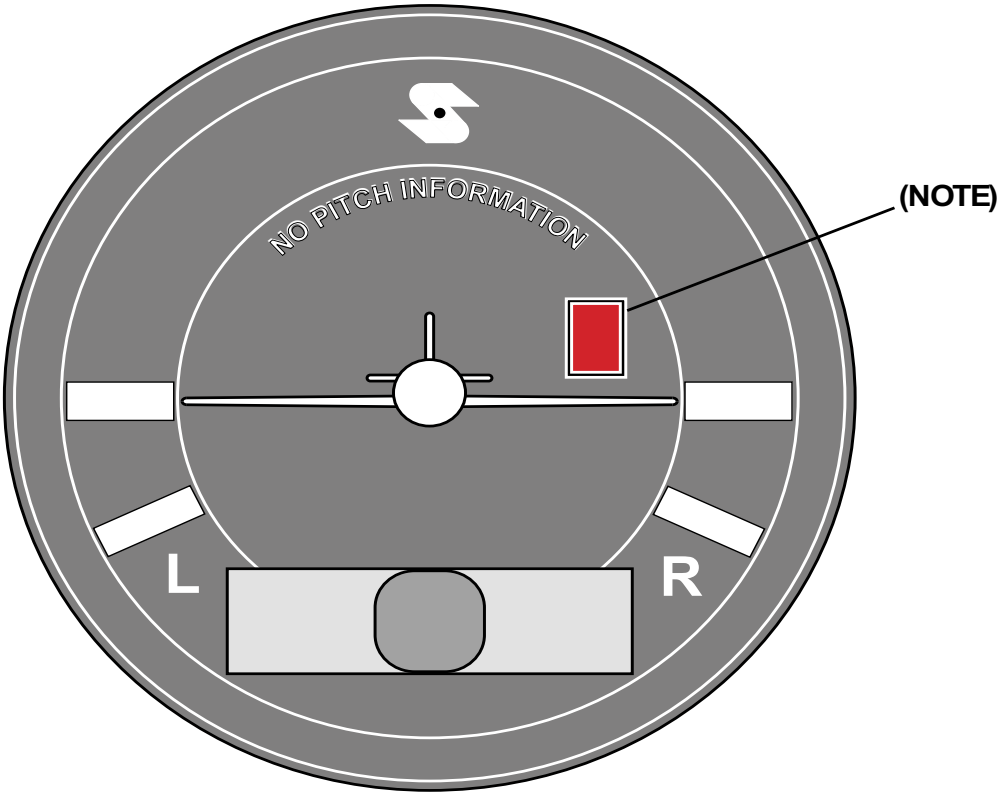
Figure 201 : Sheet 1 : Turn Coordinator Flag

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VIEW A-A

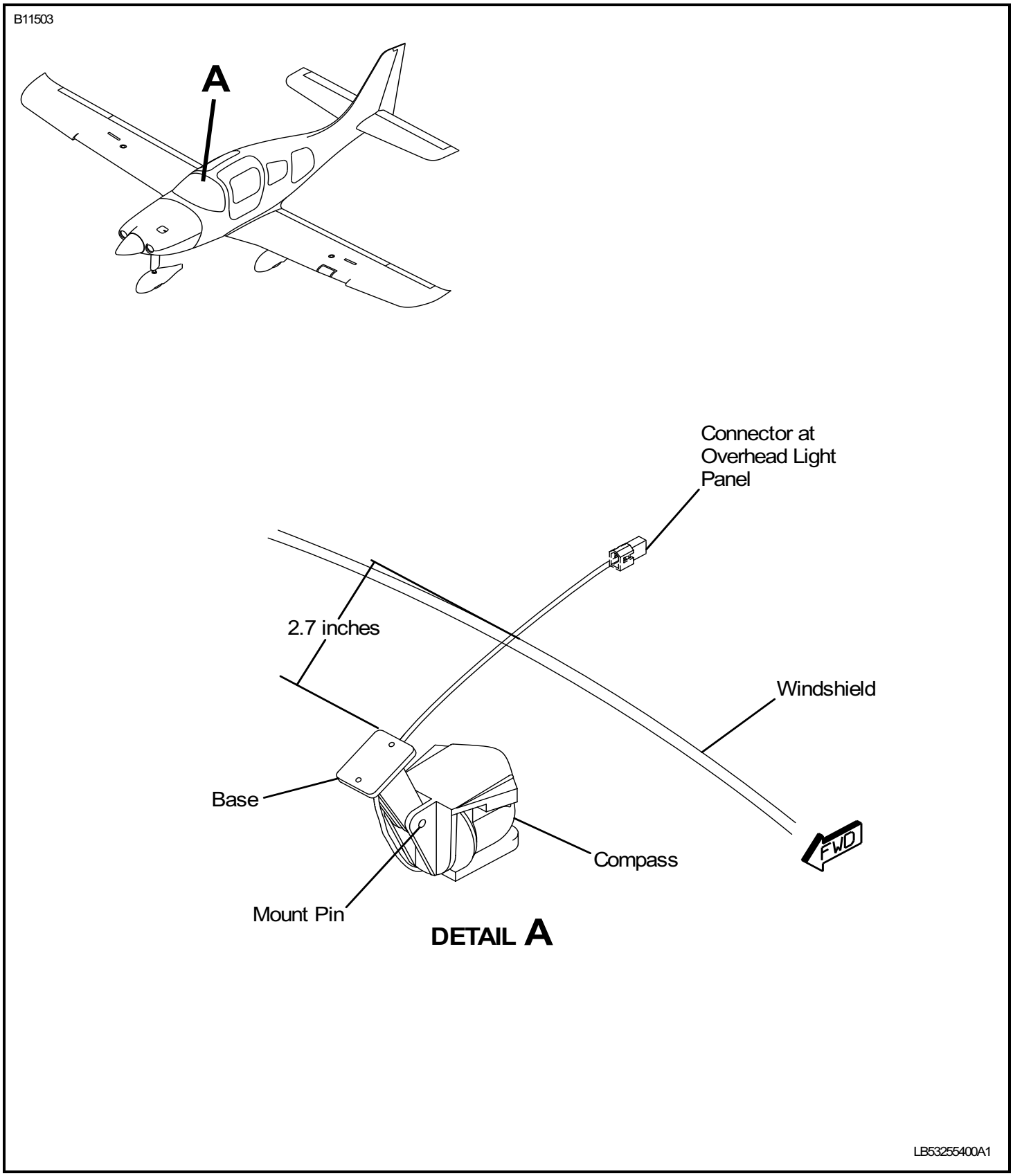


DETAIL A

NOTE: Red Flag:
Showing - Not Powered
Not Showing - Powered

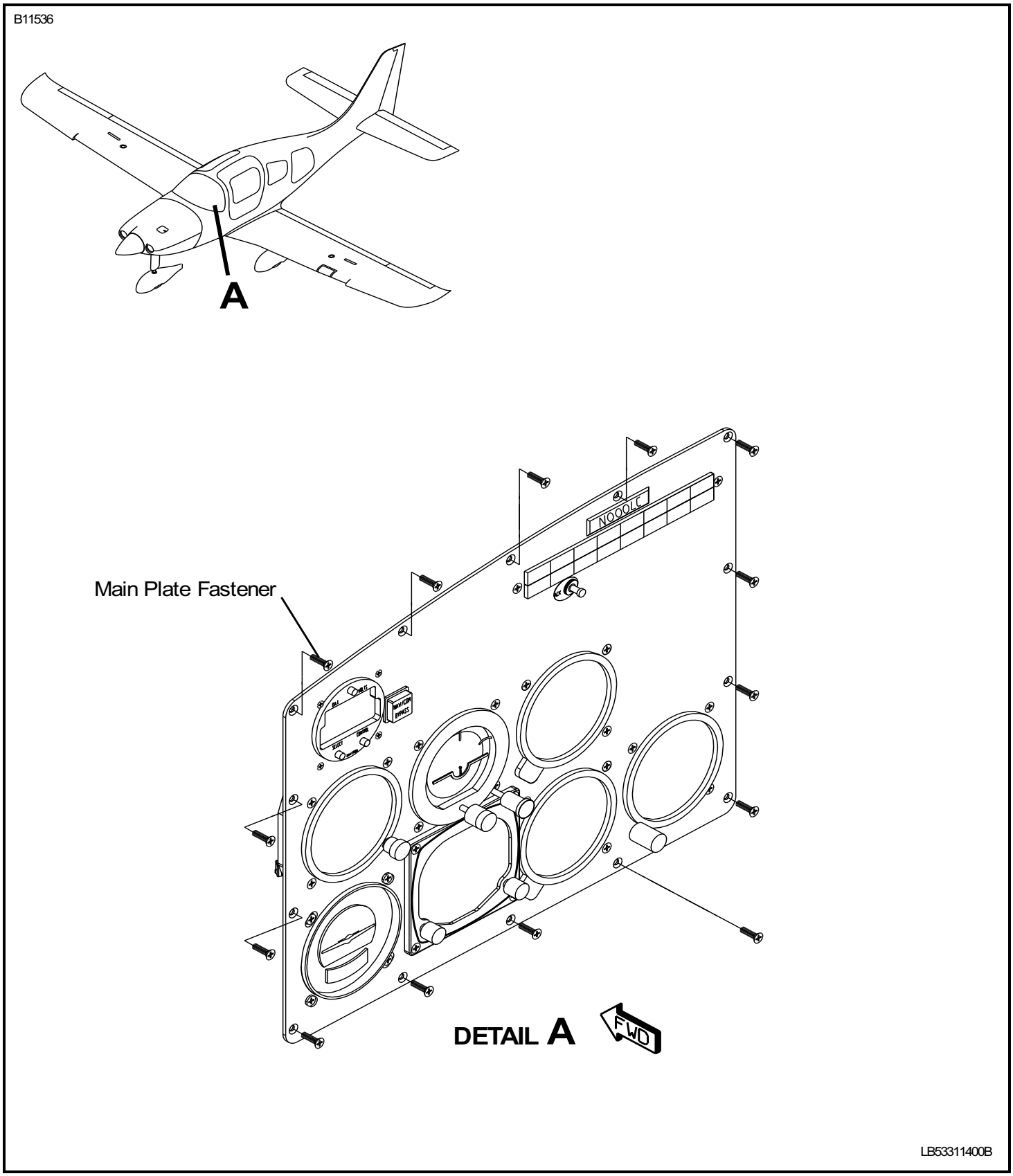
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Figure 202 : Sheet 1 : Magnetic Compass Windshield Attachment



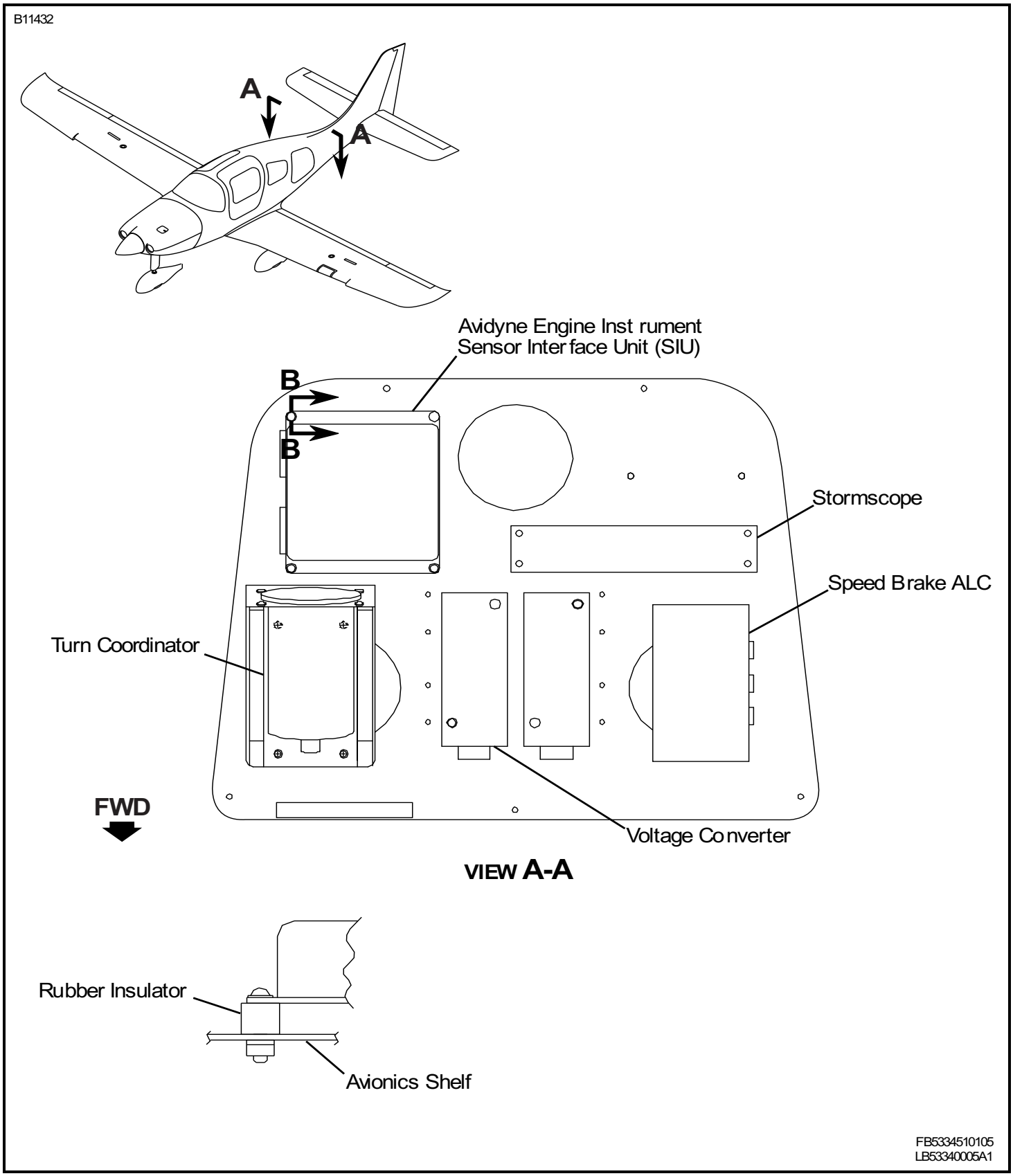
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Figure 209 : Sheet 1 : Instrument Panel Main Plate



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Figure 203 : Sheet 1 : Avionics Shelf Configuration for Avidyne Entegra Equipment



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MARKER BEACON AND LANDING SYSTEM - MAINTENANCE PRACTICES

1. Marker Beacon Receiver

A. Marker Beacon Indicators

(1) There are two marker beacon indicators:

(a) As part of the GARMIN AT Apollo SL15 Audio Amplifier or Garmin GMA 340 Audio Panel, described in Chapter 23.

(b) An S-Tec remote marker beacon light, located in the upper part of the main plate on the instrument panel. Refer to Figure 201.

B. Removal of Remote Marker Beacon

(1) Remove the glare shield and the glare shield electrical connectors.

(2) Unfasten the remote marker beacon indicator from the front of the panel. Refer to Figure 202, or Section 34-60-00, Figure 201, Avidyne Entegra Primary Flight Display and EX5000 Multi-Flight Display Installations.

(3) Disconnect the connector on the back side of the indicator.

(4) Remove the marker beacon indicator.

C. Replacement of Remote Marker Beacon

- Perform the Removal of Remote Marker Beacon procedure above in reverse order.

D. Testing of Marker Beacon Indicator Lights

(1) Both the S-Tec remote indicator and the GARMIN AT Apollo SL15 audio amplifier or the Garmin GMA 340 audio panel marker beacon lights are tested by operating the test switch on the SL15 or the GMA 340, respectively.

(2) If any light bulb fails to illuminate, take the unit to a qualified radio technician to have the bulbs replaced.

(3) If all lights fail to illuminate, check for loose connections. If the lights still fail to illuminate, take the unit to a qualified radio technician for analysis.

E. Marker Beacon - Removal of Antenna

(1) The Marker Beacon Receiver antenna is located beneath the passenger footwell. Refer to Figure 203.

(2) Removing the passenger footwell

(a) Remove all four seats.

(b) Lift the carpet in the footwell, and remove the screws connecting the footwell to the flanges.

(c) Remove the footwell from the plane.

(3) Unfasten the screws and washers connecting the antenna to the fuselage.

(4) Disconnect the antenna cable at the connector.

(5) Remove RTV white silicone sealant, if present, and remove the antenna.

F. Installation of Marker Beacon Antenna

(1) Connect the antenna cable to the antenna.

(2) Fasten the antenna to the fuselage with the screws and washers. If there is a gap between the base of the antenna and the wing skin after installation, fill the void around the antenna base with RTV white silicone sealant.

(3) Place the footwell into the flange, and fasten it.

(4) Install the carpet.

(5) Install the four seats.

2. Marker Beacon Antenna - Tuned Conditions

A. The marker beacon antenna is tuned at the factory when the airplane is manufactured. In most cases, further antenna tuning is not necessary. When the marker beacon is properly tuned, the conditions shown in Table 201 will be observed in flight. At an airplane speed of 90 knots, an altitude of 1000 ft. above the marker beacon transmitter, and the receiver set at the low-sense setting, a station passage time of 13 to 20 seconds should be observed. Station passage is indicated by both a visual and an aural indication. The times listed assume that the airplane is on the centerline of the localizer course or tracking on the extended centerline of the ILS runway. If station passage times outside the parameters shown in Table 201 are observed in flight, the antenna on the airplane can be tuned using the steps described in the next paragraph.

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Table 201. Marker Beacon Acceptable Conditions

Times for Marker Beacon Station Passage		
Ground Speed	**Time for Station Passage**	**Altitude AGL/Sense Setting**
90 Knots	13 to 20 Seconds	1000 feet/Low
110 Knots	11 to 16 Seconds	1000 feet/Low

B. **Marker Beacon Antenna Tuning Adjustments** - Construct a Test Antenna Platform (TAP). The TAP is a 2 ft., square piece of sheet aluminum (any thickness) with a CI-102 Comat marker beacon antenna attached in the exact center of the aluminum sheet. The gasket that comes with the antenna provides a template for drilling the attachment screw holes, the hole for the antenna jack, and the hole for the tuning slug. Be sure to install the gasket when mounting the antenna. Refer to Figure 204 and Figure 205 for a picture of the top and bottom of a typical TAP.

(1) When the TAP is assembled, attach a 6- to 10-ft. antenna cable to the antenna jack on the bottom side.

(2) Remove the access cover under the wing center section, just aft of the airplane's marker beacon antenna. The cover spans the width of the airplane's underside and provides access to the aileron bellcranks and crossover assembly, flap motor, and the marker beacon tuning slug.

(3) In a level area, such as an airplane hangar, place the TAP directly below the airplane's marker beacon antenna. The TAP must be placed on a box or similar type structure so that the distance between the hangar floor and the top of the aluminum sheet is exactly 6 inches. Refer to Figure 206.

(4) When the TAP is directly under the airplane's marker beacon antenna (mirror image) and in a level position, 6 inches above the hangar floor, connect an IFR Model Nav 401 (or equivalent) signal generator to the TAP antenna cable.

(5) Turn on the master switch, avionics master switch, and audio panel, and set the marker beacon receiver to the high sense position.

(6) Turn on the signal generator, and set the selector to the marker beacon mode.

(7) Turn the signal output level of the Nav signal generator down (higher dBm) until the orange MM light on the audio panel just goes out. With an observer in the cockpit, adjust the tuning slug on the airplane's marker beacon receiver antenna with a 0.25-inch socket wrench. Rotate the tuning slug until there is a visual and aural indication in the cockpit. Since the setting of the airplane's antenna is unknown, it will be necessary to experiment with the rotation direction of the tuning slug, i.e., clockwise or counterclockwise. When performing the test, ensure that both the lights on the audio panel and repeater indicator are illuminated.

CAUTION: Do not rotate the tuning slug on the antenna more than 1-1/2 turns in either direction from where the procedure began.

(8) Repeat the previous step until achieving peak performance from the Marker Beacon receiver and obtaining the lowest possible (highest dBm) signal from the test generator. When performing this adjustment, the minimum triggering level (MTL) of the audio panel should be at least -26 dBm. Further troubleshooting or adjustment of the audio panel should be made if -26 dBm MTL cannot be obtained.

(9) When the antenna is properly tuned, turn off the system and avionics master switches, and install the access cover.

C. **Marker Beacon Tuning Adjustments**

NOTE: Never adjust the tuning slug on the TAP antenna. It must remain positioned to the factory default setting.

3. Instrument Landing System - Nav Antenna

A. **Nav (ILS) Antenna Removal**

(1) The Nav Antenna is located in the vertical stabilizer. Refer to Figure 207.

(2) Remove the access panel on the vertical stabilizer by removing the screws.

(3) Loosen the set screw holding the left element in place.

(4) Grasp the antenna rod and twist counterclockwise to unscrew from the antenna body.

(5) If a set screw is present, loosen the set screw holding the opposite element in place. Repeat the previous step for the antenna on the opposite side of the tail.

(6) Remove the screws mounting the Nav Antenna body from the bulkhead.

(7) Disconnect the antenna cable from the antenna body. On aircraft equipped with the Garmin G1000 system, there is a

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filter installed between the antenna cable and the antenna.

B. **Nav (ILS) Antenna Installation**

(1) Connect the antenna cable to the antenna body. On aircraft equipped with the Garmin G1000 system, connect the antenna cable and filter to the antenna body and secure using heat-shrink wrap, ensuring heat-shrink wrap covers the receptacle, filter, and minimum of 3 inches of the antenna cable.

(2) Fasten the Nav Antenna body to the bulkhead.

(3) Insert the antenna rods through the holes, and screw clockwise into the antenna body until snug. Apply Loctite to the right element as required by the instructions for the right element below.

(4) Tighten the set screw holding the left element in place.

(5) If the right element had a set screw holding it in place, tighten the set screw. If there was no set screw, apply Loctite on the two threads closest to the antenna rod prior to installation.

(6) Install the access panel to the vertical stabilizer with the screws.

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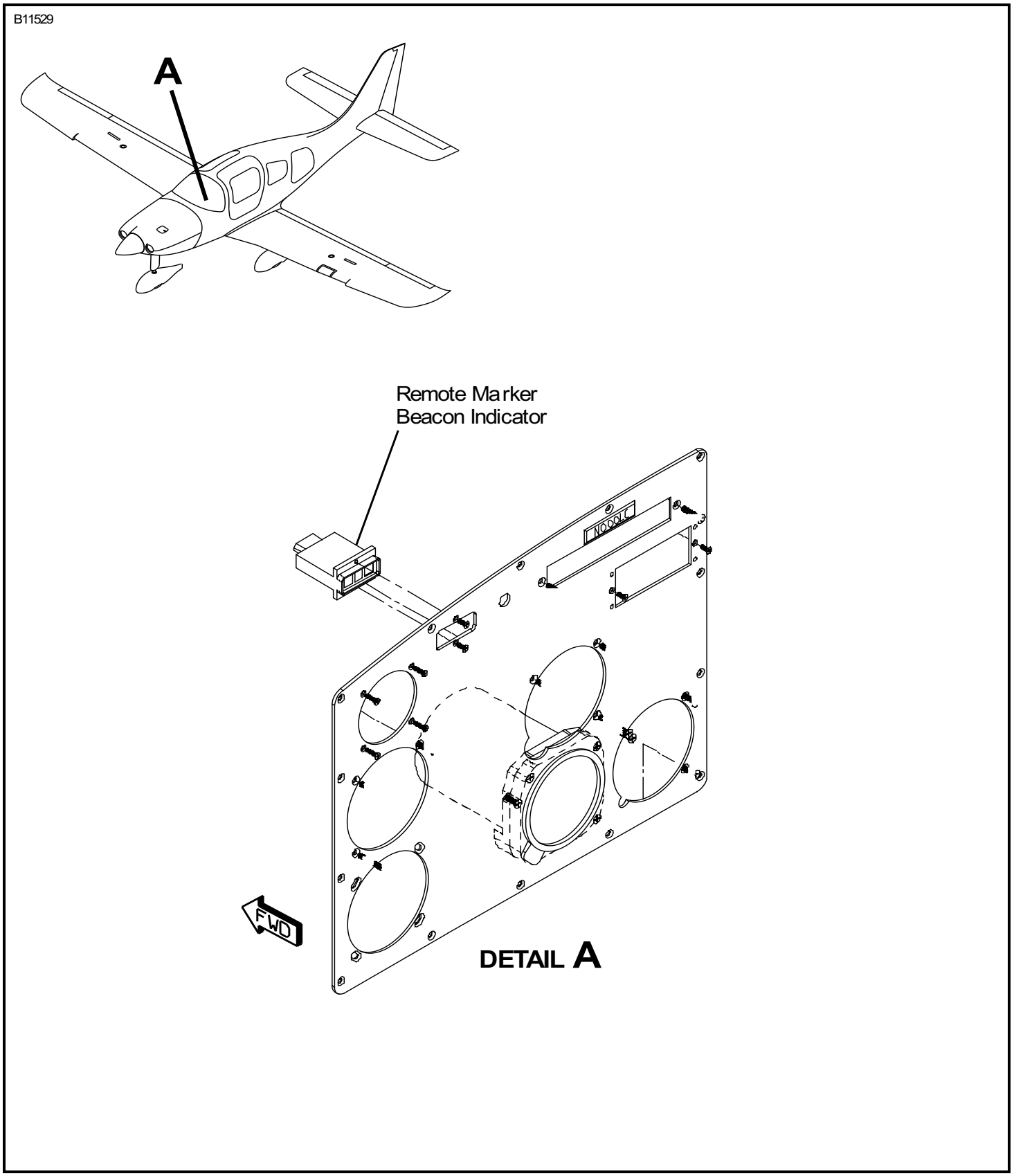
Figure 201 : Sheet 1 : S-Tec Remote Marker Beacon Indicator

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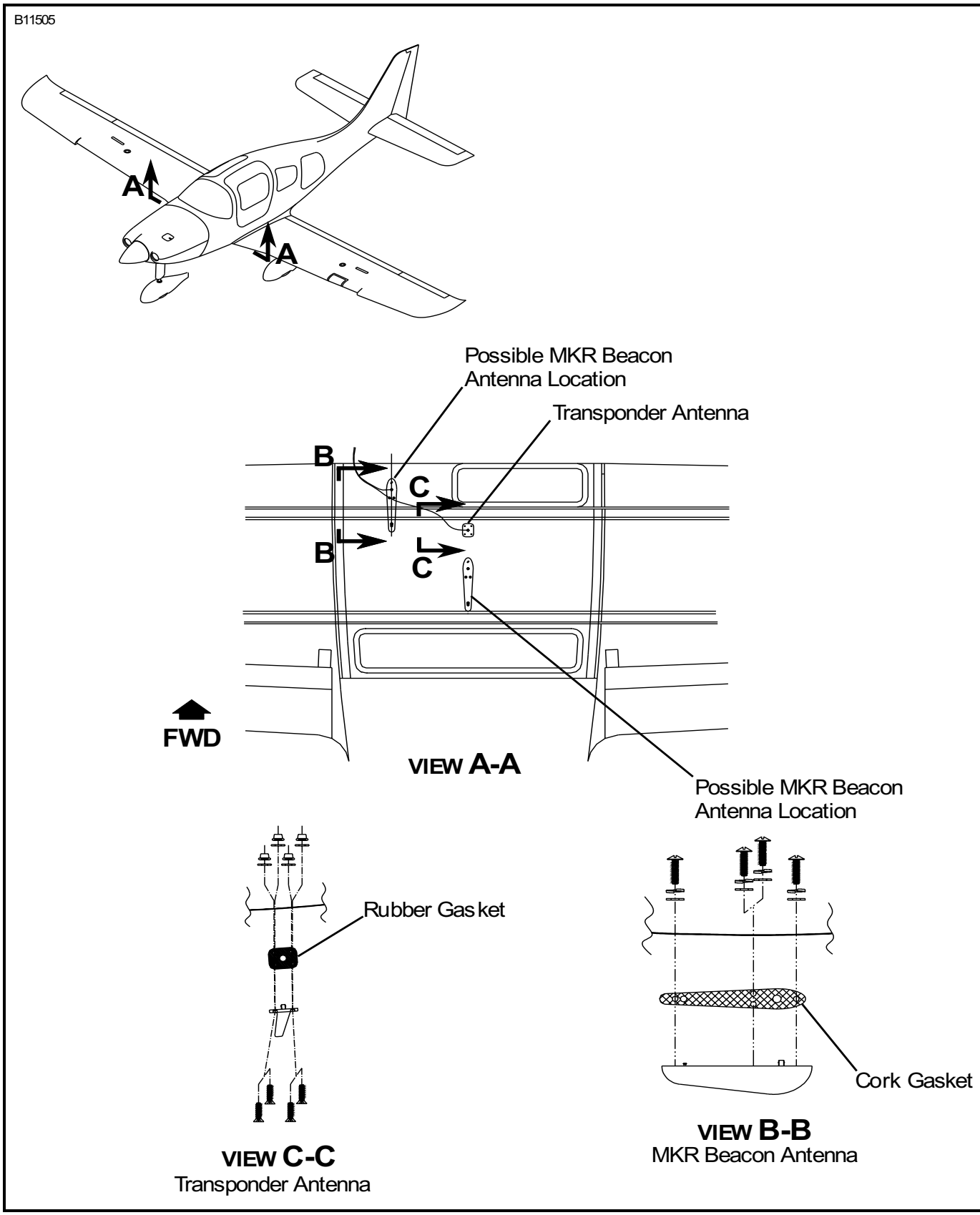
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Figure 202 : Sheet 1 : S-Tec Remote Marker Beacon Indicator



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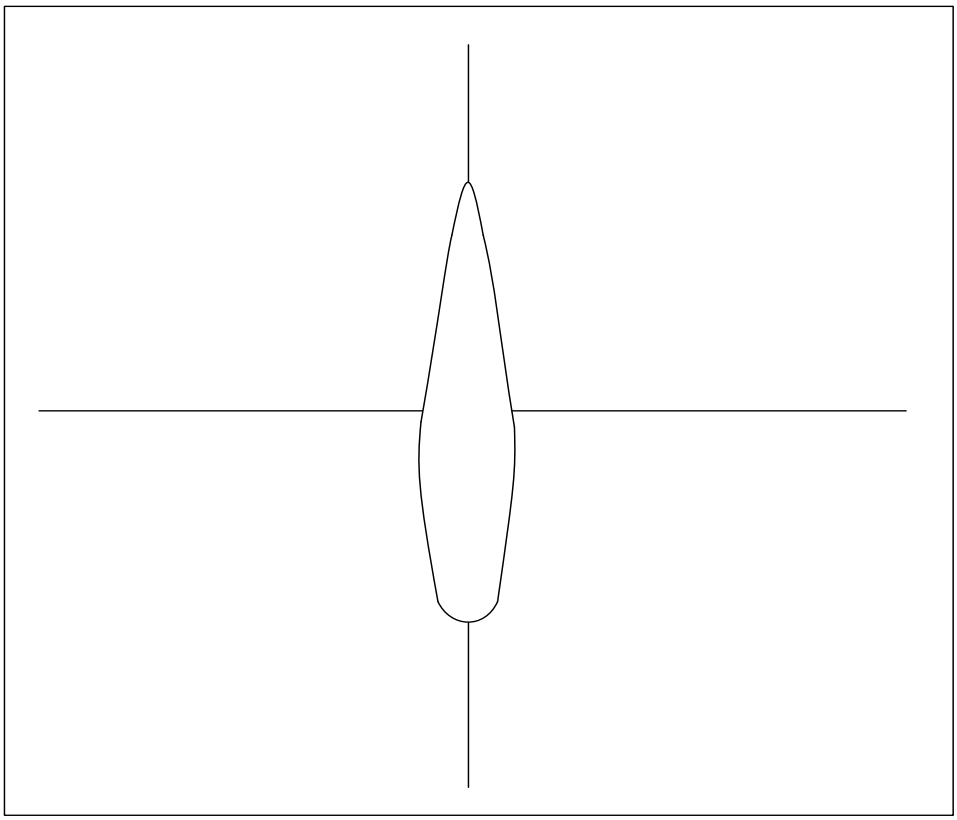
Figure 203 : Sheet 1 : Marker Beacon Antenna Location



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Figure 204 : Sheet 1 : TAP Top

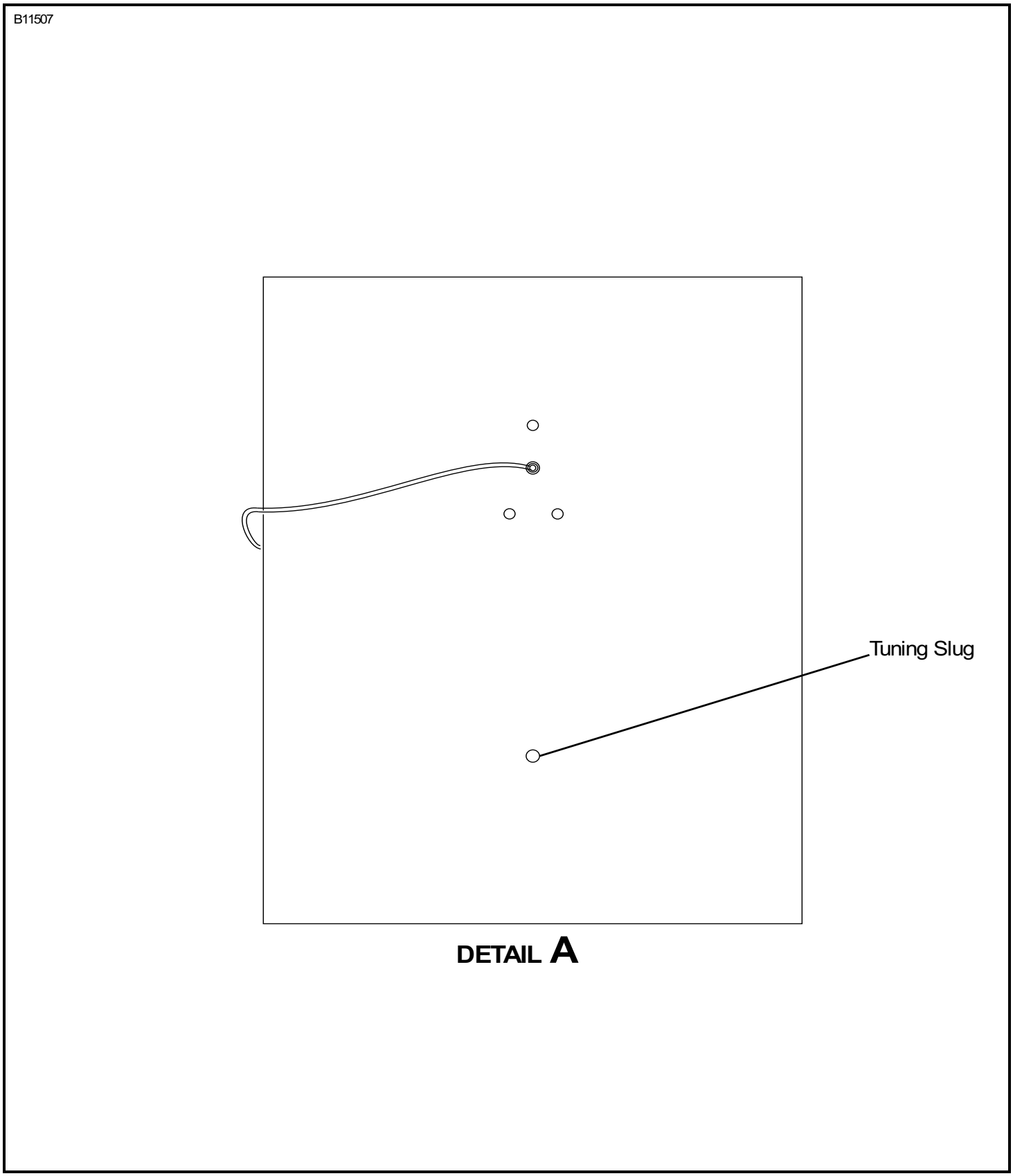
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DETAIL A

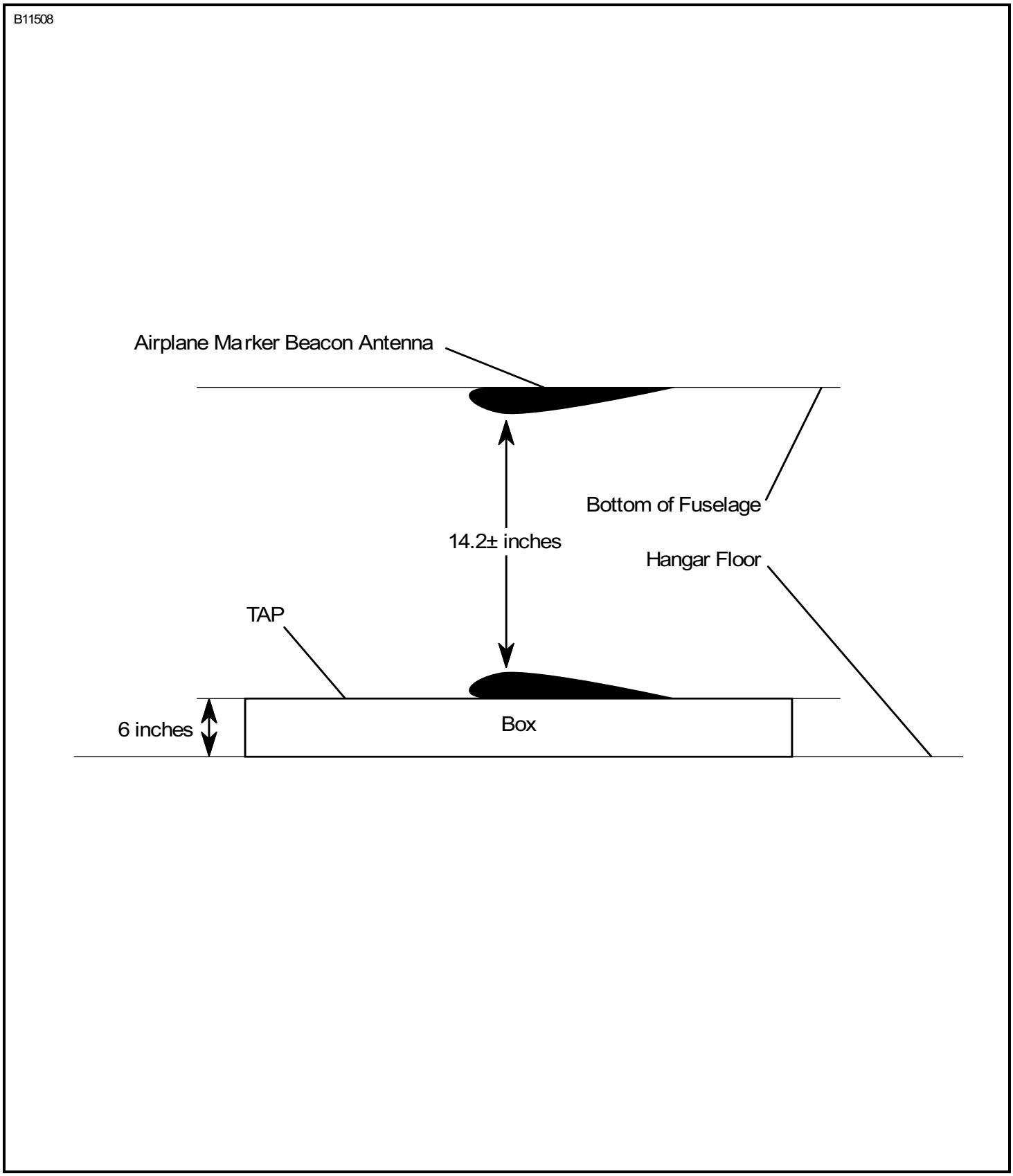
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Figure 205 : Sheet 1 : TAP Bottom



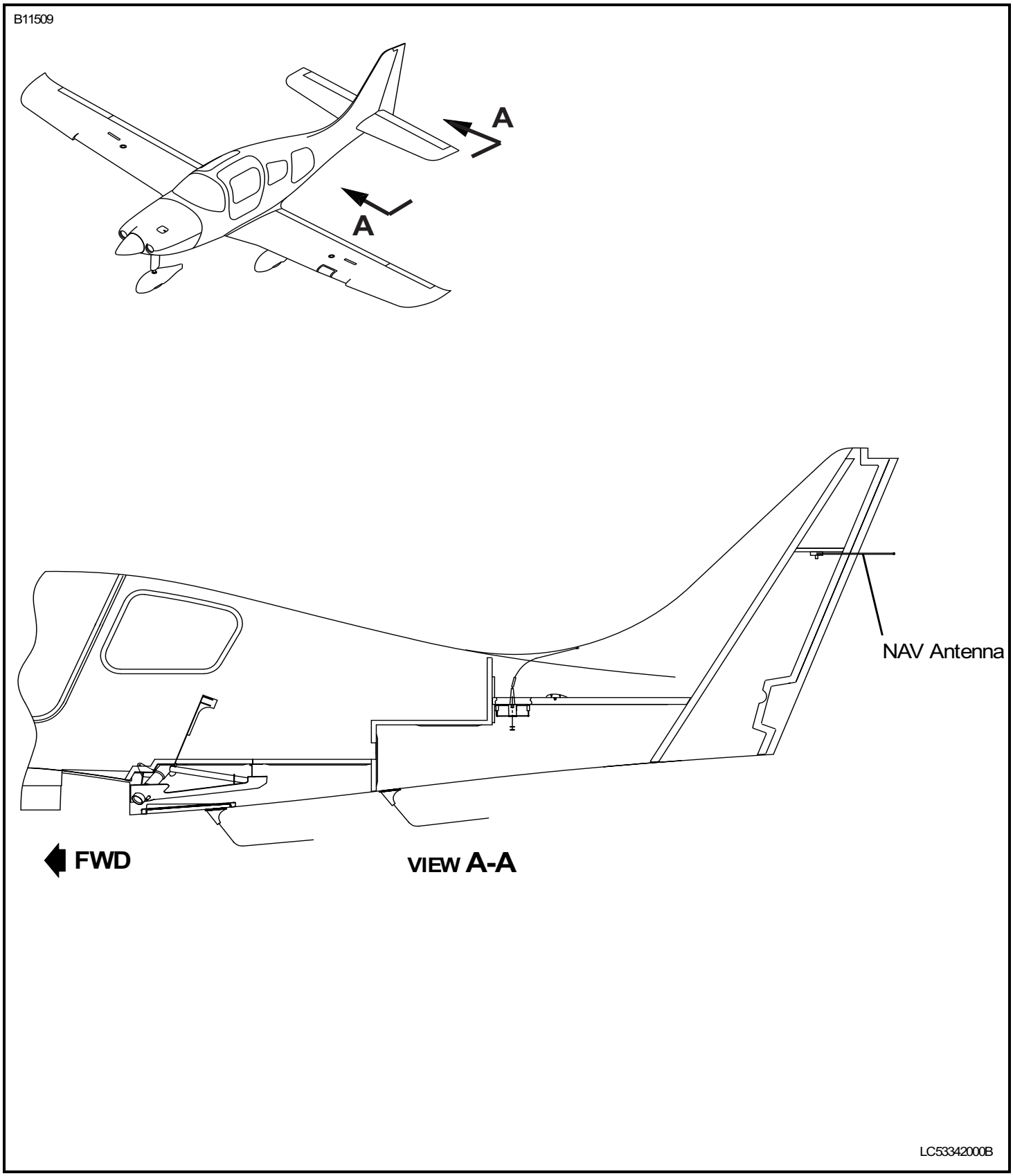
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Figure 206 : Sheet 1 : Location of Test Antenna



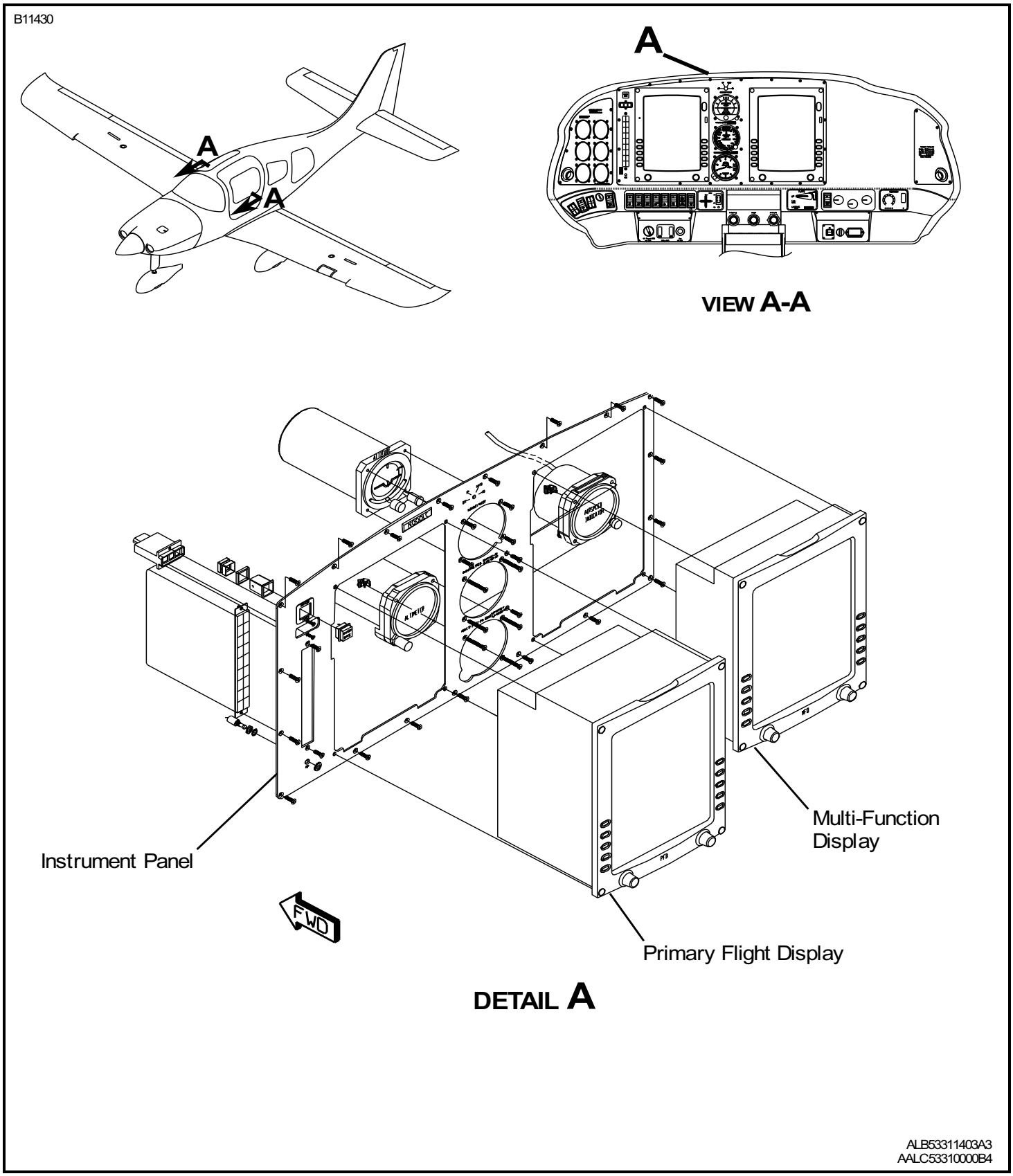
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Figure 207 : Sheet 1 : Nav Antenna Location



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Figure 201 : Sheet 1 : Avidyne Entegra Primary Flight Display and EX5000 Multi-Flight Display Installations



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GPS AND TRANSPONDER - MAINTENANCE PRACTICES

1. Garmin At Apollo GX50 GPS

A. GX50 GPS Removal

(1) Insert an Allen wrench into the hole in the faceplate and turn counterclockwise.

(2) Slide out the GX50 unit out of the rack, and remove the connectors.

B. GX50 GPS Installation

(1) Fasten the connector to the GX50 GPS.

(2) Slide the GX50 unit into its slot.

(3) Insert an Allen wrench into the hole on the cover plate, and turn clockwise until the unit is firmly seated in the rack.

C. GX50 GPS Antenna

(1) The GX50 GPS antenna is located above the ELT in the space aft of the baggage compartment. Refer to Figure 201.

(2) To service the antenna, remove the aft baggage bulkhead upholstery plug and the cover plate.

2. Garmin GNS 430 GPS System

A. GNS 430 GPS Removal

(1) Insert a 3/32 in. Allen wrench into the hole at the bottom of the faceplate, and turn counterclockwise.

(2) Slide the GNS 430 unit out of the rack, and remove the connectors.

B. GNS 430 GPS Installation

(1) Fasten the connector to the GNS 430 unit.

(2) Slide the GNS 430 unit into its slot.

(3) Insert a 3/32 inch Allen wrench into the hole at the bottom of the faceplate, and turn clockwise until the unit is firmly seated in the rack.

C. **GNS 430 Antenna** - The GNS 430 antennas are located above the ELT in the space aft of the baggage compartment as shown in Figure 201. If the Garmin avionics package is installed, there will be two antennas, one for each unit.

(1) Antenna Removal

(a) Remove the aft baggage bulkhead upholstery plug and the cover plate.

(b) Remove the four nuts on each antenna.

(c) Lift the antenna and antenna gasket from the antenna box.

(2) Antenna Installation

(a) Place the gasket on the antenna box, and align the holes.

(b) Place the antenna over the gasket with the studs through the holes in the antenna box.

(c) Install the four nuts, and torque 20 to 25 in.-lbs.

(d) Replace cover plate and aft baggage bulkhead upholstery plug.

3. Garmin At Apollo Annunciator Control Unit (ACU)

A. The following procedures for removal and installation of the GARMIN AT Apollo ACU apply only if the Avidyne Primary Flight Display is not installed in the aircraft.

(1) **ACU Remote Removal** - The ACU is located in the instrument panel on the main instrument plate. Refer to Figure 202.

(a) Remove the glare shield per instructions in Chapter 25-11-00, and remove any fittings installed into the back of the ACU.

(b) Unfasten the mounting screws on the face of the ACU.

(c) Slide the unit out of the instrument panel.

(2) **ACU Remote Installation**

(a) Install any fittings into the back of the ACU remote.

(b) Slide the unit into its slot in the instrument panel, and screw in the mounting screws.

(c) Install the glare shield.

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(3) **ACU Remote Testing**

(a) Press TEST to ensure the annunciator lights are all operative. Refer to the GARMIN AT manual for servicing to the unit.

(b) If the annunciator lights are operative, but do not operate when the GPS receiver shows it should, check wiring (refer to wiring manual) for continuity.

(c) If no apparent wiring or bulb outage is discovered, refer to the GARMIN AT GPS manual for instructions. Only qualified technicians should service this unit.

4. Mid-Continent MD41-1464A Annunciator Control Unit (ACU)

A. The following procedures for removal and installation of the Mid-Continent MD41-1464A ACU apply only if the Avidyne Primary Flight Display is not installed in the aircraft.

(1) **ACU Removal** - The ACU is located in the instrument panel on the main instrument plate. Refer to Figure 203 for a picture of the ACU.

(a) Remove the glare shield per instructions in Chapter 25-11-00.

(b) Remove the connector at the back of the ACU unit.

(c) Remove the two mounting screw on the face of the ACU.

(d) Slide the unit out of the instrument panel.

(2) **ACU Installation**

(a) Slide the unit into the instrument panel.

(b) Install the two screws securing the ACU to the instrument panel, and tighten snug.

(c) Attach the connector at the back of the ACU unit.

(d) Install the glare shield per instructions in Chapter 25-11-00.

(3) **ACU Testing**

(a) Press the test switch on the trim panel to test the annunciators on the ACU.

(b) If the annunciator lights are operative, but do not operate when the GPS receiver shows it should, check wiring (refer to wiring manual) for continuity.

(c) If no apparent wiring or bulb outage is discovered, refer to the Garmin GNS 430 system manual for instructions. Only qualified technicians should service this unit.

5. Garmin At Apollo SL30 Nav/Comm

A. **SL30 Nav/Comm Removal**

(1) The SL30 is located in the radio rack and can be removed by inserting an Allen wrench into the hole in the faceplate and turning counterclockwise. Refer to Figure 204.

(2) Slide the SL30 out of the rack.

(3) Remove the unit.

B. **SL30 Nav/Comm Installation**

(1) Slide the SL30 into its slot in the radio rack

(2) Lock it in place by inserting an Allen wrench into the hole on the cover plate and turning clockwise.

C. **SL30 Nav/Comm Testing** - Other than for checking for loose connections, all testing and service must be accomplished by a qualified radio technician.

D. **SL30 Nav 1 Initial Set-up Procedures**

Nav 1 interacts with the HSI located in the flight panel, bottom row under the Flight Director. This setup procedure should be done while seated in the left pilot seat with the ACU NAV/GPS button in the NAV position.

(1) Turn the aircraft and avionics power switches ON and the SL30 unit OFF.

(2) On the SL30 unit, press and hold the <-> and SYS buttons at the same time and turn the power knob to ON.

(3) Let go of buttons after 3 to 4 seconds.

(4) Turn the large knob one click clockwise, and follow step by step directions as given on the unit's screen for calibrating the resolver.

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(5) When done with the setup, turn the unit off and back on again.

(6) Press the NAV button, and adjust the frequency to 110.00.

(7) Press the OBS button, then rotate the left-hand knob with the green/yellow arrow located on the HSI instrument. The screen display on the unit should follow the rotation of the arrows on the HSI. If not, repeat the entire process.

E. **SL30 Nav 2 Initial Set-up Procedures**

(1) Follow the SL30 Nav 1 Initial Set-up Procedures.

(2) Press the OBS button, then rotate the OBS knob on the NAV indicator instrument. The screen display on the unit should follow the rotation of the arrows on the NAV indicator instrument. If not, repeat the entire process.

6. Navigation Indicator

A. The following procedures for removal and installation of the navigation indicator apply only if the Avidyne Primary Flight Display is not installed in the aircraft.

(1) Either a Mid-Continent MD 200-306 or a Garmin GI 106A Navigation Indicator will be installed. Removal and installation of both instruments are the same.

(2) **Removal of the Navigation Indicator** - (Refer to Figure 205) - The navigation indicator is located in the main plate of the instrument panel.

(a) Turn off the master switch.

(b) Disconnect the main plate from the instrument panel.

(c) Disconnect the connector on the back of the navigation indicator.

(d) Remove the screws connecting the navigation indicator to the instrument panel.

(3) **Navigation Indicator Installation**

(a) Place the navigation indicator in its hole in the instrument panel.

(b) Connect the electrical connectors into the back of the navigation indicator.

(c) Refasten the screws connecting the navigation indicator to the main plate of the instrument panel.

(d) Install the main plate of the instrument panel.

7. Transponder

A. Either an Apollo SL70, Garmin GTX 327, or Garmin GTX 330 Transponder will be installed. Removal and installation of the instruments are the same.

B. **Transponder Removal**

The transponder is located in the radio rack.

(1) Turn off the master switches.

(2) Remove the transponder by inserting an Allen wrench into the hole in the faceplate and turning counterclockwise. Refer to Figure 206 or Figure 207.

(3) Slide the transponder out of the rack.

(4) Disconnect the electrical and antenna connections.

C. **Transponder Installation** - Perform the steps for Transponder Removal above in reverse order.

D. **Transponder Antenna Removal** - (Refer to Figure 208.)

(1) Turn off the master switches.

(2) Remove the access plate under the center of the wing (aircraft belly area).

(3) Unfasten the antenna cable.

(4) Unfasten the antenna, and remove the antenna and the gasket.

E. **Transponder Antenna Installation**

(1) Turn off the master switches.

(2) Follow the steps for Transponder Antenna Removal above, in reverse order.

(3) Ensure proper electrical bonding of the antenna to the aircraft surface.

(4) Restore electrical power.

F. **SL70 Initial Set-up Procedure**

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(1) Turn the aircraft and avionics power switches ON and the SL70 unit OFF.

(2) On the SL70 unit, press and hold the IDENT and ALT buttons at the same time, and turn the power knob to ON while holding the buttons for 3 to 4 seconds.

(3) Rotate the large outer knob to the SL70 CONF page.

(4) Rotate the small inner knob to the ASRC page.

(5) Press the IDENT button (the altitude field will start to flash), and rotate the small inner knob to select SER.

(6) Press the IDENT button to save the selection.

(7) Rotate the small inner knob to the BAUD page.

(8) Press the IDENT button (the BAUD field will start to flash), and rotate the small inner knob to select 1200.

(9) Press the IDENT button to save the selection.

(10) When setup is complete, return to normal operation by switching the unit's power knob to OFF, then back to ON.

G. **GTX 327 and GTX 330 Initial Set-up Procedure** - Refer to the Garmin GTX 327 Transponder Installation Manual, P/N 190-002187-02, or the Garmin GTX 330 Transponder Installation Manual, P/N 190-00207-02 for set-up procedures.

8. Bendix/King KG 102A Remote Gyro And KI525A HSI Indicator

A. If the Avidyne Entegra Primary Flight Display (PFD) is installed in the aircraft the KG 102A remote gyro and KI525A HIS indicator will not be installed in the aircraft.

(1) **KG 102A Remote Gyro Removal** - (Refer to Figure 209) - The KG 102A is located in the avionics shelf under the baggage compartment floor.

(a) Remove the carpet and access panel from the baggage compartment floor.

(b) Loosen the mounting screws that hold the KG 102A in its tray. Refer to Figure 211.

(c) Disconnect all connectors to the unit.

(d) Remove the unit.

(2) **KG 102A Remote Gyro Installation** - (Refer to Figure 211.)

(a) Connect the wiring and connectors to the KG 102A.

(b) Place the KG 102A in its place in the instrument shelf.

(c) Install screws to connect the KG 102A to the tray and torque screws 10 to 12 in.-lbs.

(d) Install the access panel.

(e) Replace the carpet.

(3) **KMT 112 Slaving Transmitter Removal** - (Refer to Figure 211.)

(a) Remove the access panel on the underside of the left outboard wing. Refer to Figure 209.

(b) Disconnect the electrical connections.

(c) Unfasten the nuts, and remove the unit.

(4) **KMT 112 Slaving Transmitter Installation - Use King Installation Manual**

(a) Install the unit with the arrow on top facing forward, using the same hardware or new brass hardware only.

(b) Connect the electrical connectors.

(c) At a compass rose, adjust the unit per manufacturer's manual.

(d) Finger-tighten the nuts. Do not over-tighten.

(e) Install the access panel.

(5) **KI 525A Removal** - (Refer to Figure 212) - The KI 525A is located in main plate of the instrument panel.

(a) Unfasten the main plate, and carefully slide it out far enough to work on the back side of the KI 525A.

(b) Unfasten the mounting screws that surround the KI 525A, and slide it forward while slipping the mounting plate off the back of the unit.

(c) Disconnect connectors installed into the back of the KI 525A.

(6) **KI 525A Installation** - (Refer to Figure 212.)

(a) Slide the unit into the instrument panel, and slide the mounting bracket over the back of the unit.

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(b) Install the connectors into the back of the KI 525A.

(c) Fasten the mounting screws through the front of the instrument panel and then into the mounting plate. Torque screws 6 to 8 in.-lbs.

(7) **KA 51B Slaving Control Removal** - (Refer to Figure 210) - The KA 51B is located in the engine instrument plate of the instrument panel.

(a) Unfasten the engine instrument plate, and carefully slide it out far enough to work on the backside of the KA 51B.

(b) Unfasten the mounting screws from the KA 51B, and slide it forward.

(c) Disconnect the connectors installed into the back of the KA 51B.

(8) **KA 51B Slaving Control Installation** - (Refer to Figure 210.)

(a) Follow the KA 51B Slaving Control Removal procedure above in reverse order.

(b) At a compass rose, adjust the unit per the manufacturer's manual.

9. Terminal Junction Rail

A. **Terminal Junction Rail Removal** - (Refer to Figure 213.) - The terminal junction rail is located on the vertical brace on the backside of the instrument panel.

(1) Remove the glare shield per instructions in Chapter 25-11-00.

(2) Remove the two 6-32 flat head screws, self-locking nuts, and washers for S/N 42003 to 42023. Remove the 6-32 flat head screws for S/N 42024 and On.

(3) Remove the terminal junction rail.

B. **Terminal Junction Rail Installation** - Installation is a direct reversal of removal. Torque screws 10 to 12 in.-lbs.

10. Antenna Splitter

A. **Basic or Avidyne Option**

(1) **Garmin AT Avionics**

(a) **Antenna Splitter Removal** - (Refer to Figure 214, Sheet 1.) - The antenna splitter is located on the inside right side of the radio rack. An antenna splitter may not be present depending upon the avionics package installed in the airplane.

1 Remove the center console per instructions in Chapter 25-12-00 to gain access to the radio rack.

2 Remove two truss head 6-32 screws, self-locking nuts, and washers.

3 Remove the antenna splitter.

(b) **Antenna Splitter (Garmin AT Avionics) Installation** - Installation is a direct reversal of removal. Torque screws snug.

(2) **Garmin Avionics**

(a) **Antenna Splitter Removal** - (Refer to Figure 214, Sheet 2) - The antenna splitter is located on the inside right side of the radio rack.

1 Remove the center console per instructions in Chapter 25-12-00 to gain access to the radio rack.

2 Remove the two truss head 10-32 screws, self-locking nuts, and washers.

3 Remove the antenna splitter.

(b) **Antenna Splitter (Garmin Avionics) Installation** - Installation is a direct reversal of removal. Torque screws snug.

B. **Garmin G1000 Option**

(1) **Antenna Splitter Removal** - The antenna splitter is located under the instrument panel to the right of the tower on the co-pilot's side. It is positioned under the LRU rack to the far right.

(a) Access the antenna splitter from the underside of the instrument panel.

(b) Remove the two pan head 10-32 screws.

(c) Remove the antenna splitter.

(2) **Antenna Splitter Installation** - Installation is a direct reversal of removal. Torque screws snug.

11. Garmin Avionics Cooling Fan

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A. **Basic or Avidyne Option**

(1) **Cooling Fan Removal** - The Garmin avionics cooling fan is located on the inside of the radio rack. Refer to Figure 215.

(a) Remove the center console side panels per instructions in Chapter 25 to gain access to the radio rack.

(b) Loosen the tie wrap where the 5/8-inch hoses are connected to the fan enclosure and disconnect the hoses.

(c) Remove three pan head 6-32 screws, self-locking nuts, and washers.

(d) Unplug and remove the fan.

(2) **Cooling Fan Installation** - Installation is a direct reversal of removal. Torque screws 10 to 12 in.-lbs.

B. **Garmin G1000 Option** - There are three cooling fans. One in the LRU rack behind the PFD, one in the LRU rack behind the MFD, and one mounted under the instrument panel on the pilot's side.

(1) **Cooling Fan Removal**

(a) Remove the PFD or MFD, as required, per Chapter 34-60-00. Access the third fan from the underside of the instrument panel.

(b) Remove six Philips screws attaching the fan and its mounting bracket to the LRU rack or three pan head 6-32 screws attaching the fan to the underside of the instrument panel, as applicable.

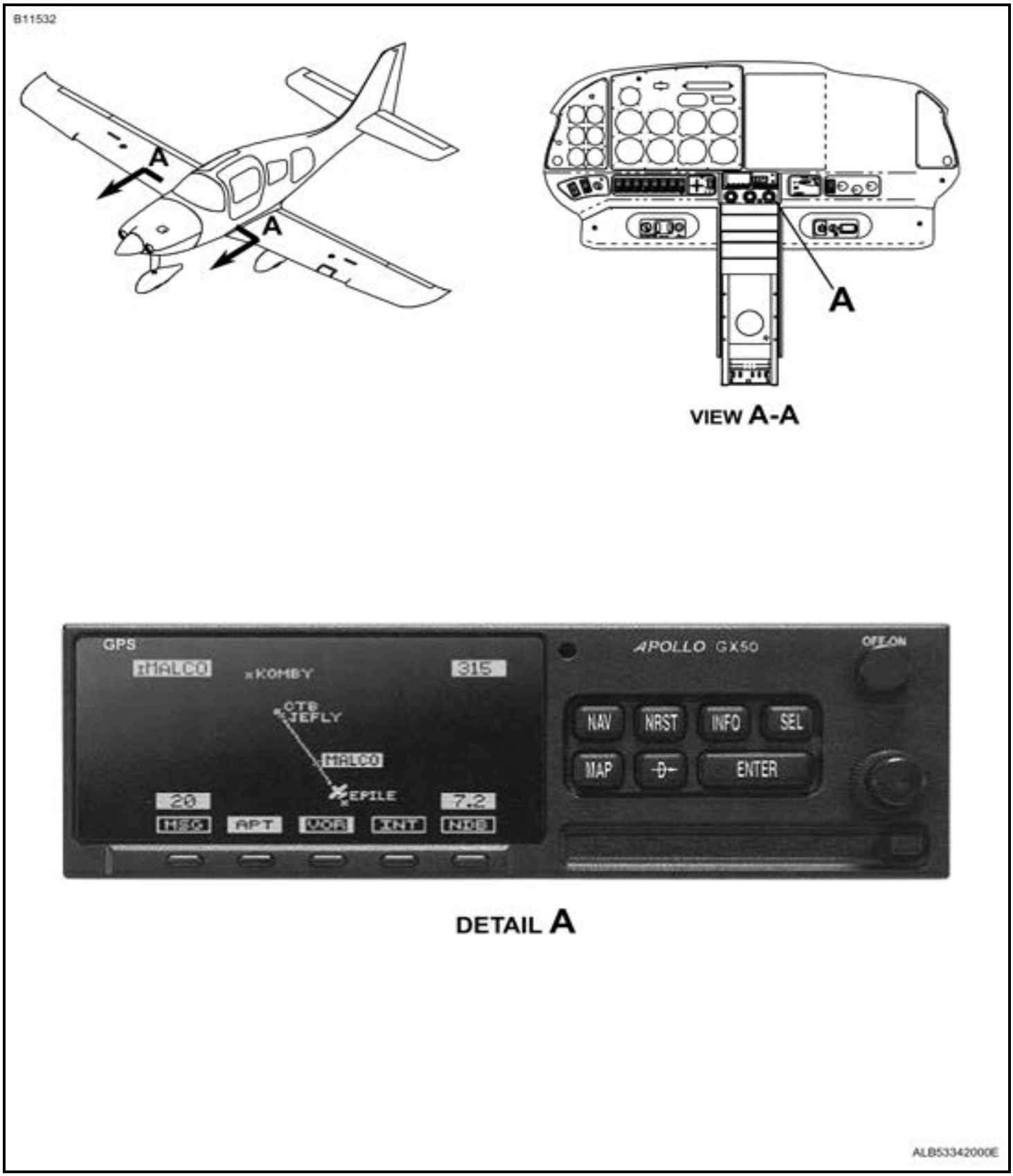
(c) Gently remove the fan.

(d) The fans have been removed from the LRU rack, they may be removed from their brackets by removing three Phillips screws.

(2) **Cooling Fan Installation** - Installation is a direct reversal of removal. Torque screws snug.

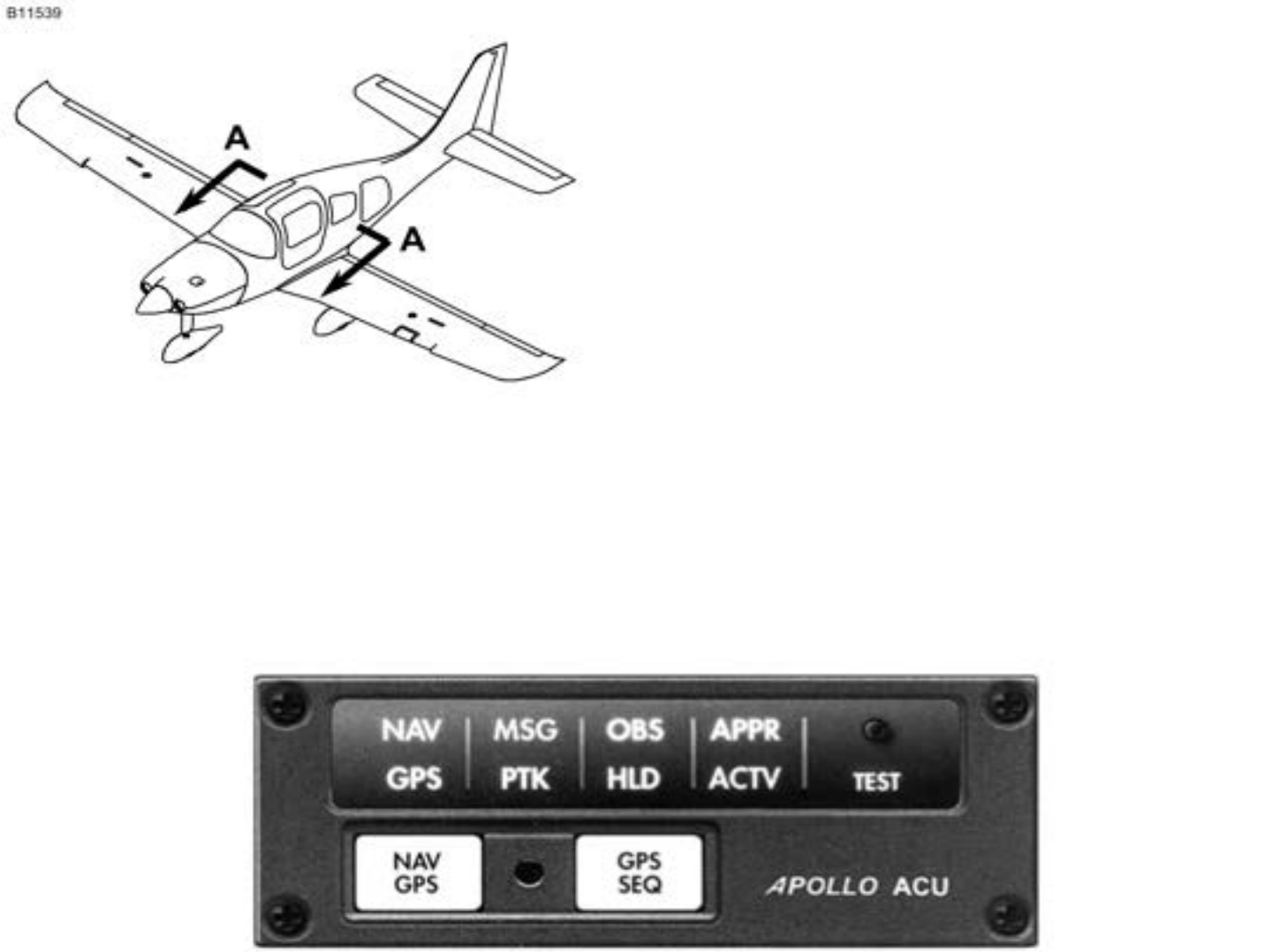
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Figure 201 : Sheet 1 : Apollo GPS GX50 System



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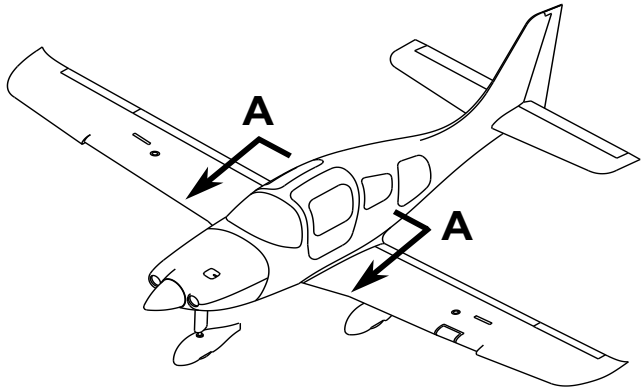
Figure 202 : Sheet 1 : Apollo ACU

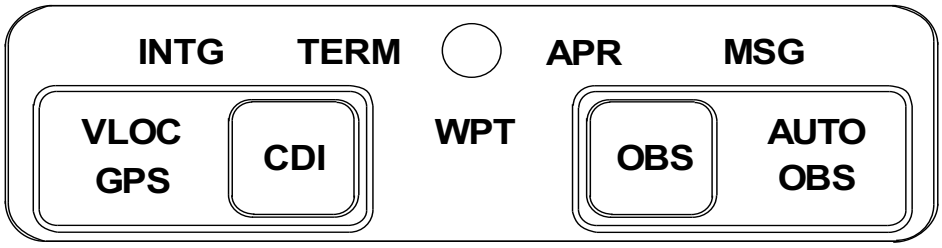


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Figure 203 : Sheet 1 : MD41-1464A ACU

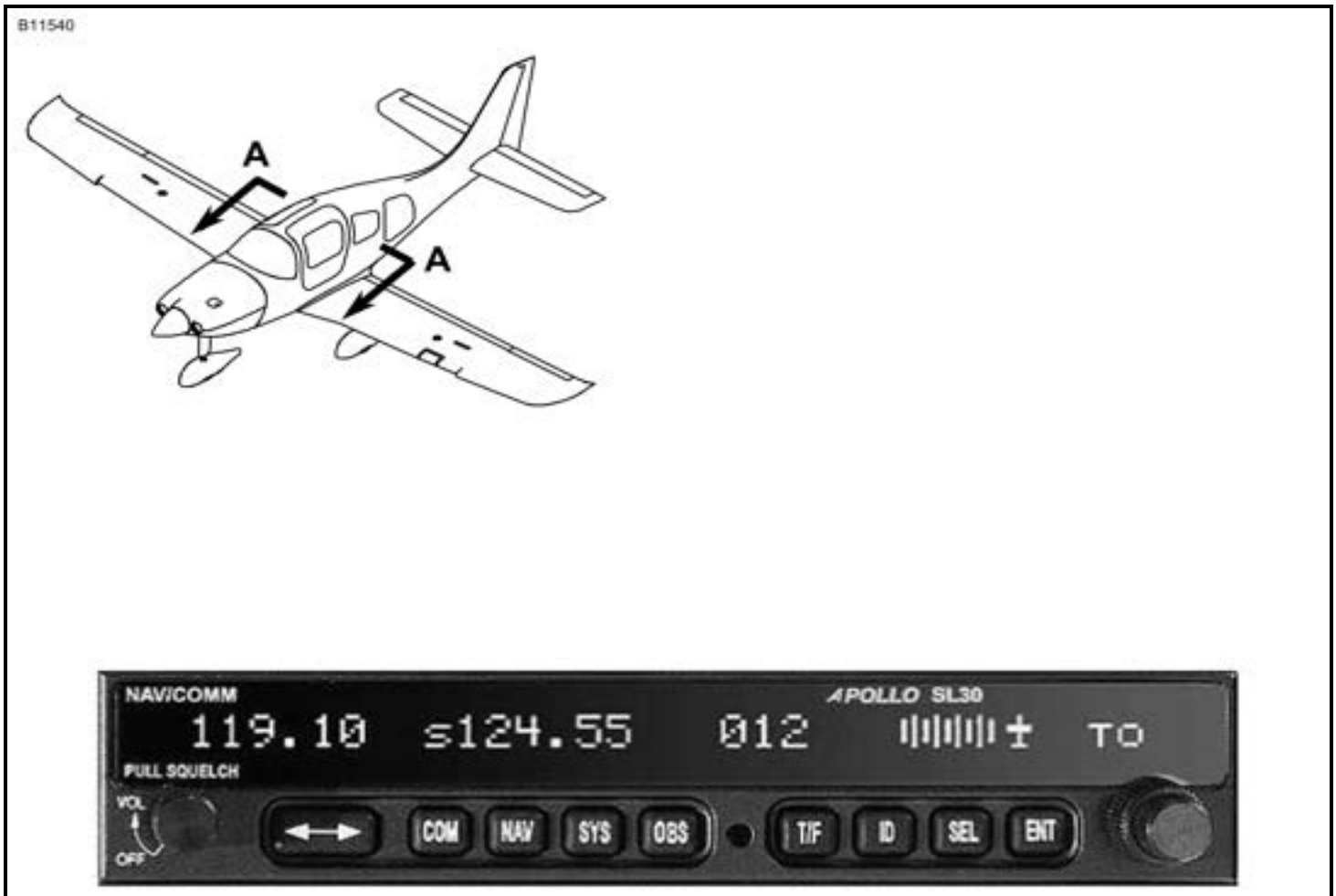
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Figure 204 : Sheet 1 : Garmin AT Apollo SL30 Nav/Comm

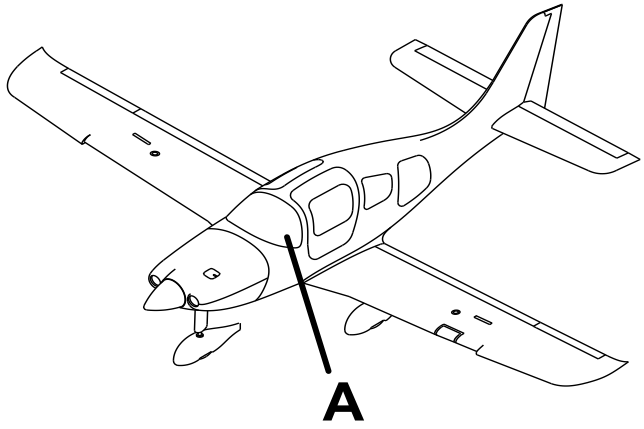


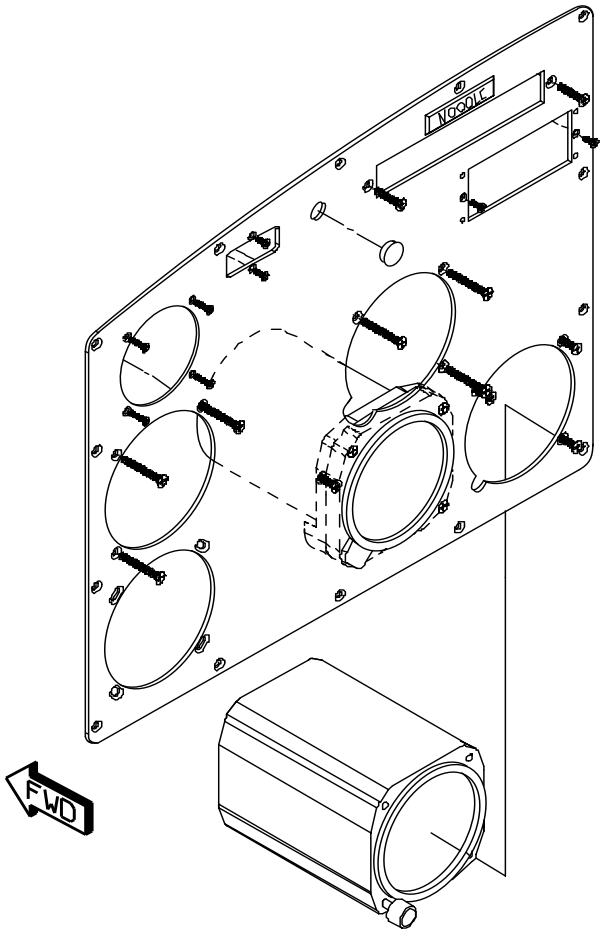
VIEW A-A

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Figure 205 : Sheet 1 : Navigation Indicator Location

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DETAIL A

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Figure 206 : Sheet 1 : GARMIN AT Apollo SL70 Transponder

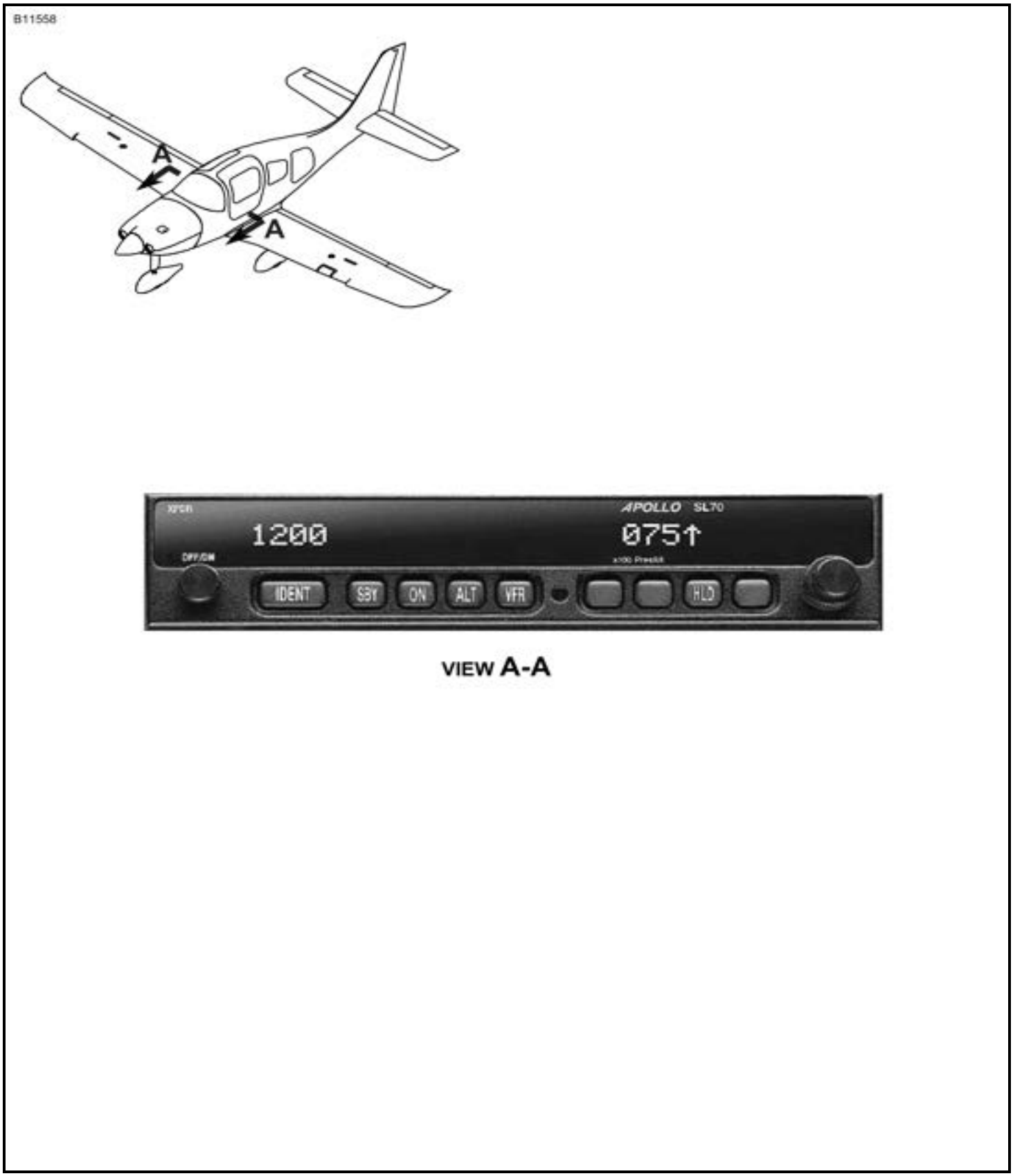
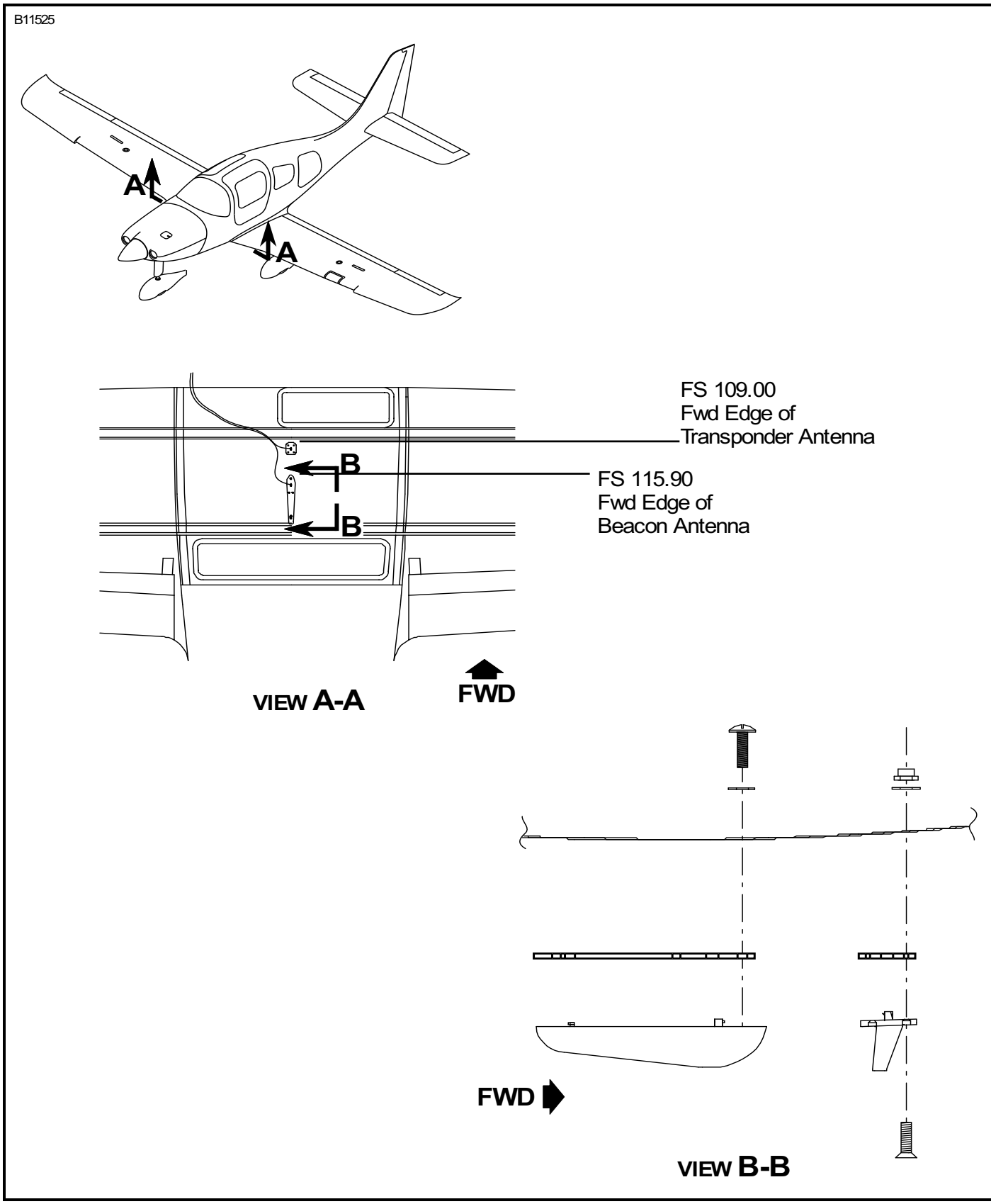


Figure 207 : Sheet 1 : Garmin GTX 327 and GTX 330 Transponder



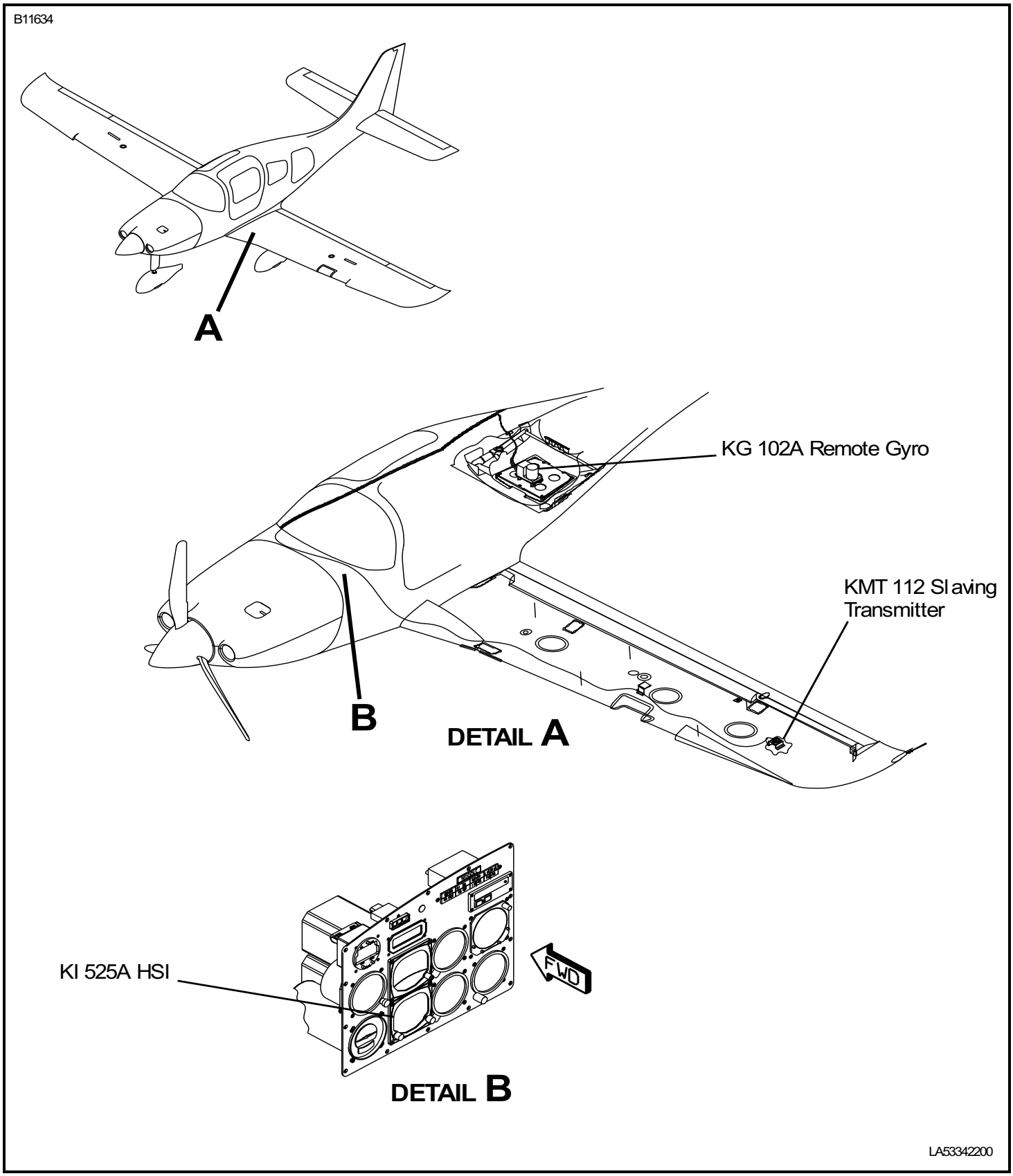
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Figure 208 : Sheet 1 : Transponder Antenna Location



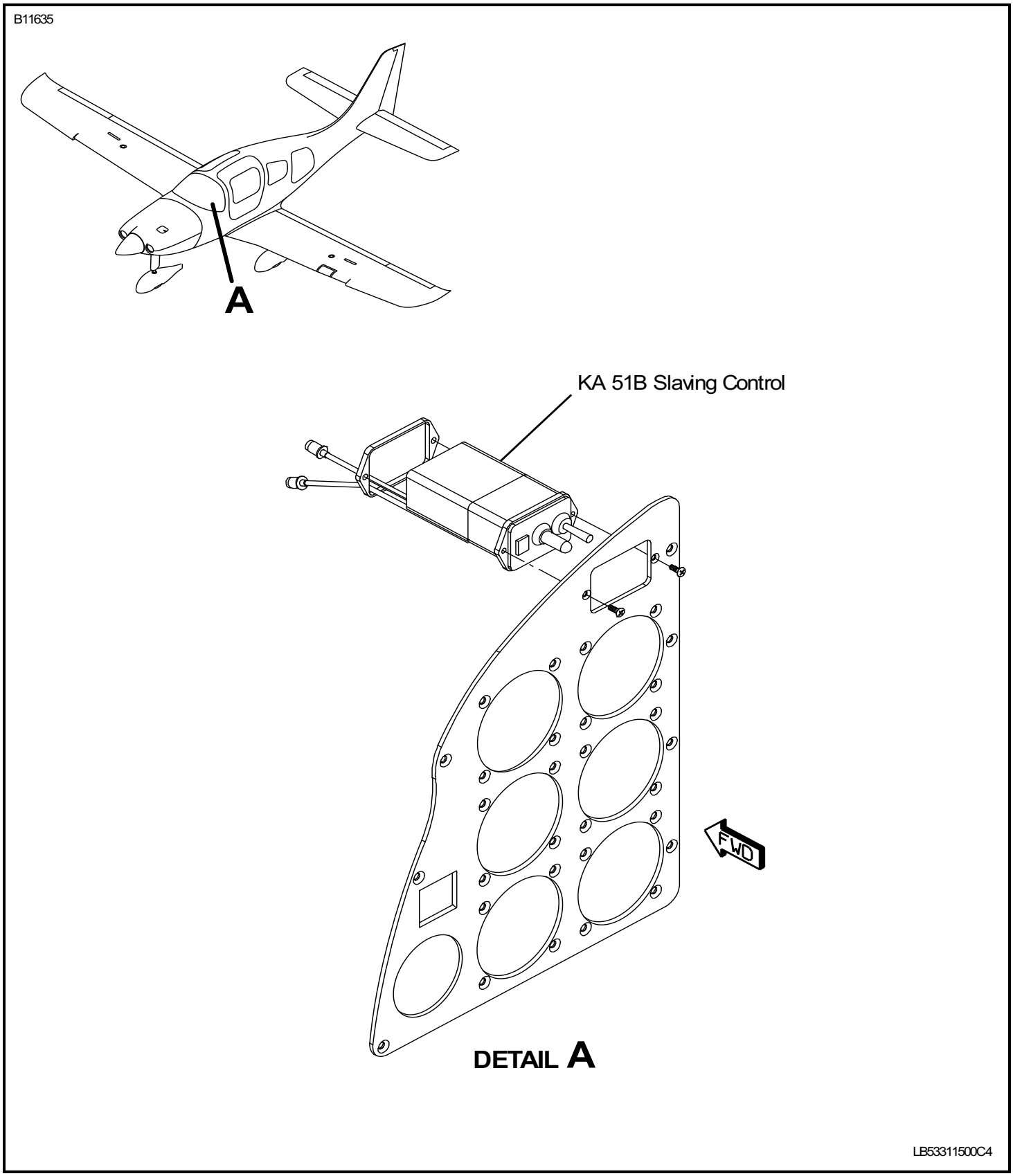
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Figure 209 : Sheet 1 : HSI - Remote Slaving System



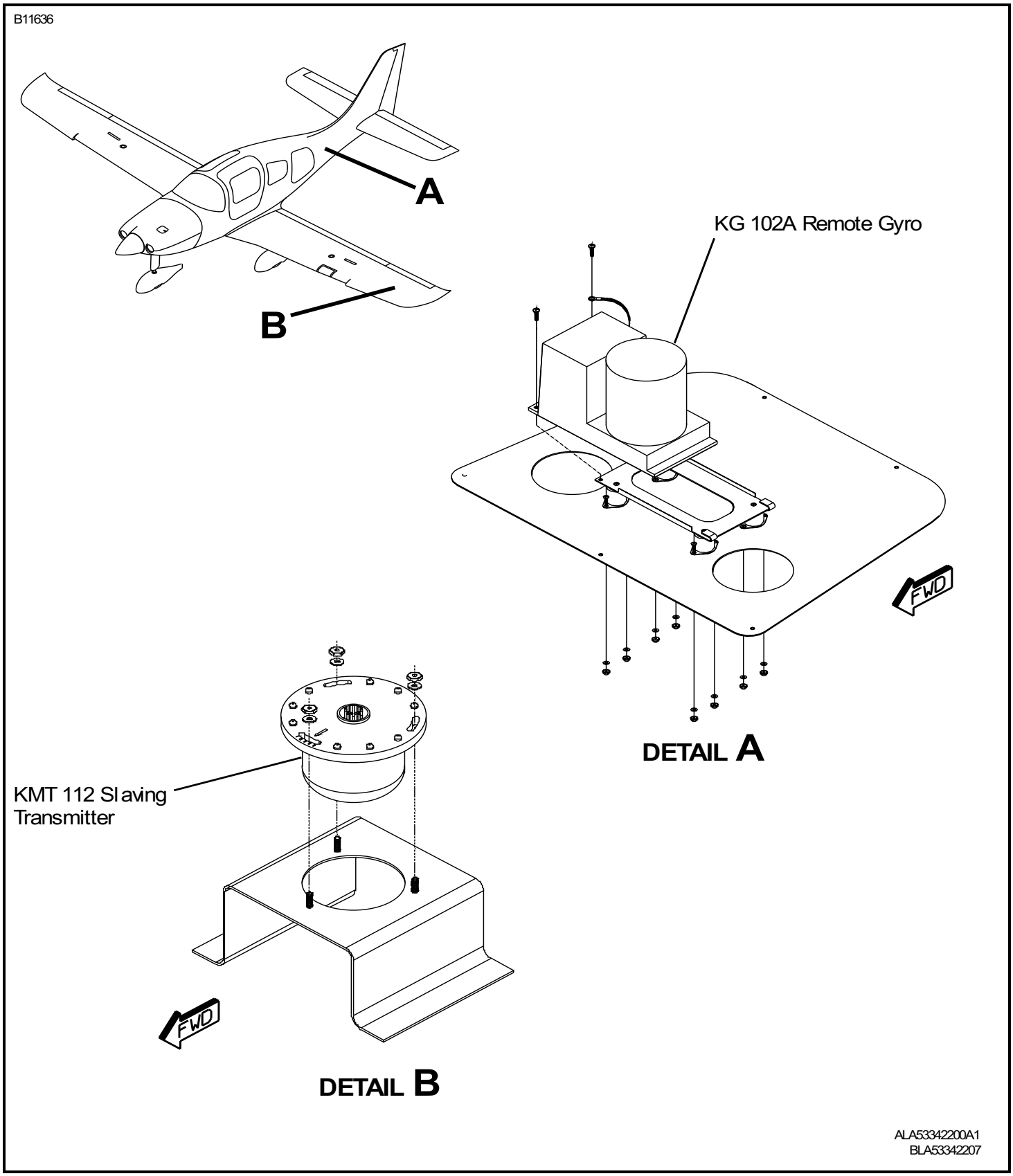
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Figure 210 : Sheet 1 : KA 51B Slaving Control Location



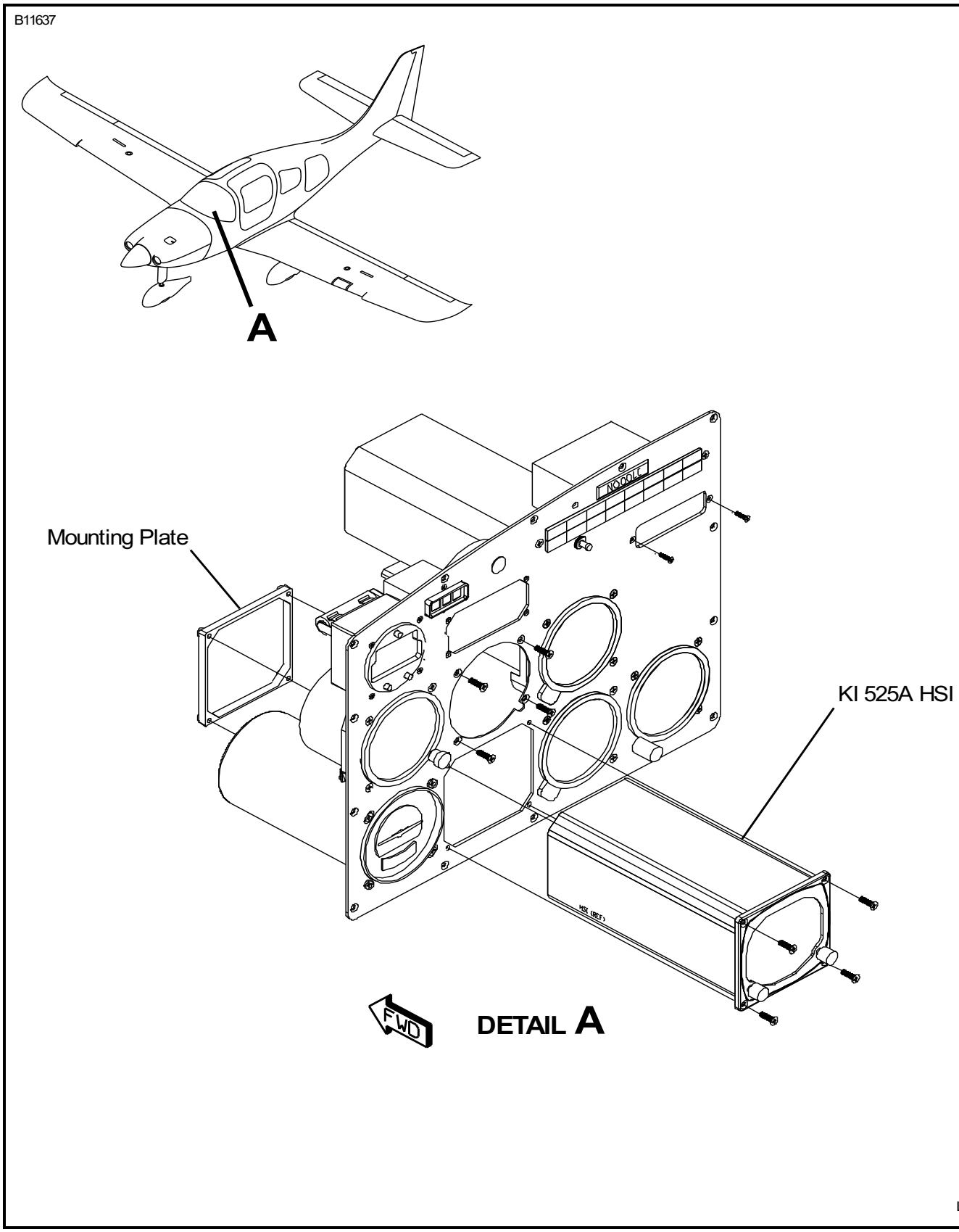
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Figure 211 : Sheet 1 : HSI Remote System Installation



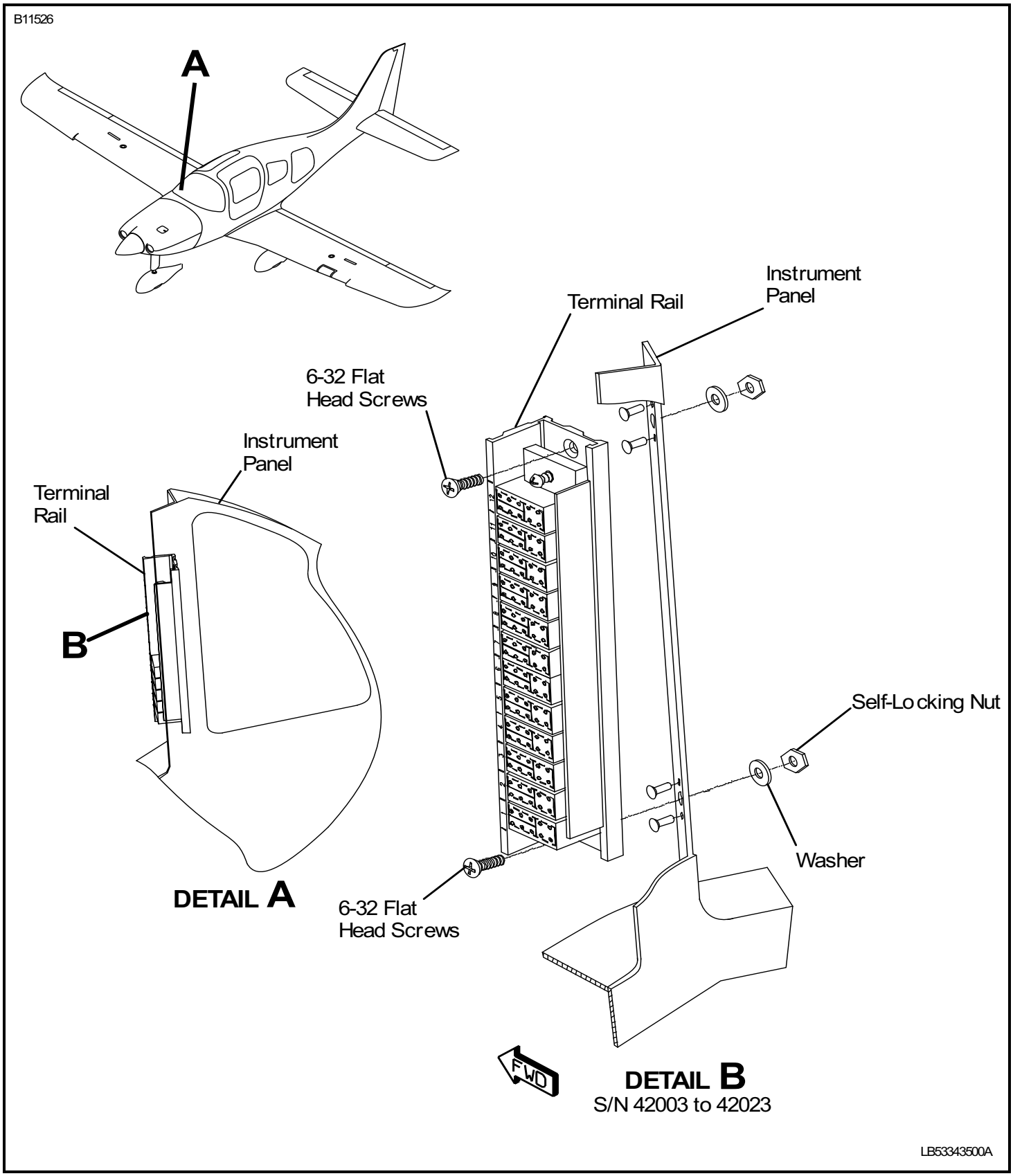
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Figure 212 : Sheet 1 : KI 525A HSI Installation



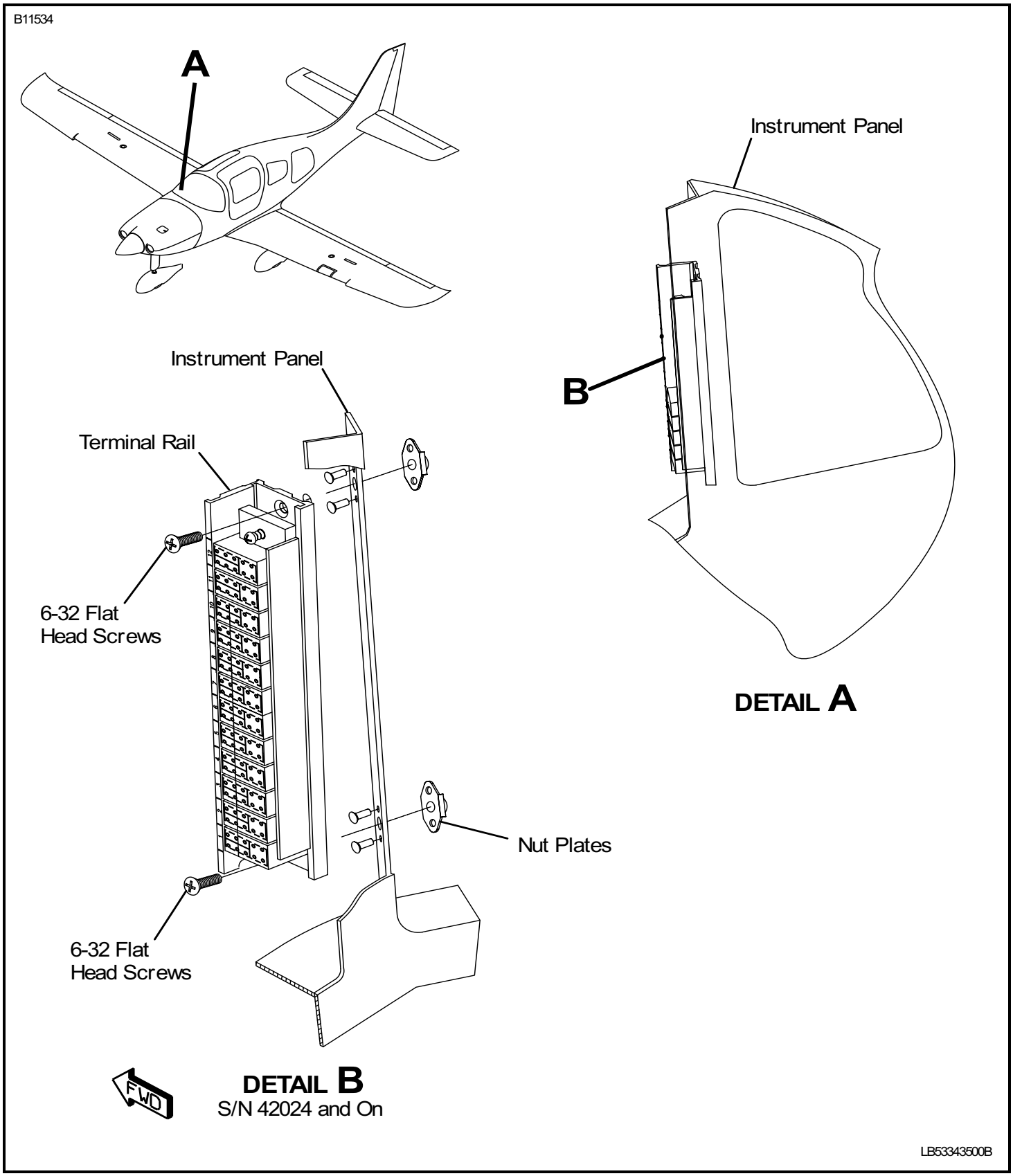
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Figure 213 : Sheet 1 : Terminal Junction Rail



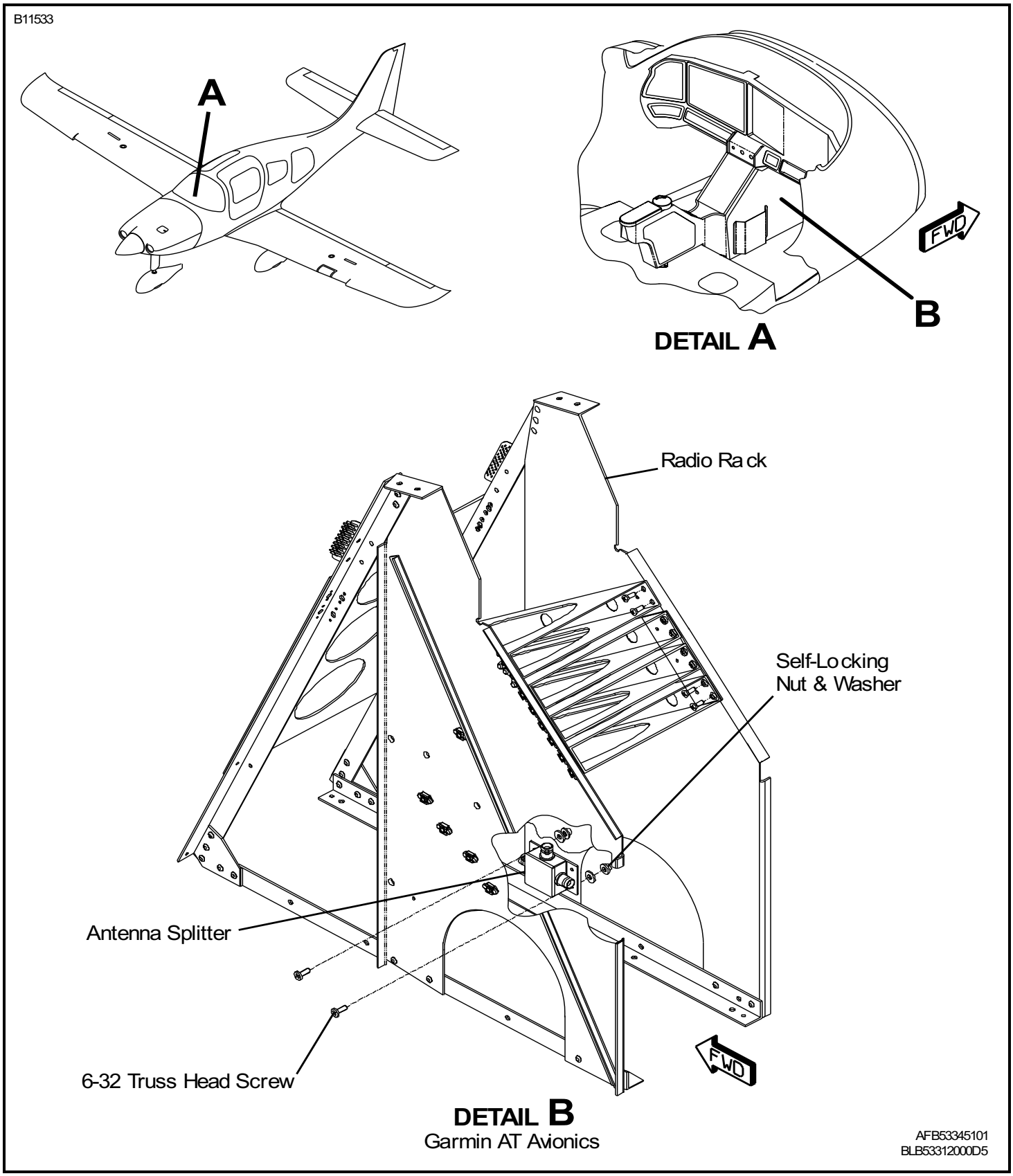
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Figure 213 : Sheet 2 : Terminal Junction Rail



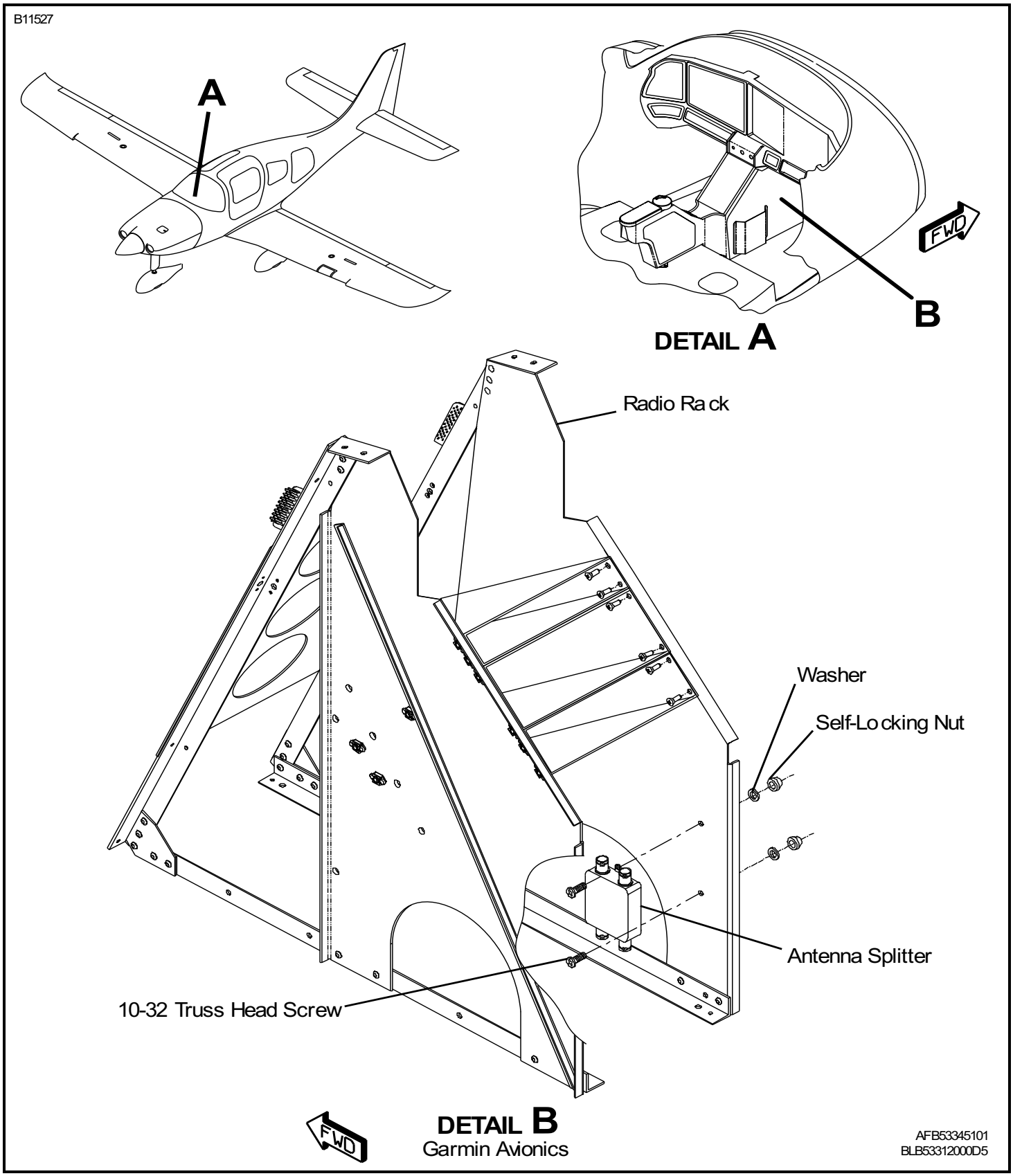
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Figure 214 : Sheet 1 : Antenna Splitter



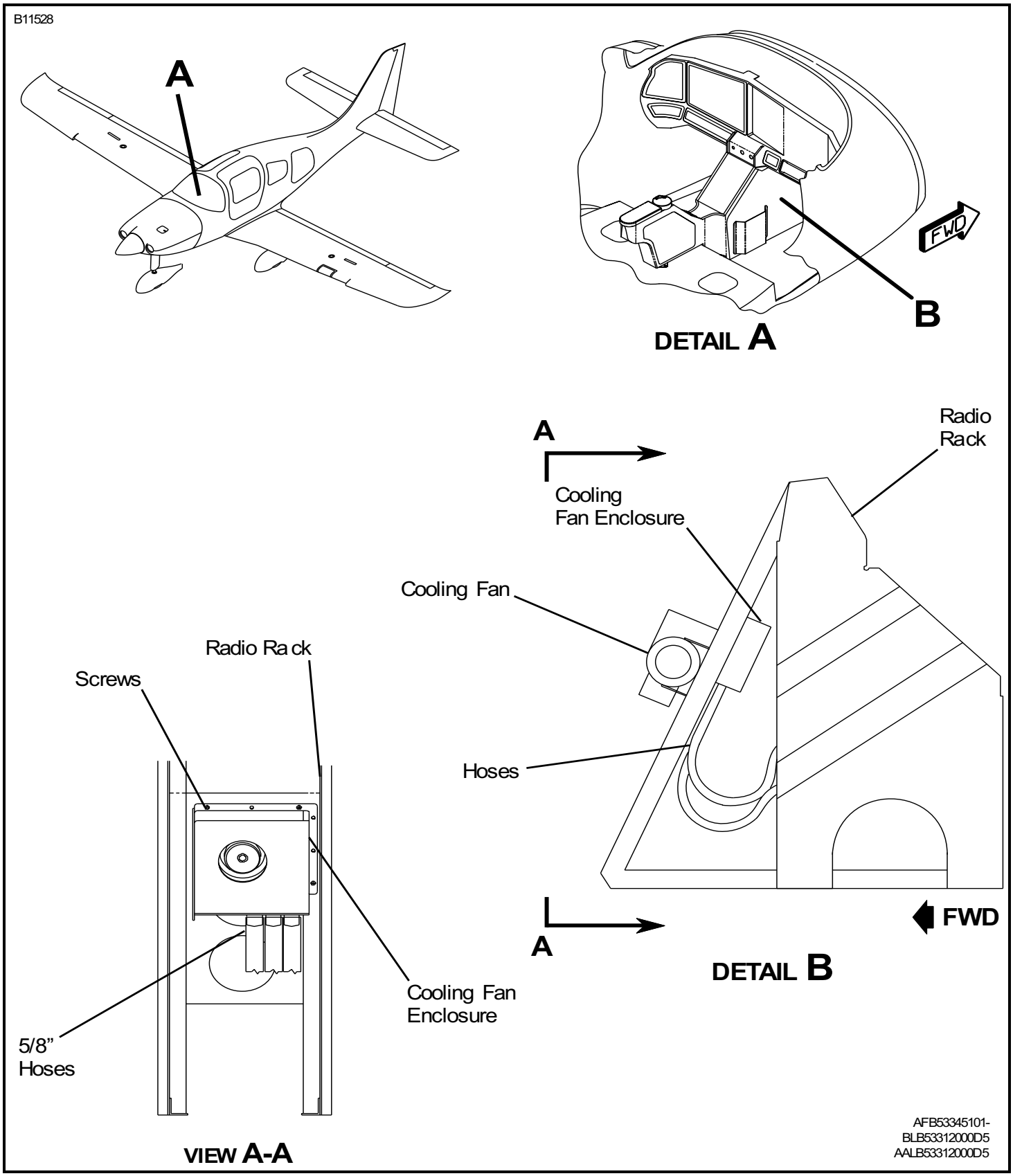
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Figure 214 : Sheet 2 : Antenna Splitter



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Figure 215 : Sheet 1 : Garmin Avionics Cooling Fan



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PRIMARY AND MULTI-FUNCTION FLIGHT DISPLAY SYSTEMS - MAINTENANCE PRACTICES

1. Avidyne Entegra Primary Flight Display (PFD)

A. **Primary Flight Display Removal** - The PFD is located in the left side of the instrument panel. Refer to Figure 201.

(1) Ensure both the master and avionics switches are in the off position.

(2) Loosen the four 5/64 inch Allen head screw fasteners at the four corners of the PFD. The fasteners will not come out of the PFD, only loosen.

(3) Slide the unit out of the instrument panel.

(4) Disconnect the pitot static hoses by pressing the quick release disconnect button on the connector of each hose.

(5) Loosen the two jackscrews on each cable connector attached to the back of the PFD and disconnect the cables.

B. **Primary Flight Display Installation**

(1) Remove the Avidyne EX5000 Multi-Function Display as described below to allow easier access to the PFD.

(2) Attach the cable connectors to the back of the PFD and tighten the jackscrews.

(3) Attach the pitot static hose connectors.

(4) Hold the PFD with one hand at the top front and the other hand supporting the bottom. Slide the PFD into the instrument panel while lifting up on the rear edge and simultaneously pushing the upper edge under the glare shield.

(5) Ensure the PFD mating clip slides over the horizontal support bracket located approximately 8 inches behind the instrument panel.

(6) Ensure that the PFD cables and pitot static hoses are routed away from any moving parts behind the instrument panel.

(7) Secure the PFD to the instrument panel with the four corner-mounted 5/64 inch Allen head screw fasteners.

(8) Perform a functional check in accordance with instructions in the System Setup and Checkout section.

C. **Troubleshooting Information**

NOTE: For troubleshooting instructions recommending wiring checks, refer to the *FlightMax Entegra Primary Flight Display Installation Manual*, P/N 600-00080-000 and/or the *FlightMax Entegra Series PFD Pilot's Guide*, P/N 600-00096-000.

Table 201. Primary Flight Display Troubleshooting

Component	Trouble	Probable Cause	Remedy
Attitude Display	Pitch ladder and horizon removed and replaced with red X.	PFD AHRS failure	Replace PFD
Altitude, Airspeed, Vertical Velocity Display	Instruments removed and replaced with red X.	PFD Air Data Computer failure	Replace PFD
Heading	Heading indication removed and replaced with red X.	Break in wire between PFD and Magnetometer	Test and repair wiring.
		Magnetometer failed	Replace Magnetometer/OAT Assembly
		PFD RS-422 or Magnetometer output power failure	Replace PFD
	Excessive heading error	Magnetometer not properly calibrated	Calibrate Magnetometer
Alignment	PFD fails to align after power up.	Aircraft was moved during power up alignment.	Cycle PFD power and remain stationary during power up alignment.
		PFD ADAHRS failure	Replace PFD

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OAT	Display indicates red dashes.	OAT probe failed	Replace Magnetometer/OAT Assembly
	Excessive temperature error.	OAT probe failed	Replace Magnetometer/OAT Assembly
True Airspeed	Display indicates red dashes.	OAT probe failed	Replace Magnetometer/OAT Assembly

Table 202. Primary Flight Display Troubleshooting: Wind Vector - VHF/GPS NAV 1&2

Component	Trouble	Probable Cause	Remedy
Wind Vector	Display indicates dashes.	Winds aloft have been calculated as light and variable or less than 5 knots.	Behavior by design, no action required.
		PFD not receiving ground speed. Break in wire between PFD and VHF/GPS NAV 1 & VHF/GPS NAV 2.	Test and repair wiring
		Aircraft has zero ground speed.	Behavior by design, no action required.
VHF/GPS NAV 1	PFD Primary Nav, Bearing, or Auxiliary indication displays dashes when GPS 1 or VLOC 1 selected.	VHF/GPS NAV 1 not configured properly	Configure unit
		Break in wire between PFD and VHF/GPS NAV 1	Test and repair wiring
		VHF/GPS NAV 1 ARINC 429 Output failed	Replace VHF/GPS NAV 1
		PFD ARINC 429 input failed	Replace PFD
	GPS1 does not slave to PFD course pointer in OBS mode.	Break in ARINC 429 wire between PFD and GPS NAV 1	Test and repair wire
		ARINC 429 Receive Input failed	Replace VHF/GPS NAV 1
		ARINC 429 Transmit output failed	Replace PFD
VHF/GPS NAV 2	PFD Primary Nav, Bearing, or Auxiliary indication displays dashes when GPS 2 or VLOC 2 selected.	VHF/GPS NAV 2 not configured properly	Configure unit
		Break in wire between PFD and VHF/GPS NAV 2	Test and repair wiring
		VHF/GPS NAV 2 ARINC 429 Output failed	Replace VHF/GPS NAV 2
		PFD ARINC 429 input failed	Replace PFD

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	GPS2 does not slave to PFD course pointer in OBS mode.	Break in ARINC 429 wire between PFD and GPS NAV 2	Test and repair wire
		ARINC 429 Receive Input failed	Replace VHF/GPS NAV 2
		ARINC 429 Transmit output failed	Replace PFD

Table 203. Primary Flight Display Troubleshooting: Autopilot Integration (Part 2)

Component	Trouble	Probable Cause	Remedy
Autopilot Integration	Airspeed, Heading, or Altitude bugs do not “fill” but autopilot modes still engage.	Break in Annunciator wires between PFD and AP Computer.	Test and repair wiring
		Autopilot Computer interface failure.	Replace unit
		PFD interface failure.	Replace unit
	Autopilot annunciators on the PFD (if equipped) do not illuminate or illuminate erratically.	Break in Annunciator wires between PFD and AP Computer.	Test and repair wiring
		Autopilot Computer interface failure.	Replace unit
		PFD interface failure	Replace unit
	Autopilot VS mode does not track PFD Vertical Speed setting, but autopilot mode engages.	Break in VS Command wires between PFD and Autopilot Computer.	Test and repair wiring
	Autopilot HDG mode does not track PFD Heading setting, but autopilot mode engages.	Break in Heading Datum wire between PFD and Autopilot Computer.	Test and repair wiring
	Autopilot HDG mode does not track PFD Heading setting, but autopilot mode engages (con.)	Autopilot Computer interface failure.	Replace unit
		PFD interface failure	Replace unit
		PFD heading bug out of alignment.	Perform PFD heading bug calibration.
	Autopilot NAV mode does not track PFD Nav Course Pointer setting or CDI.	Break in Course Datum and Cross track wires between PFD and Autopilot Computer.	Test and repair wiring
		Autopilot Computer interface failure.	Replace unit
		PFD interface failure	Replace unit
		Autopilot out of alignment	Perform autopilot alignment in NAV mode
	Autopilot NAV	Break in CDI Flag wire between PFD and Autopilot Computer	Test and repair wiring

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	Autopilot NAV annunciator flashes and FAIL indication appears.	Autopilot Computer interface failure	Replace unit
		VHF/GPS NAV 1 or 2 Nav data invalid.	Refer to VHF/GPS NAV 1 & 2 Troubleshooting procedure.
		PFD interface failure	Replace unit

Table 204. Primary Flight Display Troubleshooting: Autopilot Integration (Part 2)

Component	Trouble	Probable Cause	Remedy
Autopilot Integration (continued)	Autopilot APR mode does not track PFD CDI or VDI.	Break in CDI and VDI wires between PFD and Autopilot Computer.	Test and repair wiring
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
	Autopilot APR mode fails to engage.	Break in Glideslope flag or LOC Switch wires between PFD and Autopilot Computer.	Test and repair wiring
		Autopilot Computer interface failure.	Replace unit
		PFD interface failure	Replace unit
	Autopilot Computer display indicates vertical speed value.	Break in ALT Select wire between FD and Autopilot.	Test and repair wiring
		Autopilot Computer interface Autopilot Integration failure.	Replace unit
		PFD interface failure	Replace unit
	Autopilot GPSS mode fails to engage in flight (GPSS mode will not engage during a ground test)	Break in PFD ARINC 429 Transmit or Autopilot Relay Control wire	Test and repair wiring
		Autopilot Relay failure	Replace relay
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
	Autopilot GPSS only follows GPS 1 source	Break in Autopilot Relay Control wire	Test and repair wire
		Autopilot Relay failure	Replace relay
		PFD interface failure	Replace unit
	Flight Director command bars (if equipped) do not align to attitude reference symbol or behave erratically. (Test alignment on in level flight in smooth air with HDG and ALT modes engaged)	PFD out of alignment	Perform PFD heading bug calibration
		Break in pitch / roll command signal wires between PFD and Autopilot computer.	Test and repair wiring
		Autopilot computer interface failure	Replace unit

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		PFD interface failure	Replace unit

D. **System Setup and Checkout** - Any time the Avidyne PFD or Magnetometer/OAT Sensor Assembly is replaced with another unit, these system setup procedures should be performed to assure that the unit is properly configured for the installation. Turn on the PFD by applying power to the aircraft. Allow the PFD to initialize. This will take approximately 3 minutes.

(1) **System Setup Page Access** - Apply power to all the sensors that interface with the PFD, including: the GPS and the autopilot. System Setup Page will allow you to configure the PFD. The System Setup Page can only be accessed by pressing the line select keys in a specific manner.The System Setup Page is accessible as follows:

(a) Turn on the PFD by applying power to the aircraft via the appropriate aircraft power switches (Avionics Master, etc.) switches.

(b) The system will begin its normal start up sequence.

(c) Simultaneously press and hold the top left (L1), and 3rd down from the top left (L3) Line Select keys for until the count down timer in the lower left corner of the display indicates zero seconds.

(d) At the end of a displayed countdown clock adjacent to the L4 line select key, the System Setup Page will appear.

(2) **Aircraft Model Setup** - Perform the following steps to configure the PFD for the correct aircraft model.

(a) Enter the System Setup Page.

(b) Press LSK R4, labeled A/C Model, as required until the appropriate aircraft model is displayed in the button label. Note that the model field in the middle of the page reflects the model selected. For those aircraft models that have a + suffix displayed in the LSK R4 button label, the OAT will be suppressed from the display.

(3) **Barometric Units Setting** - Perform the following steps to configure the PFD for desired barometric units.

(a) Enter the System Setup Page.

(b) Press LSK R3, labeled Perform A/P Cal, to enter the Autopilot Calibration page.

(c) Press LSK R4, labeled Baro Unit, to cycle through barometric pressure units (inHg, hPa, Mb).

(d) Once the desired units have been selected and appear in the button label, press the Back to PFD button (LSK L4).

(4) **GNS-430 Nav/Com Setup**

(a) PFD Unit – The ARINC 429 ports on the PFD come hard-wired and pre-assigned. GNS-430 Unit no. 1 (top unit in radio rack) must be connected to ARINC port 1 (GPS output line) and port 2 (VLOC output line). GNS-430 Unit no. 2 (if installed) must be connected to ARINC port 3 (GPS output line) and port 4 (VLOC output line). Use the line select keys R1, R2 to configure the PFD in the following manner:

1 Line Select Key (LSK) R1: GPS1 Port: ARINC 1;

2 LSK R1: GPS1 Port: NONE, if GNS-430 No.1 not installed;

3 LSK R2: GPS2 Port: ARINC 3, if GNS-430 No.2 installed;

4 LSK R2: GPS2 Port: NONE, if GNS-430 No. 2 not installed.

Once complete, verify the displayed configuration in the sensor suite section of the System Setup page accurately reflects actual configuration.

(b) GNS-430 Unit(s)

NOTE: Refer to the GNS 430 users guide for GNS 430 detailed operations.

1 Place the GNS 430 unit(s) into Maintenance Mode via the following steps:

a Ensure GNS 430 unit(s) off

b Depress the ENT button as the power to the GNS 430 unit(s) is applied. Continue to depress the ENTR button until the GNS 430 self -test is completed.

c Depress the ENT button twice to display the MAIN ARINC 429 CONFIG page. Configure the GNS 430 unit(s) to have the following mandatory settings:

Table 205. Main ARINC 429 Configuration Page

Field	Garmin 1	Garmin 2 (if installed)

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IN 1	Low, Sandel EHSI	Low, Sandel EHSI
IN 2	Low, OFF	Low, OFF
Out	Low, GAMA 429 Graphics	Low, GAMA 429 Graphics
SDI	LNAV1	LNAV2

d Turn the right inner knob until the MAIN RS232 CONFIG page is displayed.

e Configure the GNS 430 unit(s) to have the following mandatory settings:

Table 206. Main RS232 Configuration Page

Field	GNS-430 No. 1		GNS-430 No. 2 (if installed)	
	Input	Output	Input	Output
CHNL 1	Off (1)	Aviation	Off (1)	Aviation
CHNL 2	Off (1)	Off (1)	Off (1)	Off (1)
CHNL 3	Crossfill	Crossfill	Off (1)	Off (1)
CHNL 4	Off (1)	Off (1)	Off (1)	Off (1)

(1) Unless other equipment is installed (i.e., Stormscope)

f Turn the right inner knob until the MAIN INPUTS 2 page is displayed. Verify CDI = GPS

g Turn the right inner knob until the MAIN DISCRETE OUTPUTS' page is displayed. Verify Discrete Toggle = APR

h Turn the right inner knob until the VOR/LOC/GS ARINC 429 CONFIG page is displayed

i Configure the GNS 430 unit(s) to have the following mandatory settings:

Table 207. VOR/LOC/GS ARINC 429 Configuration Page

Field	GNS-430 No. 1	GNS-430 No. 2 (if installed)
Speed	Low - Low	Low - Low
IN 2	VOR/ILS1	VOR/ILS2
DME Mode	Directed Freq. 1	Directed Freq. 2

(5) **Autopilot Setup**

(a) **PFD Unit** - Perform the following steps when replacing a PFD.

1 Enter the System Setup Page (refer to the System Setup and Checkout section) and press the line select key R3, labeled Perform A/P Cal and depress line select key R1 A/P Type to select the appropriate autopilot.

2 Select line select key L4 labeled Back to PFD and reenter the System Setup Page. Verify the displayed autopilot configuration in the sensor suite section of the System Setup page accurately reflects actual configuration.

(b) **S-Tec System 55x Autopilot Unit** - Perform the following steps when replacing the PFD or Autopilot unit. If the autopilot installed in the aircraft is an S-Tec System 55x, ensure it is configured as a KCS-55 compatible unit. Refer to the S-Tec System 55x Installation Manual for details.

NOTE: The following steps are to be performed in smooth air.

1 While NAV (without GPSS) mode is engaged on the System 55x, using a GNS 430 GPS source, insert an appropriately sized screwdriver into the slotted hole in the 55X faceplate.

2 Perform a NAV course alignment by slowly adjusting the screw until the aircraft closely tracks the GPS course. Allow the aircraft to stabilize between adjustments. This may take several minutes and require several adjustments.

3 Engage the 55x in HDG mode and allow aircraft to stabilize. The heading bug may be offset from the lubber line by several degrees.

4 Turn OFF all GPS/NAV systems.

5 Enter the PFD System Setup Page (refer to the System Setup and Checkout section).

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6 Turn ON all GPS/NAV systems.

7 Select Perform A/P Cal on the PFD.

8 Select Sync HDG. Autopilot should immediately command the aircraft to turn toward the heading bug.

9 Select Back to PFD.

10 Verify HDG mode accurately tracks heading bug.

11 Engage NAV (without GPSS) mode on the autopilot.

12 Verify NAV mode accurately tracks flight plan (Flight plan or waypoint will need to be entered).

13 A/P calibration is complete.

NOTE: If autopilot does not track the heading bug or NAV mode correctly repeat steps in this procedure.

(6) **PFD System Calibrations** - The PFD and Magnetometer require calibrations upon initial installation, when replaced, or bi-annual periodic inspection. The following two aircraft-level calibration procedures are listed below:

Table 208. Aircraft-Level Calibration Procedures

Calibration	When Performed	Notes	Periodic Calibration
IRU	Initial installation and replacement.	Must be performed first upon initial installation with Magnetometer.	None
Magnetometer	Initial installation, replacement, PFD replacement, or bi-annual inspection.	Must be performed after IRU calibration upon initial installation or PFD replacement.	Every 24 months

NOTE: The IRU calibration must be completed prior to the Magnetometer calibration.

(a) IRU Calibration ProcedureApproximate duration: 15 MinutesRequired Equipment: Digital Level, Resolution: 0.25 degrees, Accuracy: +/- 0.25 degreesRecommended Personnel: One avionics technicianProcedure:

1 Avionics power applied.

2 Allow the PFD to align (approximately 3 minutes) until the ADAHRS countdown timer expires and the Gyro Warm-up box is removed.

3 Wait in the aligned state for an additional 10 minutes. Enter System Setup mode on the PFD by following steps in the System Setup and Checkout section.

4 Press the IRU Install line select key (L2). The IRU Installation page is displayed.

5 Place a level 2X4-like piece of wood between the two door jams, or place an inclinometer on top (or bottom) of the PFD Bezel, aligned to display roll angle.

6 Place a digital level on the baggage compartment doorsill of the airplane.

7 Follow the directions on the IRU installation page.

NOTE: The only two axis that are changed via installation calibration input are the pitch axis and roll axis.

NOTE: The pitch attitude and roll attitude as determined by the digital level should each have a resolution of at least 1/4 degree.

8 Press the Install IRU line select key (R4) when proper pitch and roll is displayed.

9 After the display indicates the IRU calibration is done, cycle power on the PFD in order for the IRU calibration to take effect.

(b) Magnetometer Calibration Procedure

1 Approximate duration:

a 20 minutes (assuming airplane is free to rotate 360 degrees without magnetic disturbances).

2 Required equipment:

a Installed PFD

b Installed magnetometer

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c Compass: Resolution: 1 degree, Accuracy: +/- 1degree

3 Recommended personnel:

a 2 avionics technicians (one in the cockpit to push required bezel buttons, one on the wing to push the aircraft to new headings)

4 Procedure:

NOTE: Ensure the IRU installation calibration procedure has been completed.

a Airplane doors closed.

b Flaps in retracted position.

c Engine off.

d Airplane in level flight attitude.

e Install the compass onto the top surface of the left tail surface.

NOTE: A compass rose with 30 degree positioning increments may be used in lieu of a tail-mounted compass.

f Position lubber line to be parallel with longitudinal axis of the airplane.

g Position compass to be approximately 1/2 chord length back from leading edge of tail and 43 inches outboard from fuselage.

h Master switch in ON position. (Consider attaching an external power cart until ready for step K.) PFD may already be on and aligned from IRU install. If so, skip to step K. All other aircraft equipment shall be operating.

i Allow the PFD to align (approximately 3 minutes) until the ADAHRS countdown timer expires and is removed from the HSI.

j Wait in the aligned state for an additional 10 minutes.

k Enter System Setup mode on the PFD by following steps in the System Setup and Checkout section.

l Press the Perform Mag Cal line select key (L1). The Magnetometer Calibration page is displayed.

m Align centerline of airplane on magnetic north heading as indicated by tail-mounted compass (or compass rose) by pushing the aircraft as required. Ensure this is being done clear of magnetic disturbances in the pavement or immediate vicinity.

n Follow the directions on the Magnetometer Calibration page (press the Calibrate Heading button and wait until DONE is displayed).

o Align centerline to airplane on consecutive 30 degree headings as indicated by the tail-mounted compass (or compass rose) by pushing the aircraft as required and continue following the directions of the Magnetometer Calibration page until complete.

NOTE: Pressing (R3) "Redo Previous" provides undo capability.

NOTE: Aborting the calibration process (by exiting the Magnetometer Calibration Page or shutting off power to the PFD) prior to completion will result in the loss of stored calibration parameters. Restarting the calibration process from the beginning would be required.

p Press the Back to PFD line select key (L4) when complete.

q Align the centerline of the airplane with consecutive 90 degree headings as indicated on the tail-mounted magnetic compass (or compass rose) and verify the accuracy of the PFD heading display is within +/- 0.4 degrees of the tail mounted magnetic compass (or compass rose).

r Calibration complete.

(7) Post Installation Check

(a) Lighting Check - Rotate the INST instrument lighting control rheostat through its full range of motion. Ensure the PFD LED lights around all bezel keys and knobs come on and match the MFD LED lighting.

(b) GPS NAV-COMM Check

NOTE: This procedure assumes both GNS-430s are installed.

1 PFD Nav Button - Ensure GPS1 displayed. Observe that the GNS-430 No. 1 CDI button label displays

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GPS.

2 GNS-430 No. 1 CDI button - Press once and observe CDI button displays VLOC label. Observe the PFD Nav button changes to VLOC1.

3 GNS-430 No.1 CDI button - Press once and observe the CDI button label displays GPS. Note that the PFD Nav button changes to GPS1.

4 PFD Nav button - Press once and observe VLOC1 is displayed. Observe the GNS-430 No. 1 CDI button changes to VLOC.

5 PFD Nav button - Press once and note GPS2 is displayed. Observe the GNS- 430 no. 2 CDI button displays GPS.

6 GNS-430 No. 2 CDI button - Press once and observe the CDI button label displays VLOC. Observe the PFD Nav button changes to VLOC2.

7 GNS-430 No. 2 CDI button - Press once and observe the CDI button label displays GPS. Observe the PFD Nav button changes to GPS2.

8 PFD Nav button - Press once and observe VLOC2 is displayed. Observe the GNS-430 No. 2 CDI button changes to VLOC.

NOTE: Two-way communication between the PFD and both GNS-430 units is demonstrated by the procedure above.

(8) Pitot-Static Leak Check - Perform a pitot-static leak check, as described in the Flight Environment Data/Pitot-Static System section, anytime time the pitot-static ports are disconnected from the PFD.

E. **Primary Flight Display Testing and Operation** - Refer to the Avidyne FlightMax Entegra Primary Flight Display Installation Manual and FlightMax Entegra Series Pilot's Guide for testing and operation procedures.

2. Avidyne Entegra EX5000 Multi-Function Display (MFD)

A. **Multi-Function Display Removal** - The MFD is located in the right side of the instrument panel. Refer to Figure 201 or Figure 202.

(1) Ensure both the master and avionics switches are in the off position.

(2) Loosen the 5/64" allen head screw fasteners at the four corners of the MFD. The Fasteners should not come out of the MFD, only loosen.

(3) Slide the unit out of the instrument panel.

(4) Loosen the two jackscrews on the cable connector located on the back of the MFD and disconnect the cable.

B. **Multi-Function Display Installation** - Perform the steps for removal above in reverse order. During installation ensure that the MFD cable is routed away from any moving parts behind the instrument panel.

C. **Troubleshooting Information** - The Avidyne MFD incorporates a message bar located at the bottom of the display. Messages are generated by the system and displayed on the message bar and are helpful in troubleshooting system problems. The following figures present the messages that are generated by each application.

Table 209. GPS/FMS Messages

Message	Meaning
Nav Source: Not Responding	No RS 232 or ARINC 429 GPS data is being received. Check wiring and Comm port setting.
Nav Source: Position Data Not Valid	Data is being received from the external GPS. However, insufficient information is available to determine position. Aircraft could be in the hanger or the GPS may not have determined its "fix" or location.
Nav Source: Position Data Valid	The MFD is receiving valid position data from the GPS.
Nav Source: Data Format Error	Data is being received, however the MFD does not recognize the data as the format selected. May indicate a baud rate or receiver type error.

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Nav Source: No Port selected	The COMM port setting in the GPS setup dialog is set to NONE. To correct this, choose the port to which the GPS is connected, typically COM2.
Nav Source: Reconnecting	This message is displayed when data between the MFD and the GPS is being synchronized.
Nav Source: Can't Open Port	Another device is configured for the same port. Check the Setup page for all devices. Typically the GPS is configured for RS 232 Port 2 or ARINC 429 port 1. A second GPS can only be configured for ARINC port 2.
Heading Data is NOT VALID	Heading data is no longer available from the GPS. Will only appear if the GPS is being used as the heading source.
Heading Data is VALID	Heading data has been restored.

Table 210. Traffic Messages

Message	Meaning
Traffic Sensor is OFF (TAS)	Traffic data is not being received.
Traffic Sensor is in Stand-By (TAS)	The Skywatch sensor has been placed in Standby mode.
Traffic Sensor is in Self-Test (TAS)	The Skywatch sensor has been placed in Self-Test mode.
Traffic Sensor is Operating Normally (TAS)	Verification that Traffic data is valid.
TCAD Altitude Unavailable (TCAD)	Occurs when altitude data has been lost from the TCAD sensor.

Table 211. Lightning Messages

Message	Meaning
Lightning Sensor is Operating Normally	Verification that strike data to the MFD is valid.
Lightning Sensor in Demo Mode	Demo mode has been selected as the operating mode from the Lightning Setup Page.
Lightning Sensor is in Noise-Monitor Mode	Noise-Monitor mode has been selected as the operating mode from the Lightning Setup Page.
Lightning Sensor in Test Mode	Test mode has been selected as the operating mode from the Lightning Setup Page.
Lightning Sensor ERROR	The sensor system has reported an error that may mean current data is incomplete or erroneous. The error may clear.
Lightning Sensor has FAILED	The sensor system has reported an error that may mean current data is incomplete or erroneous. The error will not clear until power is removed from and reapplied to the sensor system.
Lightning Sensor is OFF	Communication of strike data from the Lightning sensor to the MFD has been lost.
Lightning Ahead	The Lightning Ahead option has been checked on the Lightning Setup page. Displayed when a Lightning Ahead condition exists.
Lightning Heading Source Failed	Heading data is no longer available from the WX-500. Strike data may still be valid. Will only appear if the WX-500 is being used as the heading source.
Lightning Heading Source OK	Heading data has been restored.
Stuck mic-PLEASE CHECK	Check the comm transmitters for indication of a stuck mike.

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Verify Antenna Location (run Setup)	There may be an inconsistency between the antenna location jumper setting and the software configuration. This message should only appear during installation.

Table 212. Datalink Messages

Message	Meaning/Action
Datalink Sensor Data Is Invalid	The EX5000 has received unreadable satellite data.
Datalink Sensor Configuration Error	The EX5000 is improperly configured for datalink.
	• Requires factory servicing. Refer to Factory Service Policies section of this manual.
Datalink Sensor Is NOT Communicating	The EX5000 is experiencing a communication failure with the internal Satellite transceiver.
	• Requires factory servicing. Refer to Factory Service Policies section of this manual.

D. **System Setup and Checkout** - Any time the Avidyne MFD is removed and sent to the factory for service, or is replaced with another unit, these system setup procedures should be performed to assure that the unit is properly configured for the installation. This section should also be referenced if it is suspected that the unit may be mis-configured. Turn on the MFD by applying power to the aircraft battery and avionics bus master switches. The green bezel key backlighting should illuminate to full intensity when the Instrument Lights dimmer control is turned fully counterclockwise. As the dimmer control is turned clockwise, the bezel key lighting should dim fully, and then increase in intensity.

(1) **Maintenance Mode Access** - Upon initial system power-up, the MFD will execute a start-up sequence. During this time, the screen will initially display the FlightMax logo, followed by a blank screen until a blue screen with the text *Initializing-Please Wait...* is displayed. The startup sequence is finished when the text *Press any bezel key to Continue...* is displayed. The initialization screen displays the loaded software version, along with NavData and Obstacle database expiration dates. Apply power to all sensors that interface with the MFD such as GPS navigator, lightning sensor, and traffic sensor. The MFD configuration is accomplished in the setup pages that are accessed through the Maintenance Mode, which is accessed as follows:

(a) Upon completion of system initialization, rotate the page knob to the *Setup* page.

(b) Simultaneously press and hold the top and middle left-hand bezel keys for at least 5 seconds until the maintenance page appears.

(2) **GPS Interface, Setup and Checkout** - The GPS is interfaced to the MFD by either of two methods selected at the time of initial installation: ARINC 429 or RS 232. The ARINC 429 interconnect requires that a shielded pair connect the MFD to the GPS navigator. The RS 232 interconnect requires that a single shielded wire connect the MFD to the GPS navigator. Refer to the FlightMax EX5000/EX3000 Installation Manual, P/N 600-00073 and/or the FlightMax EX5000 Pilot's Guide, P/N 600- 00102-000 for wiring diagram information. If it is not known which interface is utilized in a particular installation, select ARINC 429. If this selection is not correct, the message *Nav Source: Not Responding* will be displayed on the Map page after system start-up, in which case the RS 232 selection should be configured on the MFD.

NOTE: ARINC 429 is the recommended configuration.

From the Maintenance Page, press the GPS *Setup* bezel key to select the GPS Setup page.

(a) ARINC 429 Setup - When the ARINC 429 interconnect is used, GPS 1 and GPS 2 are normally connected to the MFD. To select ARINC 429, highlight each configuration line and select the appropriate MFD settings as shown in Table 213.

Table 213. MFD Settings for ARINC 429

Configuration Item	MFD Setting	Comment
Receiver	GAMA 429 Format	
Port	ARINC 1	Select ARINC 2 for GPS 2
Speed	Low	

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Press the *Save* bezel key. The MFD must be restarted for the settings to take effect. Assure the Garmin GNS 400 series navigator is configured appropriately. To display the ARINC 429 Configuration Page, apply power to the GPS while pressing and holding the ENT key. Release the ENT key when the display activates. After the data base pages, the first page displayed is the MAIN ARINC 429 CONFIG page. While in the configuration mode, pages can be selected by ensuring the cursor is off and rotating the small right knob. To change data on the displayed configuration page, press the small right knob (CRSR) to turn on the cursor. Turn the large right knob to change between data fields. Turn the large or small right knob to change a field that the cursor is on. Once the desired selection is made, press the ENT key to accept the entry. The Garmin GNS 400 series navigator should be configured as shown in Table 214.

Table 214. Garmin GNS 400 Navigator Configuration

Configuration Item	Speed	Data
IN 1	As Required	As Required
IN 2	As Required	As Required
OUT	Low	GAMA 429 Graphics w/Int

(b) RS 232 Setup - When the RS 232 interconnect is used, only GPS 1 is connected to the MFD. To select RS 232, highlight each configuration line and select the appropriate MFD settings as shown in Table 215.

Table 215. MFD Settings for RS 232

Configuration Item	MFD Setting
Receiver	King/Aviation Format
Port	COM 2
Baud	9600

Press the *Save* bezel key. The MFD must be restarted for the settings to take effect. Assure that the Garmin GNS 400 series navigator is configured appropriately. Enter the configuration mode as described in the GPS Interface, Setup and Checkout section and display the MAIN RS232 CONFIG Page. Verify that the Channel 1 RS 232 output is set to *Aviation*.

(c) GPS Checkout - After verifying the MFD GPS settings, the aircraft should be located where the GPS receiver can receive valid GPS satellite signals and generate a valid position. After the MFD has been restarted, the message *Nav Source: Position Data Valid* and the current aircraft position should be depicted on the MFD Map display. Refer to the Troubleshooting Information section for resolution of any error messages.

(3) **Lightning Sensor Interface and Setup** - The WX-500 Stormscope lightning sensor interface is optional. Interconnect details are provided in the FlightMax EX5000/EX3000 Installation Manual, P/N 600-00073 and/or the FlightMax EX5000 Pilot's Guide, P/N 600-00102-000. The MFD communicates to and from the WX-500 using RS 232 communications.

(a) Lightning Sensor Setup - To configure Lightning, access Maintenance Mode and select *Lightning Setup*. Highlight each configuration item and select the appropriate MFD settings as shown in Table 216.

Table 216. MFD Settings for Lightning Setup

Configuration Item	MFD Setting	Comment
Sensor	WX-500	
Operating Mode	Weather	
Port	COM1	
Stab Type	Synchro	Enter GPS if no other stabilization is provided.
Enable Stabilization?	Check	Leave unchecked if no stabilization is provided.
Enable Lightning Ahead Warning?	Check	Leave unchecked if not desired.
Antenna on Top?	Check	Leave unchecked if WX-500 antenna is mounted on bottom.

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Press the *Save* bezel key. The MFD must be restarted for the settings to take effect.

(b) Lightning Sensor Checkout - After the MFD has been restarted, the message *Lightning Sensor is Operating Normally* should be displayed in the message bar. If not, refer to the Troubleshooting Information section for resolution of any error messages. Select the Map display and select *Heading/Track Up*. Select the *Strike* Lightning Mode. From the MFD *Setup* page, push the *Lightning Test* bezel key. Verify that a test strike is depicted on the Map display at approximately 18 NM at 45 degrees relative bearing to the aircraft nose.Refer to the Troubleshooting Information section for resolution of any error messages.

(4) **Traffic Sensor Interface, Setup and Checkout** - The SkyWatch traffic sensor interface is optional. Interconnect details are provided in the FlightMax EX5000/EX3000 Installation Manual, P/N 600-00073 and/or the FlightMax EX5000 Pilot's Guide, P/N 600-00102-000. The MFD uses an ARINC 429 data connection to receive data from the SkyWatch TRC 497. The MFD provides mode control of the SkyWatch through discrete signals to the Softkey 1 and Softkey 4 inputs on the TRC 497.

(a) Traffic Setup - To configure SkyWatch, access Maintenance Mode as described in the Maintenance Mode Access section and select *Traffic Setup*. Highlight each configuration item and select the appropriate MFD settings as shown in Table 217.

Table 217. MFD Settings for Traffic Setup

Configuration Item	MFD Setting	Comment
Sensor	TAS	
Port	ARINC3	
External Controller?	Unchecked	
Heading Available?	Unchecked	Check if heading data is supplied to the SkyWatch and is available to the MFD.

Press the *Save* bezel key. The MFD must be restarted for the settings to take effect.

(b) Traffic Sensor Checkout - After the MFD has been restarted, the message *Traffic Sensor is Operating Normally* should be displayed in the message bar. If not, refer to the Troubleshooting Information section, for resolution of any error messages.Select the Map display and select *Heading/Track Up*. Select the *Normal* Traffic mode. From the MFD *Setup* page, push the *Traffic Test* bezel key. Verify that the traffic self-test pattern is depicted on the Map display as shown in Table 218.

Table 218. Traffic Self-Test Pattern

Symbol Type	Relative Bearing (degrees)	Approx Range	Relative Altitude
Other	30°	3.5 nm	1000 Feet Below
Other	330°	3.6 nm	1000 Feet Above
TA	270°	2 nm	200 Feet Below

The message *Traffic Sensor is in Self-Test* should appear in the message bar.Refer to the Troubleshooting Information section for resolution of any error messages.

(5) **Map Setup** - The Avidyne MFD has the capability to overlay Traffic and Lightning information onto the Map display. To utilize this feature, Map requires Heading or Track information.The MFD can receive HEADING data from one of following sources:

(a) Garmin 400 series GPS (via ARINC 429).

(b) BF Goodrich Stormscope (via RS 232). This is the most common method.

(c) BF Goodrich Skywatch (via ARINC 429) with software level 1.6 or higher.

(d) TRACK is the actual direction the aircraft is moving relative to the earth's surface, and is obtained from the GPS.

To configure the Map heading source, access Maintenance Mode as described in the Maintenance Mode Access section and select *Map Setup*. Use the right hand knob to cycle through the selections: None (Use GPS Track), FMS/GPS, Stormscope, and SkyWatch. When the appropriate setting is selected, press the *Save* bezel key. The MFD must be restarted for the setting to take effect.

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(6) **Datalink Setup** - The EX5000 is pre-configured for datalink operation. Accessing the Datalink Info page from the System Info utility, which is entered from the Maintenance Mode menu, gives details of the configuration. Verify the presence of SC Serial Number and version information on this screen.

(7) **Datalink Reception Confirmation** - ORCOMM satellite network transmits very low power VHF signals that the EX5000 must receive. If the antenna is not properly installed or if there is excessive electromagnetic interference (such as by nearby radio transmitter or inadequately grounded electronics) the system will not achieve successful reception. The following steps may be taken to assess system performance.

(a) Bring the aircraft to an area that has as few obstacles to line-of-sight viewing to the southern horizon as possible and orient the aircraft heading south.

(b) From Datalink Info table monitor for up to five minutes under the following conditions; other avionics off, avionics on, engines at idle, engines at takeoff power. With a satellite in view the signal strength and quality should peak above the following values.

1 "Signal strength" is greater than -118 dBm

2 "Estimated SNR (EbNo)" is greater than 10 dB

3 Segment errors: less than 10 percent

If the above reception levels are not achieved the following cause and corrective actions may apply.

1 The antenna field of view is obstructed. Try moving or rotating the aircraft. (if rotation works the antenna location on the aircraft may not be optimal.)

2 There is a local source of electromagnetic interference.

3 Try shutting off any nearby noise sources (such as VHF radios, alternators and magnetos). Once identified, have an authorized service center check wiring, shield terminations and component chassis grounding of the offending system.

4 Relocated the aircraft away from potential nearby sources.

5 Check electrical connections to ensure there is no improper grounding.

6 There is poor satellite coverage. Try again after fifteen minutes.

7 The antenna and cable are not properly installed.

8 The antenna may not be properly connected to the ground plane.

9 The cable may not be properly connected to the antenna or EX5000.

(c) Consult www.avidyne.com for any additional guidance.

E. **Maintenance** - Other than the periodic servicing task discussed below, maintenance of the MFD is based on condition only; no other periodic maintenance is required.

(1) **Recommended periodic scheduled servicing tasks** - The MFD contains a 3-volt lithium battery that maintains CMOS memory on an internal processor board and should be replaced after 10 years of service, or when CMOS memory fails to retain configuration data, whichever occurs first. The Avidyne MFD must be returned to an authorized FAA repair station to perform this maintenance function. Failure of the CMOS memory is indicated by the message "WARNING: CMOS battery failure. Check database expiration date. Press any bezel key to continue" on a blue background prior to system boot up. After system boot up, the MFD will function normally, but will not have retained the system date and time. If the system is interfaced to the GNS 430 through ARINC 429, it will acquire current date and time from the GNS 430 when valid satellite data is received.

F. **Multi-Function Display Testing and Operation** - Refer to the Avidyne FlightMax Entegra Multi-Function Display Installation Manual and FlightMax Entegra Series Pilot's Guide for testing and operation procedures.

3. Avidyne Magnetometer/OAT Sensor Assembly

A. **Magnetometer/OAT Sensor Assembly Removal**

(1) Ensure that the PFD power and avionics master switches are in the off position.

(2) Pull the PFD circuit breakers.

(3) Locate the magnetometer access panel located underneath the left wing, approximately 4 feet from the wing tip. This assembly replaces the KMT 112 slaving transmitter (refer to Section 34-50-00, Figure 209, Bendix/King KG 102A Remote Gyro And KI525A HSI Indicator).

(4) Remove the OAT probe nut from the mounting hole located in the magnetometer access panel.

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(5) Remove the magnetometer access panel.

(6) Remove the wire tie and cable tie securing the magnetometer connector assembly.

(7) Disconnect the connector assembly.

(8) Remove the nuts and washers securing the magnetometer to the aircraft mounted standoffs and remove the magnetometer/OAT sensor assembly from the aircraft.

B. **Magnetometer/OAT Sensor Assembly Installation** - Perform the steps for removal in reverse order. Tighten nuts snug by removing all looseness and add 1/4 turn more.

CAUTION: Ensure the magnetometer is installed with the arrow and letters "FWD" pointing forward.

C. **Magnetometer/OAT Sensor Calibration Procedures** - Refer to the System Setup and Checkout section.

4. Avidyne Engine Instrument Sensor Interface Unit (SIU)

A. **Sensor Interface Unit Removal** - The SIU is located on the avionics shelf in the avionics bay under the baggage compartment floor. Refer to Figure 203.

(1) Ensure both the master and avionics switches are in the off position.

(2) Remove the carpet and access panel from the baggage compartment floor.

(3) Disconnect connectors J900 and J901 from the unit.

(4) Remove the four mounting screws that hold the SIU to the instrument shelf. Refer to Figure 204.

(5) Remove the unit and the four rubber isolators.

B. **Sensor Interface Unit Installation**

(1) Installation is a direct reversal of removal. Hand tighten screws snug.

(2) Ensure connectors J900 and J901 are connected to the appropriate socket on the SIU.

(3) Install the access panel.

(4) Replace the carpet.

C. **SIU Checkout** - Use the MFD to checkout the SIU.

(1) Apply power to the SIU and MFD. Upon initial system power-up, the MFD will execute a start-up sequence. During this time, the screen will initially display the FlightMax logo, followed by a blank screen until a blue screen with the text *Initializing-Please Wait...* is displayed. The startup sequence is finished when the text *Press any bezel key to Continue...* is displayed. The initialization screen displays the loaded software version, along with NavData and Obstacle database expiration dates.

(2) Press any bezel key. The Initial Fuel page is displayed next.

(3) Press the Fuel Done button to exit.

D. **Avidyne Entegra Multi-Function Display (MFD) Configuration** - Refer to the Avidyne EX5000 Installation Manual. The MFD configuration is accomplished in the setup pages that are accessed through the Maintenance Mode, which is accessed as follows:

(1) Upon completion of system initialization, rotate the page knob to the Setup page.

(2) Simultaneously press and hold the top and middle left-hand bezel keys for at least 5 seconds until the maintenance page appears.

Configuration Item	Selections
Sensor Type	Not installed, Avidyne SIU
Port	Com 1, Com 2, Com *, Com 4* * Default port
Aircraft Model	Cessna 350

(3) Depress bezel key next to the Sensor Type legend until Avidyne SIU is displayed.

(4) Press the Save bezel key. The MFD must be restarted for the settings to take effect.

E. **Engine Sensor Checkout** - Refer to the Avidyne EX5000 Installation Manual and EX5000 Pilot's Guide for testing and operation procedures.

(1) After the MFD has been restarted, view the Engine page to verify data communication.

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(2) Operate the aircraft engine to confirm correct operation of all sensors. The following messages are generated by the Engine application:

Message	Meaning
Engine Sensor Unit Not Operating	No RS 232 Engine Sensor Unit data is being received by the MFD.
Engine Sensor Unit Operating Normally	Verification that Engine Sensor Unit data is valid.

5. Voltage Converters For Avidyne Entegra PFD

A. The voltage converters are located on the avionics shelf in the avionics bay under the baggage compartment floor. The voltage converters are only installed in aircraft S/N 42001 to 42500; they are not required on aircraft S/N 42501 and On. Refer to Figure 203.

(1) **Voltage Converter Removal**

(a) Ensure both the master and avionics switches are in the off position.

(b) Remove the carpet and access panel from the baggage compartment floor.

(c) If the XM weather receiver is installed, remove per 0.

(d) Unplug the converter.

(e) If applicable, remove all four voltage converter mounting screws and the XM weather receiver support bracket, otherwise remove the two voltage converter mounting screws.

(f) Remove the converter(s).

(2) **Voltage Converter Installation** - Installation is a direct reversal of removal. Hand tighten screws snug.

6. Voltage Converter For Avidyne Entegra EX5000 MFD

A. The voltage converter is located inside the radio rack at the base. Refer to Figure 205. The voltage converter is only installed in aircraft S/N 41002 to 41500; it is not required on aircraft S/N 41501 and On.

(1) **Voltage Converter Removal**

(a) Ensure both the master and avionics switches are in the off position.

(b) Remove the center console per instructions in Chapter 25-12-00 to gain access to the radio rack.

(c) Remove the connector.

(d) Remove the four hex head bolts attaching the power converter plate to the bottom of the radio rack and the floor.

(e) Lift out the converter.

(f) Remove heat shrink material (if present) on the nuts and washers and remove the two 8-32 x 2.5 inch screws holding the voltage converter to the mounting plate.

B. **Multi-Function Display Power Converter Installation**

(1) Perform the steps for removal in reverse order.

(2) Torque the 8-32 x 2.5 inch screws 20 to 25 in.-lbs. Apply new heat shrink material.

(3) Torque the four mounting plate bolts 30 to 36 in.-lbs.

7. Garmin G1000 System

A. This section describes how to remove and replace G1000 equipment in the Cessna 350. After removal and replacement, LRUs must be configured and tested per the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

CAUTION: When removing and/or replacing any G1000 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies. Before removing any G1000 LRU, it is required that the technician verify the LRU software part number and version against the Required Equipment List RB011002.

To check an LRU software part number and/or version:

(1) Start the G1000 system in configuration mode as described in Section 3.4 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

(2) The System Status page shows a list of LRUs in the LRU window. Activate the cursor and highlight the LRU window. Refer to Figure 206.

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(3) Use the FMS knob to scroll through the list in the window and select the desired LRU.

(4) The software part number and version is displayed in the DATA window. Compare this to the data in the Required Equipment List.

NOTE: If a faulty LRU is not reporting its software version and part number, check aircraft maintenance logs for last software version loaded and verify against the Required Equipment List. The Software Manifest page may also be used to check part numbers and versions.

B. **Troubleshooting** - For troubleshooting procedures for the Garmin G1000 System see the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

C. **GDU 1040 PFD and GDU 1042/GDU 1044 MFD** - Two Garmin GDUs are installed in the instrument panel. The PFD is a GDU 1040 and the MFD is a GDU 1042/GDU 1044 (required for Wide Area Augmentation System (WAAS) utilization). Both displays provide control and display of nearly all functions of the G1000 integrated cockpit system. The displays are located side-by-side, with the GMA 1347 Audio Panel located in-between.Both displays are installed in the instrument panel using built-in 1/4-turn fasteners. Each display uses a single Garmin 62-pin connector. Electrical power to the PFD is from the essential bus, whereas the MFD receives power from the Avionics bus. On the panel, traditional attitude, altitude, and airspeed indicators are provided as backup instruments. These are mounted in a vertical configuration to the left of the PFD. Refer to Figure 207.

(1) **Removal of GDU 1040 PFD/GDU 1042 MFD**

(a) Using a 3/32nd hex tool, rotate all four 1/4-turn fasteners counter-clockwise until they reach their stops.

(b) Carefully remove the display from the panel.

(c) While supporting the display, disconnect the connector.

(2) **Replacement of GDU 1040 PFD/GDU 1042 MFD**

(a) Visually inspect the connector and pins for signs of damage. Repair any damage. While supporting the display, connect the connector to the rear of the unit.

(b) Carefully insert the display into the panel cutout, ensuring that all four 1/4-turn fasteners align with the corresponding holes.

(c) Seat the display in the panel cutout. Do not use excessive force while inserting the display.

(d) Once seated, rotate all four 1/4-turn fasteners clockwise to lock the display to the panel.

(e) Load software, configure, and test the MFD and/or PFD according to Section 6.1 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

D. **GMA 1347 Audio Panel** - The Garmin GMA 1347 Audio Panel is a digital audio panel with integrated marker beacon receiver. The GMA 1347 provides control of all cockpit intercom/mic systems as well as NAV/COM/ILS audio. The unit also provides display reversion mode control through a large red button. Power is received from the Avionics bus; consequently the unit only powers up when the avionics master switch is turned on. The GMA 1347 interfaces with the existing marker beacon antenna as well as existing mic and phone jacks. Refer to Figure 208.

(1) **Removal of GMA 1347 Audio Panel**

(a) Using a 3/32nd hex tool, turn the hex nut counter-clockwise until the GMA 1347 is unlocked from its location.

(b) Carefully remove the GMA 1347 from its rack.

(2) **Replacement of GMA 1347 Audio Panel**

(a) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.

(b) Gently insert the GMA 1347 into the rack until the locking tab engages the rack.

(c) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not over-tighten the nut.

(d) Load software, configure, and test the GMA 1347 according to Section 6.2 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

E. **GIA 63/GIA 63W Integrated Avionics Unit** - Two Garmin GIA 63/GIA 63W IAUs provide VHF COM, VHF NAV, GPS NAV and other various navigation functions, as well as control logic for the Garmin GFC 700 autopilot. GIAs provide a communication interface to all other G1000 LRUs in the system. The GIA 63W is required for Wide Area Augmentation System (WAAS) utilization. Only one type of GIA either GIA 63 or GIA 63W will be installed. Both GIAs are located directly behind the MFD and installed into their respective LRU racks. The no. 1 GIA is powered through the essential bus and thus

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immediately powers up when the left battery or right battery master switch is turned on. The no. 2 GIA receives power through the Avionics bus and powers up when the avionics master switch is turned on. Refer to Figure 209. Both GIA 63/GIA 63Ws interface to the following equipment:

- VOR/LOC/Glideslope Antenna System
- VHF COM 1 & 2 Antennas
- GPS Antennas
  WAAS: GA 35 (no. 1 GIA only); GA 37 (no. 2 GIA only)
  Non WAAS: Two GA 56 (one for no. 1 GIA, the other for no. 2 GIA)
- WX-500 StormScope, if installed (no. 2 GIA Only)
- GFC 700 Autopilot
- Ryan TCAD, if installed (no. 2 GIA Only)
- CO Detector (no. 2 GIA Only)

(1) **Removal of GIA 63/GIA 63W Integrated Avionics Unit**

(a) Remove the MFD per the GDU 1040 PFD/GDU 1042 MFD section.

(b) Unlock the GIA 63/GIA 63W handle by loosening the Phillips screw on the handle.

(c) Pull the handle upward to unlock the GIA 63/GIA 63W. Gently remove the unit from the rack.

(2) **Replacement of GIA 63 Integrated Avionics Unit**

(a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

(b) Gently insert the GIA 63/GIA 63W into its rack. The handle should engage the dogleg track.

(c) Press down on the GIA 63/GIA 63W hand to lock the unit into the rack.

(d) Lock the handle to the GIA 63/GIA 63W body using the Phillips screw.

(e) Reinstall the MFD per the GDU 1040 PFD/GDU 1042 MFD section.

(f) Load software, configure, and test the GIA 63/GIA 63W(s) according to Section 6.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

F. **GEA 71 Engine/Airframe Unit** - The Garmin GEA 71 Engine/Airframe Unit provides engine/airframe data to the G1000 system. Data received from transducers/sensors is processed, then sent via RS-485 to a GIA 63/GIA 63W, and subsequently to the GDU 1042/GDU 1044 MFD. The GEA 71 is connected to both GIA 63/GIA 63Ws for redundant communications. In display reversionary mode, engine instrumentation is displayed on the PFD as well. The GEA is located directly behind the PFD in its LRU rack. Power is received from the essential bus. Refer to Figure 210.The GEA interfaces to the following:

- Manifold Pressure Sensor
- Tachometer Sensor
- Oil Pressure Sensor And Low Pressure Switch
- Oil Temperature Sensor
- Cylinder Head Temperature (CHT) Sensors
- Exhaust Gas Temperature (EGT) Sensors
- Turbine Inlet Temperature (TIT) Sensors (*Cessna 400 only*)
- Fuel Level Sensors (Four)
- Low Fuel Switches (L And R)
- Fuel Selector Position Switches
- Fuel Flow Sensor
- Electrical System Sensors And Alternator Inop Switches
- Open Door Detection Switch
- Aileron And Elevator Position Sensors
- Oxygen System Sensors And Switches
- Starter Engage System

(1) **Removal of GEA 71 Engine/Airframe Unit**

(a) Remove the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(b) Unlock the GEA 71 handle by unscrewing the Phillips screw.

(c) Pull the handle upward to unlock the GEA 71.

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(d) Gently remove the GEA 71 from its rack.

(2) **Replacement of GEA 71 Engine/Airframe Unit**

(a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

(b) Gently insert the GEA 71 into the rack. The handle should engage the dogleg track.

(c) Press down on the handle to lock the unit into place.

(d) Lock the handle to the GEA 71 body using the Phillips screw.

(e) Reinstall the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(f) Load software, configure, and test the GEA 71 according to Section 6.4 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

G. **GTX 33 Mode S Transponder** - The Garmin GTX 33 provides Mode A, C, and S altitude and position reporting information from the G1000 system. The unit is mounted directly behind the PFD in its LRU rack. Power is received from the Avionics Bus. The GTX 33 sends data via RS-232 directly to a GIA 63/GIA 63W.Information is then sent to the PFD, where the pilot can control the transponder. The GTX 33 is connected to both GIA 63/GIA 63W(s) for redundant communications. The GTX 33 interfaces with the existing transponder antenna. Refer to Figure 211.Test according to Title 14 CFR 91.411 and 91.413 as well as Part 43 Appendix F.

(1) **Removal of GTX 33 Transponder**

(a) Remove the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(b) Unlock the GTX 33 handle by loosening the Phillips screw on the handle.

(c) Pull the handle upward to unlock the GTX 33. Gently remove the unit from the rack.

(2) **Replacement of GTX 33 Transponder**

(a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

(b) Gently insert the GTX 33 into its rack. The handle should engage the dogleg track.

(c) Press down on the GTX 33 hand to lock the unit into the rack.

(d) Lock the handle to the GTX 33 body using the Phillips screw.

(e) Reinstall the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(f) Load software, configure, and test the GTX 33 according to Section 6.4 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

H. **GDC 74A Air Data Computer** - The Garmin GDC 74A provides digital air data to the G1000 system. The unit is mounted to its rack directly behind the PFD. Power is received from the Essential Bus. The GDC 74A connects to existing pitot/static ports. Air data is sent to the PFD for display. Four redundant data paths exist to the PFD, MFD, and both GIAs. Test according to Title 14 CFR §§ 91.411 and 91.413 as well as Part 43 Appendix F. The static system and altimeter must be verified every 2 years in accordance with 14 CFR §§ 91.411 and Part 43 Appendix E. Refer to Figure 212.Testing procedures per Section 6.6 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

(1) **Removal of GDC 74A Air Data Computer**

(a) Remove the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(b) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.

(c) Loosen each thumbscrew on the hold-down clamp and remove the clamp.

(d) Carefully remove the unit from its mount.

(2) **Replacement of GDC 74A Air Data Computer**

(a) Place the unit in the mounting tray.

(b) Position the locking clamp and fasten using the thumbscrews.

(c) Connect the pitot/static plumbing.

(d) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.

(e) Reinstall the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(f) Load software, configure, and test the GDC 74A according to Section 6.6 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

I. **GTP 59 Oat Probe** - The Garmin GTP 59 OAT Probe provides the GDC 74A with air temperature data. The OAT probe is

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mounted to the bottom side of the left wing.

(1) **Removal of GTP 59 Oat Probe**

(a) Remove the OAT probe cover assembly from the underside of the left wing by removing the Phillips screws.

(b) Disconnect the connector.

(c) Use a deep-socket to hold the probe in place on the outside of the cover. On the inside of the cover, loosen the GTP 59 mounting nut and remove the GTP 59.

(2) **Replacement of GTP 59 Oat Probe**

(a) Replacement is the reverse of removal.

(b) No configuration is required for the GTP 59. Test according to Section 6.6.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

J. **GRS 77 Attitude And Heading Reference System (AHRS)** - The Garmin GRS 77 AHRS provides attitude and heading information to the G1000 system. The unit is mounted on the center of the cockpit floor just aft of the firewall. Power is received from the Essential Bus. The GRS 77 interfaces with and provides power to the GMU 44 Magnetometer. The GRS 77 supplies attitude and heading information directly to the PFD, MFD, and to both GIAs. Refer to Figure 213.
The GRS 77 utilizes an Earth magnetic field model which is updated once every five years. The update is expected to be available from Garmin by July 1 of each of the following years: 2005, 2010, 2015, and every five years thereafter.

(1) **Removal of GRS 77 AHRS**

(a) Remove the instrument panel tower per Chapter 25-11-00.

(b) Disconnect the AHRS connector.

(c) Remove the four Phillips thumbscrews with a screwdriver and set them aside.

(d) Gently lift the GRS 77 from the mounting plate. (If the supports for the mounting plate are removed, the GRS 77 must be re-calibrated. See Section 6.7.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03. If the mounting plate is removed and reinstalled in exactly the same position, the GRS 77 does not need to be recalibrated.)

(2) **Replacement of GRS 77 AHRS**

(a) Place the GRS 77 on the mounting plate, ensuring the orientation is correct.

(b) Fasten the unit to the plate using the Phillips thumbscrews.

(c) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 77.

(d) Load software, configure, and test the GRS 77 according to Section 6.7 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

(e) Reinstall the instrument panel tower per Chapter 25-11-00.

K. **GMU 44 Magnetometer** - The GMU 44 provides horizontal and vertical magnetic field information to the GRS 77 AHRS. This allows heading to be calculated and provides assistance during AHRS alignment. The GMU 44 is mounted at the left wingtip. Refer to Figure 214.

(1) **Removal of GMU 44 Magnetometer**

(a) Remove left wing access cover by unscrewing the Phillips screws.

(b) Unscrew the three screws, or remove the three self-locking nuts and washers that hold the GMU 44 to its mounting rack.

(c) Carefully lift the GMU 44 from the rack.

(d) Disconnect the wiring harness.

(2) **Replacement of GMU 44 Magnetometer**

(a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the wiring harness to the GMU 44.

(b) Lower the GMU 44 into the rack and secure with three Phillips screws or three self-locking nuts and washers, as applicable. Tighten screws and/ or nuts snug by removing all looseness and add 1/4 turn more.

(c) Reinstall left wing access cover.

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(d) Load software, configure, and test the GMU 44 according to Section 6.7 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

L. **GDL 69A Data Link Receiver** - The GDL 69A is an XM Satellite Radio data link receiver with the addition of XM Satellite Radio audio entertainment. For display of weather information and control of audio channel and volume, the GDL 69A is interfaced to the GDU 1042/GDU 1044 via an Ethernet link. Audio volume and channel changes may also be controlled with remotely mounted switches located in the center console. The GDL 69A is also interfaced to a Garmin audio panel for amplification and distribution of the audio signal. The GA 55 XM Satellite Radio antenna receives the XM Satellite Radio data signal and passes it to the GDL 69A. On aircraft equipped for Wide Area Augmentation System (WAAS), the GA 55 is not installed and the XM Satellite Radio data signal is received by the GA 37 (combined GPS and XM satellite radio) antenna located on the antenna bay behind the hat rack. Refer to Figure 215.

(1) **Removal of GDL 69A Data Link Receiver**

(a) Remove the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(b) Unlock the GDL 69A handle by unscrewing the Phillips screw.

(c) Pull the handle upward to unlock the GDL 69A.

(d) Gently remove the GDL 69A from its rack.

(2) **Replacement of GDL 69A Data Link Receiver**

(a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

(b) Gently insert the GDL 69A into the rack. The handle should engage the dogleg track.

(c) Press down on the handle to lock the unit into place.

(d) Lock the handle to the GDL 69A body using the Phillips screw.

(e) Reinstall the PFD per the GDU 1042 PFD/GDU 1044 MFD section.

(f) Load software, configure, and test the GDL 69A according to Section 6.8 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

M. **GCU 476 Remote Keypad** - The GCU 476 interfaces with the GDU 104x PFD/MFD. The GCU 476 Remote Keypad provides alphanumeric, softkey, and flight planning function keys used to interface with the G1000. In addition to alphanumeric, softkey, and flight planning function keys the GCU 476 provides COM/NAV tuning capabilities. The GCU 476 mounts on the center console using a single jackscrew. Refer to Figure 216.

(1) **Removal of GCU 476 Remote Keypad**

(a) Remove the setscrew securing the keypad with a 3/32nd hex tool and carefully remove keypad.

(b) While supporting the keypad disconnect the connector.

(c) Remove the unit.

(2) **Replacement of GCU 476 Remote Keypad**

(a) Connect the connector to the back of keypad.

(b) Gently place keypad into position.

(c) Secure with setscrew using a 3/32nd hex tool. Do not over-tighten.

(d) Reload configuration files and verify software version.

N. **GFC 700 Autopilot System** - Refer to Chapter 22-01-00.

O. **GA 35 Antenna (WAAS)**

The GA 35 antenna is one of the components of the Garmin G1000 System that allows utilization of the Wide Area Augmentation System (WAAS). The antenna is mounted to a leveling wedge on top of the aircraft and forward of the vertical stabilizer at FS 185.51. The leveling wedge is permanently affixed to the aircraft.

(1) **Antenna Removal**

(a) Remove the aft center overhead light panel (ACOLP) per instructions in Chapter 25-30-00.

(b) Disconnect the antenna cable from the connector on the base of the antenna. Keep the antenna connector clean with a plastic cover cap or other means of protection.

(c) Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four screws attaching the antenna to the leveling wedge.

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(d) Stow the braided wire/terminal from the antenna.

(e) Remove the antenna.

(2) **Antenna Installation**

(a) Remove all old RTV-162 silicone from the leveling wedge to antenna mounting surface and apply a fresh layer.

(b) Unstow the braided wire/terminal.

(c) Place the antenna on the wedge and install the mounting screws. Ensure the braided wire/terminal is aligned to the antenna mounting hole. Torque 17 to 21 inlbs. Wipe away any excess RTV-162.

(d) Fill the voids over the screw heads with RTV-162 and smooth it to the contours of the antenna.

(e) Connect the antenna cable to the connector on the base of the antenna.

(f) Install the aft center overhead light panel (ACOLP) per instructions in Chapter 25-30-00.

P. **GA 37 Antenna (WAAS)**

The GA 37 is a combined GPS and XM satellite radio antenna, and is one of the components of the Garmin G1000 System that allows utilization of the Wide Area Augmentation System (WAAS). The antenna is behind the hat rack mounted on the antenna box (S/N 42001 thru 421014) or on a platform attached to the antenna box (S/N 421015 and on).

(1) **Antenna Removal**

(a) Remove the aft baggage bulkhead upholstery plug and the cover plate.

(b) Disconnect the GPS antenna cable and the XM satellite radio data cable from the connectors on the base of the antenna. Keep the antenna connectors clean with a plastic cover cap or other means of protection.

(c) Remove the four screws, self-locking nuts, and washers attaching the antenna to the antenna box. For aircraft S/N 421015 and on, remove the four screws attaching the antenna to the platform.

(d) Lift the antenna and the O-ring from the antenna box.

(2) **Antenna Installation**

(a) Place the O-ring on the antenna.

(b) Place the antenna on the antenna box.

(c) Install the fours screws, self-locking nuts, and washers, as applicable. Torque 28 to 34 in-lbs.

(d) Connect the GPS antenna cable and the XM satellite radio data cable.

(e) Replace the cover plate and aft baggage bulkhead upholstery plug.

Q. **Configuration Module Removal And Replacement**

(1) **Removal of Configuration Module** - (Refer to Figure 217.)

(a) Disconnect connector from LRU.

(b) Remove 2 screws (8) from cover (7) and remove cover.

(c) Unplug connector from configuration module (1).

(d) Remove configuration module.

(2) **Replacement of Configuration Module** - (Refer to Figure 217.)

(a) Inspect connector for damaged pins (4).

(b) Place configuration module (1) in position.

(c) Insert connector into configuration module (1).

(d) Assembly of the connector is the reverse of disassembly.

(3) **If the GRS 77 AHRS Configuration Module is replaced** - All three GRS 77/GMU 44 calibration procedures must be performed. Refer to Section 6.6.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

(4) **If GDC 74 Configuration Module is replaced** - Configuration settings must be reloaded to the GDC 74A. Refer to Section 6.5.2 of the Garmin G1000 System Maintenance Manual, P/N 190-00577- 03.

(5) **If the Master Configuration Module is replaced**

(a) Start the G1000 system in configuration mode.

(b) Go to the Configuration Upload Page on the PFD.

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(c) Press the UPDT CFG softkey.

(6) **If both the PFD and Master Configuration Module are replaced** - The G1000 system (except GRS 77/GMU 44 and GDC 74A) must be re-configured. Refer to Section 6.1.1 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

R. **GEA 71 Backshell Thermocouple Removal And Replacement**

(1) **Removal of GEA 71 Backshell Thermocouple** - (Refer to Figure 218.)

(a) Remove GEA 71 per GEA 71 Engine/Airframe Unit section.

(b) Remove GEA connector backplate.

(c) Remove connector J701 (5) from the backplate.

(d) Remove cover (6) from the backshell.

(e) Unscrew thermocouple from boss on backshell. Extract the thermocouple pins from the connector.

(2) **Replacement of GEA 71 Backshell Thermocouple** - (Refer to Figure 218 and Table 219.)

(a) Crimp pins (2) onto each of the thermocouple wires (1). Ensure that pre-stripped wire length is 1/8" prior to crimping.

(b) Insert newly crimped pins and wires into the appropriate connector housing location (4) as specified by the 350/400 avionics wiring diagram of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

(c) Place thermocouple body (1) onto the backshell boss (5). Place the thermocouple as shown in Figure 218 so that the wires exit towards the bottom of the backshell.

(d) Fasten thermocouple tightly to backshell using the provided screw (3).

(e) Fasten cover (6) to backshell using the provided screws (7).

Table 219. Thermocouple Kit (011-00981-00)

Item No.	Description	Quantity Needed	Garmin Part Number
1	3 in. Thermocouple, K type	1	925-L0000-00
2	Pins no. 22 AWG	2	336-00021-00
3	Screw	1	211-60234-08

S. **Inspection**

(1) **Annual**

(a) Power up the aircraft by turning on the left & right batteries/alternators, start the engine and then turn on the Avionics Master Switch. - ensure the cross tie switch is OFF, A/P Trim Master ON.

(b) Wait approximately one minute for the system to initialize.

(c) Verify the system obtains attitude and altitude, and no functions on the PFD or MFD are inoperative. Inoperative functions are X out in red.

(d) Allow the system to acquire GPS position. Verify that both GIAs have GPS position via the AUX page.

(e) Ensure the system does not generate any failures or alerts.

(f) Verify all engine parameter and temperature sensors are operational.

(g) Verify aircraft charging system is operation, both left and right bus.

(h) Turn off the left alternator and verify the G1000 annunciates a failure of the left alternator and plays an aural message *Left Alternator Out*.

(i) Monitor the full screen engine page on the MFD, turn off the left battery switch. Verify the left battery voltage drops off and the G1000 annunciates a failure of the left bus; restore the left battery and alternator switches to the on position.

(j) Turn off the right alternator and verify the G1000 annunciates a failure of the right alternator and plays an aural message *Right Alternator Out.*

(k) Monitor the full screen engine page on the MFD, turn off the right battery switch. Verify the right battery voltage drops off and the G1000 annunciates a failure of the right bus; restore the right battery and alternator switches to the on position.

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(l) Verify the selected full tank value position, agrees with the indicator on the MFD (full screen EIS, or engine strip). Switch fuel tanks, and verify the indication on the G1000 follows. Move the fuel selector between tanks momentarily and verify the G1000 annunciates fuel valve along with an aural warning.

(m) Leave the avionics switch on, and turn off the engine with the mixture control (lean to cut-off)

(n) Unlatch the pilot door, and verify the G1000 annunciates door, latch the pilot door, and repeat for the copilot door and the aft cargo door.

(o) If speed breaks are installed actuate the speed breaks and verify the G1000 annunciates an advisory to indicate the speed breaks are deployed.

(p) If the rudder hold option is installed actuate the hold function and verify the G1000 annunciates an advisory to indicate the rudder is being held.

(q) Turn off the avionics switch, ensure the G1000 enters reversionary mode (composite engine/PFD).

(r) Run the elevator trim to the full up position, and verify the trim indicates full up. Repeat for down.

(s) Run the aileron trim to the full right position, and verify the indication. Repeat for left.

(t) Turn off the trim master switch and verify elevator and aileron trim are disabled and G1000 annunciates a pitch trim failure.

(2) **1000 Hour** - Perform a visual inspection every 1000 airframe hours in conjunction with the next scheduled inspection and check for corrosion, damage, or other defects for each of the items listed below. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

(a) **GDU 1040PFD and GDU 1042/GDU 1044 MFD** - Remove the PFD & MFD. Inspect the mounting surface and connector for corrosion, heavy oxidation, or other damage.

(b) **GMA 1347 Audio Panel** - Inspect the GMA 1347 unit, rack, and connectors for corrosion or other defects.

(c) **GIA 63/GIA 63W Integrated Avionics Unit** - Inspect the GIA 63/GIA 63W #1 and #2 unit, rack, and connectors for corrosion or other defects.

(d) **GEA 71 Engine/Airframe Unit** - Inspect the GEA 71 unit, rack, and connectors for corrosion or other defects.

(e) **GTX 33 Mode S Transponder** - Inspect the GTX 33 unit, rack, and connectors for corrosion or other defects.

(f) **GDL 69A XM Radio Data Link Receiver** - Inspect the GDL 69A unit, rack, and connectors for corrosion or other defects.

(g) **GRS 77 AHRS** - Inspect the GRS 77 unit, rack, and connector for corrosion or other defects.

8. WX-500 Stormscope Processor

A. The WX-500 Stormscope Processor is located on the avionics shelf under the baggage compartment floor; it's location on the avionics shelf depends on other equipment installed on the aircraft.

(1) WX-500 Stormscope Processor and Tray Removal. Refer to Figure 219 or Figure 220.

(a) Ensure both the master and avionics switches are in the off position.

(b) Remove the carpet and access panel from the baggage compartment floor.

(c) Disconnect the wiring connectors.

(d) Remove the locking nut on front of the mounting tray.

(e) Slide out the stormscope from the mounting tray.

(f) Stormscope Mounting Tray Removal (if required)

1 S/N 42001 to 42500) - Remove the four 10-32 screws and spacers attaching the tray to the avionics shelf. Remove the tray.

2 S/N 42501 and On - Remove the XM Weather Receiver per the Turn Coordinator section. Remove the three 10-32 screws attaching the tray to the avionics shelf. Remove the tray.

(2) WX-500 Processor and Tray Installation (refer to Figure 219) - Installation is a direct reversal of removal. Hand tighten screws snug.

9. WX-500 Stormscope Antenna

A. The WX-500 Stormscope Antenna is located under the hat rack.

(1) WX-500 Stormscope Antenna Removal. Refer to Figure 221.

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(a) Ensure both the master and avionics switches are in the off position.

(b) Remove the hat rack per Chapter 25-30-00 to gain access to the baggage bulkhead access panel. Remove the baggage bulkhead access panel.

(c) Remove the antenna wiring connector.

(d) Remove RTV silicon caulk around antenna base, if present.

(e) Remove two screws attaching the antenna and gasket to the fuselage and antenna plate.

(f) If necessary, remove two screws attaching the antenna plate to the fuselage. If these screws do not go through the gasket, they must be removed.

(2) WX-500 Stormscope Antenna Installation. Refer to Figure 221.

(a) If necessary, install two screws attaching the antenna plate to the fuselage. Ensure these screws go through the gasket. Torque 30 to 36 in.-lbs.

(b) Install two screws attaching the antenna and gasket to the fuselage and antenna plate. Torque 30 to 36 in.-lbs.

(c) Apply and smooth RTV silicon caulk around the antenna base.

(d) Attach antenna wiring connector.

(e) Replace the baggage bulkhead access panel and the hat rack per Chapter 25-30-00.

10. MX20 Multi-Function Display (Optional)

A. **Display Removal** - The map displays are located in the instrument panel, and the controller is located on the display unit. Refer to Figure 222.

(1) Insert a 3/32 inch hex wrench through the hole at the bottom center of the unit you want to remove.

(2) Turn the wrench counterclockwise at least 5 turns, and gently pull the unit away from the instrument panel until free.

B. **Display Installation**

(1) Slide the unit into the tray and insert a 3/32 inch hex wrench through the hole at the bottom center of the unit you want to remove.

(2) Turn the wrench clockwise while pushing on the unit. Do not over-torque the locking screw.

C. **Initial Set-up Procedure**

(1) Turn the aircraft power and MX20 power to ON.

(2) Immediately after the self-test of the unit is complete, press line select keys 1, 4, and 6 in sequence. Do not press any other buttons before starting the sequence. (Line select keys are located vertically along the right hand side of the MX20 unit below the rotary power knob, and are numbered in order with the top key as 1.

(3) Depress the FN function key, one or more times until INSTL is displayed above the smart keys.

(4) Select the INSTL function by pressing the smart key directly below the INSTL label on the screen. The Enable/Disable Functions will be the first screen to appear and FUNC will be highlighted above the smart keys.

(5) Set the Enable/Disable Functions per Figure 223 by pressing the line key next to the function label.

(6) Press the smart key below the DATA label while still in the INSTL function. Set the External Data Sources per Figure 224 by pressing the line key next to the function label.

(7) Press the smart key below the MISC label while still in the INSTL function. Leave everything on these pages at factory defaults as shown in Figure 225.

NOTE: After making final setting adjustments, do not power off for 30 seconds.

(8) Press Enter/Menu to confirm all settings.

(9) After the 30-second waiting period, turn the MX20 power off to apply all configuration settings.

11. ORBCOMM Antenna (Basic or Avidyne Option Only)

A. The ORBCOMM antenna is not installed with the Garmin G1000 System.

(1) **ORBCOMM Antenna Removal**

(a) The ORBCOMM antenna is located on top of the aircraft and forward of the vertical stabilizer. Refer to Figure 226 and Figure 227.

(b) Remove the aft center overhead light panel (ACOLP) per instructions in Chapter 25-30-00.

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(c) Disconnect the antenna cable from the connector on the base of the antenna. Keep the antenna connector clean with a plastic cover cap or other means of protection.

(d) Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four machine screws attaching the antenna to the fuselage.

(e) With an assistant within the aircraft immobilizing the self locking nuts, remove the screws mounting the ORBCOMM antenna to the fuselage.

(f) Remove the antenna.

(g) Discard the antenna gasket.

(2) **ORBCOMM Antenna Installation**

(a) If necessary, remove all paint and primer from the entire contact area of the antenna gasket to expose the lightning mesh.

(b) Position the ORBCOMM antenna and antenna gasket (PIC Wire and Cable part #PGP103-02 or PVF110-10) on top of the fuselage.

(c) Attach the antenna to the fuselage using the four machine screws, washers and nuts. Have an assistant inside the fuselage immobilize the nuts while the screws are tightened.

(d) Do not exceed 15 in-lbs of torque and tighten the hardware in a progressive pattern.

(e) Allow the antenna gasket to cure for 24 hours per the gasket manufacturer's instructions.

(f) Fill the voids around the antenna base and over the screw heads with RTV white silicone sealant conforming to Mil-A-46146. Smooth the sealant to the contours of the antenna base.

(g) Connect the antenna cable to the connector on the base of the antenna.

(h) Install the aft center overhead light panel (ACOLP) per instructions in Chapter 25-30-00.

12. Ryan TCAD Processor

A. The Ryan TCAD Processor is located on the avionics shelf under the baggage compartment floor. Refer to Figure 228.

(1) **Processor Removal**

(a) Ensure both the master and avionics switches are in the off position.

(b) Remove the carpet and access panel from the baggage compartment floor.

(c) Disconnect all connectors to the unit.

(d) Loosen the two mounting screws that hold the processor onto the tray railings in the mounting bracket attached to the avionics shelf.

(e) Slide the processor off the tray railings and remove the unit.

(2) **Processor Installation**

(a) Perform steps a-d of the Processor Removal section, in reverse order.

(b) Perform a functional check of the MFD in accordance with the System Setup and Checkout section.

13. Ryan TCAD Antenna

A. **Antenna Removal**

(1) Top Antenna - The top antenna is located on top of the aircraft and forward of the vertical stabilizer. Refer to Figure 229 and Figure 230.

(a) Remove the aft center overhead light panel (ACOLP) per instructions in Chapter 25.

(b) Disconnect the cabling from the base of the antenna. Keep the antenna connectors clean with a plastic cap or other means of protection.

(c) Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four machine screws attaching the antenna to the fuselage.

(d) With an assistant within the aircraft immobilizing the self locking nuts, remove the screws mounting the antenna to the fuselage.

(e) Remove the antenna.

(f) Discard the antenna gasket.

(2) Bottom Antenna - The bottom antenna is on the aileron bellcranks and crossover assembly access panel located on

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the bottom of the aircraft. Refer to Figure 229 and Figure 231.

(a) Remove the aileron bellcranks and crossover assembly access panel.

(b) Disconnect the cabling from base of the antenna. Keep the antenna connectors clean with a plastic cap or other means of protection.

(c) Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four machine screws attaching the antenna to the fuselage.

(d) Remove the screws mounting the antenna to the access panel.

(e) Remove the antenna.

(f) Discard the antenna gasket.

B. **Antenna Installation**

(1) Top Antenna

(a) If necessary, remove all paint and primer from the entire contact area of the antenna gasket to expose the lightning mesh.

(b) Position the antenna and antenna gasket (PIC Wire and Cable part #PTS109-01) on top of the fuselage.

(c) Attach the antenna to the fuselage using the four machine screws, washers and nuts. Have an assistant inside the fuselage immobilize the nuts while the screws are tightened.

(d) Do not exceed 15 in-lbs of torque and tighten the hardware in a progressive pattern.

(e) Allow the antenna gasket to cure for 24 hours per the gasket manufacturer's instructions.

(f) Fill the voids around the antenna base and over the screw heads with RTV white silicone sealant conforming to Mil-A-46146. Smooth the sealant to the contours of the antenna base.

(g) Connect the antenna cables to the connectors on the base of the antenna.

(h) Install the aft center overhead light panel (ACOLP) per instructions in Chapter 25.

(2) Bottom Antenna

(a) If necessary, remove all paint and primer from the entire contact area of the antenna gasket to expose the lightning mesh.

(b) Position the antenna and antenna gasket (PIC Wire and Cable part no. PTS109-01) on the aileron bellcranks and crossover assembly access panel.

(c) Attach the antenna using the four machine screws.

(d) Do not exceed 15 in-lbs of torque and tighten the hardware in a progressive pattern.

(e) Allow the antenna gasket to cure for 24 hours per the gasket manufacturer's instructions.

(f) Fill the voids around the antenna base and over the screw heads with RTV white silicone sealant conforming to Mil-A-46146. Smooth the sealant to the contours of the antenna base.

(g) Connect the antenna cables to the connectors on the base of the antenna.

(h) Install the aileron bellcranks and crossover assembly access panel.

14. Ryan TCAD Transponder Coupler

A. The Ryan TCAD transponder coupler is located on the fuselage behind the instrument panel on the co-pilot's side for the basic or Avidyne option, and on the center wing aft spar behind the rear seats footwell for the Garmin G1000 option.

(1) **Basic or Avidyne Option**

(a) **Transponder Coupler Removal**

1 Ensure both the master and avionics switches are in the off position.

2 Remove the co-pilot's side kidney panel per Chapter 25.

3 Disconnect the cabling to the coupler.

4 Depending upon the mounting configuration, there are either two screws, washers and nuts; or only two screws to be removed for detachment of the coupler from the mounting bracket. Refer to Figure 232, Sheet 1 and Figure 232, Sheet 2.

(b) **Transponder Coupler Installation** - Installation is the direct reverse of removal.

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(2) **Garmin G1000 Option**

(a) **Transponder Coupler Removal**

1 Ensure both the master and avionics switches are in the off position.

2 Remove the rear seats footwell per Chapter 25.

3 Disconnect the cabling to the coupler.

4 Remove two screws, washers and nuts attaching the coupler to the spar. Refer to Figure 233.

(b) **Transponder Coupler Installation** - Installation is the direct reverse of removal.

15. XM Weather Receiver

A. The XM Weather Receiver is located on the avionics shelf under the baggage compartment floor. Refer to Figure 234 and Figure 235.

(1) **Receiver Removal**

(a) Ensure both the master and avionics switches are in the off position.

(b) Remove the carpet and access panel from the baggage compartment floor.

(c) Disconnect all connectors to the unit.

(d) Remove the four screws and washers that hold the receiver onto the mounting bracket attached to the top of the two PFD voltage converters.

(e) Remove the receiver.

(2) **Receiver Installation**

(a) Perform the steps of the Receiver Removal section, in reverse order.

(b) Perform a functional check of the MFD in accordance with the System Setup and Checkout section.

16. XM Weather Antenna

A. **Antenna Removal** - The antenna is located behind the instrument panel on the co-pilot's side. Refer to Figure 236 and Figure 237.

(1) Remove the glare shield per instructions in Chapter 25-11-00.

(2) Disconnect the antenna cable from the connector on the base of the antenna. Keep the antenna connector clean with a plastic cover cap or other means of protection.

(3) Remove the screws, washers, and self-locking nuts mounting the antenna to the mounting bracket.

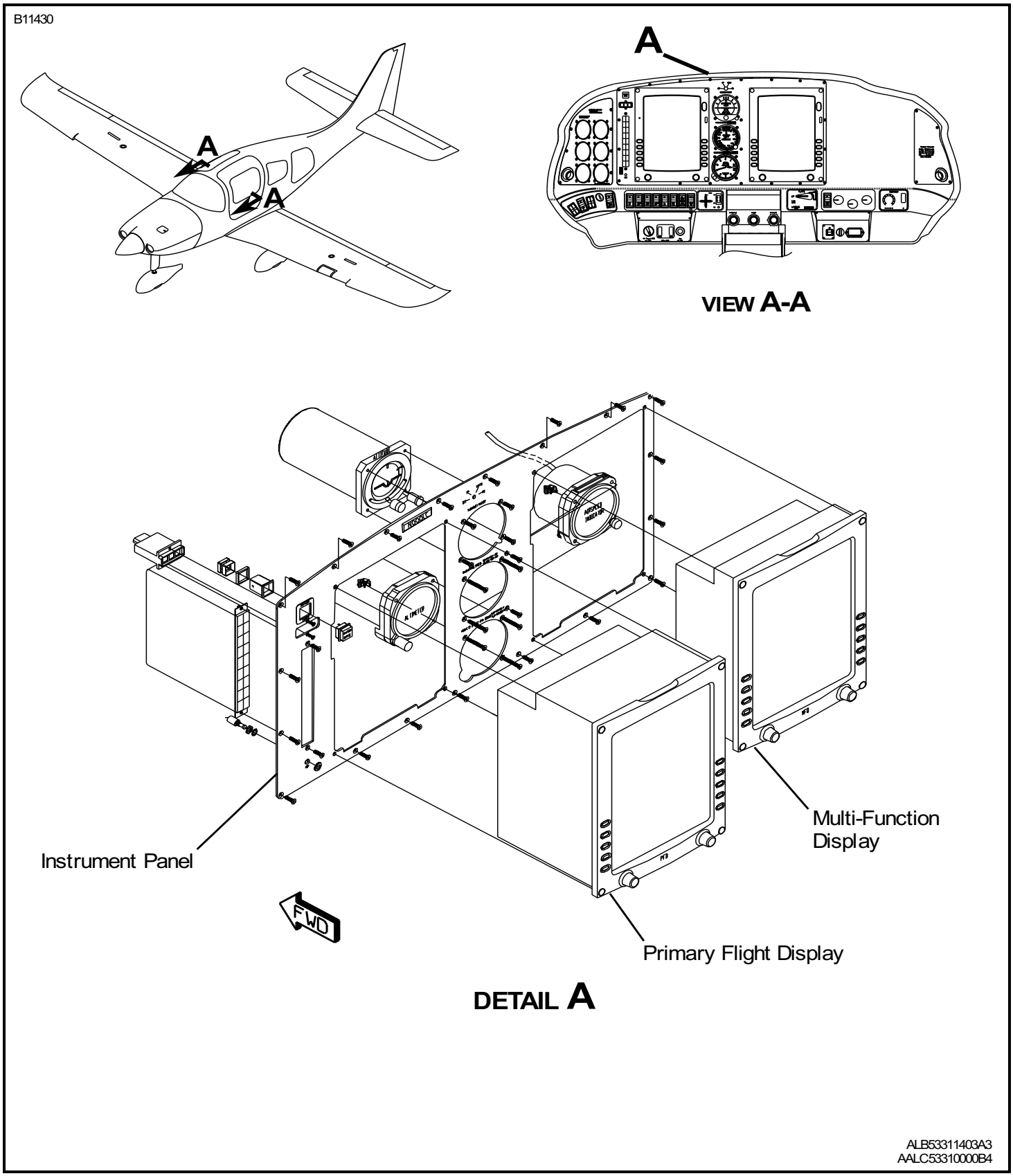
(4) Remove the antenna.

B. **Antenna Installation**

(1) Perform the steps of the Antenna Removal section, in reverse order.

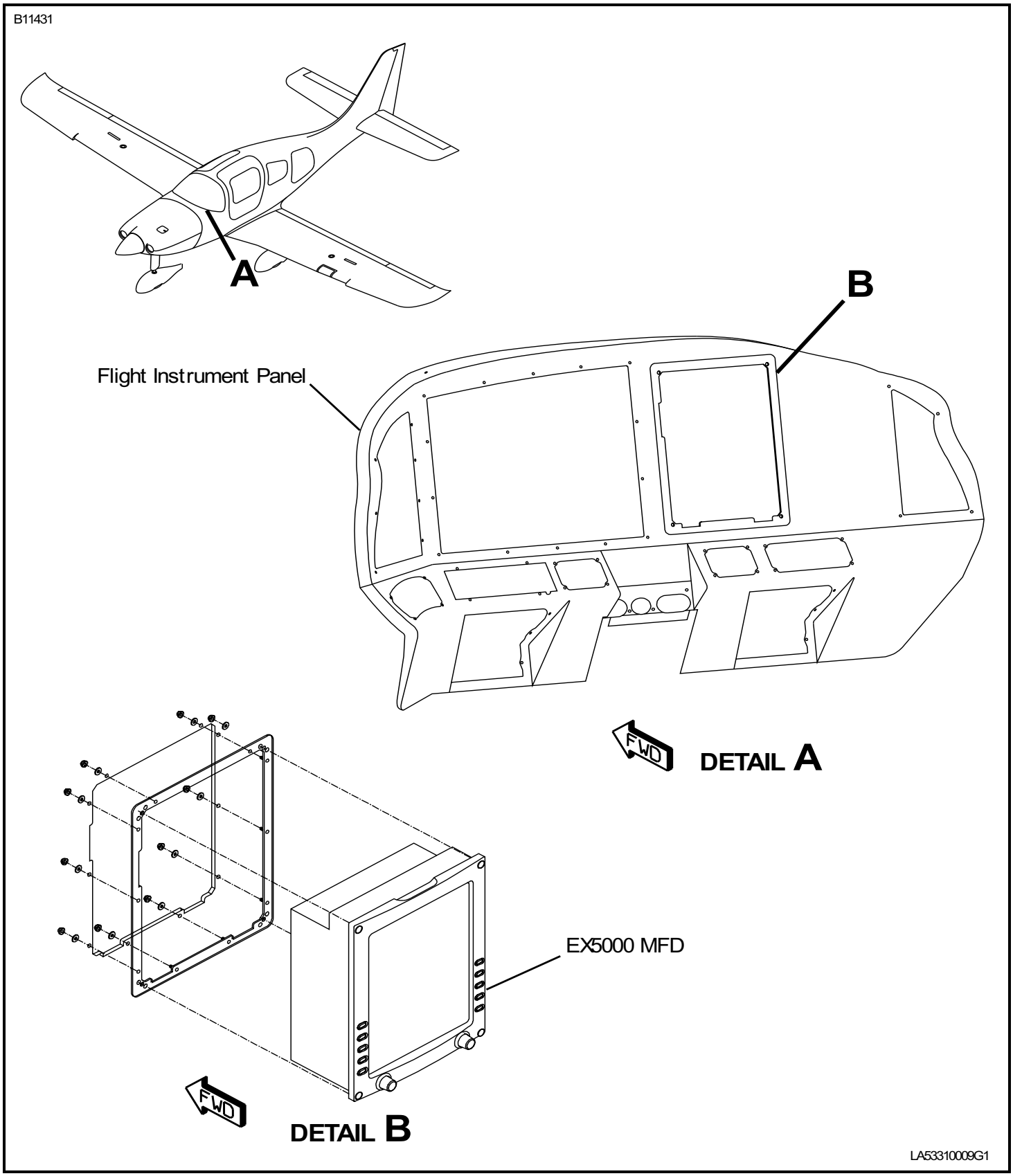
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Figure 201 : Sheet 1 : Avidyne Entegra Primary Flight Display and EX5000 Multi-Flight Display Installations



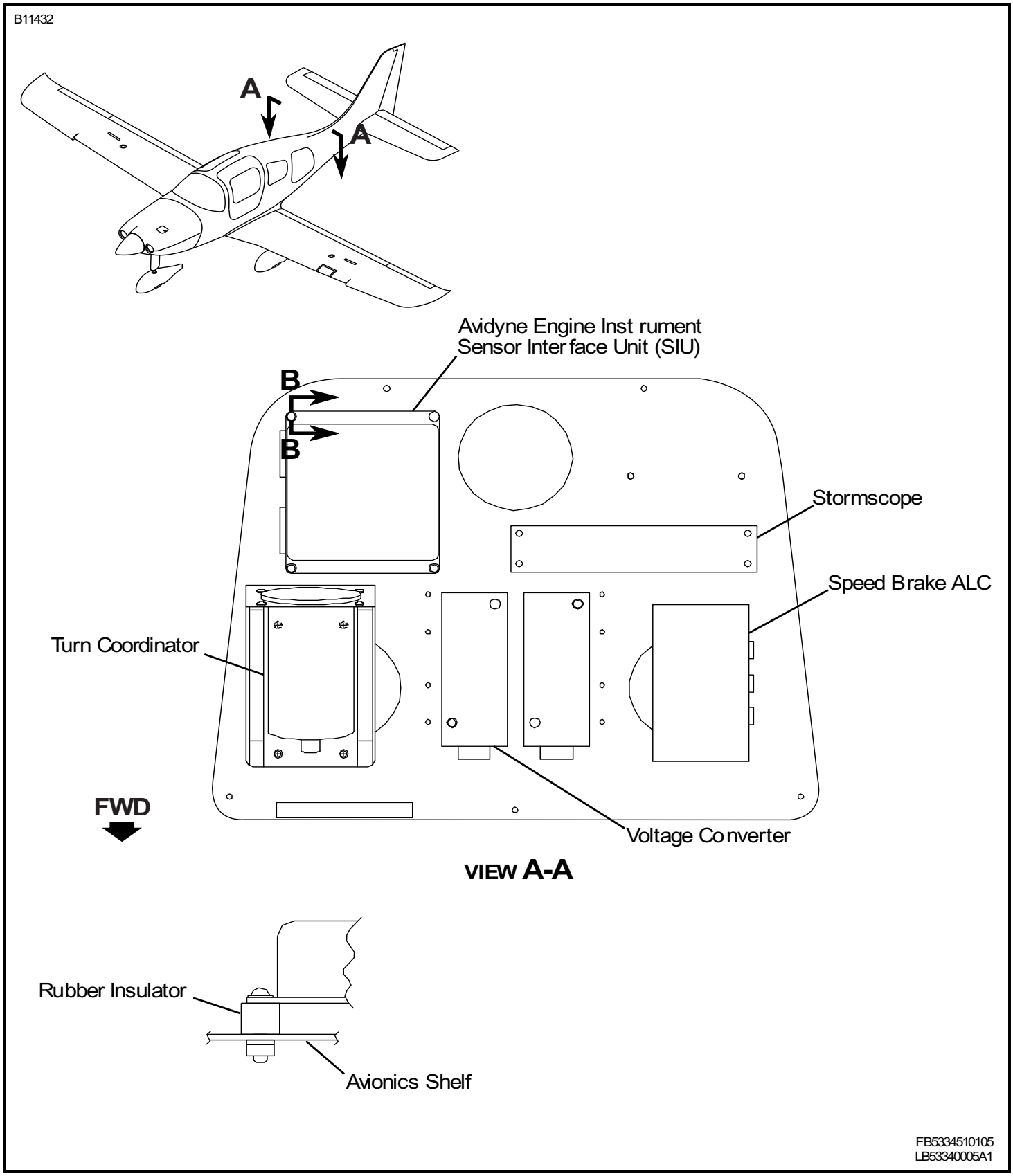
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Figure 202 : Sheet 1 : Avidyne Entegra EX5000 Multi-Flight Display Installation Without PFD Option



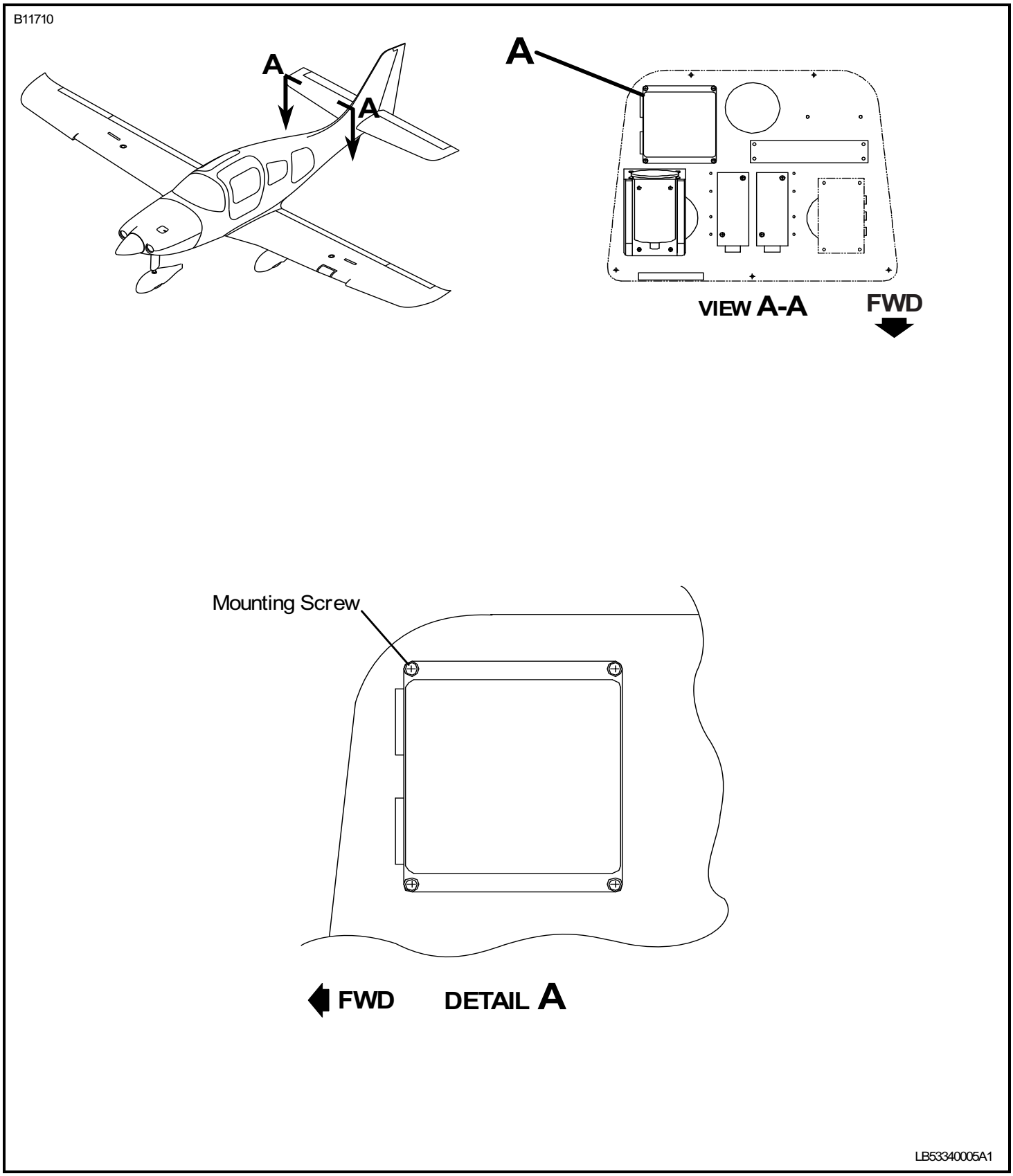
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Figure 203 : Sheet 1 : Avionics Shelf Configuration for Avidyne Entegra Equipment



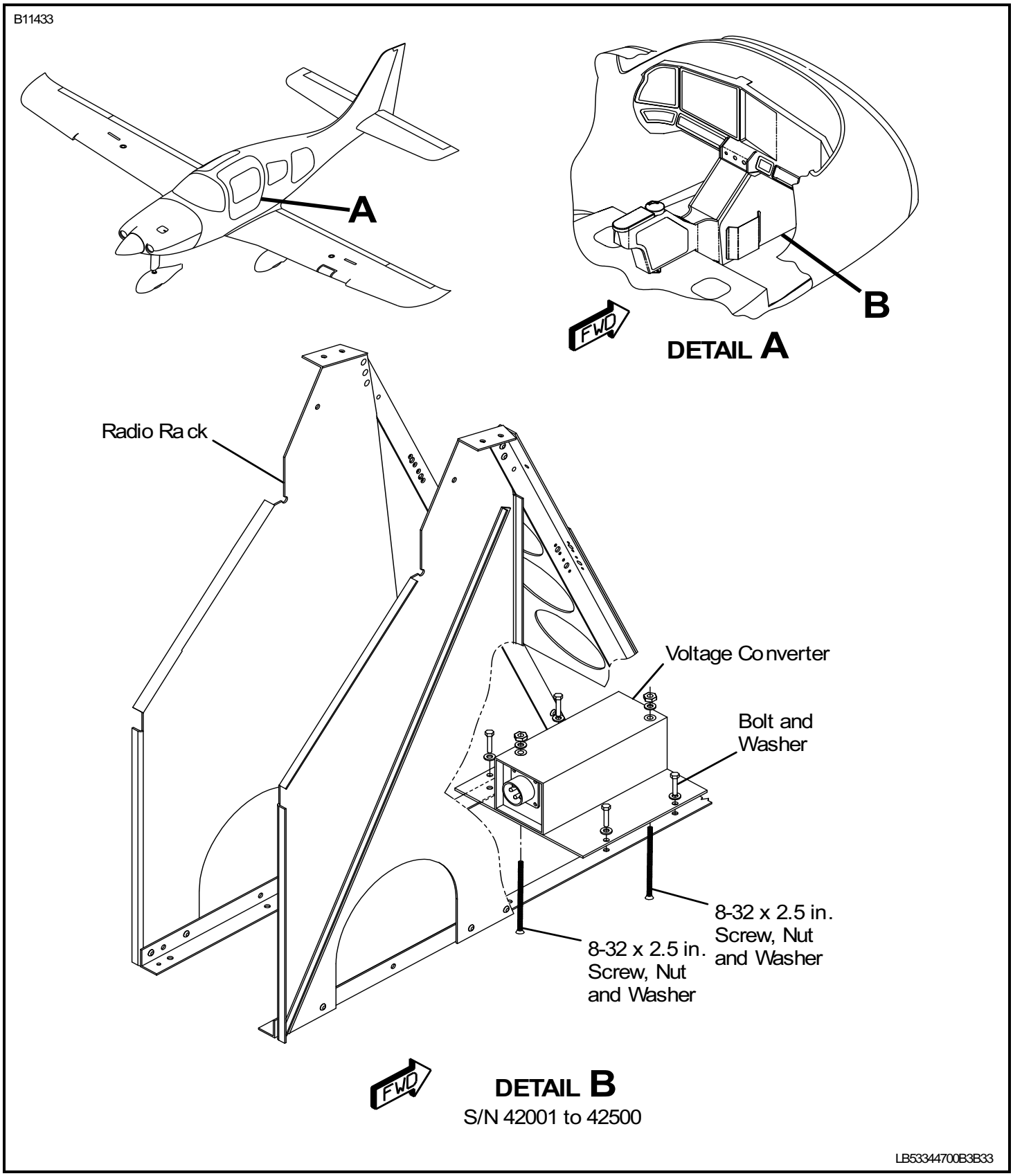
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Figure 204 : Sheet 1 : Engine Instrument Sensor Interface Unit



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Figure 205 : Sheet 1 : EX5000 Voltage Converter Installation (S/N 42001 to 42500)



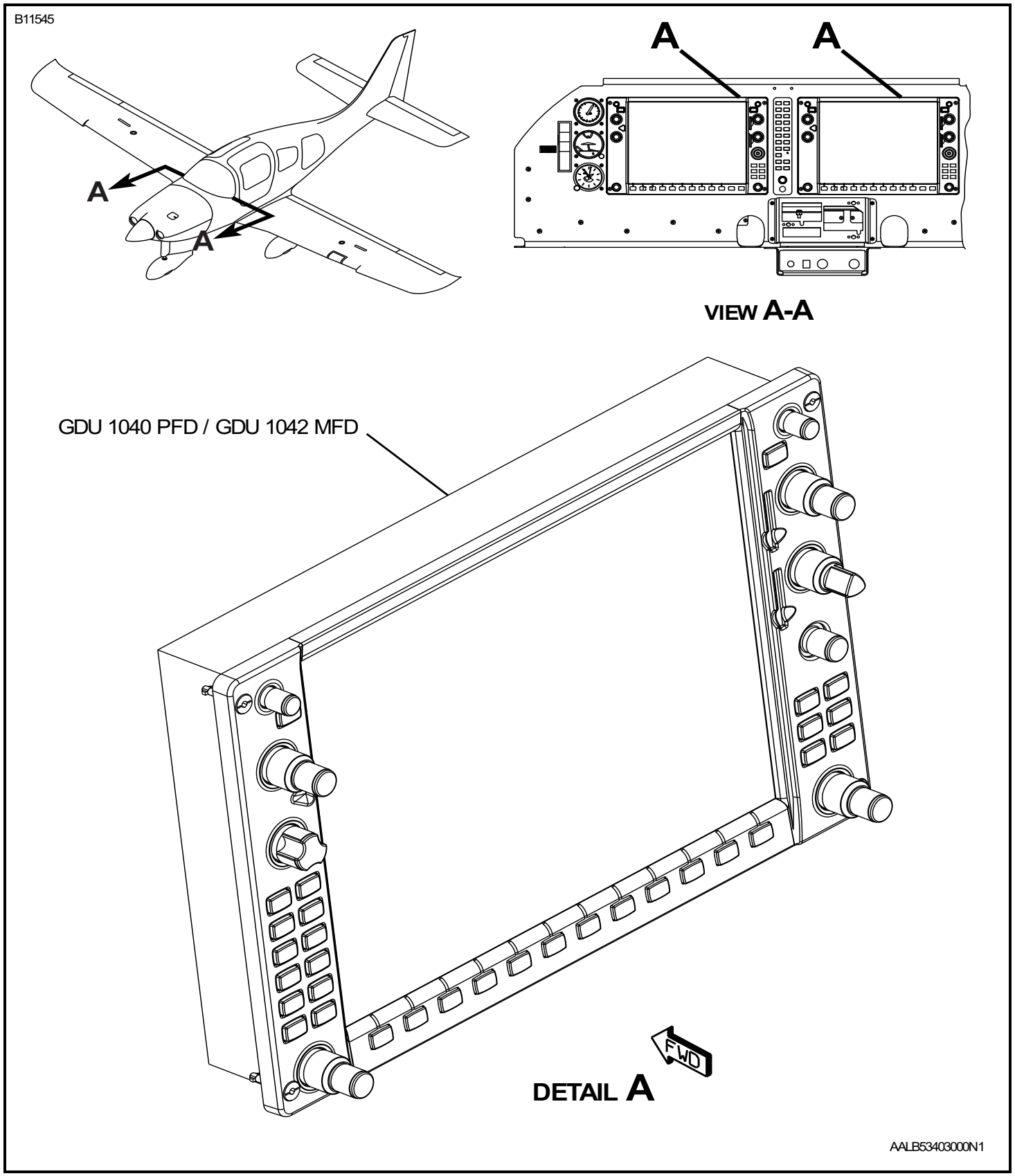
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Figure 206 : Sheet 1 : System Status Page (Configuration Mode)



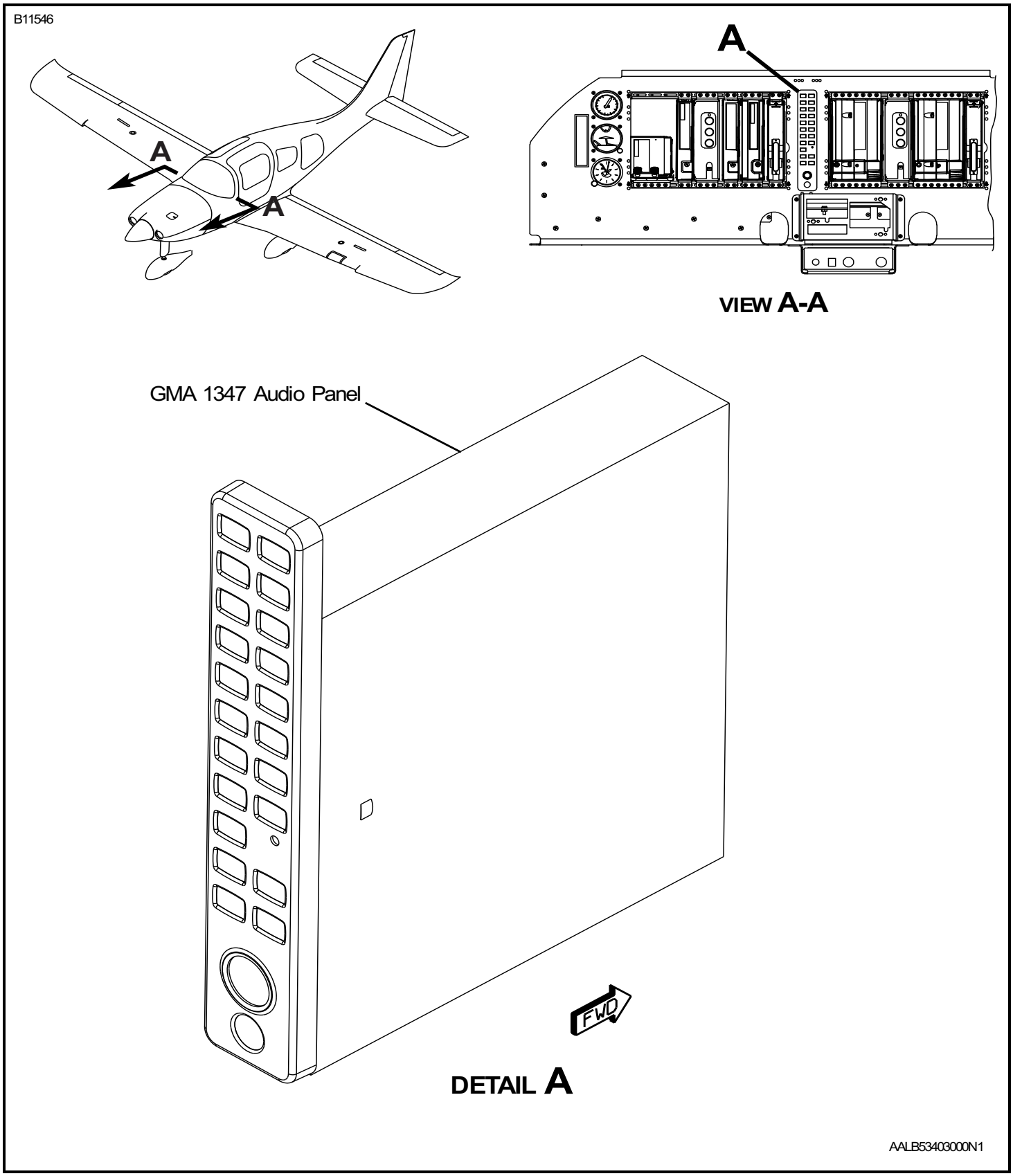
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Figure 207 : Sheet 1 : GDU1040 PFD/GDU 1042 MFD (GDU 1040 PFD shown)



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Figure 208 : Sheet 1 : GMA 1347 Audio Panel



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Figure 209 : Sheet 1 : GIA 63/GIA 63W Integrated Avionics Unit

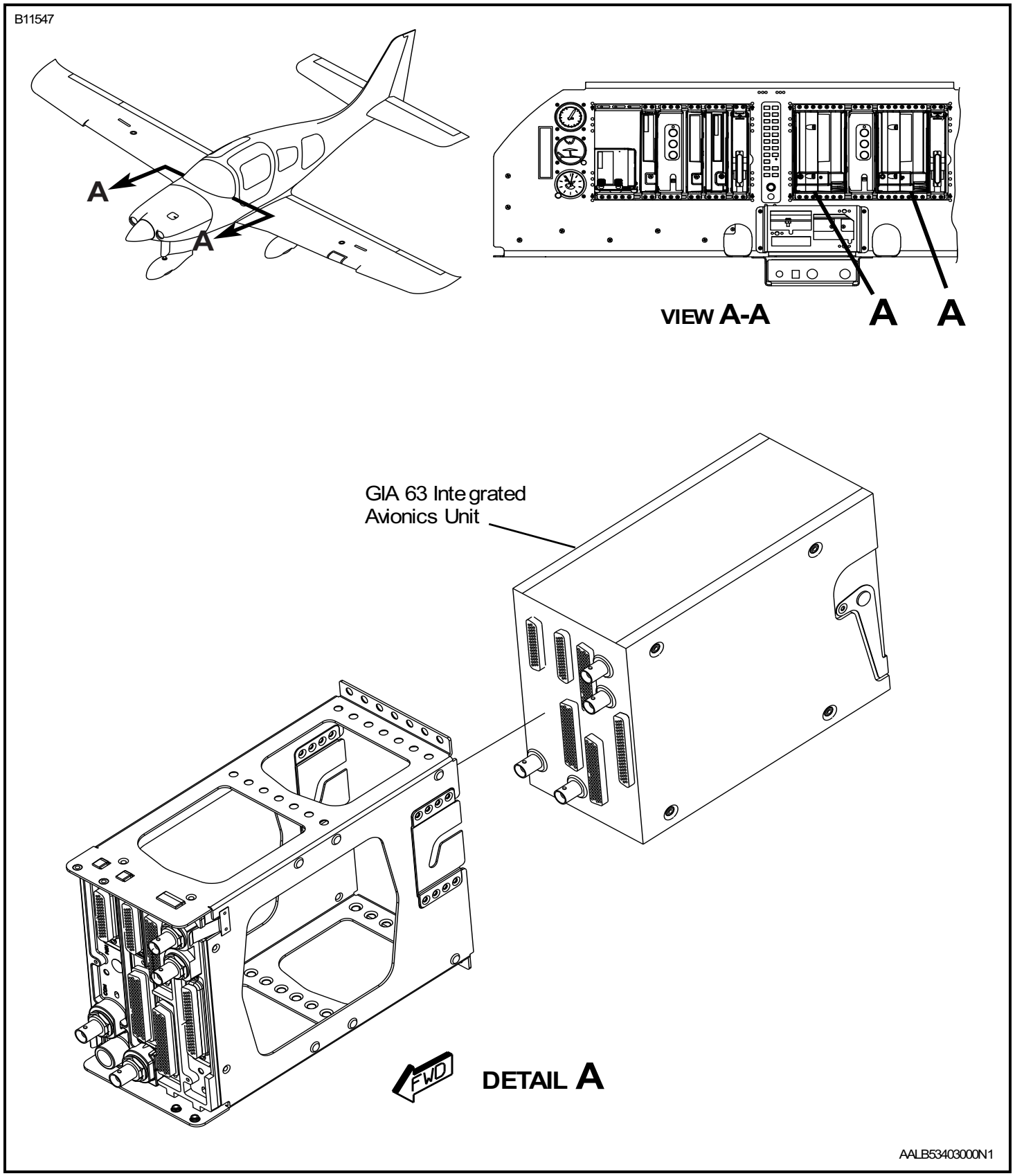
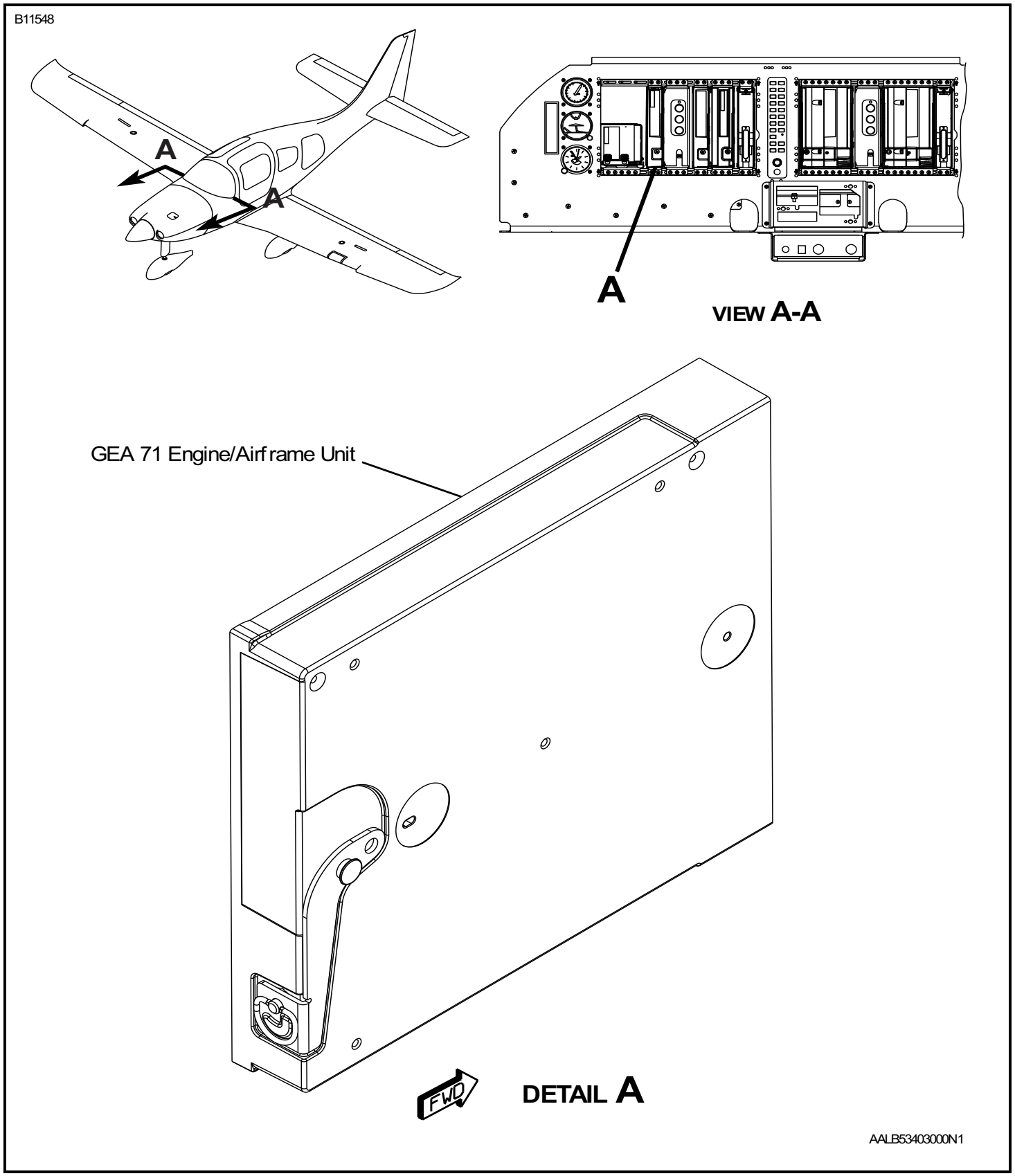


Figure 210 : Sheet 1 : GEA 71 Engine/Airframe Unit



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Figure 211 : Sheet 1 : GTX 33 Transponder

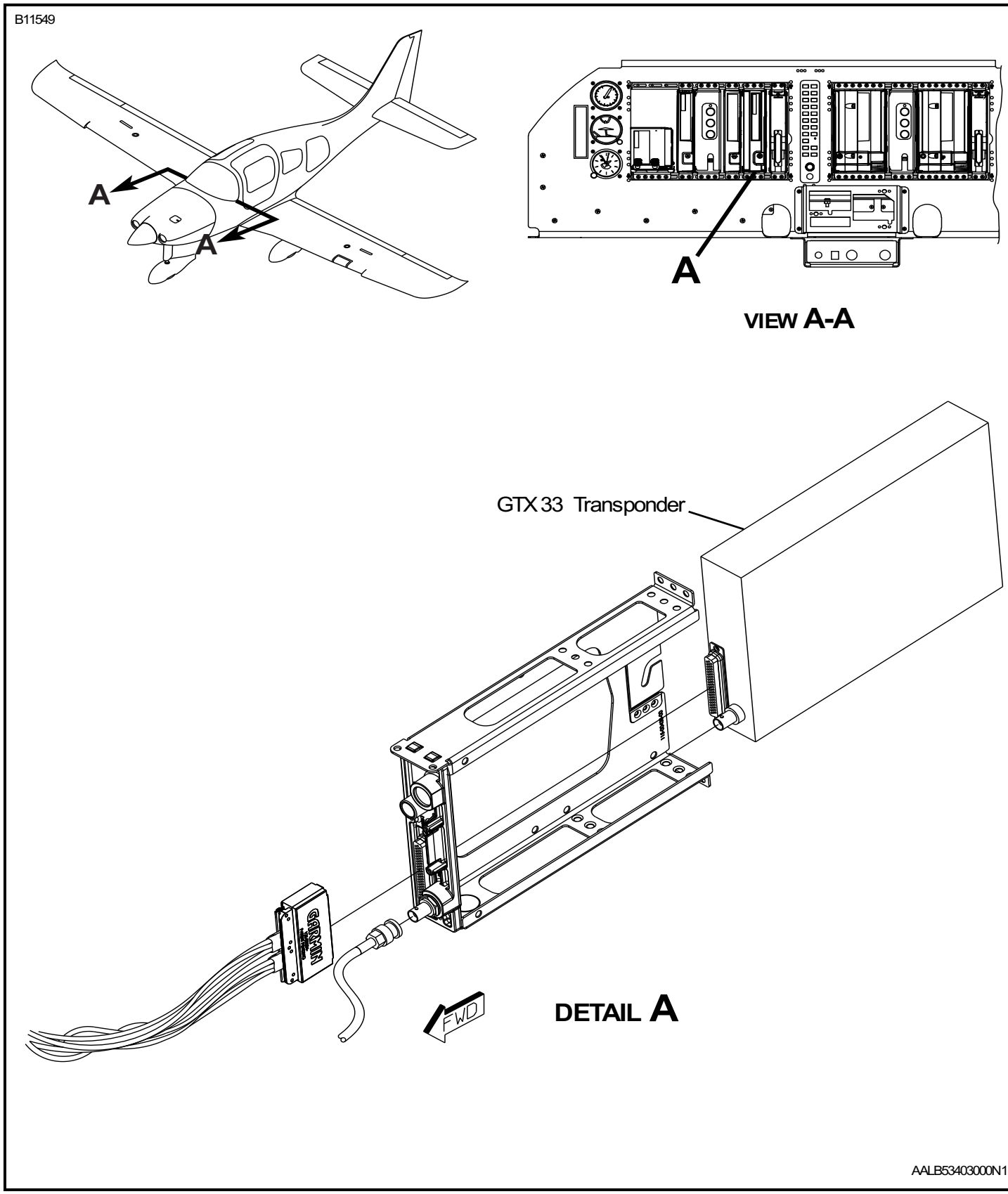
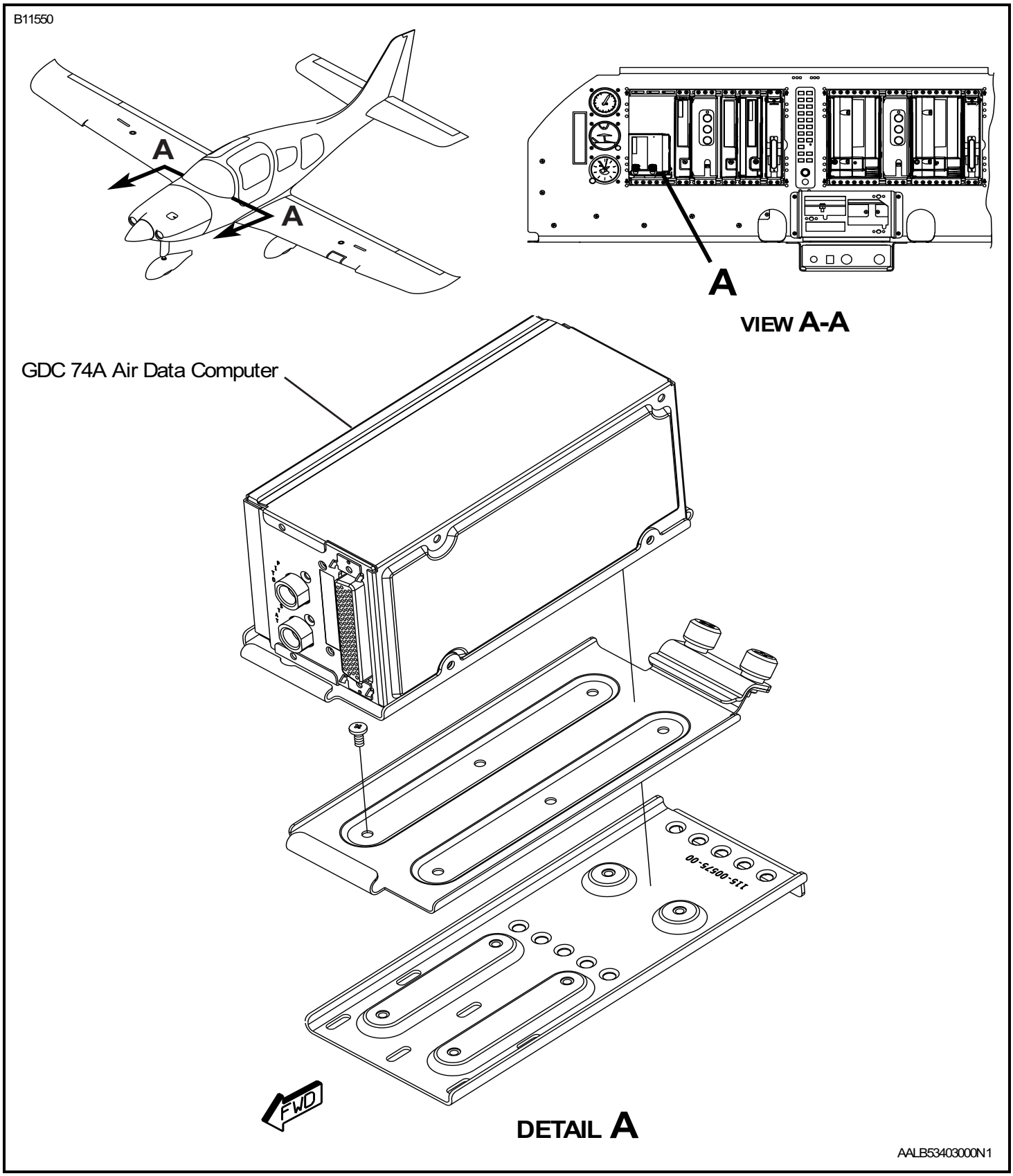
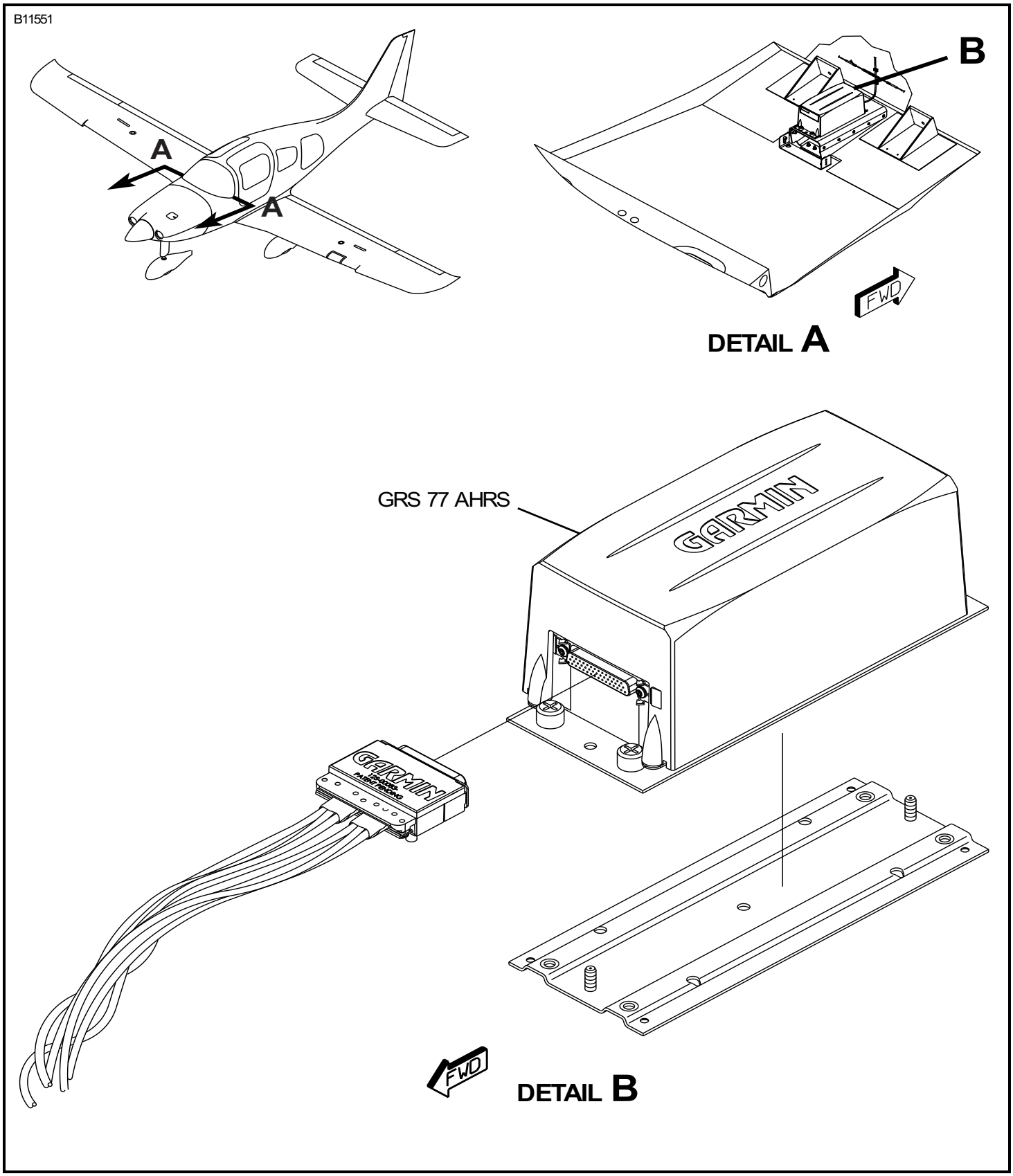


Figure 212 : Sheet 1 : GDC 74A Air Data Computer



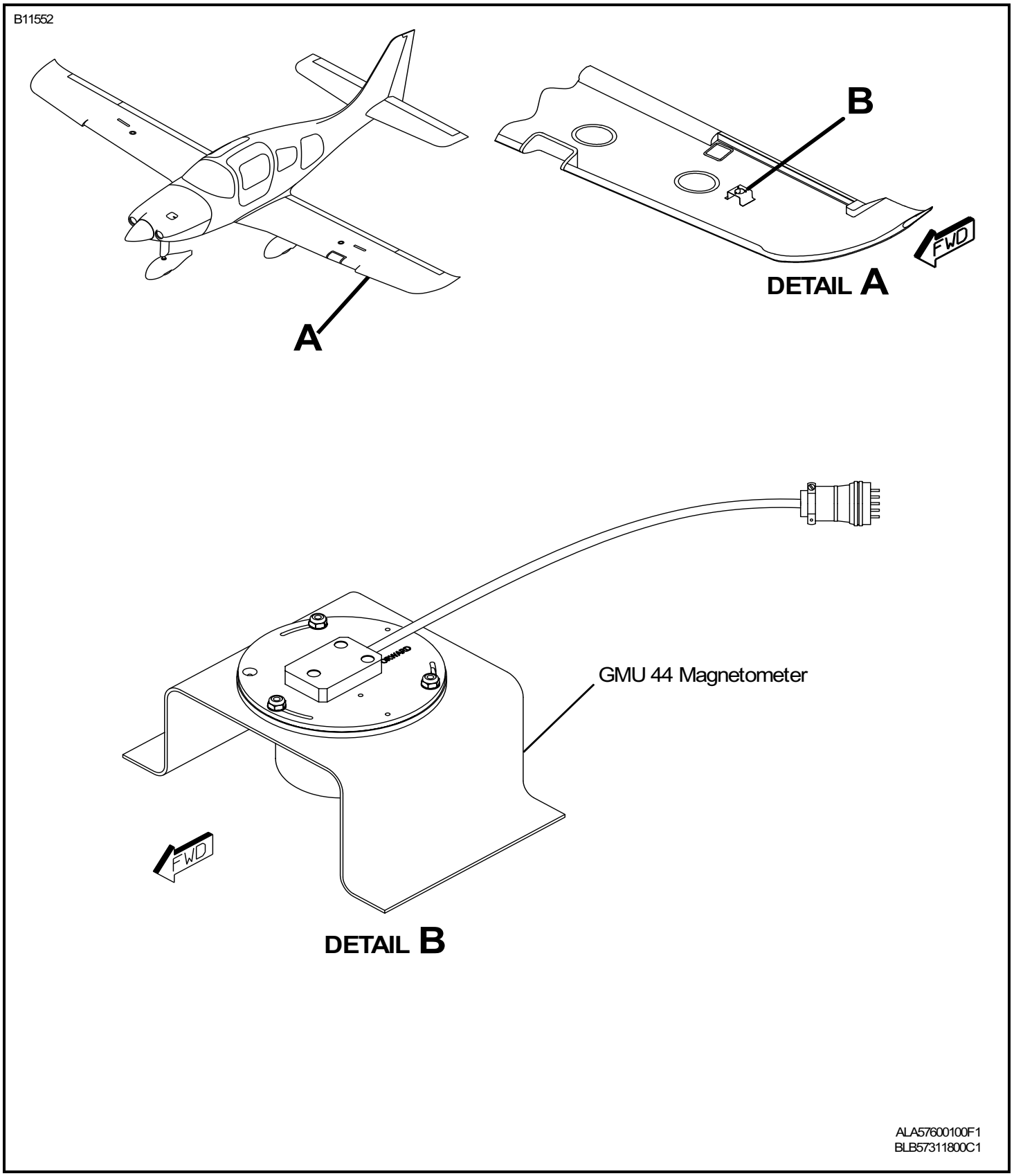
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Figure 213 : Sheet 1 : GRS 77 AHRS



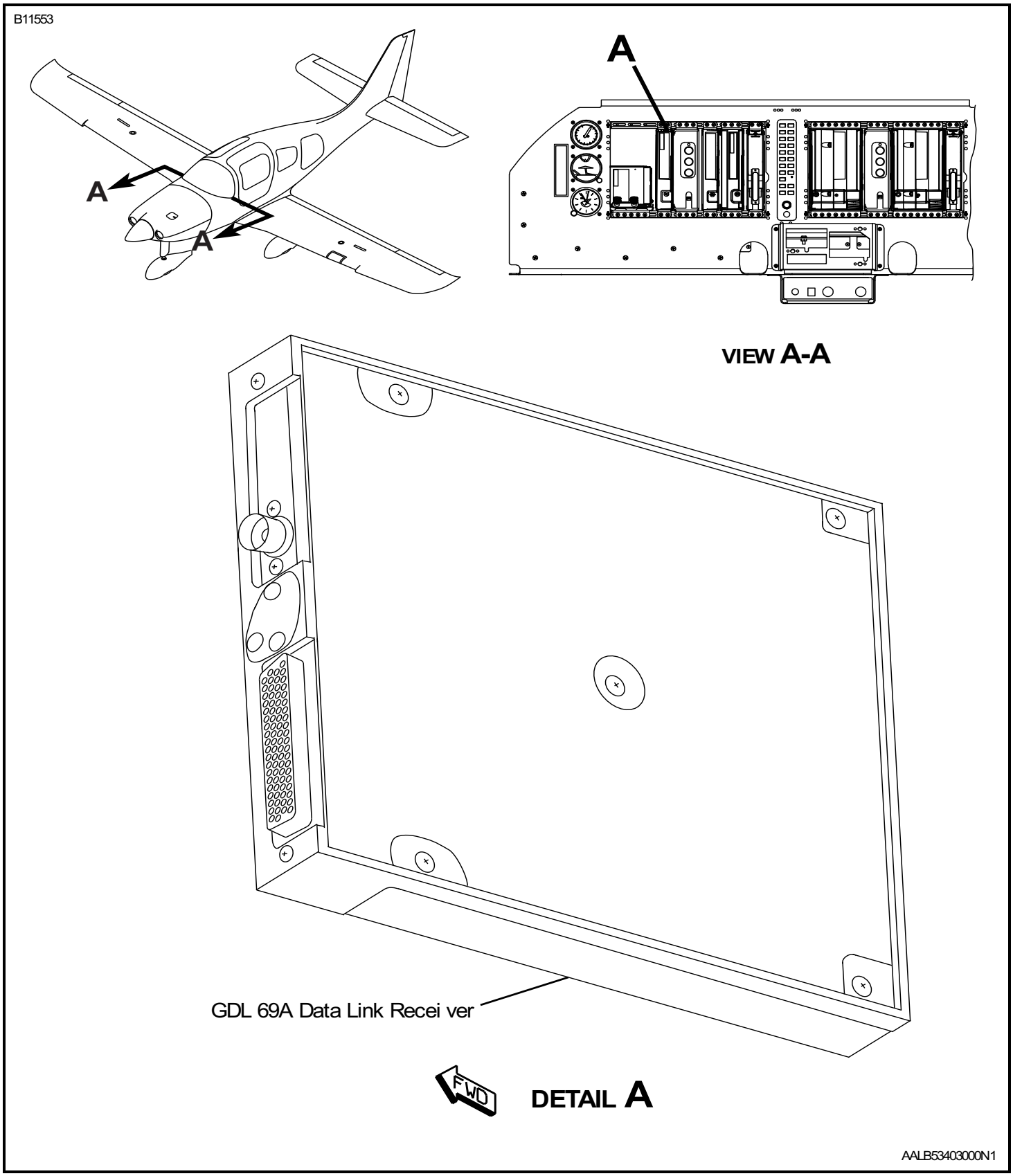
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Figure 214 : Sheet 1 : GMU 44 Magnetometer



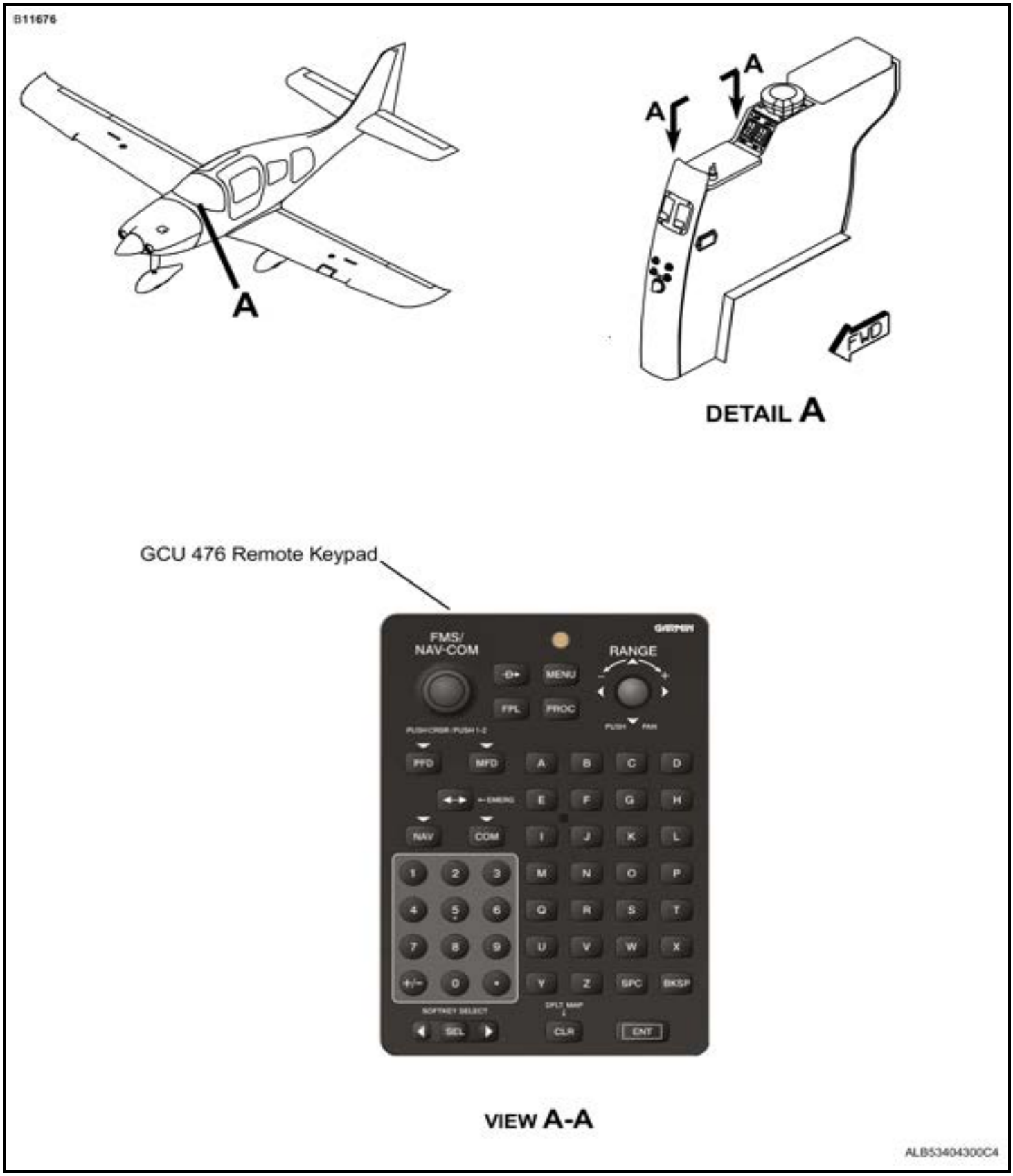
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Figure 215 : Sheet 1 : GDL 69A Data Link Receiver



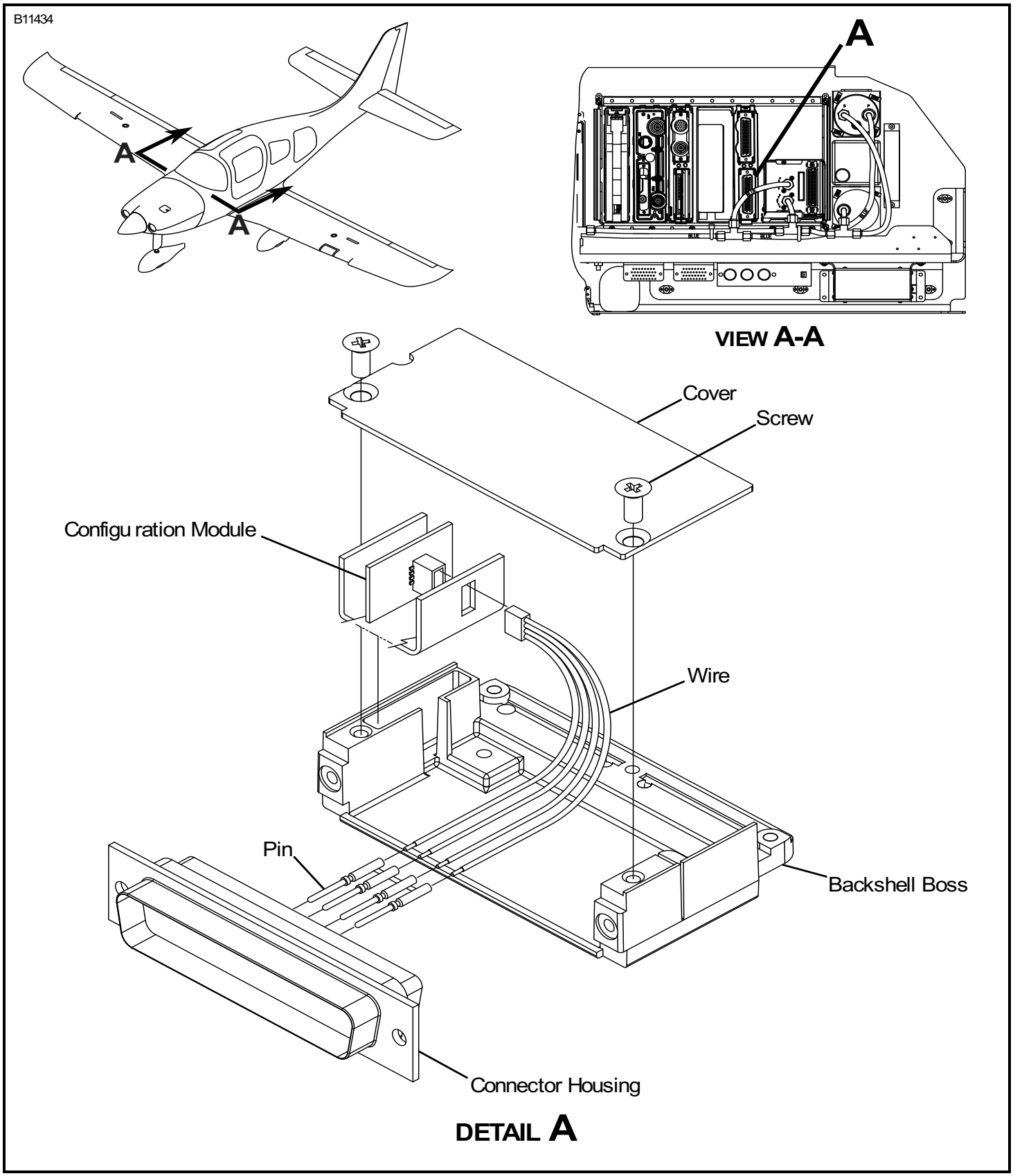
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Figure 216 : Sheet 1 : GCU 476 Remote Keypad



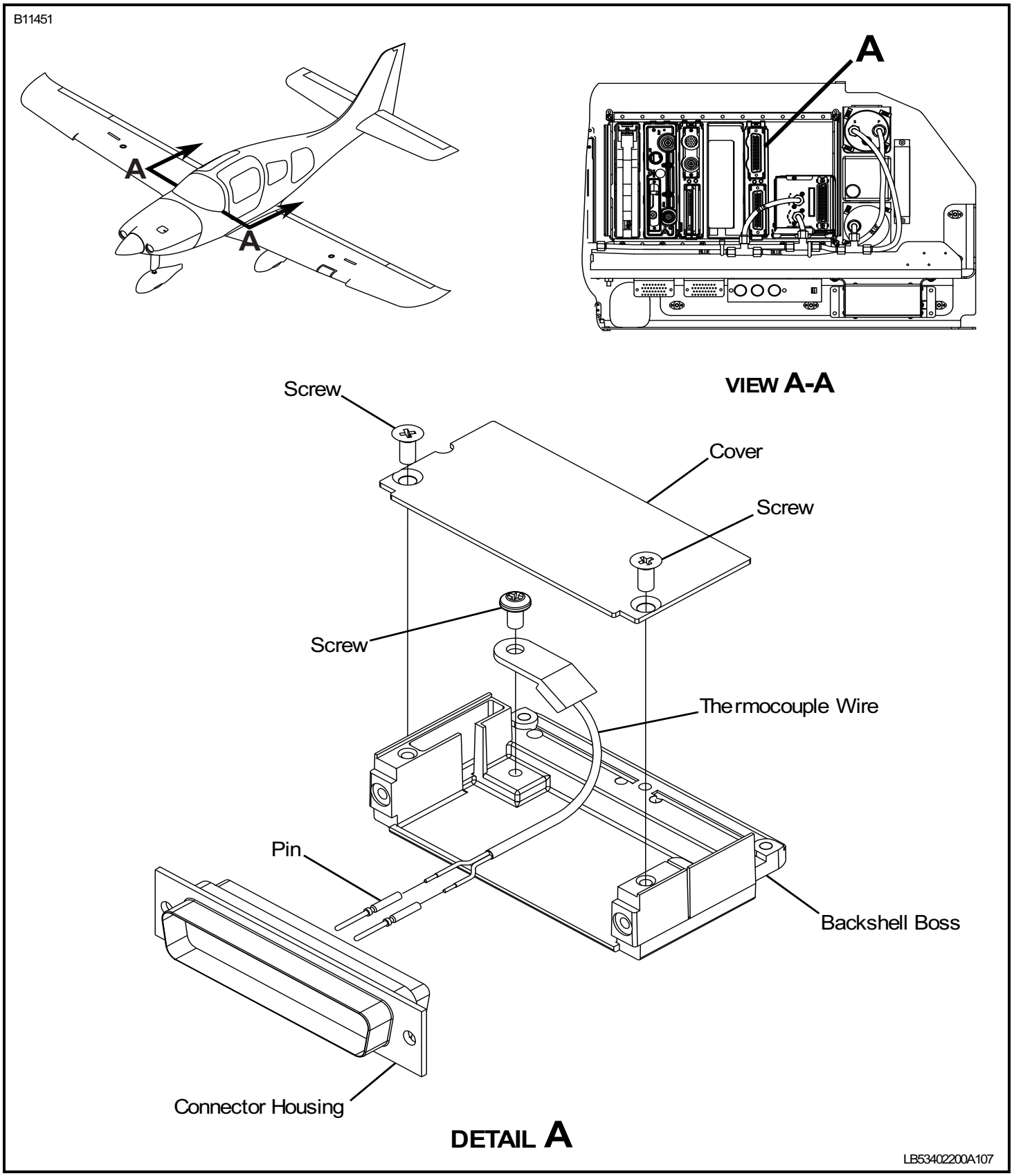
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Figure 217 : Sheet 1 : Configuration Module Installation



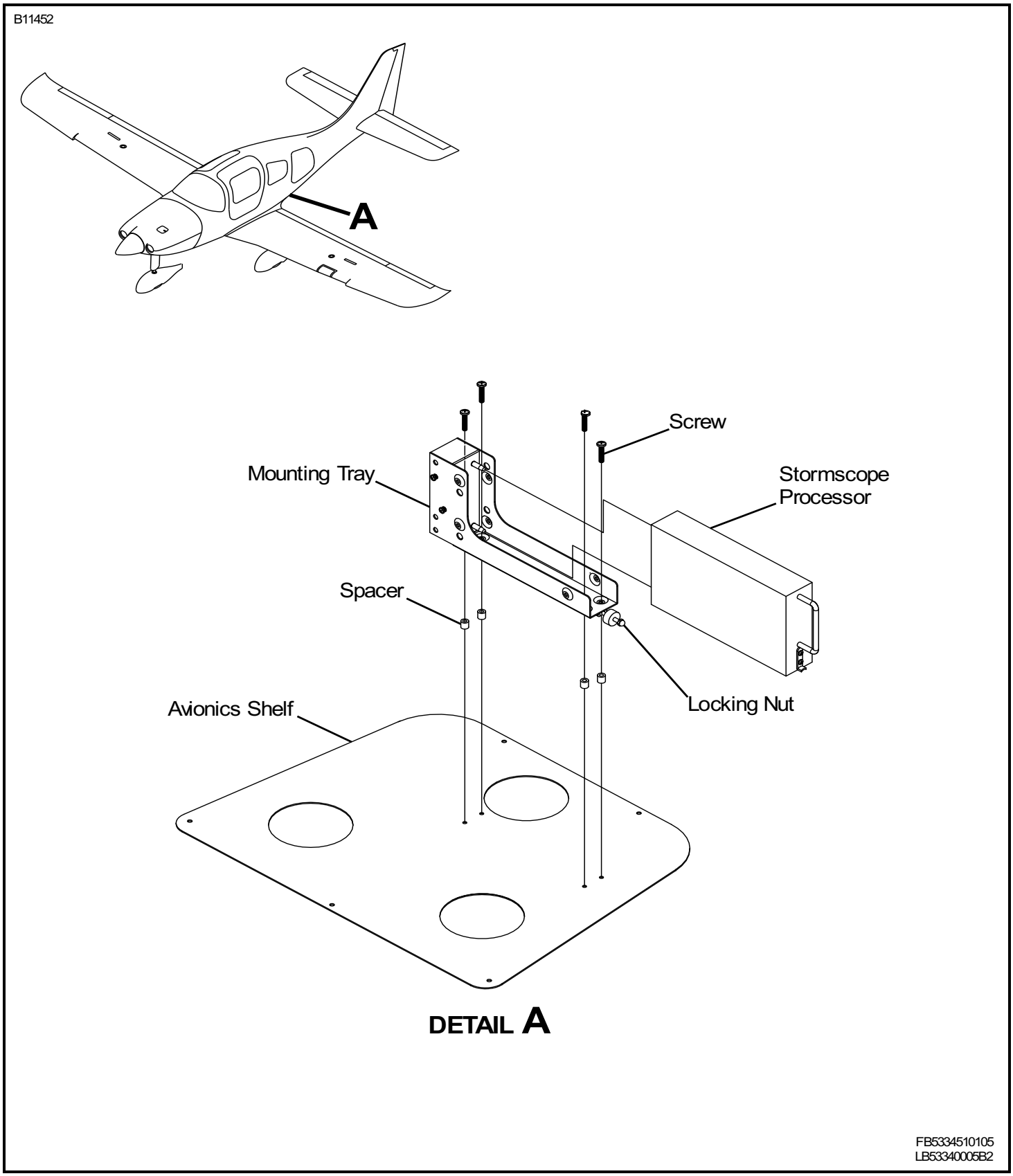
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Figure 218 : Sheet 1 : GEA Backshell Thermocouple



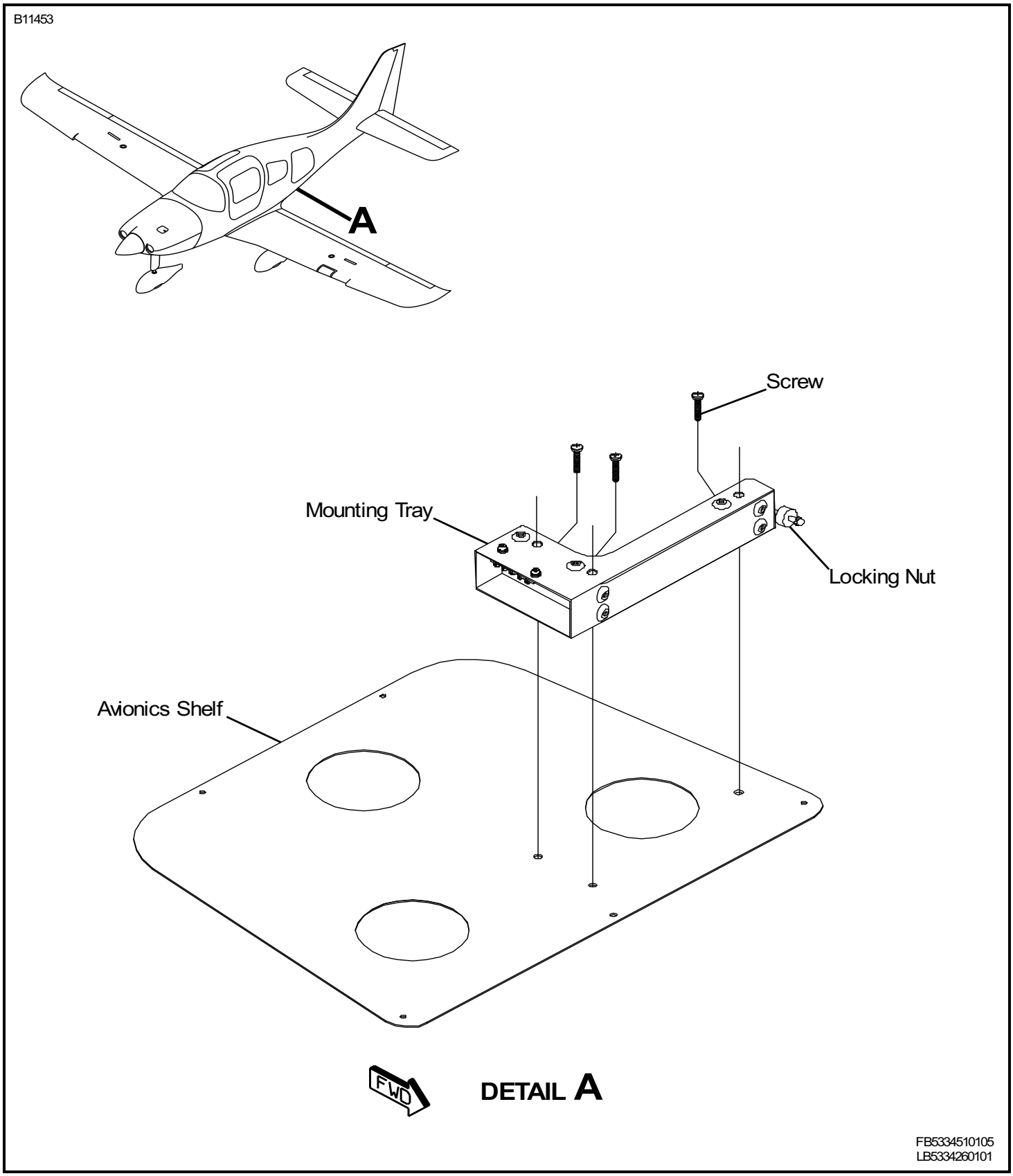
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Figure 219 : Sheet 1 : Stormscope Processor and Mounting Tray (S/N 42002 to 42500)



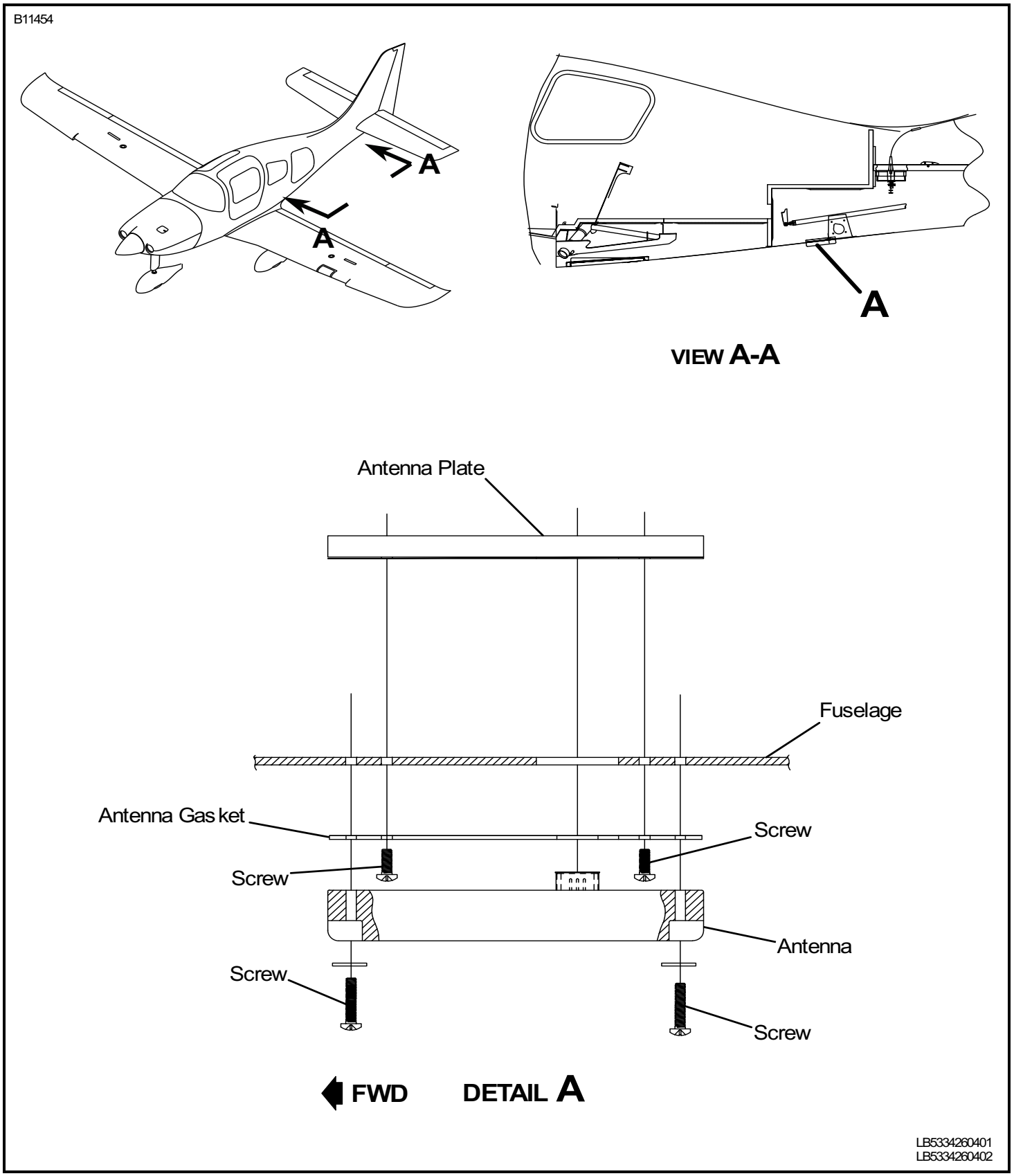
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Figure 220 : Sheet 1 : Stormscope Mounting Tray (S/N 42501 and On)



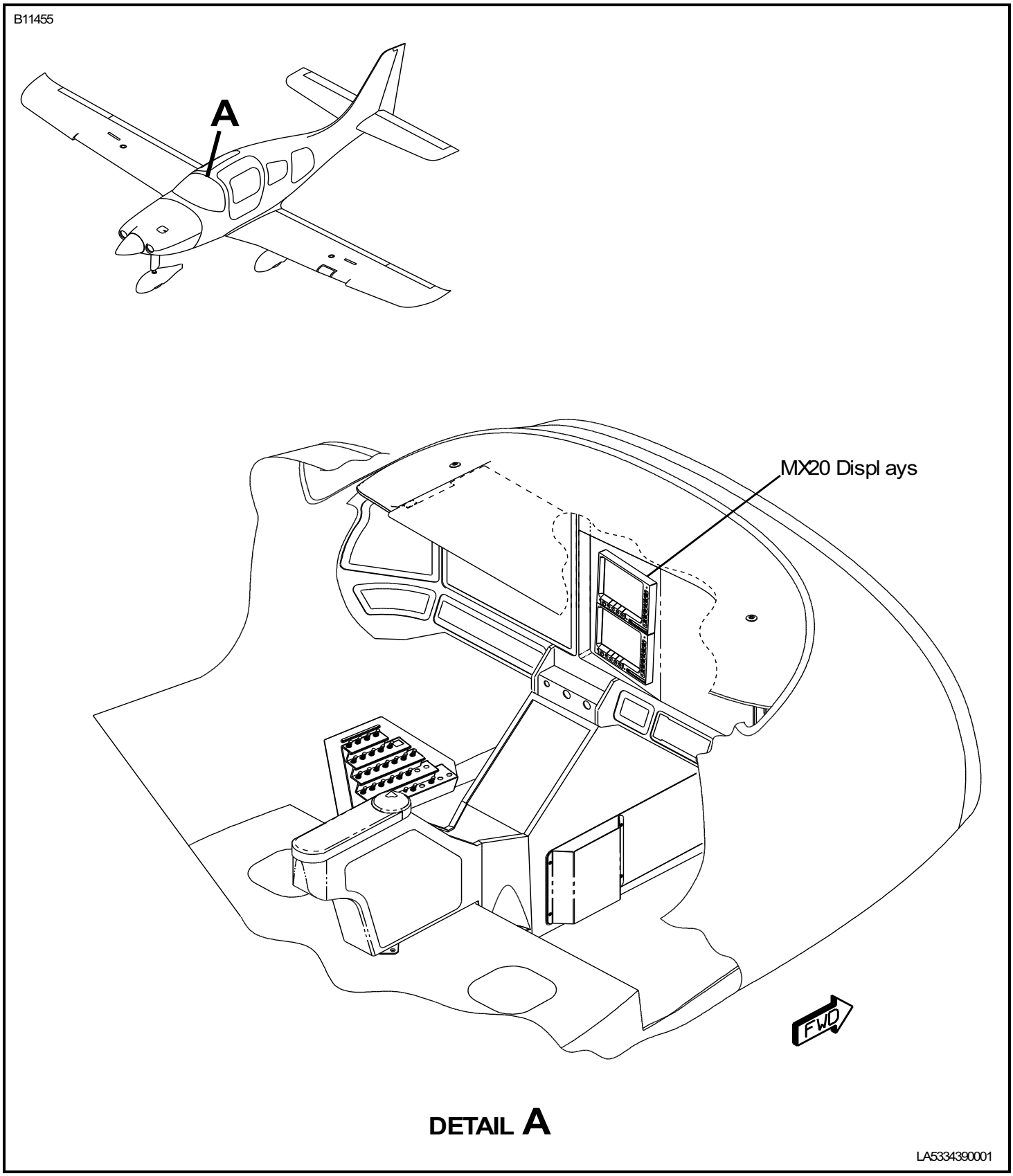
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Figure 221 : Sheet 1 : Stormscope Antenna Installation



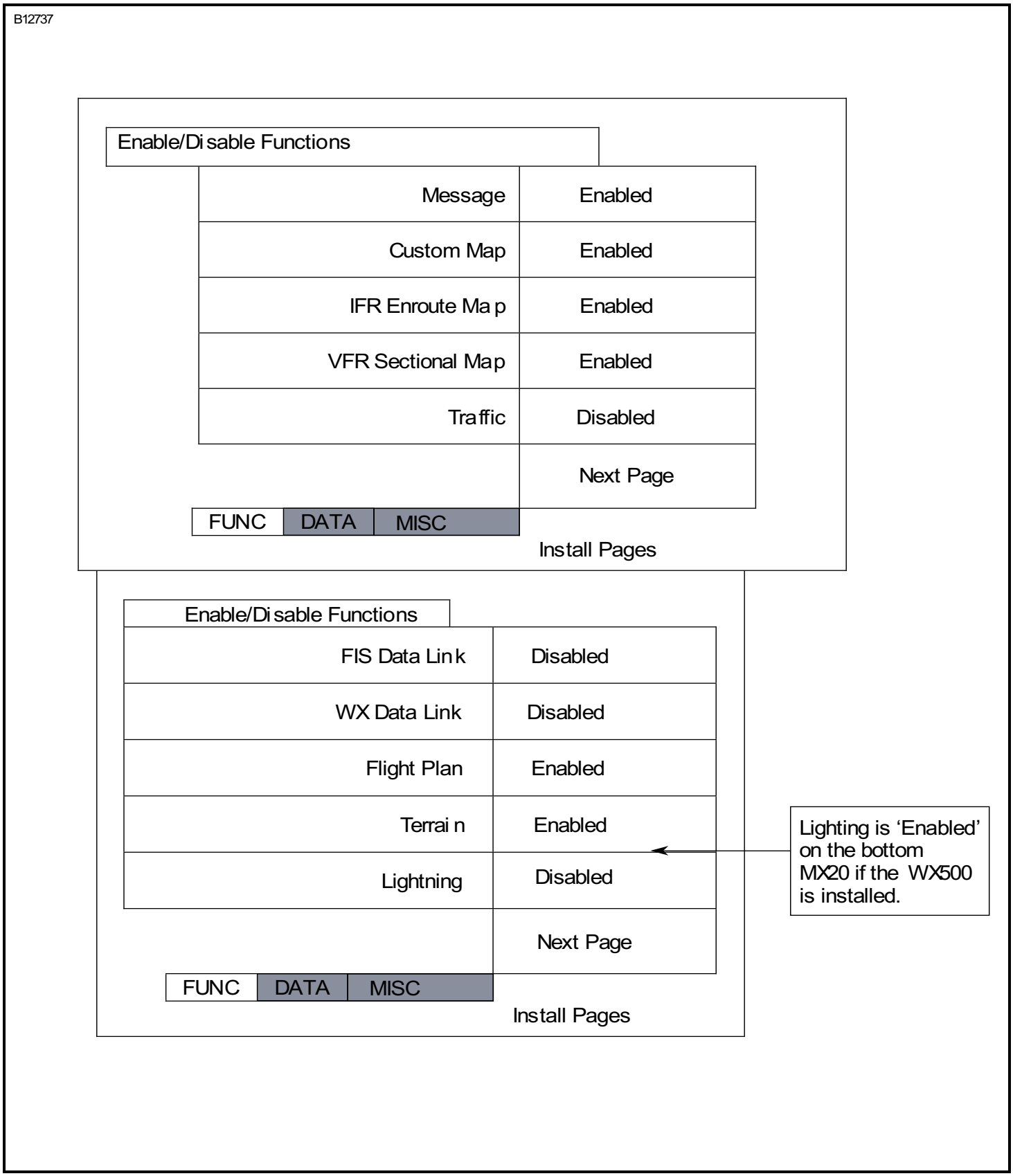
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Figure 222 : Sheet 1 : MX20 Displays



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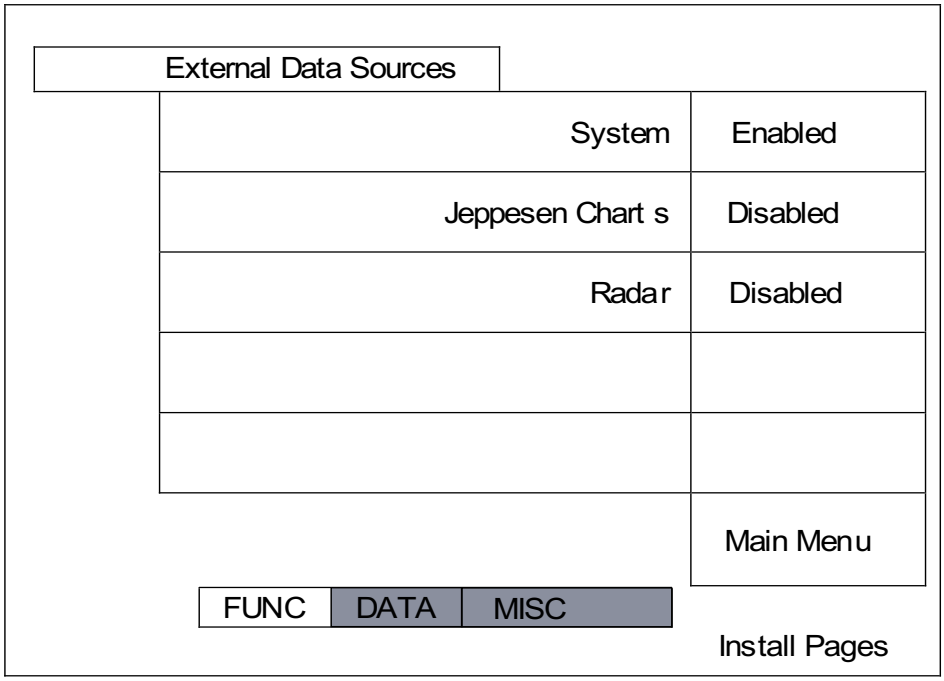
Figure 223 : Sheet 1 : MX20 Enable/Disable Functions Screen



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Figure 223 : Sheet 2 : MX20 Enable/Disable Functions Screen

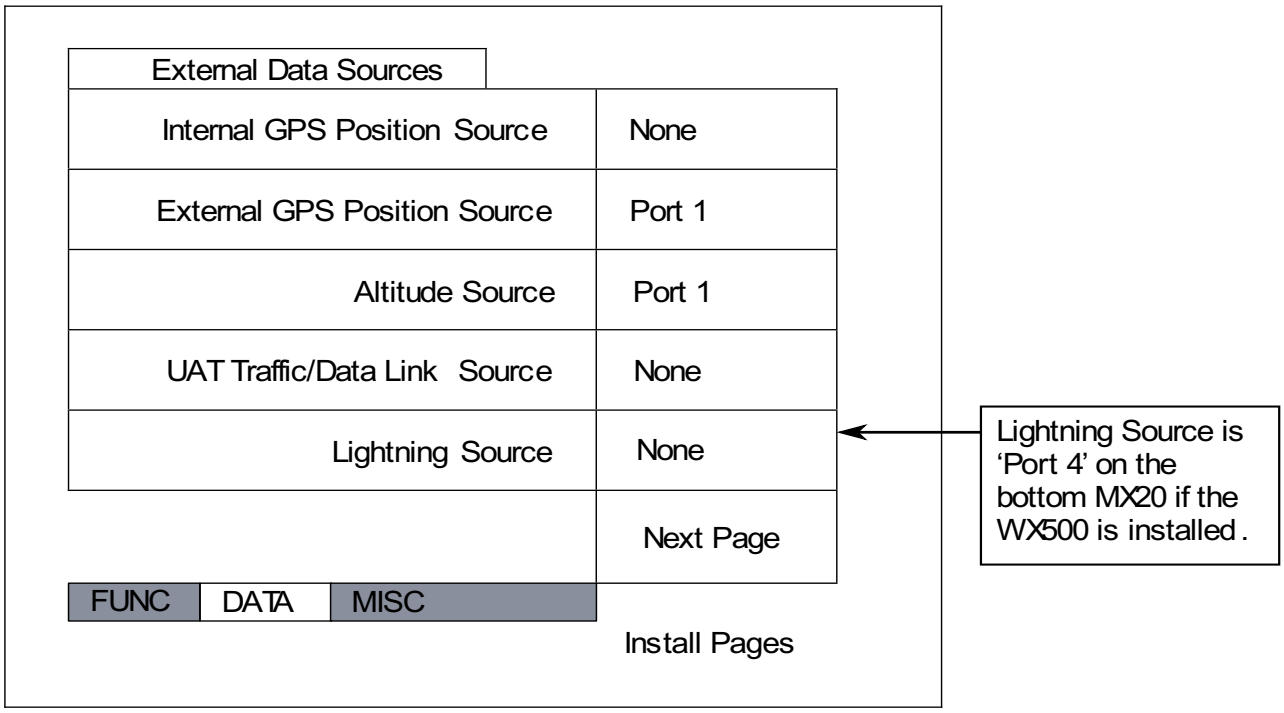
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Figure 224 : Sheet 1 : MX20 External Data Sources Screen

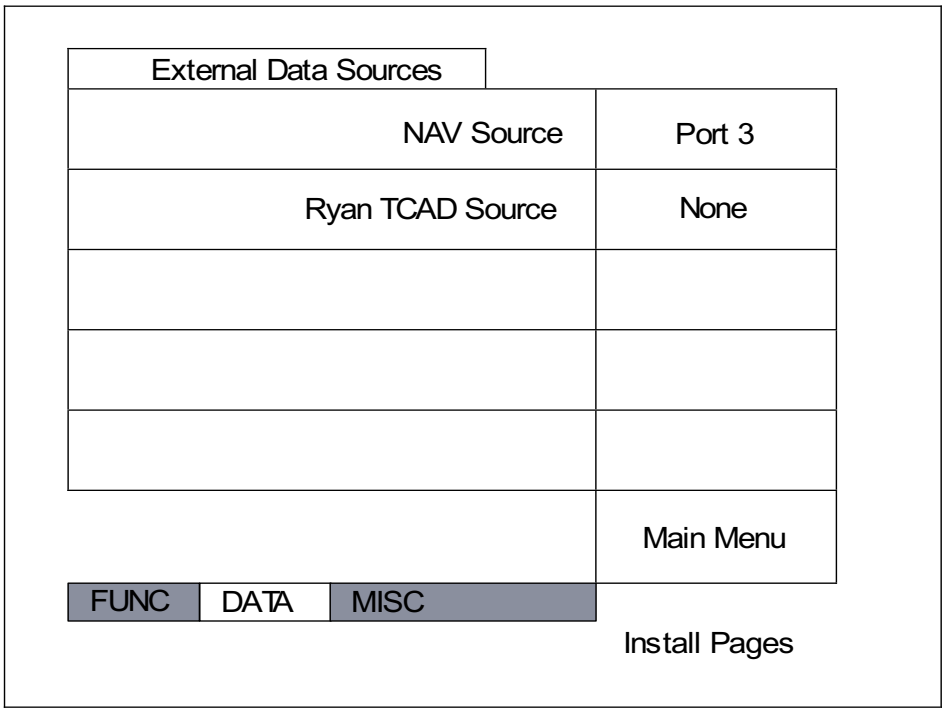
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Figure 224 : Sheet 2 : MX20 External Data Sources Screen

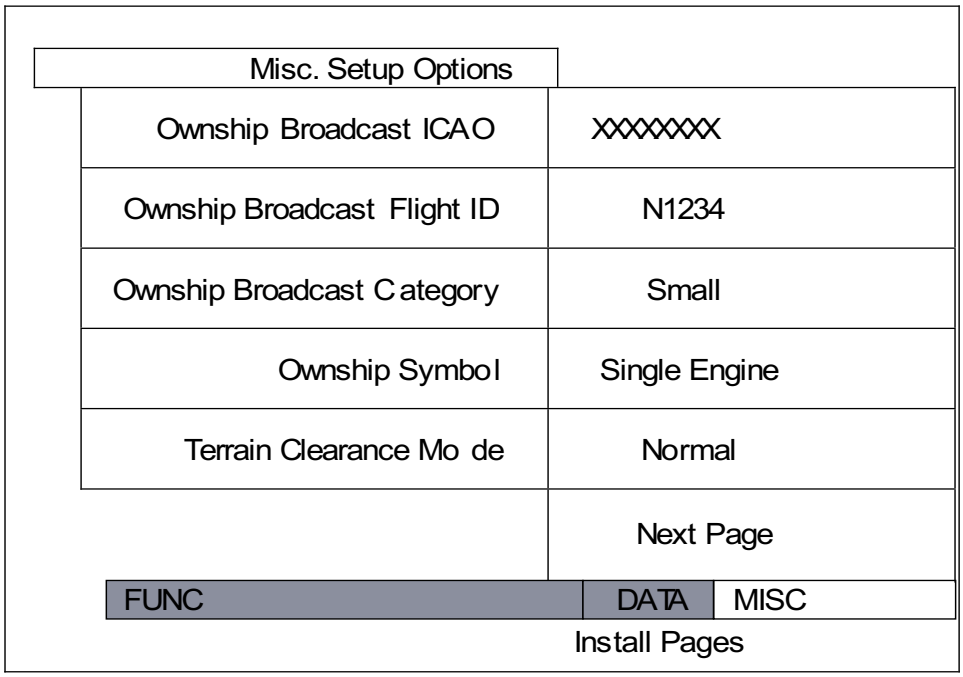
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Figure 225 : Sheet 1 : MX20 Miscellaneous Setup Options

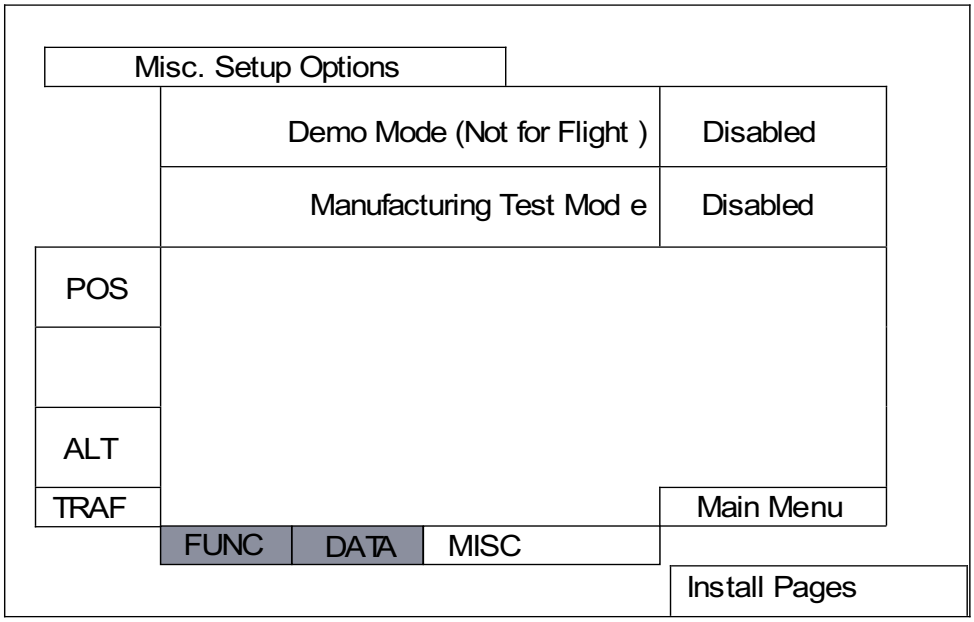
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Figure 225 : Sheet 2 : MX20 Miscellaneous Setup Options

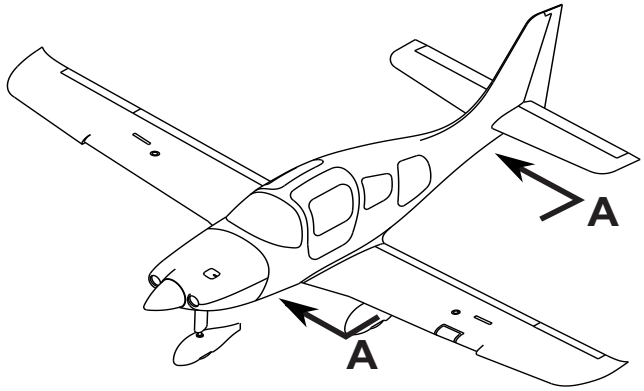
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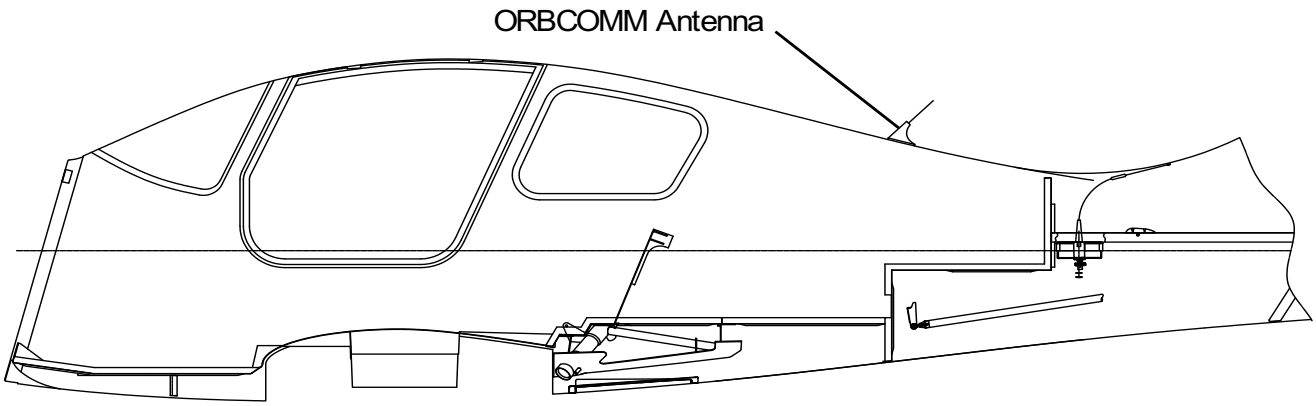


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Figure 226 : Sheet 1 : ORBCOMM Antenna Location

B11456



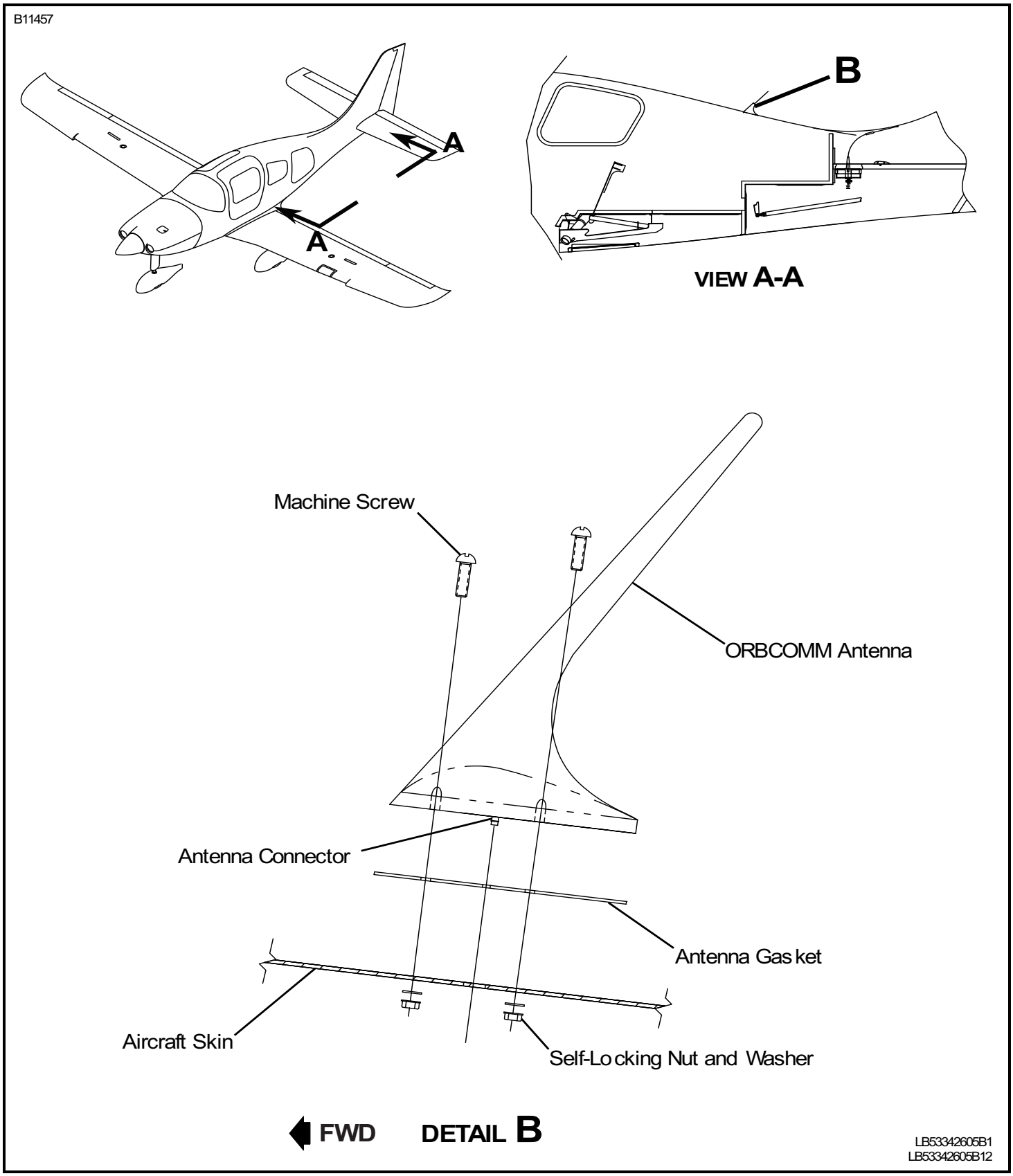


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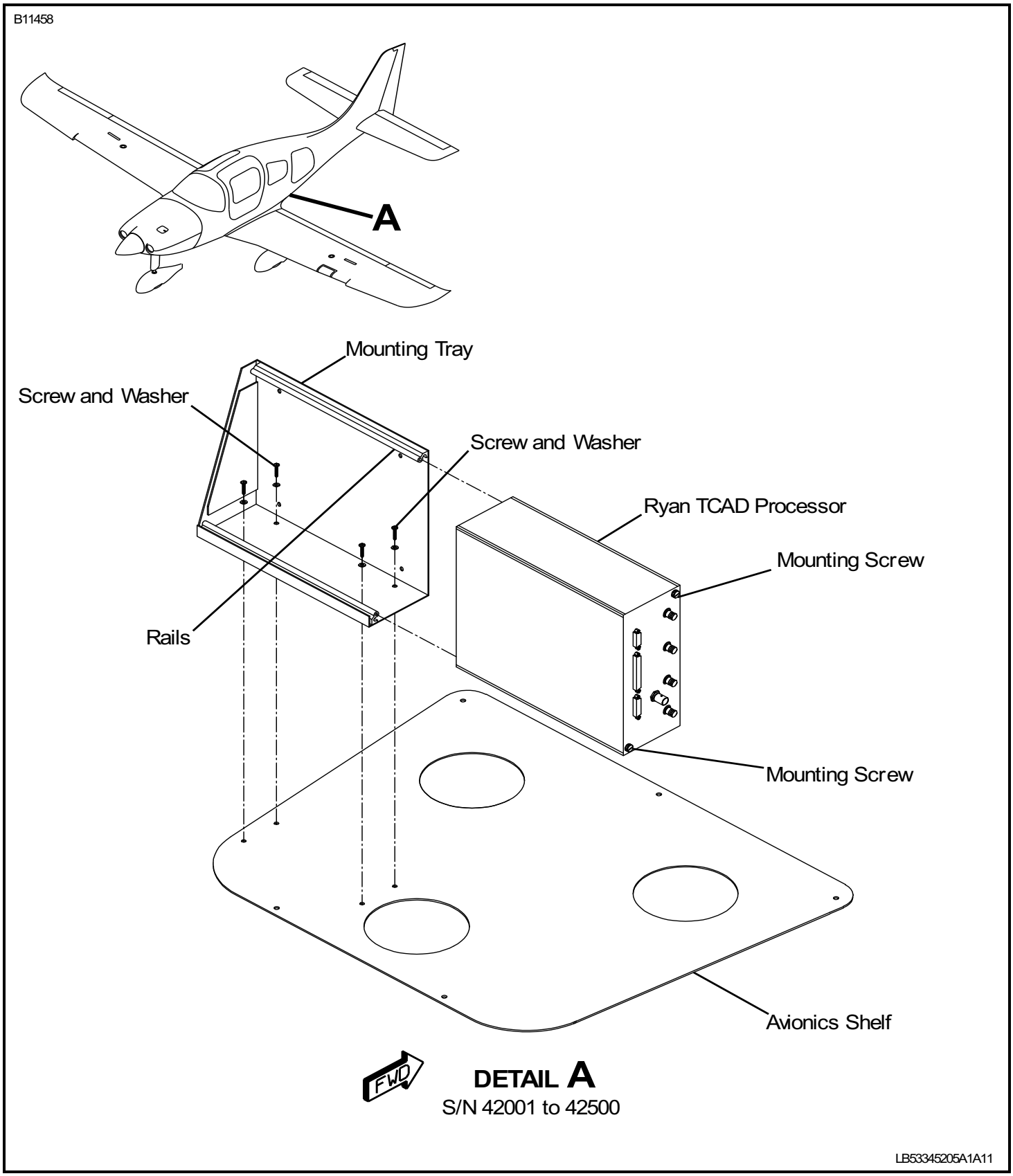
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Figure 227 : Sheet 1 : ORBCOMM Antenna



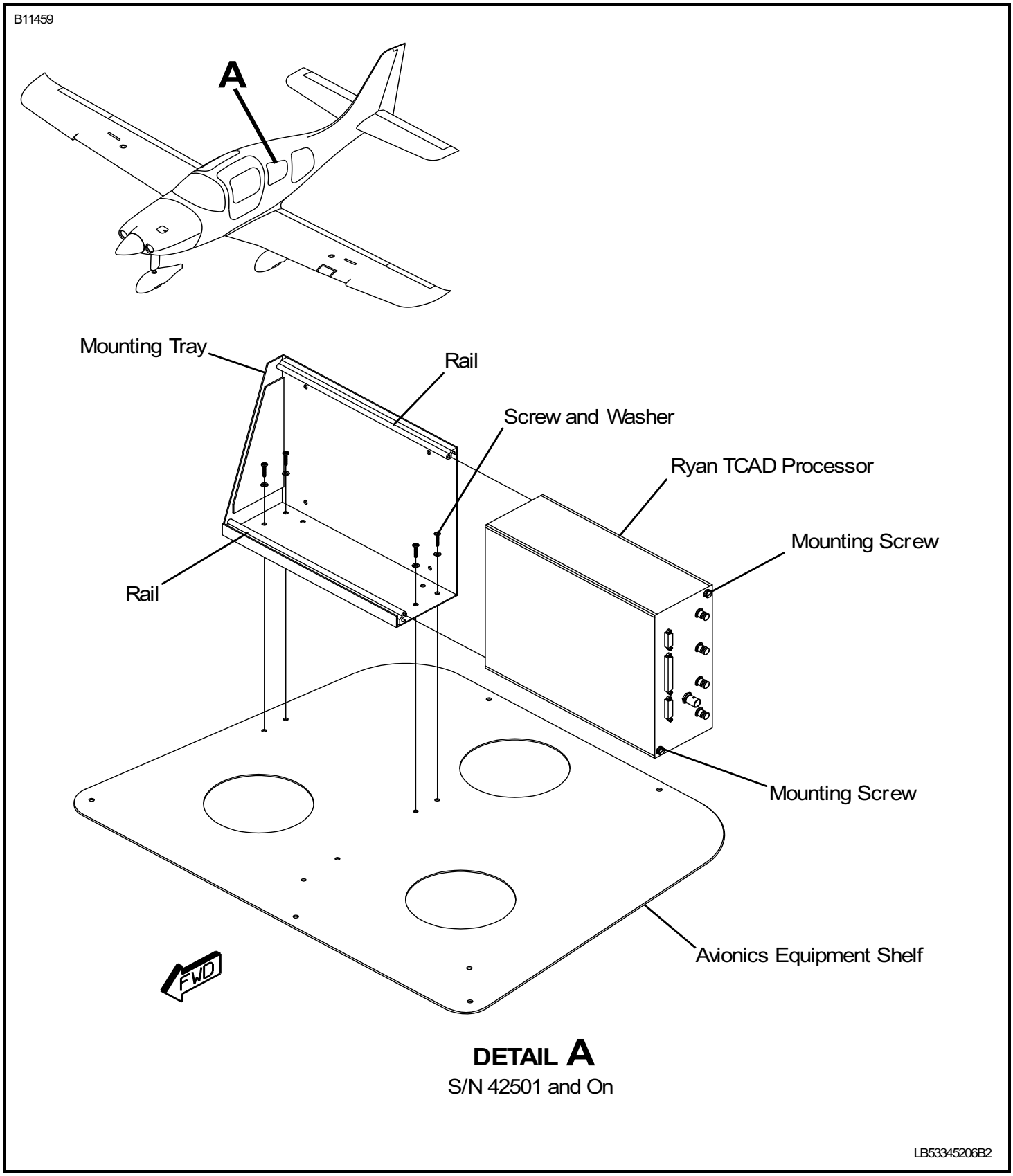
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Figure 228 : Sheet 1 : Ryan TCAD Processor



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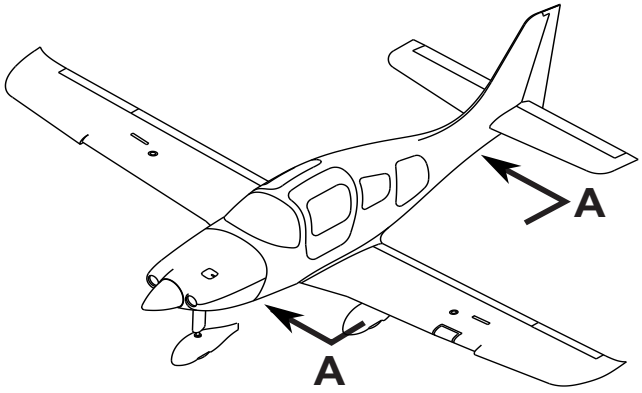
Figure 228 : Sheet 2 : Ryan TCAD Processor

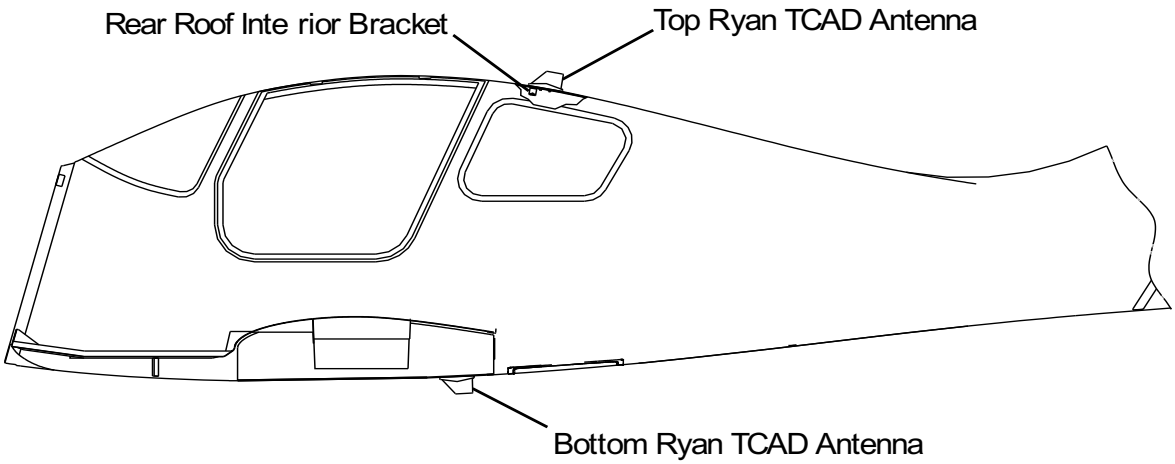


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Figure 229 : Sheet 1 : Ryan TCAD Antenna Locations

B11491



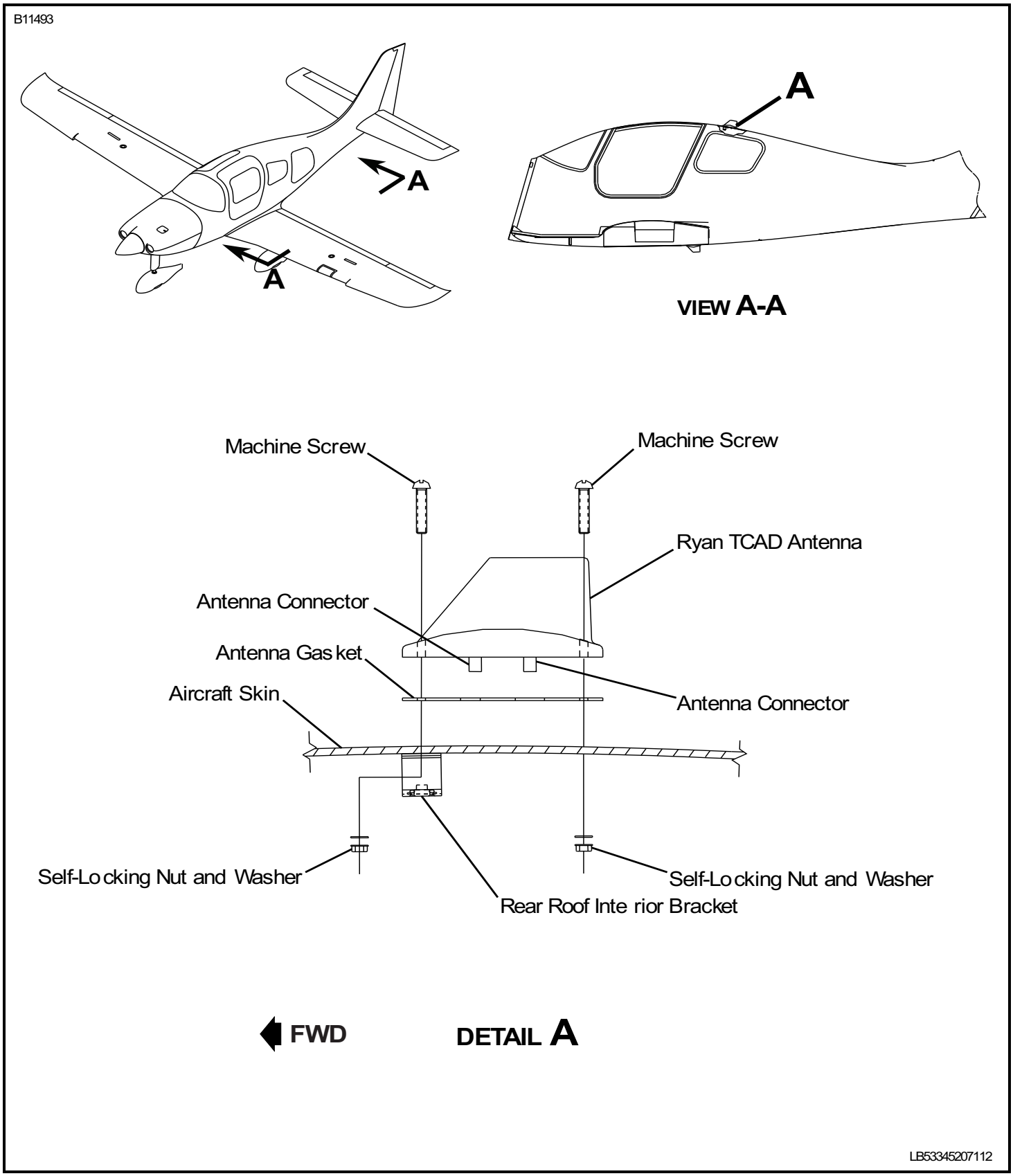


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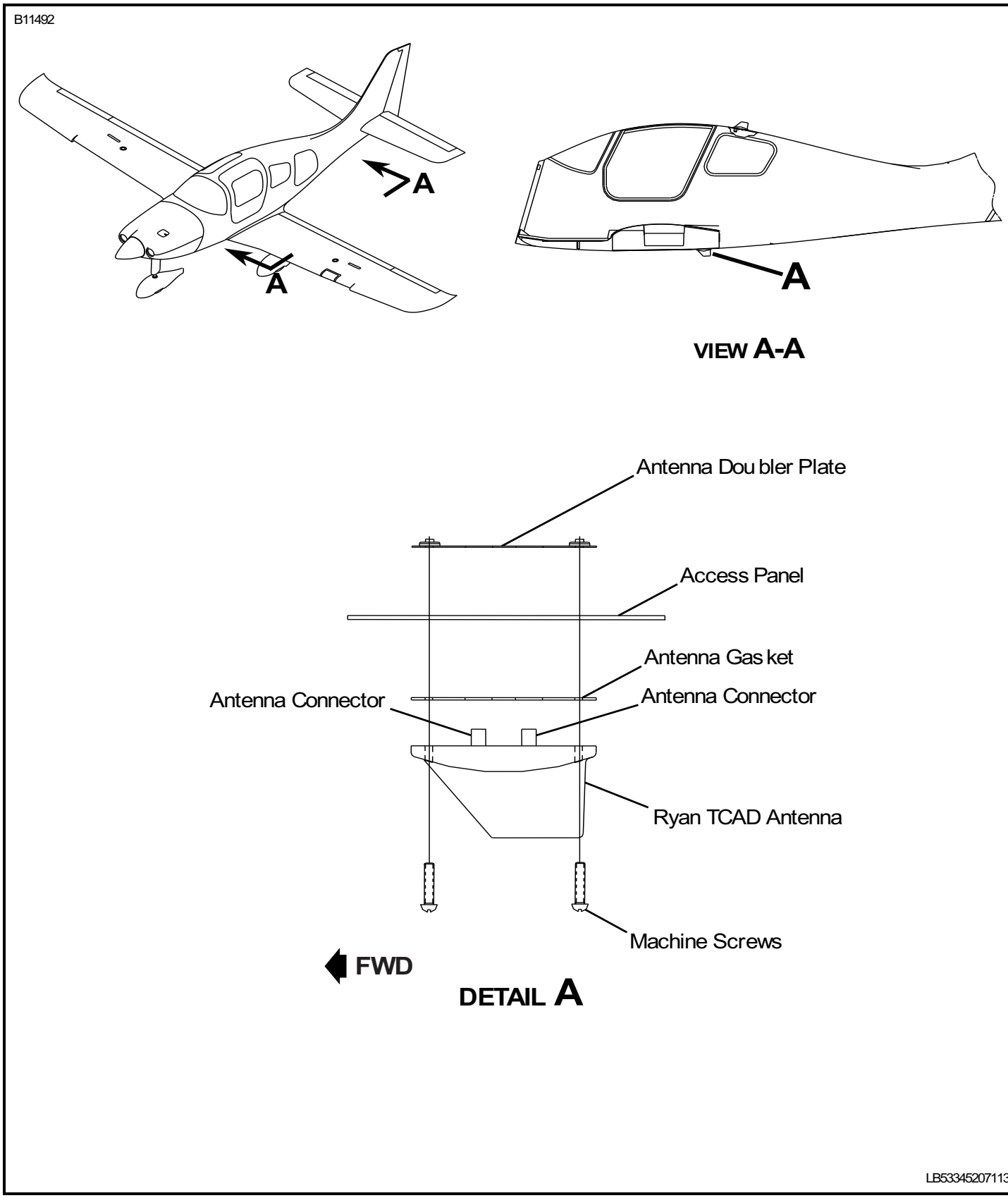
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Figure 230 : Sheet 1 : Top Ryan TCAD Antenna



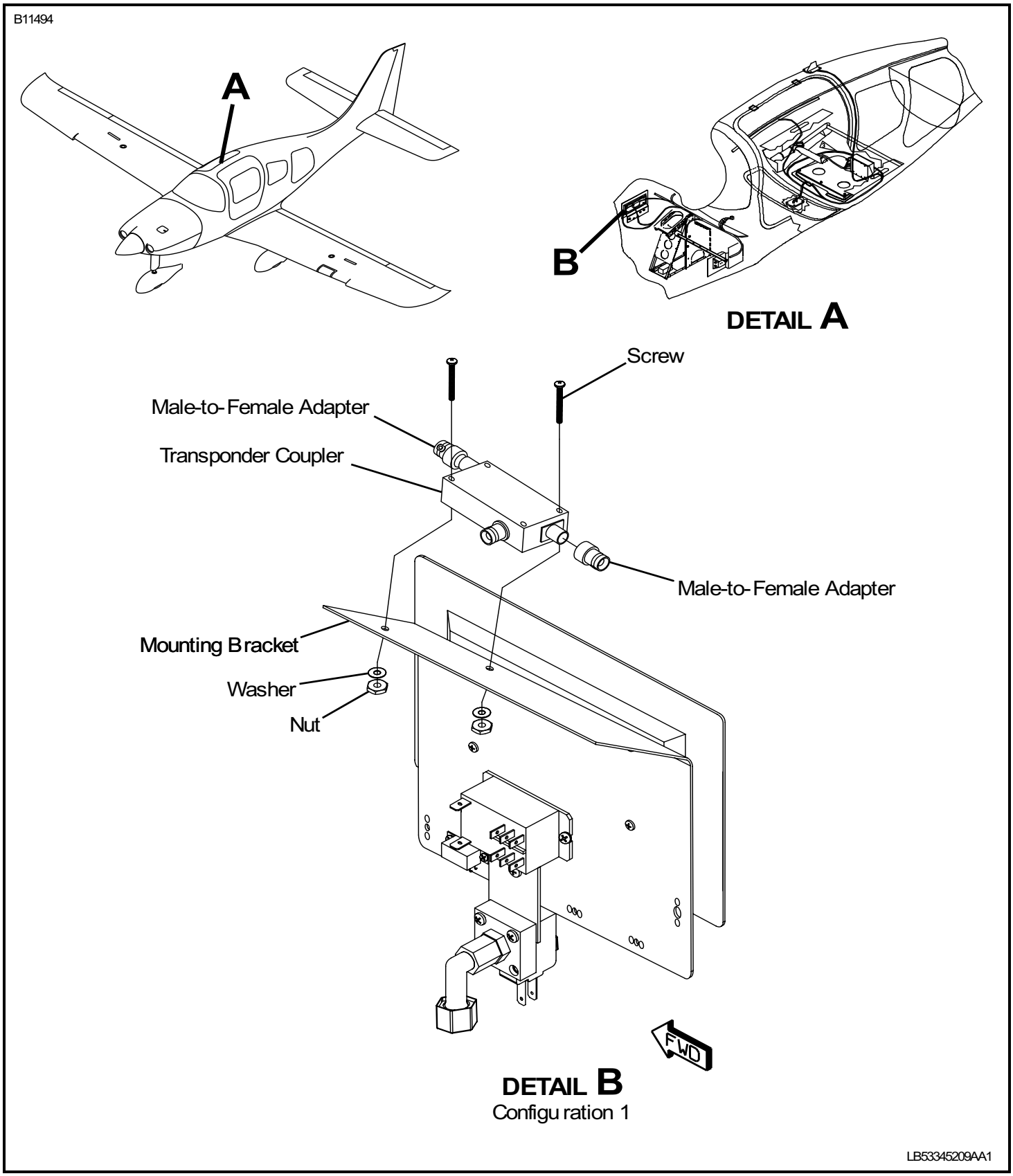
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Figure 231 : Sheet 1 : Bottom Ryan TCAD Antenna



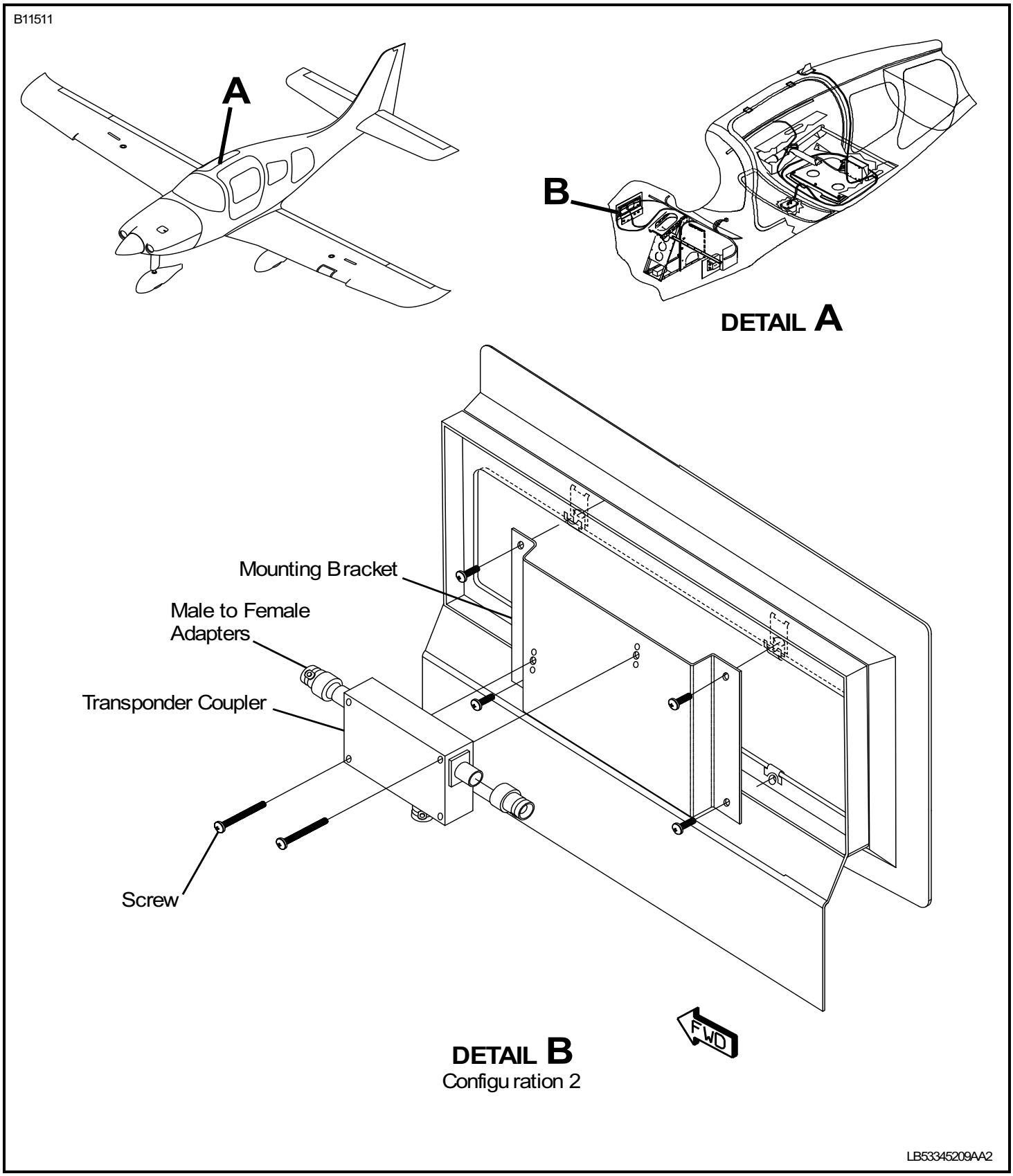
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Figure 232 : Sheet 1 : Ryan TCAD Transponder Coupler (Basic or Avidyne Option)



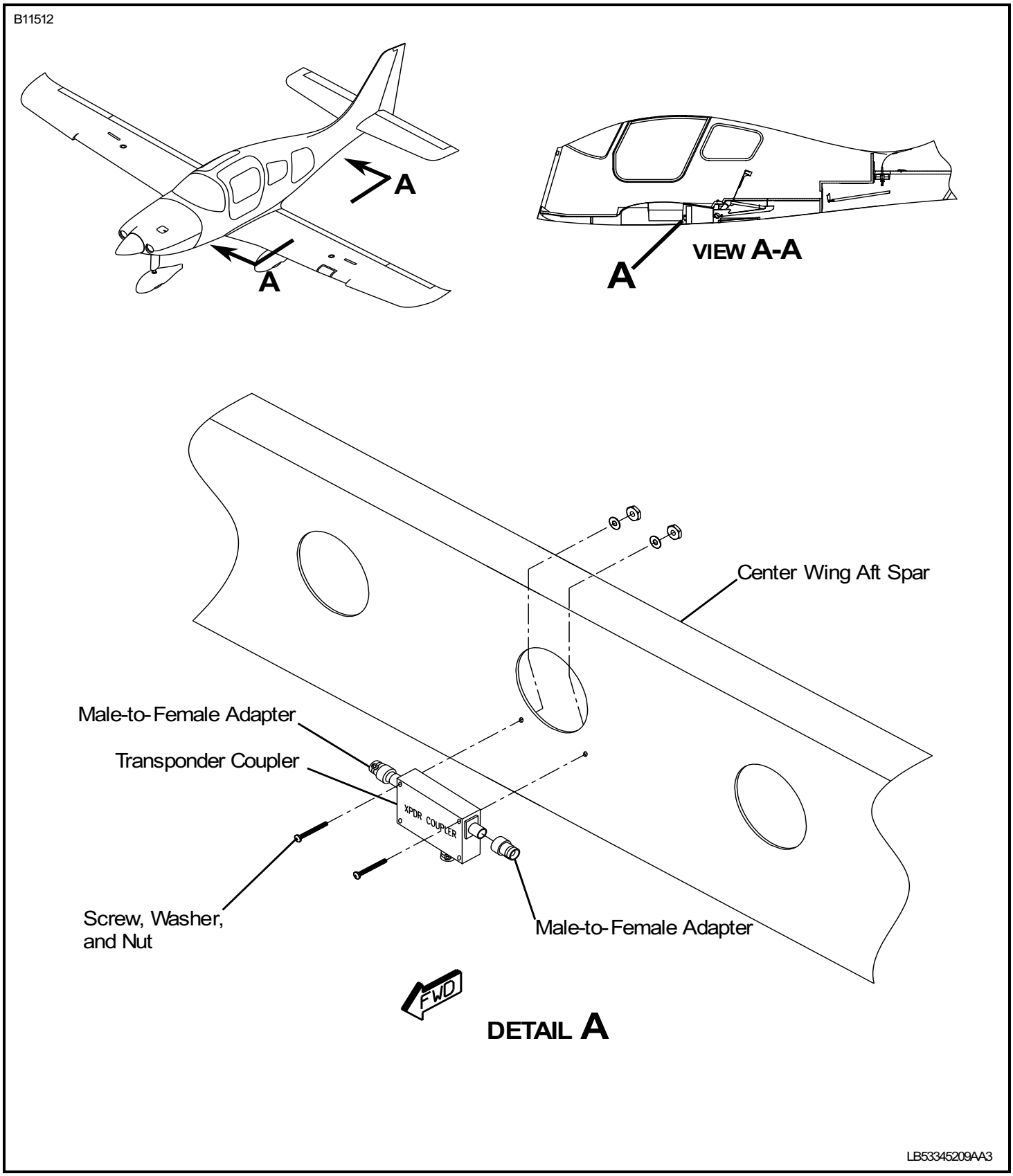
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Figure 232 : Sheet 2 : Ryan TCAD Transponder Coupler (Basic or Avidyne Option)



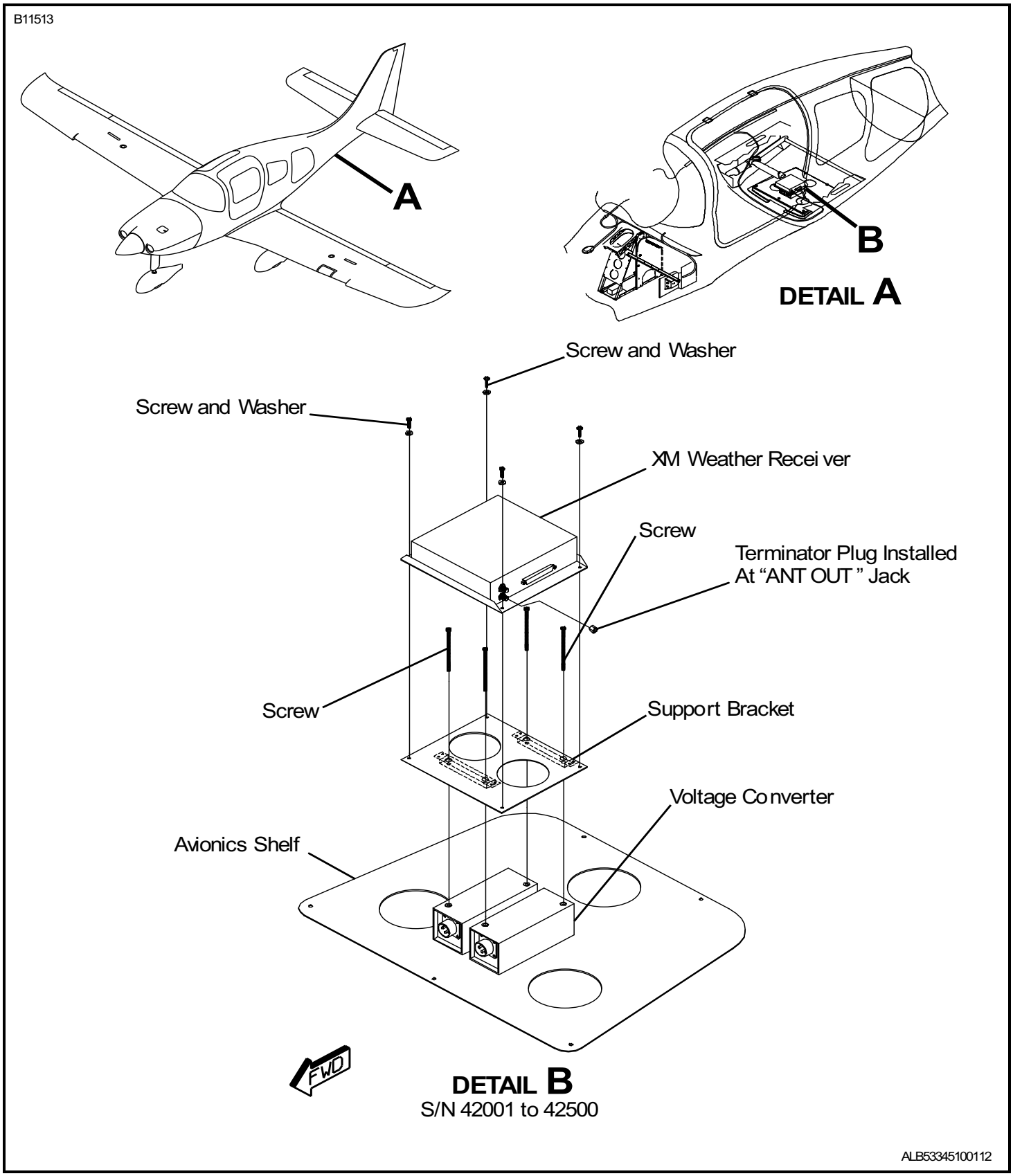
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Figure 233 : Sheet 1 : Ryan TCAD Transponder Coupler (Garmin G1000 Option)



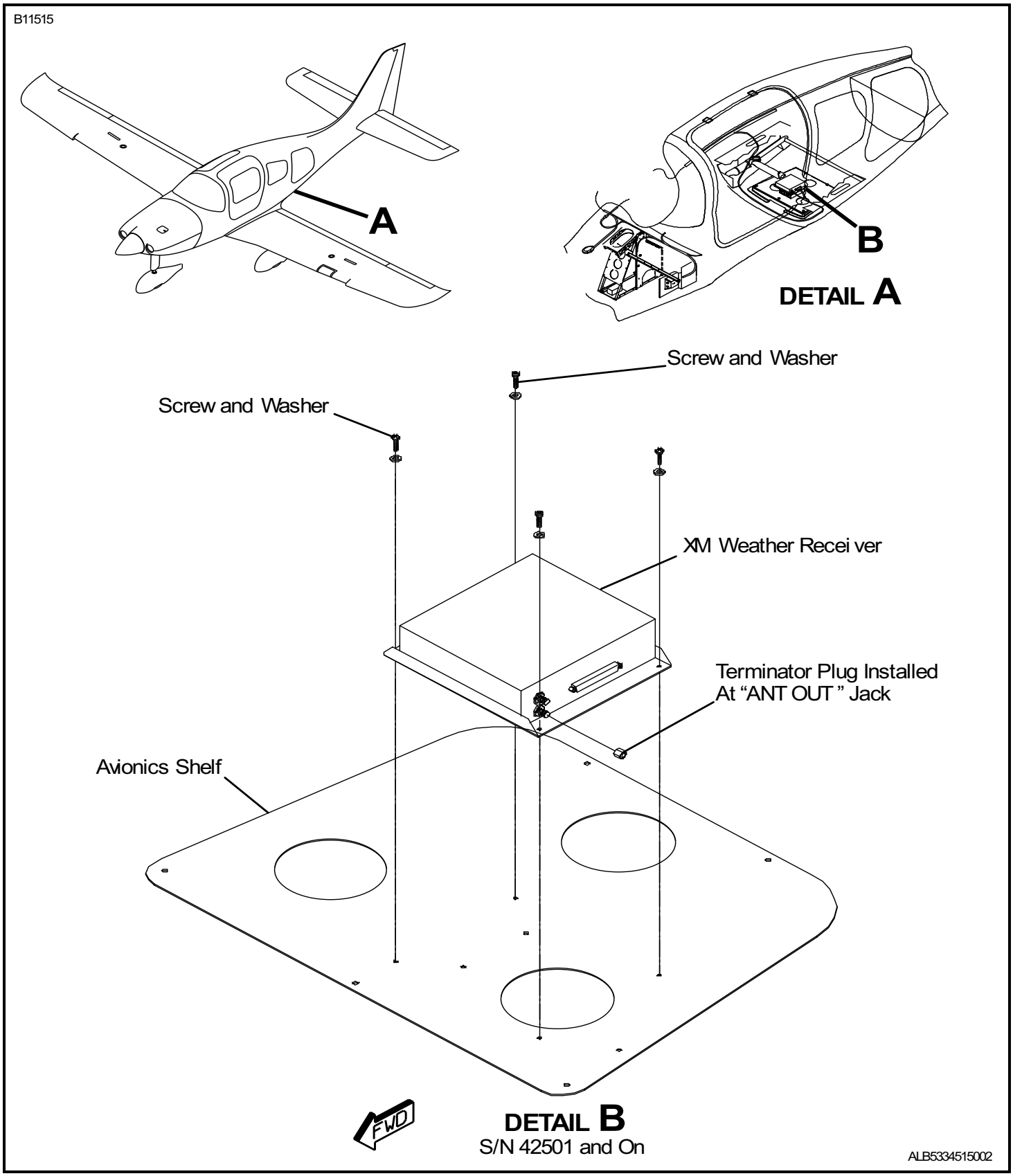
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Figure 234 : Sheet 1 : XM Weather Receiver



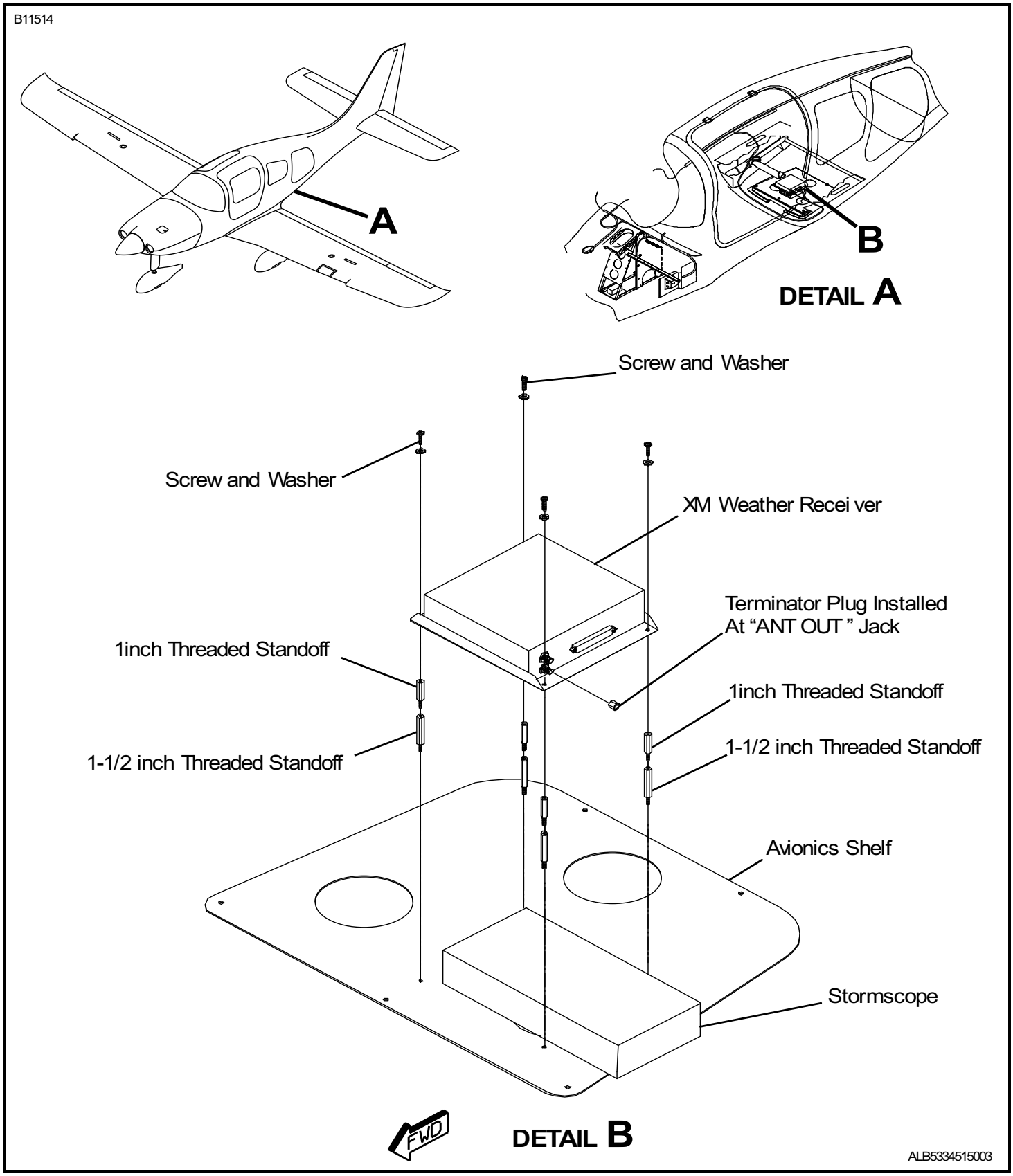
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Figure 234 : Sheet 2 : XM Weather Receiver



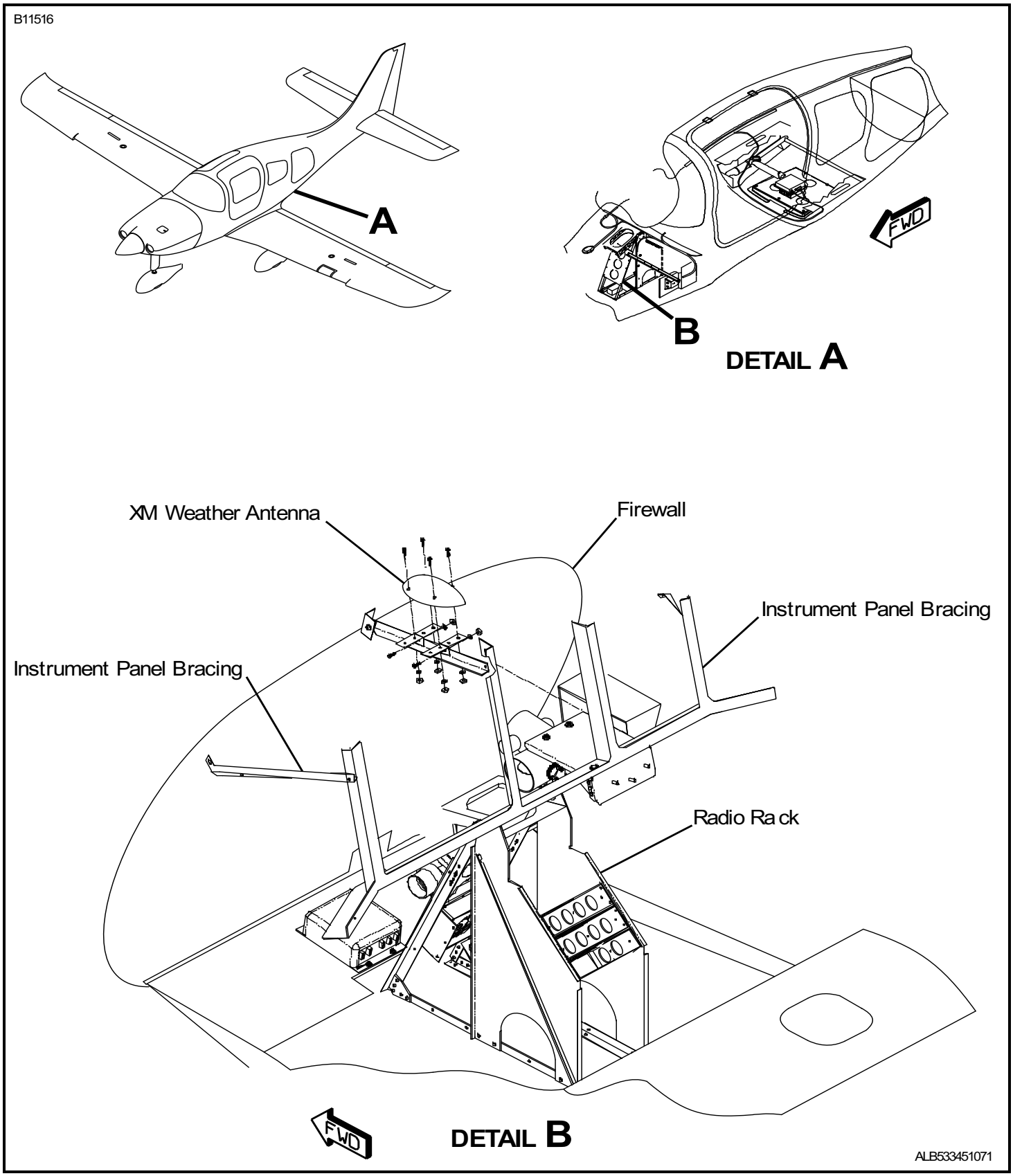
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Figure 235 : Sheet 1 : XM Weather Receiver if Stormscope Installed (S/N 42501 and On)



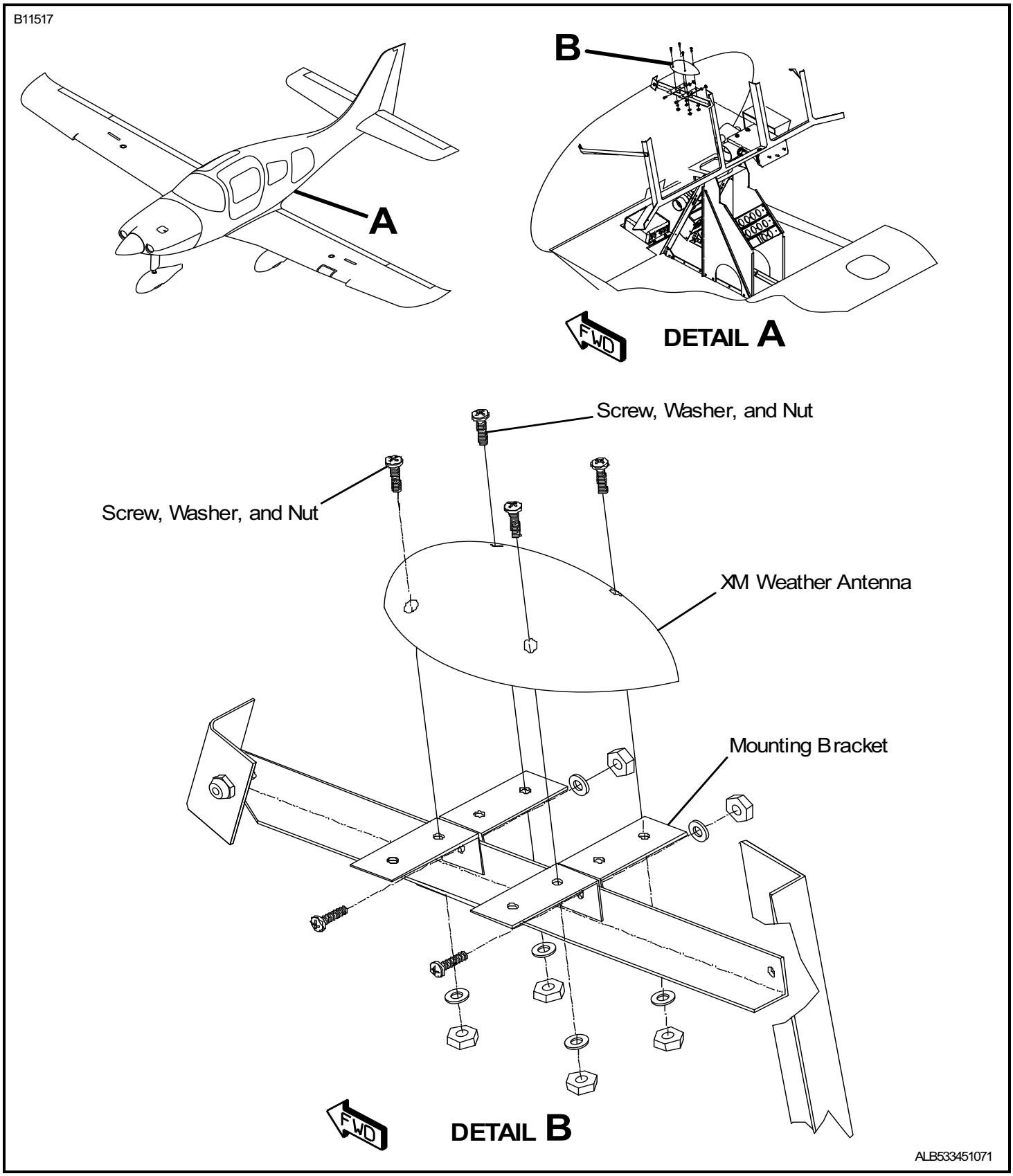
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Figure 236 : Sheet 1 : XM Weather Antenna Location



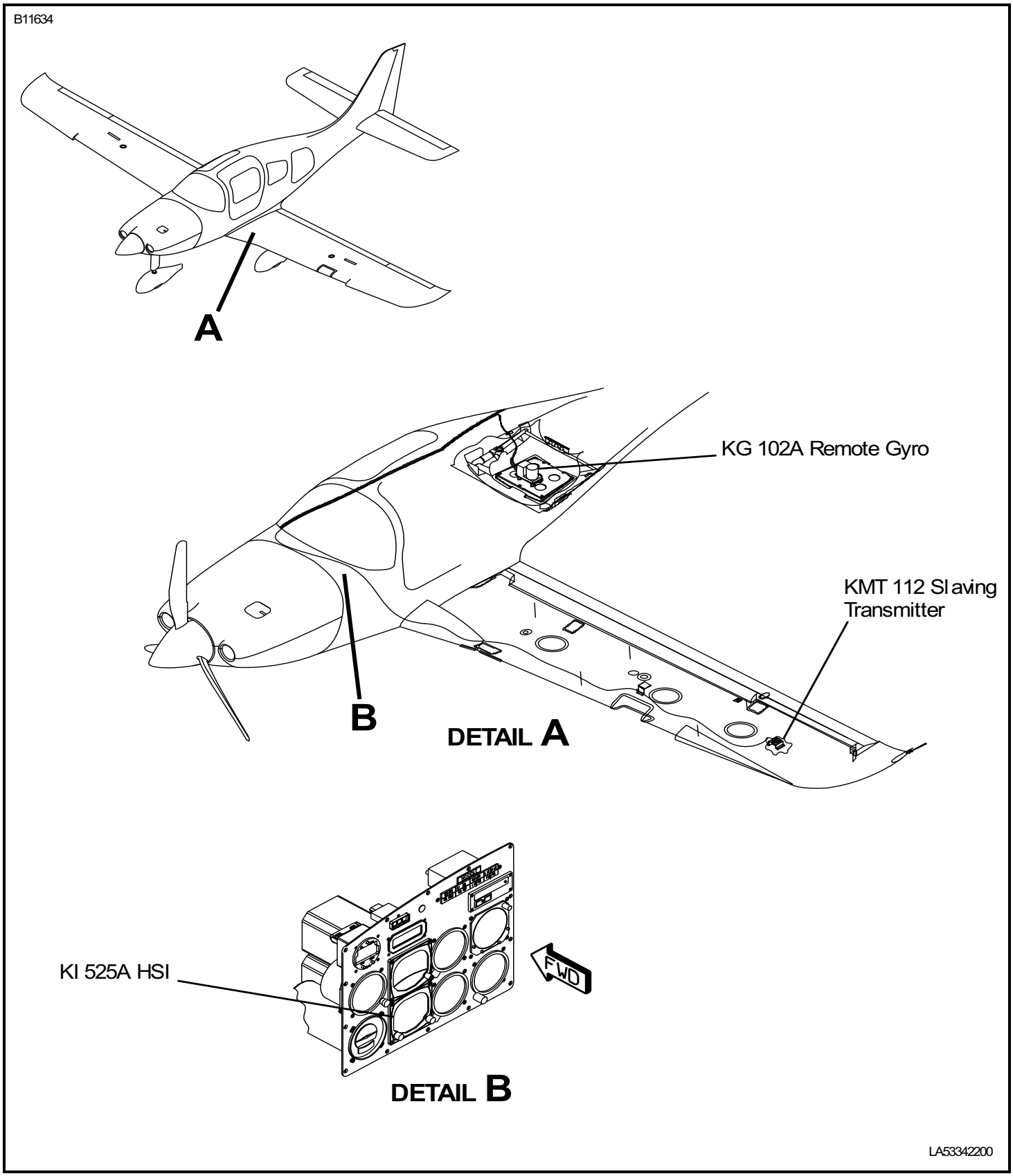
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Figure 237 : Sheet 1 : XM Weather Antenna



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Figure 209 : Sheet 1 : HSI - Remote Slaving System



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ENVIRONMENTAL CONTROL SYSTEM - GENERAL

1. General - Scope and Definition

A. Environmental control in the Cessna 350 is accomplished with either an Environmental Control System (ECS) or the optional air conditioning system also known as the Automatic Climate Control System (ACCS). For information concerning the ACCS, refer to Section 21-65-00. The ECS, shown in Figure 1, controls temperature throughout the front and rear cabin area. The ECS incorporates the use of an air-to-air heat exchanger, ram intake air, and an electric fan to distribute heated and outside air to various outlets within the cabin. The ECS consists of two subsystems, heated air and fresh air. Heated air is sent to the floor vent system and defroster, and fresh air is ducted through the eyeball vents. Ram air enters through a duct on the right side of the engine cowling and flows to either the heat exchanger (located on the right exhaust manifold) or the fresh air manifold. Air to the heat exchanger, depending on the control settings, is mixed with outside air in the heater box. The air then passes through a fan unit before entering the distribution system. Operating the single speed fan will increase the airflow through the system (not including the eyeball vents). The electrical system for the ECS is shown in Figure 2.

B. An air mixing valve takes both incoming hot and fresh air and mixes them according to the needs of the passengers.

C. At the windshield area, there is a defrost plenum directly built into the fuselage. This will allow the pilot or copilot to defrost the windshield during cold weather conditions and supply fresh air to that location during periods of high ambient temperature.

D. Located in the instrument panel are adjustable eyeball vents, one each for the pilot and copilot. These vents supply only fresh air to the front seat positions and may be manually directed to a desired location.

E. Near the pilot and copilot's feet is a floor vent tube. Like the defrost plenum, this tube can direct hot, fresh, or mixed air to the lower front cabin area.

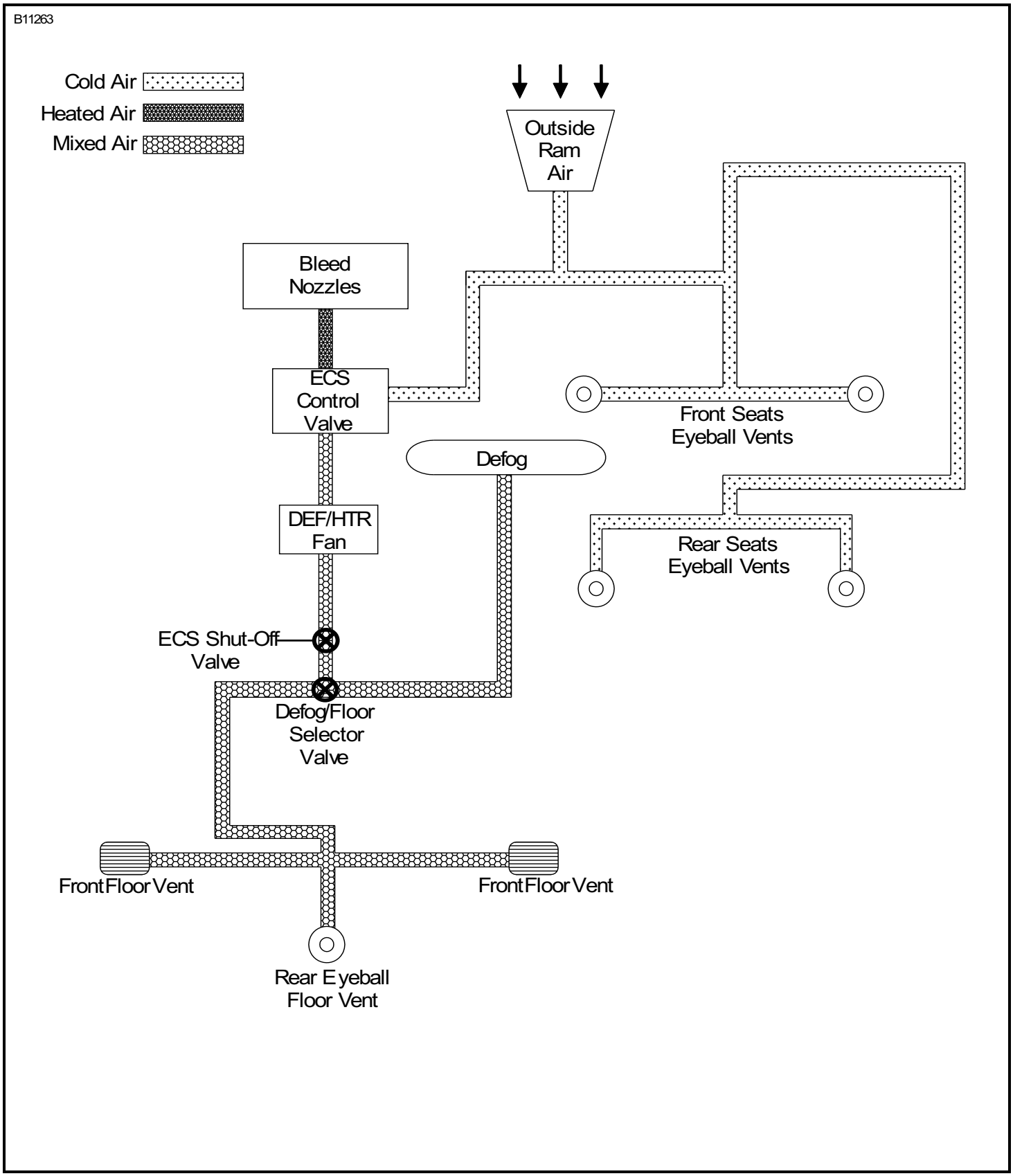
F. Directly tied into the floor vent tube and located at the aft end of the center console is an adjustable eyeball vent. This vent supplies the rear passengers the same air temperature as the floor and defrost vents but with the ability to be manually turned off independently or directed to a desired location.

G. Just to the outside of each rear passenger and located on the interior side panel is a fresh air eyeball vent. Like the front, these vents supply only fresh air and may be manually adjusted or directed to a desired location.

H. An electrically controlled fan may be turned on while on the ground or during flight to supply cooler fresh air through the defrost plenum, floor tube, and rear passenger center console eyeball vent during the summer or aid in deicing the windshield before and after takeoff during the winter months.

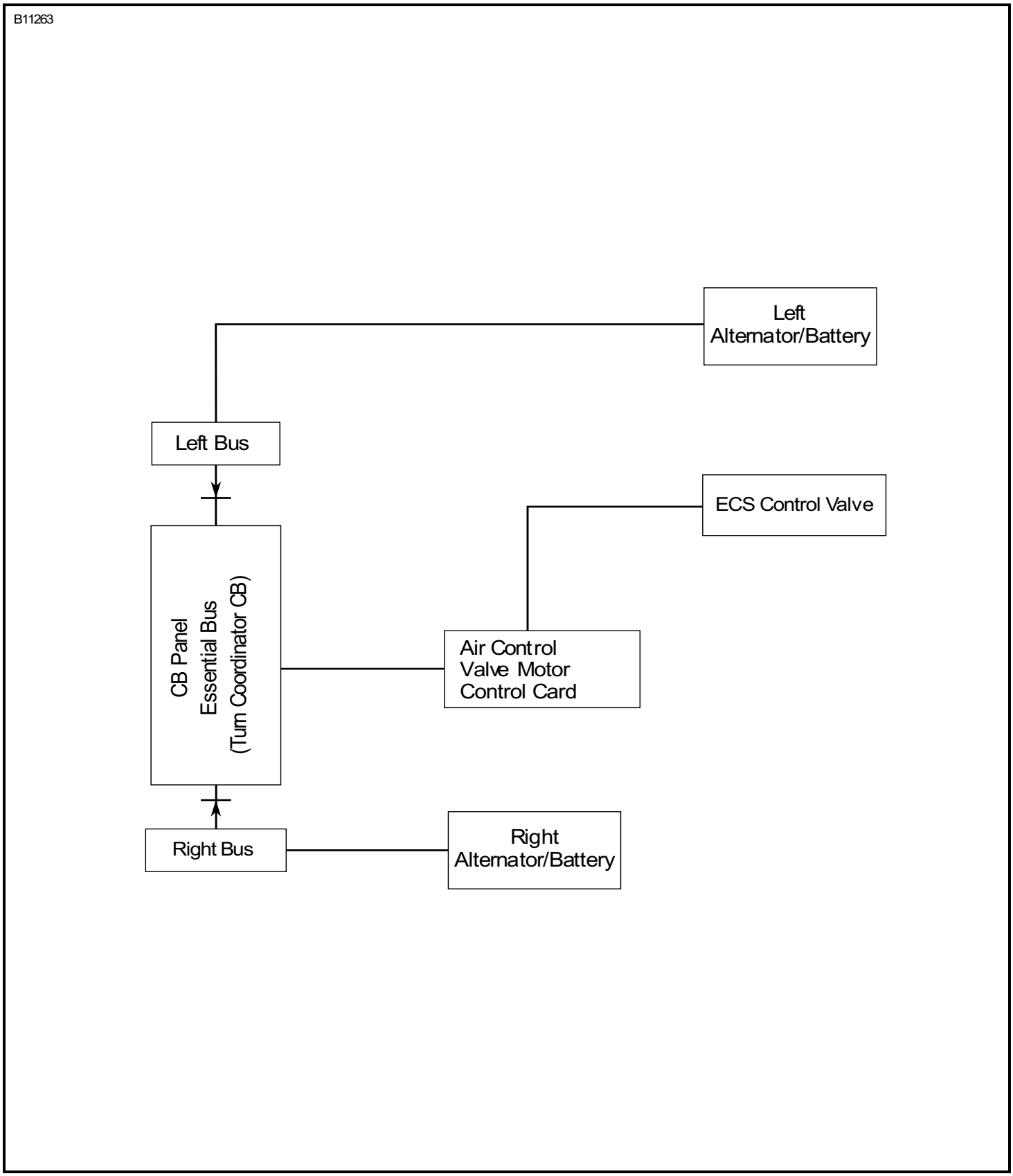
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Figure 1 : Sheet 1 : Environmental Control System Diagram



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Figure 2 : Sheet 1 : Schematic of ECS Electrical System Connection



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AIR DISTRIBUTION - MAINTENANCE PRACTICES

1. Distribution System

A. The components in the distribution system depend upon whether Avidyne avionics or the Garmin G1000 system is installed in the aircraft.

B. **Basic or Avidyne Option without Air Conditioning** - The distribution system includes the fan, ECS control valve, servo motor, defrost plenum, control panel, and eyeball and floor vents.

C. **Basic or Avidyne Option with Air Conditioning** - The distribution system includes the fan, ECS control valve, servo motor, ECS shut-off valve assembly, defrost/floor valve assembly, defrost plenum, control head, and eyeball and floor vents.

D. **Garmin G1000 Option with or without Air Conditioning** - The distribution system includes the fan, ECS control valve, servo motor, ECS shut-off valve assembly, defrost/floor valve assembly, defrost plenum, control head, and eyeball and floor vents.

2. ECS Fan

A. Basic or Avidyne Option without Air Conditioning

(1) ECS Fan Removal

(a) Unplug the electrical connector plug, then cut and remove the plastic tie straps located at the inlet and outlet ports of the fan.

(b) Pull the flexible ducting away from the fan.

(c) Loosen and disconnect the main mounting clamp until the fan can be freed from the cradle housing.

(d) Remove the fan by pulling it out from underneath the instrument panel. Refer to Figure 204.

(2) ECS Fan Installation

(a) Position the fan in the cradle leaving enough room between the heater control box and the fan inlet port for flexible ducting.

(b) Connect and tighten the main mounting clamp around the fan motor housing.

(c) Slide the flexible ducting over the inlet and outlet ports of the fan, and secure using plastic tie straps.

(d) Reconnect the ELC connector.

(e) Turn on the power, and test the fan for proper operation.

B. Garmin G1000 Option, Basic Option, or Avidyne Option with Air Conditioning

(1) ECS Fan Removal

(a) Unplug the electrical connector plug, then cut and remove the plastic tie straps at the inlet port, and remove the hose clamp at the outlet port of the fan.

(b) Pull the ECS shut-off valve assembly away from the outlet port of the fan. Refer to Figure 203 and Figure 204.

(c) Pull the flexible ducting away from the fan.

(d) Loosen and disconnect the main mounting clamp until the fan can be freed from the cradle housing.

(e) Remove the fan by pulling it out from underneath the instrument panel.

(2) ECS Fan Installation

(a) Position the fan in the cradle.

(b) Connect and tighten the main mounting clamp around the fan motor housing.

(c) Slide the flexible ducting over the inlet port of the fan, and secure using plastic tie straps.

(d) Slide the end of the ECS shut-off valve assembly over the outlet port of the fan. Apply RTV5818 silicone all around and secure with a hose clamp. Refer to Figure 204.

(e) Reconnect the ELC connector.

(f) Turn on the power, and test the fan for proper operation.

C. ECS Fan Maintenance - The electrically controlled ECS fan is a self-contained unit and requires only a periodic check for proper operation.

3. Servo Motor

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A. **Servo Motor Removal**

(1) With power on, adjust the temperature knob in the instrument panel to the full cold position, then turn power off.

(2) Remove the ECS control valve per Section 21-40-00 of this manual.

(3) Remove the servo motor to ECS control valve attachment screws, washers, and nuts.

(4) Pull the servo motor back and away from the control valve until the servo motor drive arm to linear drive rod clevis is exposed.

(5) Remove the cotter pin, washer, and clevis pin, and remove the servo motor from the control valve.

B. **Servo Motor Installation**

(1) With the servo motor arm in the fully extended position, install the servo motor drive arm into the linear actuator clevis, and reinstall the clevis pin, washer, and a new cotter pin.

(2) Align the mounting holes in the servo motor with the mounting holes in the ECS control valve. Install screws, washers, and nuts only tight enough so that servo motor can be pushed forward by hand.

(3) With the heater control valve secured, hand push the servo motor forward until the silicone sealing cushion inside of ECS control valve is fully seated against metal sealing ring. Tighten the servo mounting hardware.

C. **Servo Motor Maintenance** - The ECS control servo motor is a self-contained unit and requires only a periodic check for proper operation.

4. ECS Shut-Off Valve Assembly

A. **ECS Shut-Off Valve Removal**

(1) Unplug the electrical connector plug.

(2) Remove the hose clamps at the inlet and outlet ports of the ECS shut-off valve. Refer to Figure 203 and Figure 204.

(3) Pull the flexible ducting away from the outlet port.

(4) Pull the ECS shut-off valve assembly away from the outlet port of the ECS fan.

B. **ECS Shut-Off Valve Installation**

(1) Apply RTV5818 silicone all around the outlet port of the ECS fan and the outlet port of the ECS shut-off valve.

(2) Slide the inlet port of the ECS shut-off valve over the outlet port of the ECS fan. Rotate the valve as required to clear the rudder pedal tubes and ECS servo motor actuator screw and secure with a hose clamp.

(3) Slide the flexible ducting over the outlet port of the ECS shut-off valve, and secure with a hose clamp.

(4) Reconnect the electrical connector.

(5) Turn on the power, and test the ECS shut-off valve for proper operation.

C. **ECS Shut-Off Valve Maintenance** - The ECS shut-off valve assembly is a self-contained unit and requires only a periodic check for proper operation.

5. Defrost/Floor Valve Assembly

A. The defrost/floor valve assembly is only present with the Garmin G1000 option or the Avidyne option with air conditioning.

B. **Defrost/Floor Valve Removal**

(1) Unplug the electrical connector plug.

(2) Remove the hose clamps at the inlet and outlet ports of the defrost/floor valve. Refer to Figure 203 and Figure 204.

(a) On Garmin G1000 aircraft, the defrost/floor valve assembly may be attached directly to the defrost channel. With this condition, remove the hose clamp and a MS51861-25C screw. There is a length of silicone sheeting between the defrost/floor valve and the defrost channel.

(3) Pull the flexible ducting away from the inlet and outlet ports.

(4) Remove the defrost/floor valve.

C. **Defrost/Floor Valve Installation**

(1) Defrost/floor valve attached directly to the defrost channel:

(a) Fill the gap between the inner diameter of the valve flange and the outer diameter of the defrost channel flange with 0.062-inch or 0.093-inch AMS3320 silicone sheet.

(b) When installing a new defrost/floor valve, match drill a 0.166-inch diameter (No. 32) hole in the valve flange.

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(c) Install a MS51861-25C screw through the valve flange, the defrost channel flange, and the silicone sheet. Dry torque hand tight plus 1/4 turn.

(d) Apply RTV5818 silicone all around the inlet port, install flexible ducting, and secure with a hose clamp.

(e) Reconnect the electrical connector.

(f) Turn on the power, and test the defrost/floor valve for proper operation.

(2) Defrost/floor valve not attached directly to the defrost channel:

(a) Apply RTV5818 silicone all around the inlet and outlet ports of the defrost/floor valve.

(b) Slide the flexible ducting over the inlet and outlet ports of the defrost/floor. Rotate the valve as required to clear the rudder pedal tubes and ECS servo motor actuator screw and secure with a hose clamps.

(c) Reconnect the electrical connector.

(d) Turn on the power, and test the defrost/floor valve for proper operation.

D. **Defrost/Floor Valve Maintenance** - The defrost/floor valve assembly is a self-contained unit and requires only a periodic check for proper operation.

6. Defrost Plenum

A. **Defrost Plenum Removal/Installation** - The defrost plenum is an integral part of the fuselage and contains no moving parts; therefore, no removal is required.

B. **Defrost Plenum Maintenance** - The defrost plenum is an integral part of the fuselage; therefore, no maintenance is required.

7. Eyeball Vents (Instrument Panel)

A. **Eyeball Vent Removal (Instrument Panel)** - To remove the left and/or right eyeball vent(s) from the instrument panel, use the following instructions:

(1) Remove the left side instrument panel and/or the right side kidney panel screws. (Does not apply to the Garmin G1000 system instrument panel.)

(2) Gently pull the left side instrument panel and/or the right side kidney panel away from the main panel until the eyeball vent flexible ducting can be seen. (Does not apply to the Garmin G1000 system instrument panel.)

(3) Remove the cable tie securing the tubing to the back of the plenum cap. For the Garmin G1000 system, the back of the vent may be accessed from underneath the instrument panel.

(4) Loosen the clamp securing the plenum cap to the backside of the eyeball vent, and gently pull away from the vent.

(5) Loosen and remove the threaded eyeball vent retaining collar from the backside of the panel.

(6) Pull the eyeball vent out from the front of the panel. Refer to Figure 201 for a schematic of the eyeball vent.

B. **Eyeball Vent Installation (Instrument Panel)** - To install the left and/or right eyeball vent(s) into the instrument panel, use the following instructions:

(1) Insert the eyeball vent(s) into the front of the left side instrument panel, into the front of the right kidney panel, or into the lower instrument panel (Garmin G1000 system), as applicable.

(2) Install the threaded eyeball vent retaining collar to the backside of eyeball vent.

(3) Press the plastic plenum cap onto the backside of the eyeball vent, and secure with clamp. When the clamp is secured, the screw housing must remain at the 10 to 12 o'clock position for the left eyeball vent and at the 12 to 3 o'clock position for the right eyeball vent, as viewed from the rear.

(4) Slide the tubing over the back of the plenum cap and secure with a new cable tie.

(5) Install the instrument panel and/or the kidney panel to the main panel with screws. (Does not apply to the Garmin G1000 system instrument panel.)

8. Eyeball Vents (Rear Interior Side Panels)

A. **Eyeball Vent Removal (Rear Interior Side Panels)** - To remove the rear interior side panel eyeball vents, use the following instructions:

(1) Remove the rear interior panel attachment screws, then gently pull the interior side panel up and away from the fuselage until access is gained to the rear of the eyeball vent.

(2) Loosen and remove the threaded eyeball vent retaining collar from the backside of the panel.

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(3) Pull the eyeball vent out from the front of the panel.

B. **Eyeball Vent Installation (Rear Interior Side Panels)** - To install eyeball vents into the rear interior side panel, use the following instructions:

(1) Insert the eyeball vent into the side panel hole.

(2) Install the threaded eyeball vent retaining ring to the backside of the eyeball vent.

(3) Press the panel back into place ensuring the vent is seated in the ducting channel.

(4) Install the rear interior panel to the fuselage, and attach it with screws.

C. **Eyeball Vent Maintenance (Rear Interior Side Panels)** - The rear interior side panel eyeball vents are self-contained; therefore, no maintenance is required.

9. Floor Vent (Front)

A. **Floor Vent (Front) Removal/Installation** - The front floor vent is an integral part of the radio rack frame with no moving parts; therefore, no removal is required.

B. **Floor Vent (Front) Maintenance** - The front floor vent is self-contained; therefore, no maintenance is required.

10. Floor Vent (Rear)

A. **Floor Vent (Rear) Removal** - To remove the rear floor vent, use the following instructions (refer to Figure 202):

(1) Gently pull the center console panel out and away from the main console.

(2) With the rear heat vent flexible ducting exposed, loosen the clamp securing the plastic plenum cap to the backside of the eyeball vent, and gently pull the plastic housing away from vent.

(3) Loosen and remove the threaded eyeball vent retaining collar from the backside of the panel.

(4) Pull the eyeball vent out from the front of the panel.

B. **Floor Vent (Rear) Installation** - To install eyeball vents onto the center console, use the following instructions:

(1) Insert the eyeball vent into the side panel hole.

(2) Install the threaded eyeball vent retaining collar to the backside of the eyeball vent.

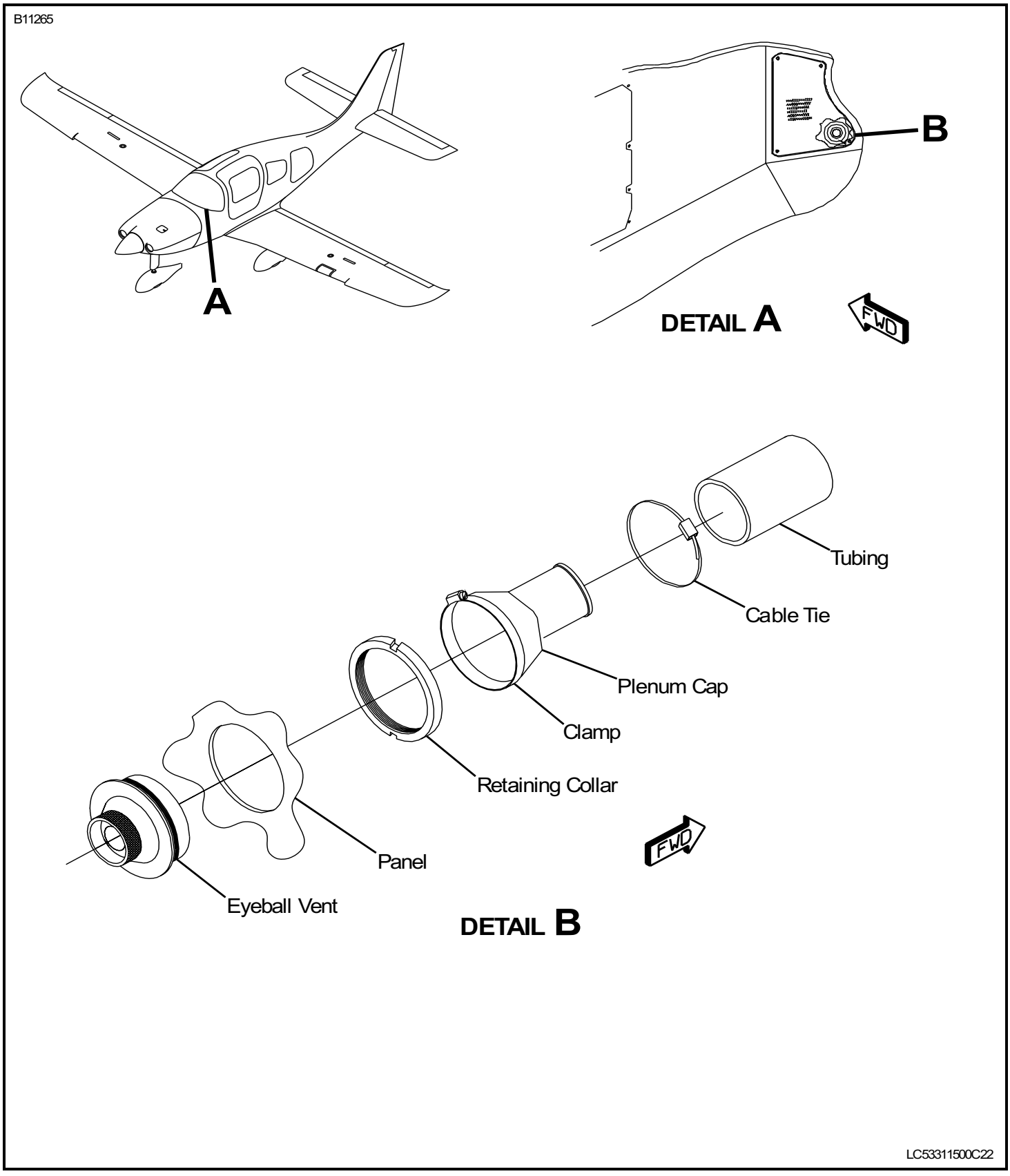
(3) Press the plastic plenum cap onto the backside of the eyeball vent, and secure with a clamp.

(4) Install the center console panel onto the main panel until the duel lock stripping is fully engaged.

C. **Floor Vent (Rear) Maintenance** - The rear floor vent is self-contained; therefore, no maintenance is required.

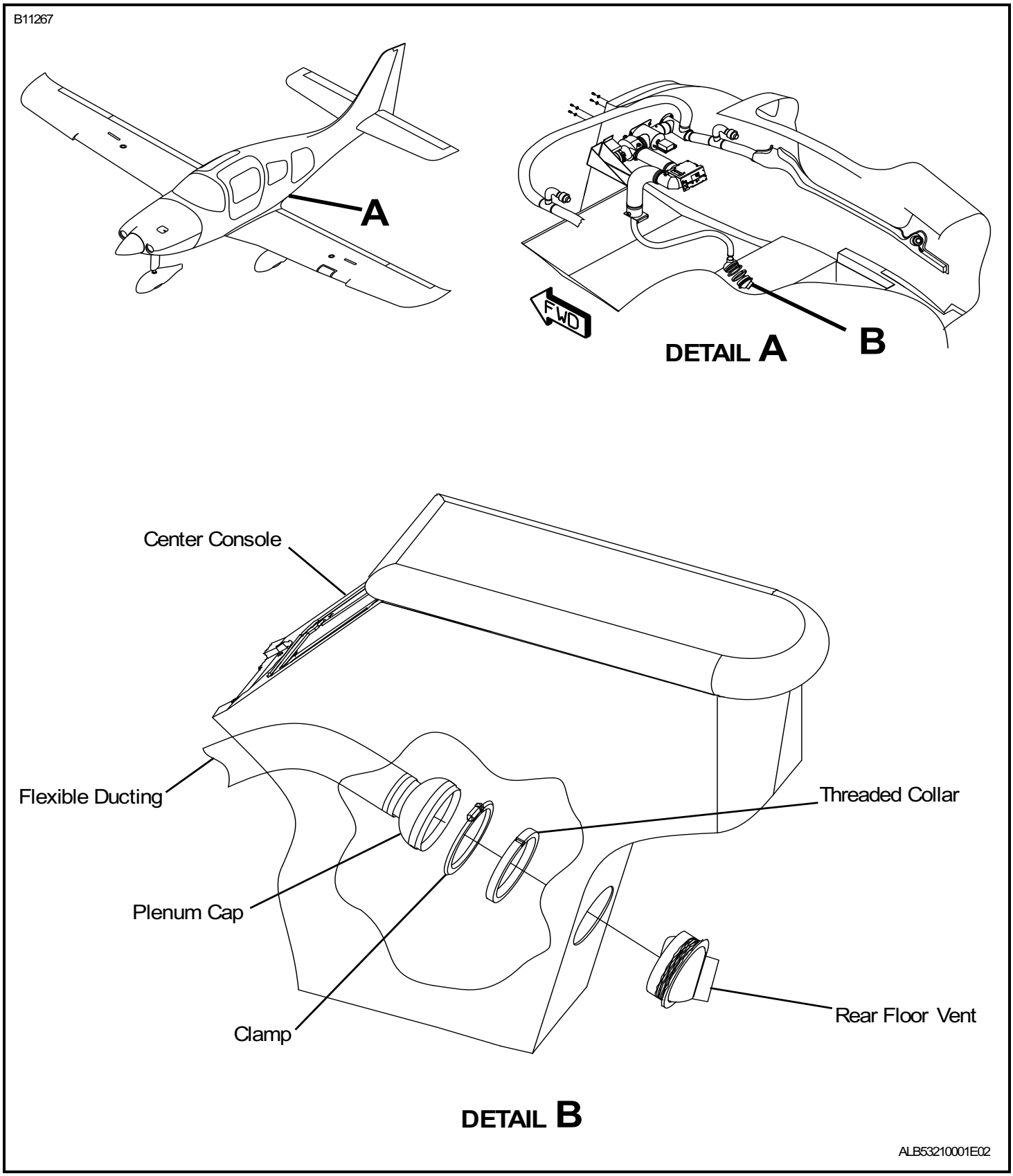
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Figure 201 : Sheet 1 : Eyeball Vent Installation



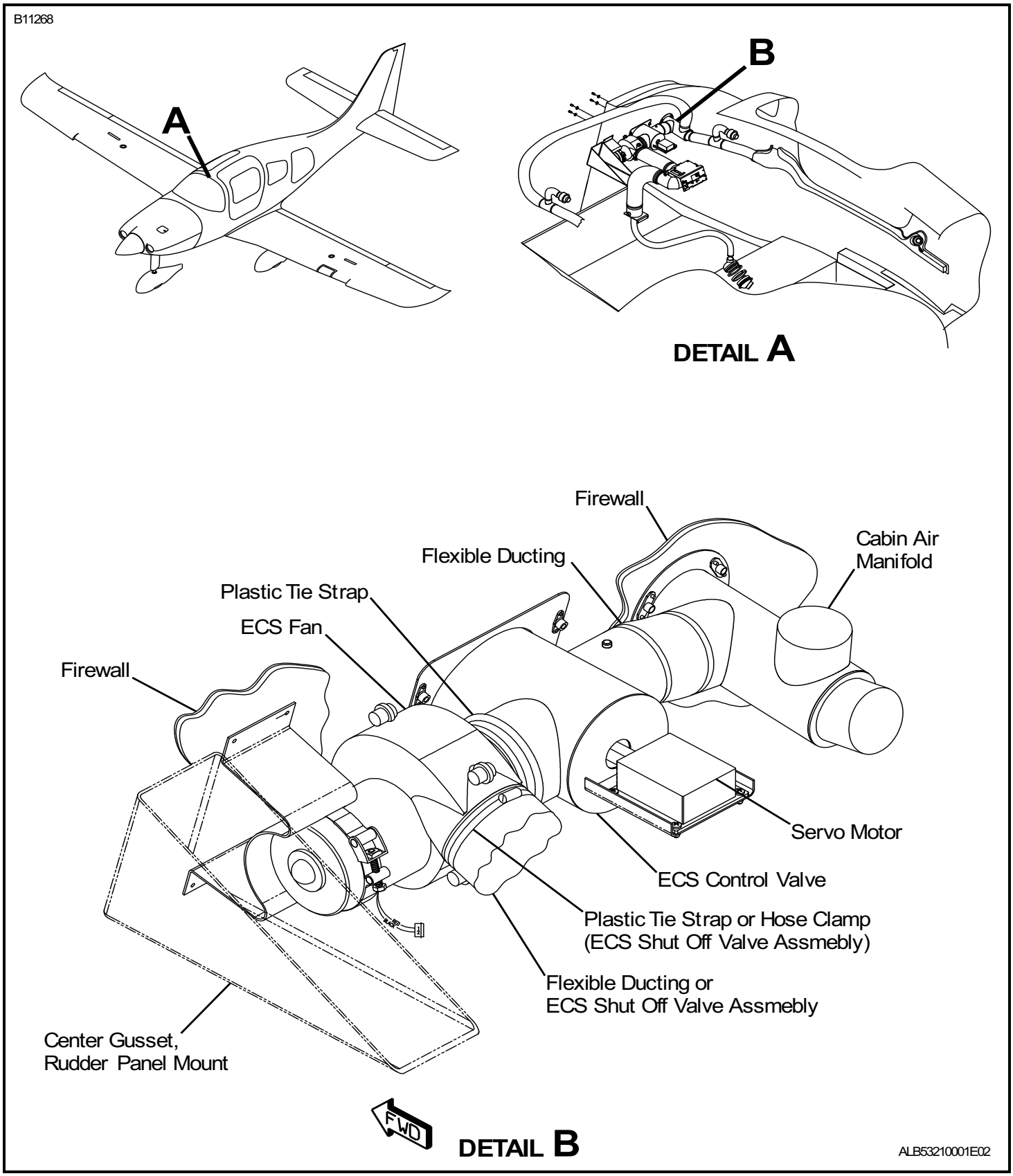
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Figure 202 : Sheet 1 : Rear Floor Vent



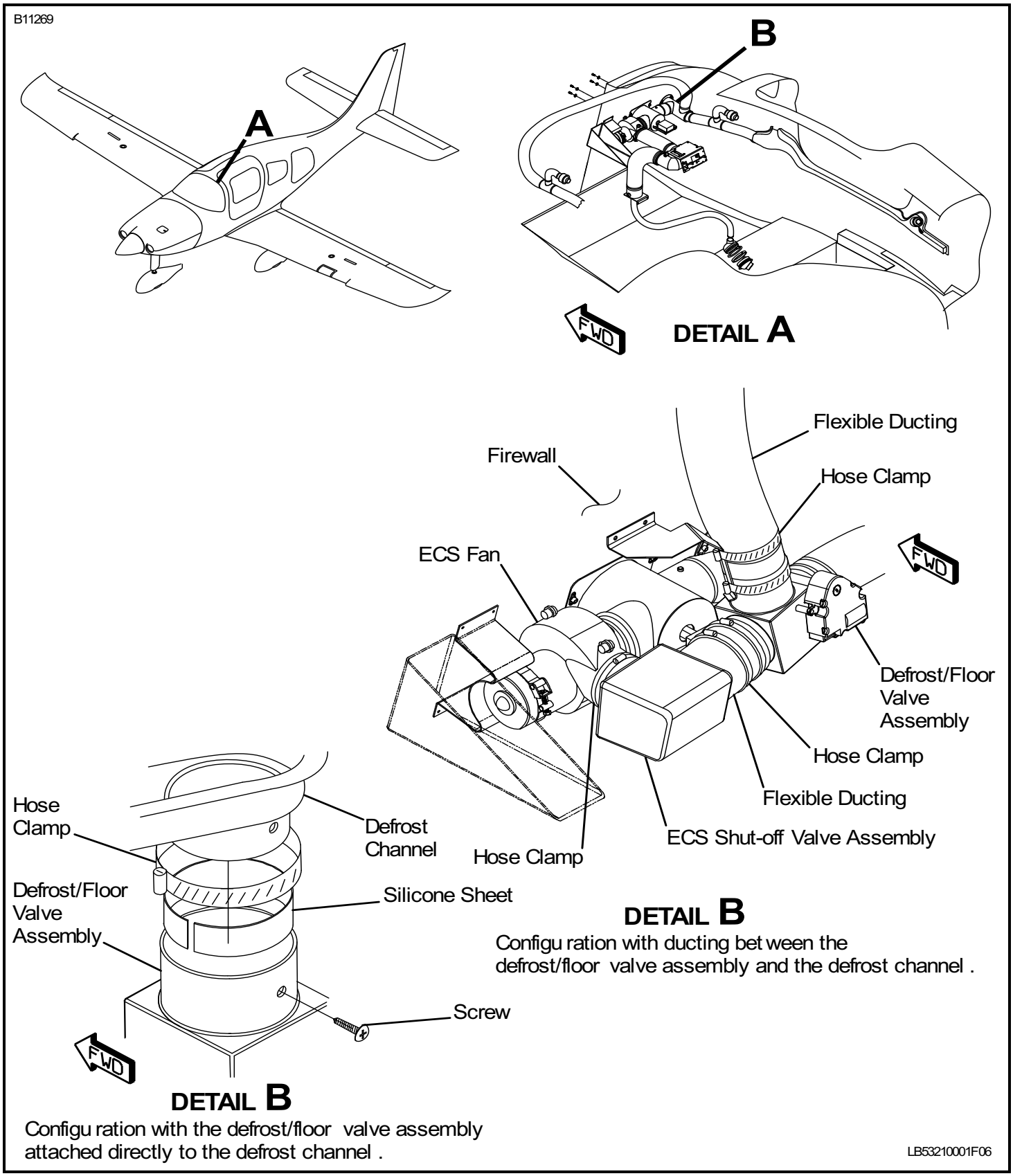
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Figure 203 : Sheet 1 : ECS System Attachment



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Figure 204 : Sheet 1 : ECS Shut-Off Valve and Defrost/Floor Valve Assemblies



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HEATING AND HEATER CONTROL VALVE - MAINTENANCE PRACTICES

1. Heating and ECS Control Valve

A. The following procedures are applicable to the heater and ECS control valve.

2. ECS Control Valve

A. **ECS Control Valve Removal** - To remove the ECS control valve from the airplane, use the following instructions:

(1) Loosen the steel clamp securing the flexible ducting to the ECS control valve inlet tube located in the engine compartment, and pull the ducting away from the tube. Refer to Figure 201.

(2) Cut and remove the plastic tie straps from the inlet and outlet tubes of the ECS control box, and pull the flexible ducting away from the tubes. Refer to Section 21-20-00, Figure 204.

(3) Disconnect the heater control valve servo motor ELC plug.

(4) Remove the faceplate screws and washers, and pull the faceplate outward until the inner plunger seal can be seen.

(5) Remove the control box bolts and washers. Pull the control box out from under the instrument panel. Refer to Figure 202.

B. **ECS Control Valve Installation** - To install the heater control box use the following instructions:

(1) Insert the control box into the opening in the firewall from inside of the cabin area with the servo motor facing up.

(2) Install the control valve mounting bolts and washers from the engine compartment side of the firewall.

(3) Install the faceplate onto the front of the control box, then install the faceplate mounting screws and washers.

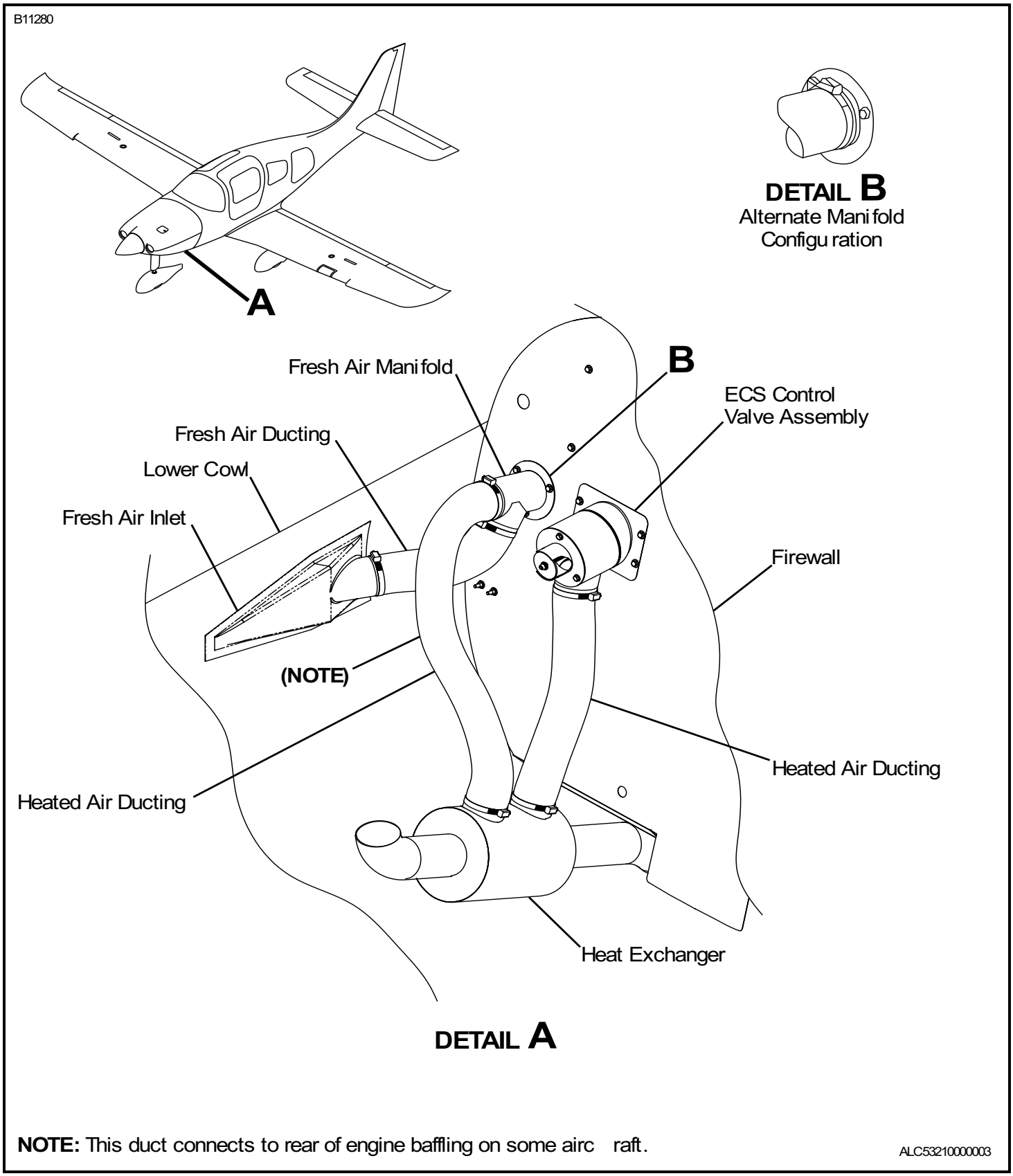
(4) Reattach the flexible ducting to the faceplate inlet tube with a steel clamp.

(5) Turn on the power and test the heater control box for proper valve operation and adjust the servo motor per the instructions in Section 21-20-00, Servo Motor , if necessary.

C. **ECS Control Valve Maintenance** - The heater control box is a self-contained unit. Only a periodic check for proper valve operation is required.

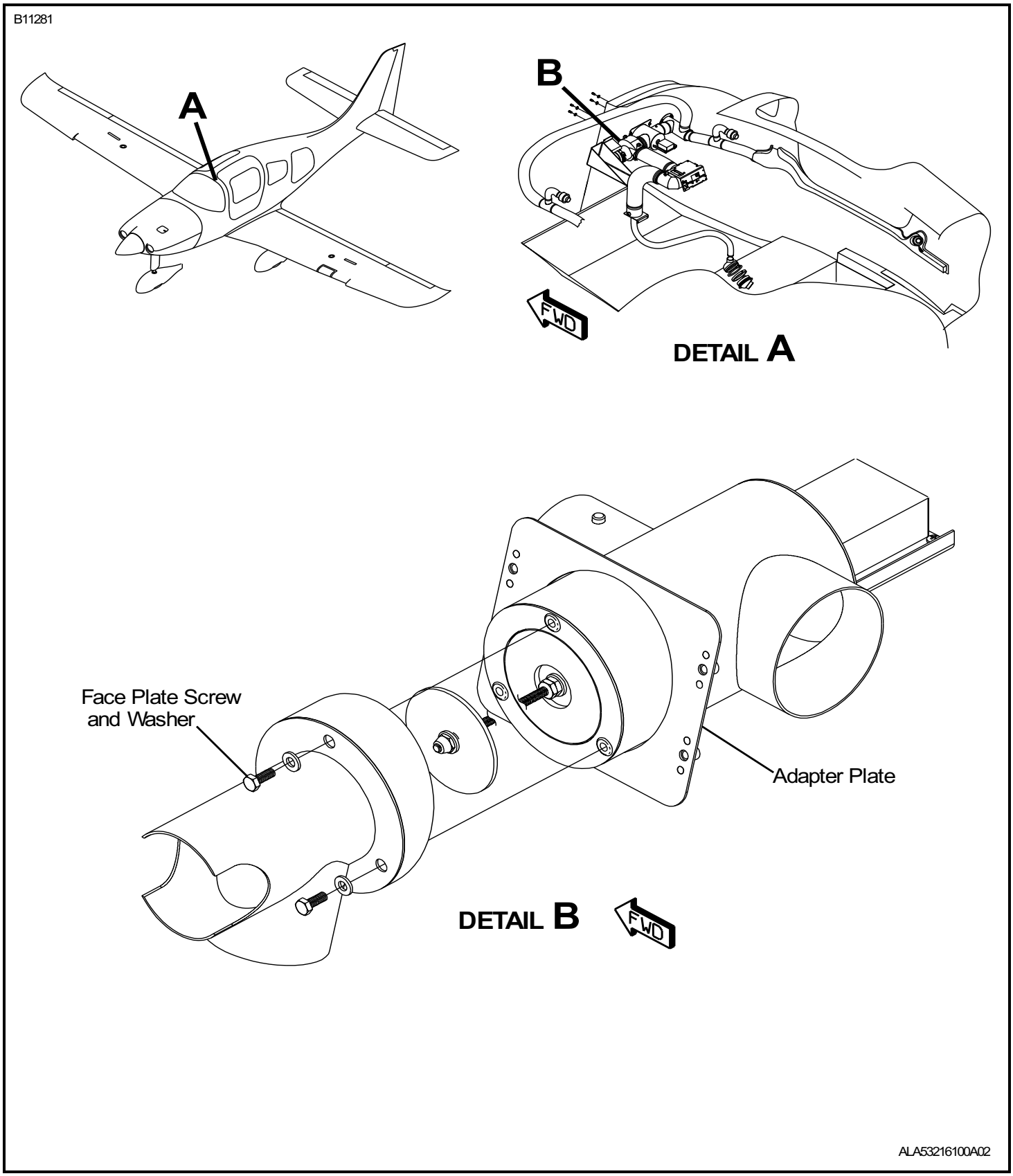
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Figure 201 : Sheet 1 : ECS Assembly in Engine Compartment



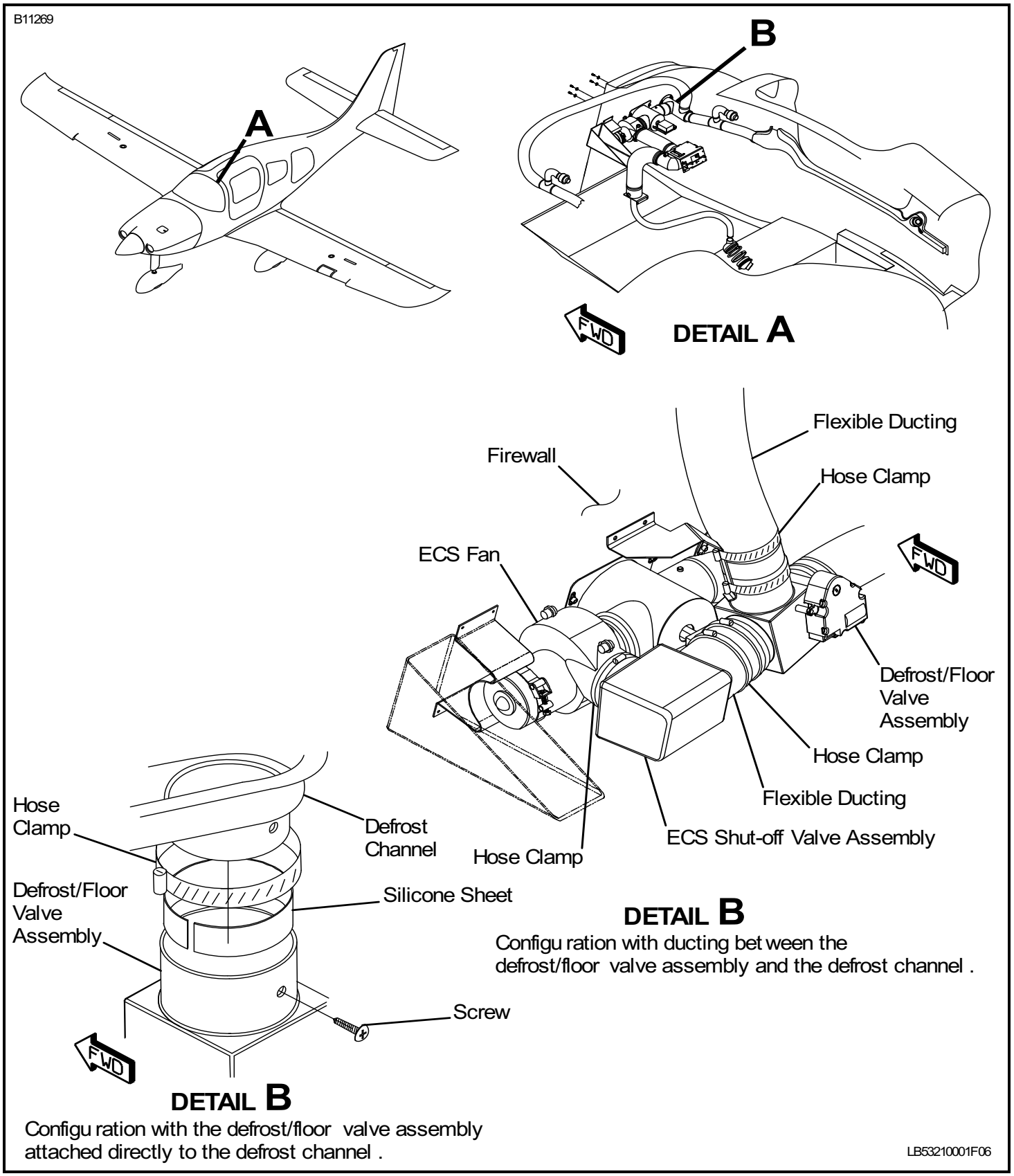
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Figure 202 : Sheet 1 : ECS Control Valve and Faceplate Attachment



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Figure 204 : Sheet 1 : ECS Shut-Off Valve and Defrost/Floor Valve Assemblies



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TEMPERATURE AND AIR CONTROL - MAINTENANCE PRACTICES

1. Temperature and Air Control Unit

A. One of three types of temperature and air control units will be present in the aircraft, depending upon avionics installed (basic, Avidyne or Garmin G1000) and upon installation of optional Automatic Climate Control System (ACCS). For aircraft with Garmin G1000 avionics and without ACCS, the temperature and air control unit is similar to the control head of the ACCS with identical removal and installation procedures. Refer to Section 21-65-00. For aircraft with basic instruments only or Avidyne avionics, and without ACCS, see below. For aircraft with optional ACCS, refer to Section 21-65-00.

B. **Removal of Air Control Unit** - To remove the temperature and air control unit, use the following instructions:

(1) Turn the power off and remove the temperature and air control knobs from the control unit faceplate located on the instrument panel by removing the set screws at the bottom of each knob. Refer to Figure 201.

(2) Remove the fan rocker switch by gently squeezing together the locking tabs located on the outer perimeter of the switch from behind the instrument panel.

(3) Carefully pull the rocker switch away from the instrument panel until wiring is exposed.

(4) Disconnect the wiring from the back side of the rocker switch, and remove the switch.

(5) Disconnect the ELC plug between the motor control card and the heater control box servo motor.

(6) Cut the plastic tie straps located at the air control unit inlet and outlet tube locations.

(7) Pull the flexible ducting away from the air control unit.

(8) Remove the four 6-32 self-locking nuts and washers attached to the Pem studs on the face of the air control unit. Refer to Figure 202.

(9) From the back side of the instrument panel, remove the nuts and washers securing the air control unit tray to the back side of the instrument panel as shown in Figure 202 and Figure 203.

(10) Pull the air control unit forward (toward the firewall) and away from the instrument panel until the knob shafts are behind the instrument panel.

(11) Remove the air control unit by pulling it out from underneath the instrument panel.

C. **Installation of Air Control Unit** - To install the air control unit, use the following instructions.

(1) Place the unit onto the cover panel studs from the backside of the instrument panel, and secure with nuts and washers

(2) Slide the flexible ducting onto the air control unit inlet and outlet tubes, and secure using plastic tie straps.

(3) Reconnect the motor control card to the servo motor ELC connector.

(4) Reconnect wiring to the backside of the fan rocker switch, and gently press the switch into the opening until the switch locking tabs are fully engaged.

(5) Install the temperature and air control knobs to the control shafts, checking for free knob rotation. Install set screws at bottom of knobs as shown in Figure 201.

(6) Turn the power on, and check for proper fan and heater control valve operation.

D. **Maintenance of Air Control Unit** - The temperature and air control unit is self-contained. A periodic check for fan, air control unit butterfly valve, and heater control valve operation is required. A functional check should be completed at each 100 hour or annual inspection using the following procedure. Refer to Section 21-00-00 for description on control operation.

(1) With cabin fan off, check adequacy of airflow for the front floor vents, rear floor vent, and defroster.

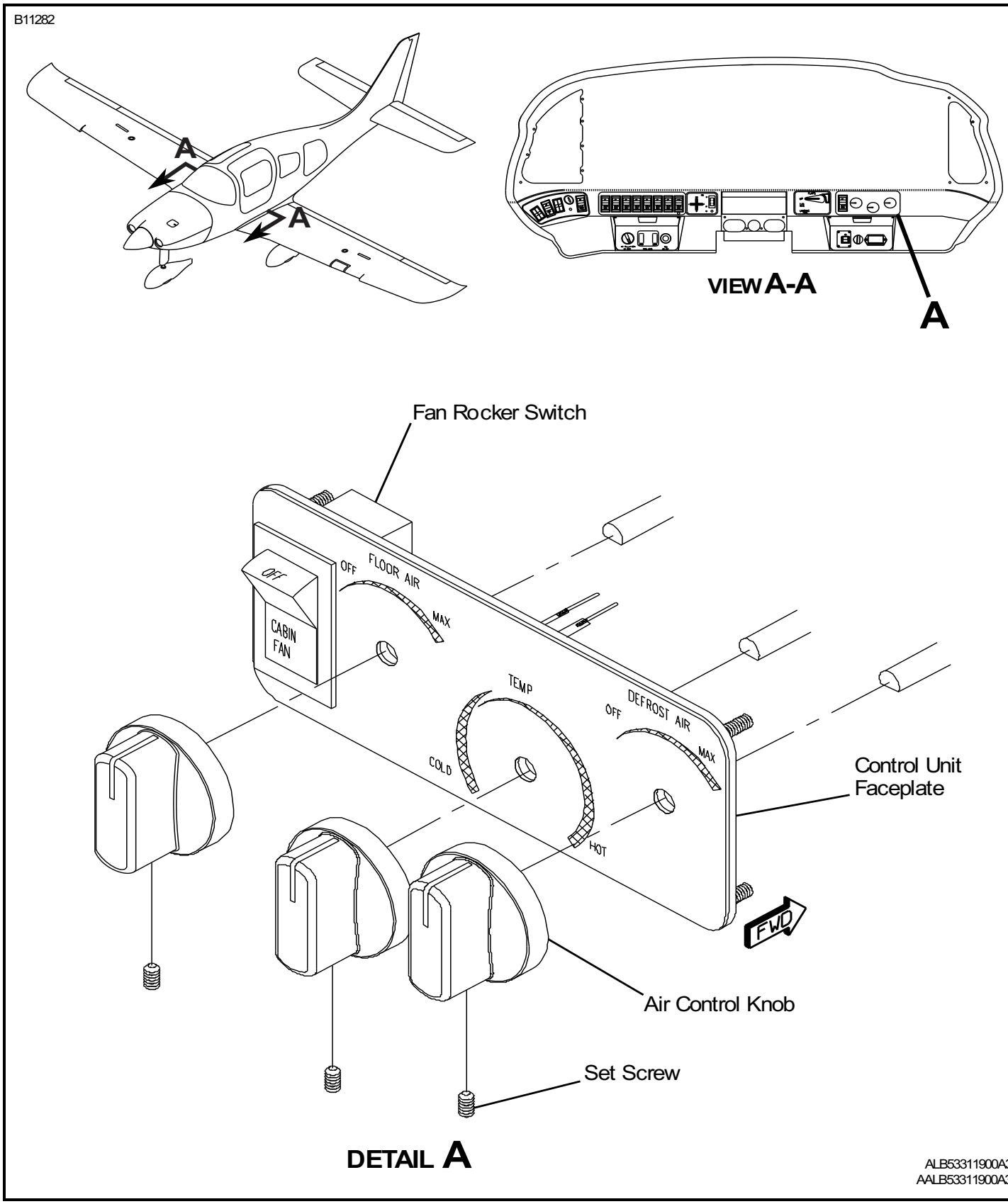
(2) Check for proper operation of the air volume control, temperature control, and air output control.

(3) Evaluate eyeball fresh air vents for proper operation.

(4) Evaluate the adequacy of the system in terms of crew and passenger comfort, as well as the defroster's ability to control fogging of the windscreen.

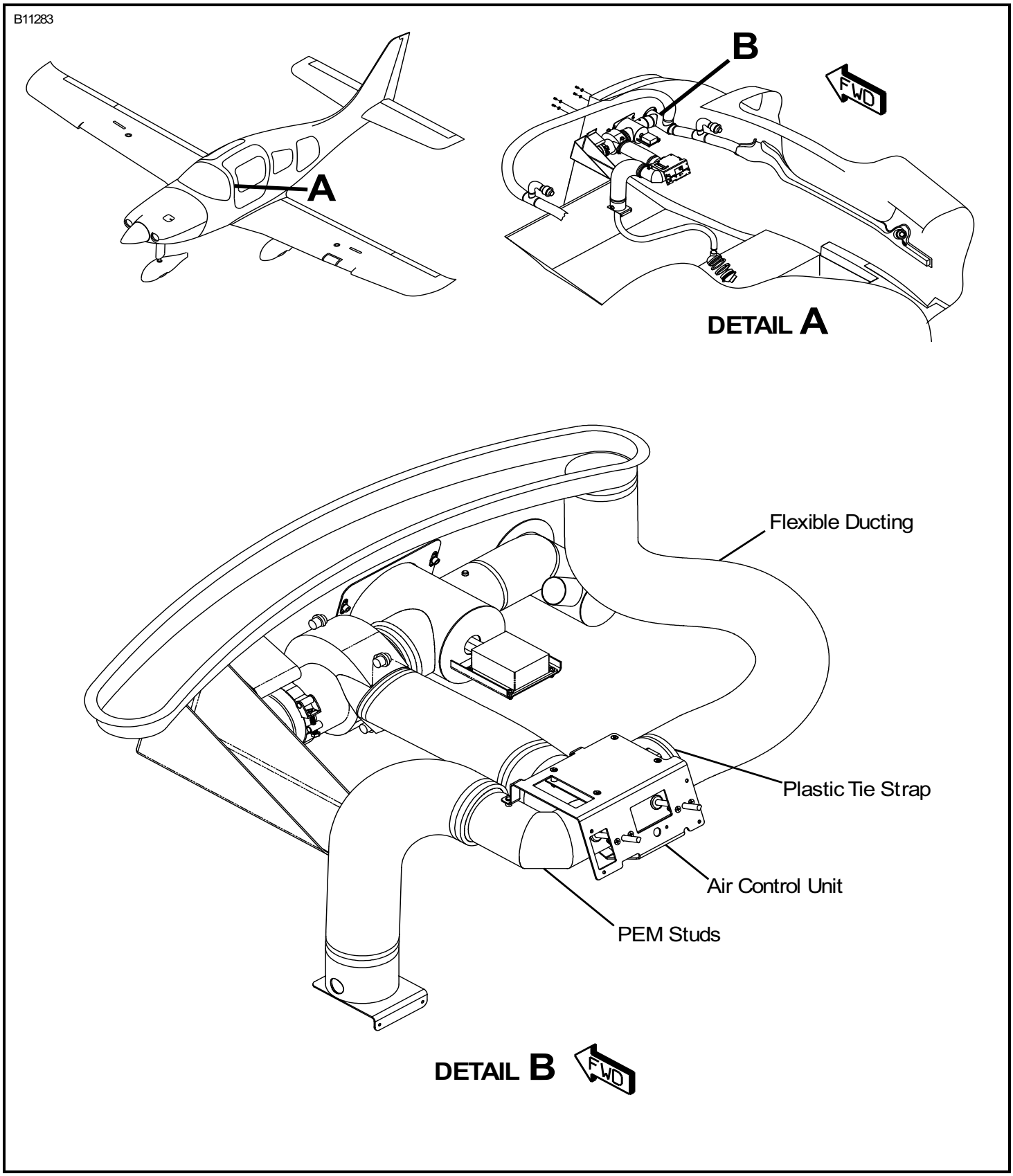
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Figure 201 : Sheet 1 : ECS Faceplate



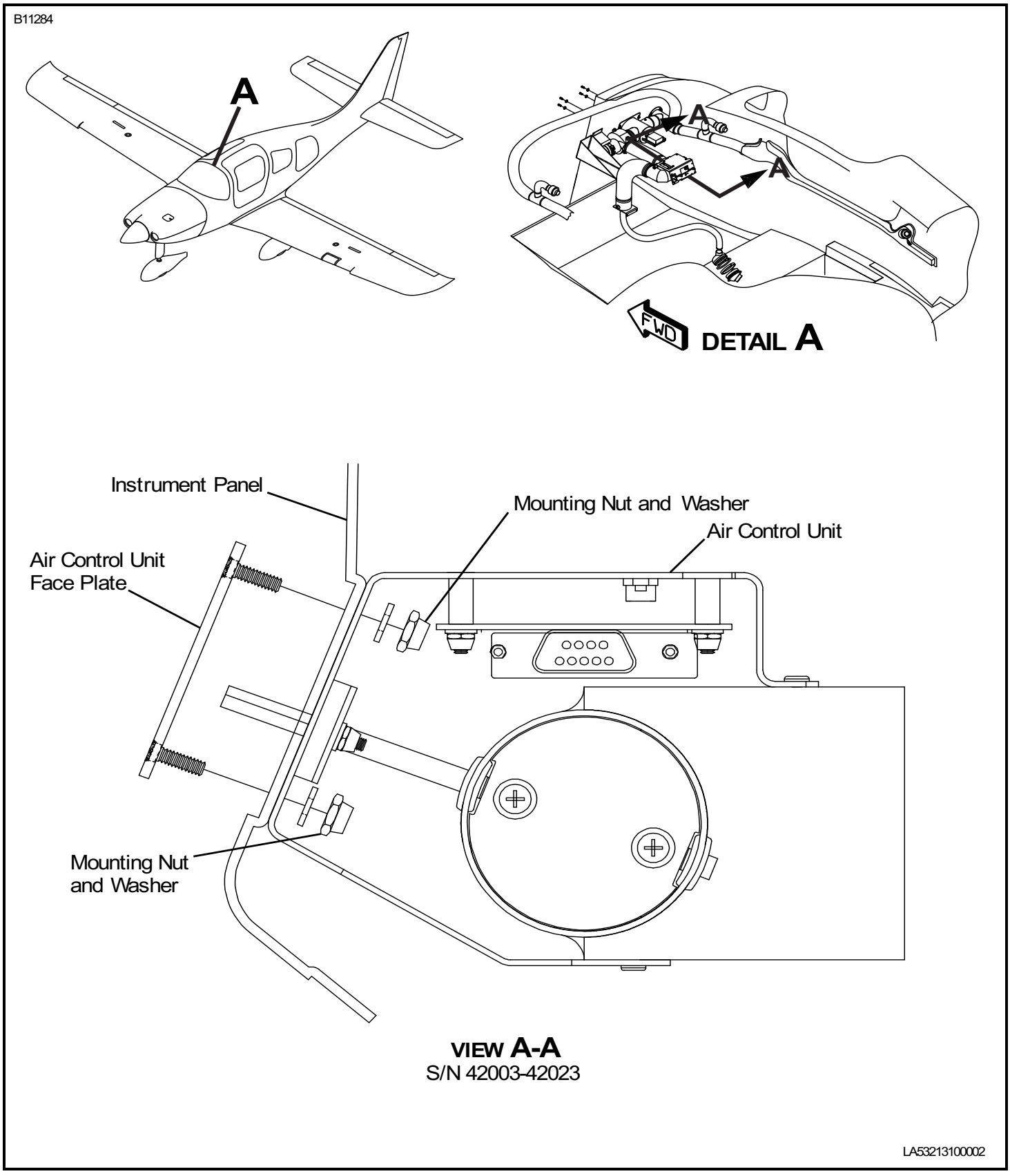
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Figure 202 : Sheet 1 : Air Control Unit Attachment



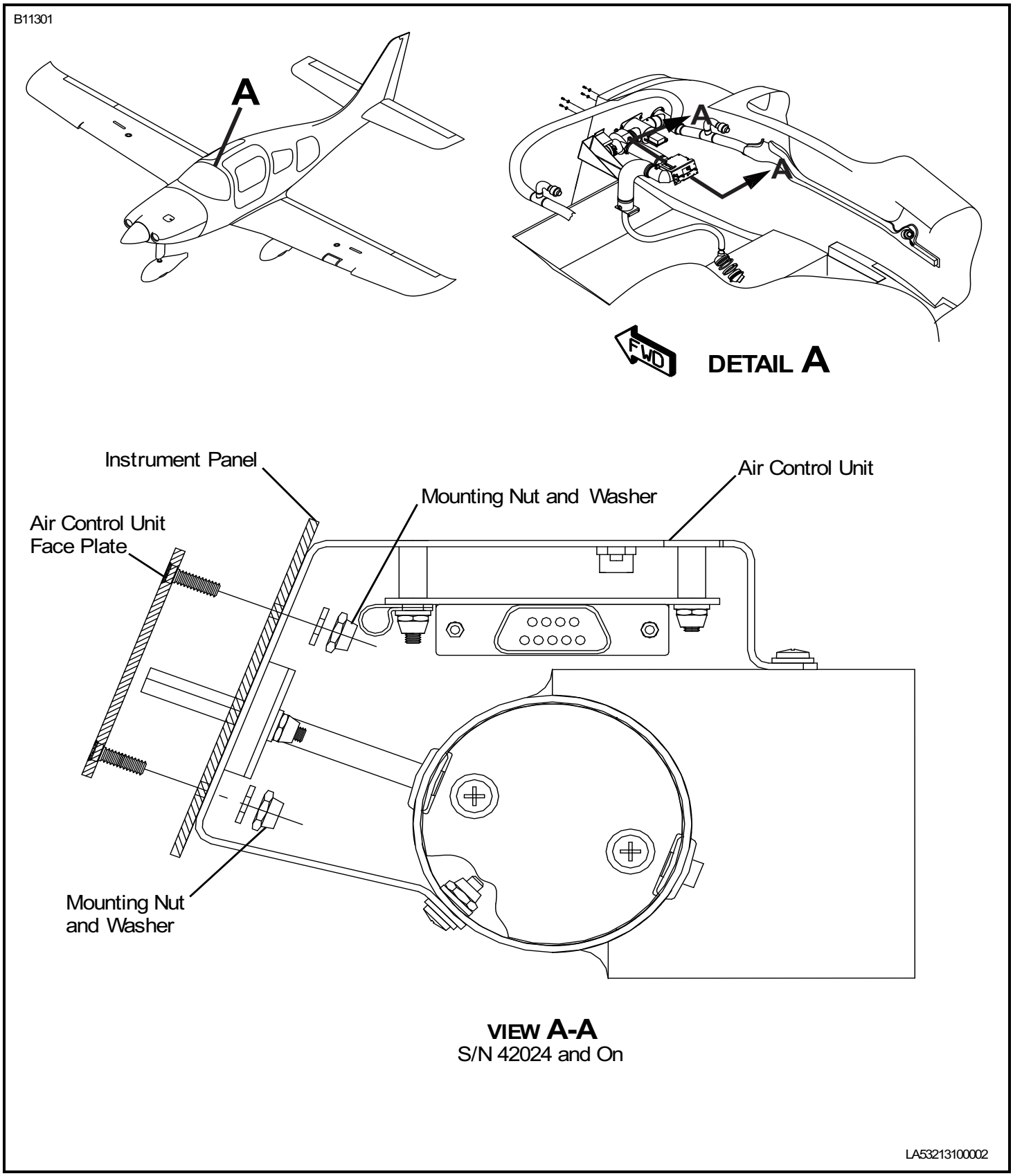
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Figure 203 : Sheet 1 : Air Control Unit Attachment



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Figure 203 : Sheet 2 : Air Control Unit Attachment



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AUTOMATIC CLIMATE CONTROL SYSTEM (ACCS) - MAINTENANCE PRACTICES

1. Automatic Climate Control System (ACCS)

A. **System Description** - The Automatic Climate Control System (ACCS) is a fully automatic system with manual operation capabilities comprised of a Control Head, Electronic Control Unit (ECU), temperature sensors, a Vapor Cycle System (VCS) air conditioning system, valves, hoses, and tubing. The air conditioning system consists of a compressor, a condenser with fans, a receiver-dryer with trinary pressure switch and an evaporator with an expansion valve and evaporator coil temperature sensor. Barrier hoses connect the components. A combination of steel and aluminum fittings are used. Refer to Figure 201.In manual or automatic operation, the system offers 11 choices for heater and air conditioning blower speed.On 28 VDC aircraft with ACCS equipped with the optional electric compressor, precooling of the aircraft cabin is possible using a ground power supply.

NOTE: Ground power operation of the air conditioning will require a ground power source that can deliver 100 amps during use.

After connecting a ground power source and switching the unit on, the ACCS can be activated by pushing the external switch found near the ground power receptacle. When activated, the aircraft power grid is disabled and the electric compressor and evaporator blower will run continuously while the condenser fans automatically cycle as needed. The external ACCS switch does not function when a battery master is on. The ACCS control panel is disabled during external power air conditioning operation and the ACCS cools at maximum capability. External power ACCS operation can be deactivated by pushing the external switch, removing the ground power source, or turning on either battery master switch. With a battery master on, the ACCS will be fully functional except the electric compressor will be off. An electric A/C interlock assembly allows this integration of the ACCS with the airplanes electrical system. Refer to Section 24-50-00 for removal and installation of the interlock assembly.The ACCS has self-diagnostic capabilities which can be displayed on the screen of the Control Head. Refer to Table 201.

B. **Air Distribution** - Air distribution is controlled by the ACCS with pilot, or co-pilot, input through the control head. Refer to Figure 202.

(1) Air Conditioning - When air conditioned air is required, the evaporator blower is actuated and air is distributed through the overhead ceiling console vents and associated flood ducts.

(2) Heat - When heat is required, the ACCS shuts off the compressor and evaporator blower and regulates the heater/defog blower in the forward cabin. Heated air is distributed approximately 90 percent to the floor ducts with the remaining air being diverted to the windscreen defog ducts.

(3) Defog - When the Defog mode on the Control Head is selected, the heater/defog blower is activated at maximum speed and all air is distributed to the windscreen. Temperature selection on the Control Head regulates the temperature of the defog air.

C. **Diagnostics** - To display the Diagnostics Fault Code display, turn on the ACCS (or ECS on Garmin G1000 equipped aircraft) and depress the ON button three times in succession. The digital display on the Control Head will display any active fault codes. Repeatedly pressing the ON button will scroll through the active codes. Each code displayed should be addressed in turn until the No Faults code E00 is displayed. Refer to Table 201. With the Garmin G1000 option and no ACCS, fault codes E04 and E08 will always display due to the absence of an evaporator temperature sensor or OAT sensor in the system. In this case, codes E04 and E08 may be ignored.

Table 201. Fault Display Codes

Fault Code	Information	Correction
E00	No Faults	-
E01	Cabin Temp Sensor-Shorted	Check cabin temp sensor wiring and sensor for correct operation. Refer to Figure 203.
E02	Cabin Temp Sensor-Open	Check cabin temp sensor wiring and sensor for correct operation. Refer to Figure 203.
E03	Evap Temp Sensor-Shorted	Check evaporator coil temp sensor wiring and sensor for correct operation. Refer to Figure 205.
E04	Evap Temp Sensor-Open	Check evaporator coil temp sensor wiring and sensor for correct operation. See Figure 205.
E05	Not Currently Used	-

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E06	Not Currently Used	-
E07	OAT Sensor-Shorted	Check OAT sensor wiring and sensor for correct operation. Refer to Figure 204.
E08	OAT Sensor-Open	Check OAT sensor wiring and sensor for correct operation. Refer to Figure 204.
E09	Not Currently Used	-
E10	Heater Blend Air Actuator-Open	Check actuator sensor wiring and sensor for correct operation.
E17	No Communication with ECU	Check ECU wiring and system circuit breakers for correct operation.

(1) **Advanced Diagnostics** - The Advanced Diagnostics display mode is activated by pressing and holding the ON button for five seconds while the Fault Code display is active.

Press the "ON" button to cycle through the following displays:

CAb	Cabin Temperature Sensor temperature reading
EUAP	Evaporator Coil Temperature Sensor temperature reading
HEAt	ECS mixing valve position (percentage open)
A-C	VCS compressor command (ON or OFF)
FAn	Blower Speed-Evaporator or Heater/Defog (percentage of max)

Press the OFF button to return to the normal display mode.

(2) **Firmware Version** - The Firmware Version display mode is activated by pressing and holding the ON button for five seconds while the Advanced Diagnostics display is active.

Press the ON button to cycle through the following displays:

ECU	ECU Module Firmware Version
CtrL	Control Head Firmware Version

Press the OFF button to return to the normal display mode.

(3) **Parameter Function** - The parameter function is activated by pressing and holding the OFF button for five seconds while the advanced diagnostics display mode is active.

(a) Press the FAN + or the FAN - buttons to select configuration (config) setting: Pf-x (x may be displayed as 0,1, or 2 depending on setting).

(b) Press the TEMP + or the TEMP - buttons to change the selected config setting to: Pf-1.

(c) Exit and save the settings in the programming mode by pressing the OFF button.

NOTE: Do not change any other settings within the configuration screen or damage to the system may occur.

D. **Safety and System Service Precautions**

(1) **Cabin Sealing** - All of the following items must be sealed to ensure that carbon monoxide does not enter the cabin of the aircraft. Use McMaster-Carr Foam Tape P/N 8512K24 or LP Aero Plastic Foam Tape P/N SP-FT114, 3M 425 foil tape, and PR 1428 B1/2 or RTV5818 sealant as indicated.

(a) Two clear access panels, or one plug, located on the aft condenser bulkhead - Ensure that foam tape is intact around the edges of the panels. When reinstalling the panels apply foam tape as required to form an airtight seal. When reinstalling the plug, apply Wacker Elastosil 952, Dow Corning 732, or GE RTV-108 Sealant.

(b) Baggage floor at the aft condenser attachment - Sealed with .25 in.+/- bead of sealant. Apply sealant as required.

(c) Components penetrating the forward condenser bay bulkhead, and hollow shank rivets in the condenser compartment - Apply sealant as required.

(d) Lower corners of condenser compartment and gear member seals - Apply sealant as required.

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(e) Between gearbox and fuselage on exterior near forward side of gear leg attachment - Apply sealant as required.

(f) Air conditioning system bay perimeter - Ensure foam tape is intact and installed on both sides of the attachment holes. Apply foam tape as required.

(g) Air conditioning system bay access cover - Seal the air conditioning system bay access cover to the floor using foil tape to create a leak-proof seal.

(2) **Warnings and Cautions for R134a Refrigerant**

CAUTION: Liquid refrigerant is corrosive to metal surfaces. Follow the operation instructions supplied with equipment being used.

CAUTION: Never add R-12 refrigerant to an air conditioning system designed to use R-134a. Damage to the system may result.

CAUTION: R-12 compressor oil cannot be mixed with the R-134a compressor oil. They are NOT compatible.

CAUTION: Do NOT use R-12 servicing equipment or parts on an R-134a air conditioning system.

(a) Avoid breathing R-134a refrigerant, lubricant vapor, or mist. Exposure may irritate eyes, nose, and throat. Wear eye protection when servicing the air conditioning system. Serious eye injury can result from eye contact with refrigerant. If eye contact is made, seek medical attention immediately.

(b) If accidental system discharge occurs, ventilate the work area before resuming service. Large amounts of R-134a refrigerant will displace oxygen and cause suffocation. Work only in well-ventilated areas.

(c) Do not heat refrigerant containers above 125° F or expose refrigerant to open flame. Do not use open flame to heat refrigerant containers.

(d) Do not intentionally drop, puncture, or incinerate refrigerant containers.

(e) The evaporation rate of R-134a refrigerant at average temperature and altitude is extremely high. As a result, anything that comes in contact with the refrigerant will freeze. Always protect skin or delicate objects from direct contact with refrigerant. For personal protection, goggles and protective gloves should be worn and clean cloth wrapped around fittings, valves and connections when doing work that includes opening the refrigerant system. If R-134a refrigerant comes in contact with any part of the body, severe frostbite and personal injury can result. Flush exposed zone immediately with cold water and obtain prompt medical assistance.

(f) R-134a service equipment or aircraft air conditioning system should not be pressure tested or leak tested with compressed shop air. Though R-134a is considered non-flammable, some mixtures of air and R-134a have been shown to be combustible at elevated pressures. These mixtures are potentially dangerous and may result in fire or explosion.

(g) Never add R-134a refrigerant to a system that has not been evacuated to 29 inches Hg. vacuum pressure.

(3) **System Service Precautions**

(a) Never open or loosen a connection before discharging the system refrigerant.

(b) Do NOT open a refrigerant system or uncap a replacement component unless its temperature is as close as possible to room temperature. This will prevent condensation from forming inside of a component which is cooler than the surrounding air.

(c) Before disconnecting a component from the system, clean the outside of the fittings thoroughly.

(d) Immediately after disconnecting a component from the system, seal the open fittings with a cap or plug.

(e) Before connecting an open fitting always install a new seal or gasket designed for the specific component. Coat the fitting and seal with clean refrigerant oil before connecting.

(f) Do NOT remove the sealing caps from a replacement component until ready to install.

(g) When installing a refrigerant line avoid bends which produce radii smaller than specified below for the hose being installed.

Table 202. Minimum Bend Radii for Various Hose Sizes

Hose Size	Minimum Bend Radius
1/2 in. 13mm (No. 10)	3.0 inch
13/32 in. 10mm (No. 8)	2.5 inch

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5/16 in. 8mm (No. 6)	2.0 inch

(h) Position refrigerant lines to avoid exhaust components, flight controls, or items which may chafe the line. Refer to A.C. 43-13, *Acceptable Methods, Techniques, and Practices,* for compliant methods of installation.

(i) Apply a light coat of Ester RL-500S (POE) (engine driven compressor) or FVC68D (PVE) (electric compressor) refrigerant oil to the O-rings of all fittings. Tighten fittings only to the specified torque listed below. Use an anti-torque backup wrench on ALL component fittings. The refrigerant fittings will not tolerate over-tightening.

Table 203. Specific Torques for Various Fittings

Tube O.D.	Thread Size	Torque (Ft-lbs.)
3/8 in. (.375 in.)	5/8-18	11-13
1/2 in. (.500 in.)	3/4-16	15-20
5/8 in. (.625 in.)	7/8-14	21-27

(j) When disconnecting a fitting, use a wrench on BOTH halves of the fitting. This will prevent twisting of the refrigerant lines or tubes.

(k) Refrigerant oil will absorb moisture from the atmosphere if left uncapped. Do NOT open an oil container until ready to use and install the cap immediately after using. Store the oil only in a clean moisture-free container.

(l) Keep service tools and the work area clean. Contamination of the VCS air conditioning system through careless work habits must be avoided.

(m) Plumbing systems in aircraft are subject to vibration and extreme temperature and atmospheric pressure changes. Therefore, strict adherence to procedures and correct use of material is mandatory.

E. System Components

(1) Refrigerant Oil - There is either POE or PVE refrigerant oil used in the air conditioning system depending upon the type of compressor installed on the airplane. POE refrigerant oil is used with the engine driven compressor and PVE refrigerant oil is used with the electrically driven compressor. It is important to have the correct amount of oil in the air conditioning system. This will ensure proper lubrication of the compressor. Too little oil will result in damage to the compressor. Too much oil will reduce the cooling capacity of the system.

CAUTION: Do not mix POE and PVE refrigerant oils. Verify the type of oil used in the refrigerant system before starting any work on the system. If the type of oil used is in doubt, contact Cessna.

(a) The oil used in the system is either Ester RL-500S (POE) which is wax-free synthetic ester refrigerant oil or FVC68D (PVE) which is a polyvinyl ether refrigerant oil. Do NOT use any other oil when servicing or repairing the system. The oil container should be kept tightly capped until it is ready for use and then capped immediately after use to prevent contamination. Refrigerant oil will quickly absorb any moisture it comes in contact with.

(b) When an air conditioning system is assembled at the factory all components except the compressor are refrigerant oil-free. After the system has been charged and operated, the oil in the compressor is dispersed through the system. The evaporator, condenser, receiver/dryer, and compressor will retain a significant amount of oil.

(c) When a receiver/dryer, condenser, or evaporator core is replaced, 1 fluid oz. (29 cc) of refrigerant oil must be added.

(d) When the compressor is replaced, the new compressor is factory filled with the correct amount of oil. It will not be necessary to check the oil level in the compressor or to add oil unless there has been an oil loss. This may be due to a rupture or leak from a line, shaft seal, evaporator, or condenser. Oil loss at a leak point will be evident by the presence of a wet shiny surface around the leak.

(2) VCS Compressor - The air conditioning compressor will be either engine driven or electrical, depending upon the configuration of the aircraft.

(a) Engine Driven Compressor - The compressor is a swash or wobble plate axial compressor with a displacement of 147 cu. cm. The compressor cycles to regulate cabin temperature and to avoid evaporator coil freeze-up. The compressor is engaged by a magnetic clutch driven by a V-Belt off the accessory drive at the back of the engine. The magnetic clutch is switched off during air conditioning operation anytime the refrigerant system pressure drops below 29 psi (2 bar) at the trinary switch or anytime the high-pressure side of the refrigerant system rises above 397 psi (27 bar). Refer to Figure 206.

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1 Engine Driven Compressor Removal - To remove the compressor use the following instructions (refer to Figure 206):

a Remove the upper and lower engine cowling per the instructions in Chapter 71.

b Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

c Remove the evaporator/blower assembly cover. Refer to Figure 211.

d Evacuate the air conditioning system.

e Remove the belt guard.

f Loosen the V-belt idler sheave and remove the V-belt.

g Disconnect the two refrigerant hoses from the compressor and seal all open fittings with a cap or plug.

h Remove 3 AN6-16A bolts, washers, and self-locking nuts attaching the compressor to the engine.

i Remove the compressor.

2 Engine Driven Compressor Installation - To install the compressor use the following instructions:

a Attach the compressor to the engine using 3 AN6-16A bolts, washers, and self-locking nuts. Torque 20 to 25 ft.-lbs.

b Place the V-belt onto the compressor pulley.

c Install the belt guard using AN6H-10A and AN6H-14A bolts, and washers. Torque 275 to 325 in.-lbs. Safety wire the bolts.

d Tighten the V-belt idler sheave to provide the belt tension recommended by Teledyne Continental Motors service procedures.

e Connect the two refrigerant hoses to the compressor.

f Replace the receiver/dryer if required per the Removal and Installation of the Receiver/Dryer procedures.

g Evacuate and charge the system per the System Evacuation and System Charging procedures.

h Replace the Evaporator/Blower Assembly cover.

i Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

3 Maintenance - The compressor is self-contained and maintenance is not required. Inspection of the drive belts and pulleys should be completed at each 100 hour or annual inspection.

CAUTION: Do NOT overtighten the belt. After approximately 5 hours of initial operation, recheck the belt tension and adjust as required .

a Inspect the compressor drive belt for chafing and/or wear. The belt must be replaced if there is any indication of fraying or dry rot. Check the belt tension and adjust per Teledyne Continental Motors recommended service procedures as required.

b Inspect the compressor drive belt pulleys for security, tightness, rotational smoothness, and freedom.

(b) Electrical Compressor - The electric compressor is a brushless, hermetic DC variable speed compressor. The compressor operates at a fixed high speed determined by an electronic motor controller which cycles the compressor between high speed and idle speed depending on requirements of the system. Electric power to run the compressor is provided by an accessories alternator mounted on the aft of the engine. The compressor is protected by the trinary switch which senses out of limit system pressures. Any time the system pressure drops below 29 psi or rises above 397 psi, the compressor is shut down. Refer to Figure 207.

1 Electrical Compressor Removal

a Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

b Remove the elevator interconnect access panel per instructions in Chapter 25.

c Remove the evaporator/blower assembly cover. Refer to Figure 211.

d Evacuate the air conditioning system.

e Disconnect all electrical connections to the compressor.

f Disconnect the two refrigerant hoses from the compressor and seal all open fittings with a cap or plug.

g At each of three locations remove the AN4-10 bolt, washers, isolation ring, isolation bushing, and bolt

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sleeve spacer attaching the compressor to the shelf.

h Remove the compressor.

2 Electrical Compressor Installation

a Ensure that the wet weight of the compressor is 14.68 lbs. when installed in a dry system. Add refrigerant oil as required.

b At each of three locations attach the compressor to the shelf using the AN4-10 bolt, washers, isolation ring, isolation bushing, and bolt sleeve spacer.

c Connect the two refrigerant hoses to the compressor.

d Connect all electrical connections to the compressor.

e Evacuate and charge the system per the System Evacuation and System Charging procedures.

f Replace the Evaporator/Blower Assembly cover.

g Replace the Elevator Interconnect Access Panel per instructions in Chapter 25.

h Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

3 Electrical Compressor Maintenance - The VCS compressor is self-contained and maintenance is not required.

(3) Condenser/Fan Assembly - The condenser receives high-temperature, high-pressure vapor from the compressor. The vapor enters at the upper fitting of the condenser. As the vapor flows through the condenser, it releases heat to the cooler ambient air flowing over the condenser. As the vapor releases heat it changes to a liquid. Under average load, 2/3 of the condenser contains refrigerant vapor and the other 1/3 contains liquid refrigerant. The liquid refrigerant in the condenser has lost much of its heat, but still remains at a relatively high temperature and pressure. The assembly is located beneath the baggage compartment floor with vents for inlet and outlet directly below, through the belly of the aircraft. Refer to Figure 208.

(a) Removal of the Condenser/Fan Assembly - To remove the condenser/fan assembly use the following instructions:

1 Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

2 Remove the evaporator/blower assembly cover, the avionics shelf access panel and the air conditioning system bay access cover. Refer to Figure 211.

3 Evacuate the air conditioning system.

4 Disconnect the wiring harness.

5 Disconnect the two refrigerant hoses from the condenser/fan assembly and seal all open fittings with a cap or plug.

6 Remove 5 screws and 3 bolts mounting the condenser/fan assembly to the fuselage bulkheads.

7 Lift the Condenser/Fan assembly out of the compartment.

(b) Installation of the Condenser/Fan Assembly - To install the condenser/fan assembly use the following instructions:

1 Install PR1428 B1/2 or RTV5818 sealant, as required, at all openings between the air conditioning bay and the cabin area (including underlying floors) to completely seal off the cabin area from the air conditioning bay.

2 Replace the Condenser/Fan assembly in the compartment.

3 Reinsert the 5 screws and 3 bolts mounting the condenser/fan assembly to the fuselage bulkheads.

4 Reconnect the wiring harness.

5 Remove caps and plugs from any openings that were sealed and reconnect the two refrigerant hoses from the condenser/fan assembly.

6 Install McMaster-Carr foam tape P/N 8512K24, as required, around the perimeter of the air conditioning system bay.

7 Install a .25 in. +/- bead of PR1428 B1/2 or RTV5818 sealant between the baggage floor and the aft condenser attachment.

8 Evacuate and charge the system. Add 1 fluid oz. (29 cc) of refrigerant oil when installing a new

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condenser/fan assembly.

9 Replace the air conditioning system bay access cover. Seal the air conditioning system bay access cover to the floor using 3M 425 adhesive foil tape to create a leak proof seal.

10 Replace the avionics shelf access panel, and the evaporator/blower assembly cover.

11 Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

(c) Maintenance - The condenser/fan assembly is a self-contained unit and maintenance is not required.

(4) Expansion Valve - The expansion valve is located at the inlet to the evaporator. It is the connection point between the high-pressure and low-pressure side of the refrigeration system. The expansion valve determines the flow rate of the refrigerant and provides a pressure drop, reducing pressure on the liquid refrigerant before it enters the evaporator. Refer to Figure 209 and Figure 211.

(a) Removal of the Expansion Valve - To remove the expansion valve use the following instructions:

1 Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

2 Remove the evaporator/blower assembly cover. Refer to Figure 210.

3 Evacuate the air conditioning system.

4 Disconnect the refrigerant hose from the expansion valve and the expansion valve from the evaporator/blower assembly, and seal all open fittings with a cap or plug.

(b) Installation of the Expansion Valve - to install the expansion valve use the following instructions:

1 Connect the expansion valve to the evaporator/blower assembly and connect the refrigerant hose to the expansion valve. Torque the fittings.

2 Evacuate and charge the system.

3 Replace the evaporator/blower assembly cover.

4 Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

(c) Maintenance - The expansion valve is a self-contained unit and maintenance is not required.

(5) Evaporator/Blower Assembly - The evaporator uses a plate and fin evaporator coil. The tubing is hydraulically expanded into the fins to ensure good thermal conductivity. The evaporator removes heat and moisture from air entering the aircraft cabin. Low-pressure liquid refrigerant from the expansion valve enters the evaporator. Due to the reduced pressure, the refrigerant boils into a vapor. The boiling refrigerant absorbs the heat from the incoming air as it changes from a liquid to a vapor. The unit is located under a cover against the aft wall of the baggage compartment. Refer to Figure 211.

(a) Removal of the Evaporator/Blower Assembly - To remove the Evaporator/Blower Assembly, use the following instructions:

1 Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

2 Remove the Evaporator/Blower Assembly cover and the elevator interconnect cover. Refer to Figure 211.

3 Evacuate the air conditioning system.

4 Disconnect all electrical connections to the Evaporator/Blower Assembly.

5 Remove the Expansion Valve.

6 Disconnect the low-pressure refrigerant hose from the Evaporator/Blower Assembly and seal all open fittings with a cap or plug.

7 Remove 2 screws and washers connecting the overhead air duct to the Evaporator/Blower Assembly.

8 Remove 3 screws attaching the Evaporator/Blower Assembly to the aircraft.

9 Remove the Evaporator/Blower Assembly.

(b) Installation of the Evaporator/Blower Assembly - To install the Evaporator/Blower Assembly, use the following instructions:

1 Slide the Evaporator/Blower Assembly outlet into the overhead air duct.

2 Install 3 screws attaching the Evaporator/Blower Assembly to the aircraft.

3 With 2 screws and washers, attach the overhead air duct to the Evaporator/Blower Assembly.

4 Install the Expansion Valve.

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5 Attach the low-pressure refrigerant hose to the evaporator/blower.

6 Reconnect all electrical connections to the evaporator/blower assembly.

7 Evacuate and charge the system. Add 1 fluid oz. (29 cc) of refrigerant oil when installing a new evaporator/blower assembly.

8 Replace the elevator interconnect cover and evaporator/blower assembly cover.

9 Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

(c) Maintenance - The evaporator/blower assembly is a self-contained unit and maintenance is not required.

(6) Receiver/Dryer with Trinary Switch - The receiver/dryer acts as a storage and filter unit for refrigerant oil. The trinary switch serves two functions. One portion of the switch cycles the condenser fans once the refrigerant pressure reaches 235 psi (16 bar), and turns off the fans at 190 psi (13 bar). The other portion of the switch protects the system from damage due to low refrigerant or excessive pressures by turning off the compressor. Power to the compressor is interrupted once the refrigerant pressure drops below 29 psi (2 bar) or exceeds 397 psi (27 bar) at the trinary switch. The trinary switch automatically resets once pressures are restored to a safe level.The Trinary Switch is considered an LRU (Line Replaceable Unit) which allows it to be replaced independently of the receiver/dryer assembly. The switch is sealed with an O-ring on the receiver/dryer and releases a Schrader valve upon removal, thus allowing replacement without evacuating the refrigeration system. It is important to note that the 14VDC switch and the 28VDC switch, although identical electrically, have different electrical interface connectors. The correct switch must be ordered based on the configuration (14VDC or 28VDC) of the aircraft being serviced.The receiver/dryer assembly is attached to the landing gear tab located beneath the baggage compartment, forward of the condenser bay on the copilot's side. Refer to Figure 212.

(a) Removal of the Receiver/Dryer and/or the Trinary Switch - To remove the receiver/dryer and/or the trinary switch use the following instructions:

1 Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

2 Remove the evaporator/blower assembly cover and the avionics shelf access panel. Refer to Figure 211.

3 Disconnect the electrical connections from the Trinary Switch.

4 Remove the trinary switch, if desired. The trinary switch may be removed without evacuating the system. Refer to the Receiver/Dryer with Trinary Switch section above. For removal of the Receiver/Dryer continue with the following steps.

5 Evacuate the air conditioning system.

6 Disconnect the refrigerant hoses from the receiver/dryer and seal all open fittings with a cap or plug.

7 Remove two bolts, washers, and self locking nuts and remove the receiver/dryer.

(b) Installation of the Trinary Switch Only - To install the trinary switch use the following instructions:

1 Screw in the trinary switch. The tightening torque for the trinary switch is 7 ft.-lbs.

2 Reconnect the electrical connections to the trinary switch.

3 Replace the avionics shelf access panel and the evaporator/blower assembly cover.

4 Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

(c) Installation of the Receiver/Dryer - To install the receiver/dryer use the following instructions:

1 Attach the Receiver/Dryer to the landing gear tab with two bolts, washers, and self locking nuts.

2 Attach the refrigerant hoses to the receiver/dryer.

3 Reconnect the electrical connections to the trinary switch.

4 Evacuate and charge the system. Add 1 fluid oz. (29 cc) of refrigerant oil when installing a new receiver/dryer.

5 Replace the avionics shelf access panel and the evaporator/blower assembly cover.

6 Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

(d) Maintenance - The receiver/dryer with trinary switch assembly is a self-contained unit and maintenance is not required.

(7) Cabin Temperature Sensor - The cabin temperature sensor assembly contains a calibrated thermistor and fan used to draw cabin air over the thermistor. The sensor assembly provides input data to the ECU. Cabin temperature is

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determined by comparing the inside temperature of the aircraft with the desired cabin temperature setting. The temperature is regulated by cycling the compressor and regulating the evaporator blower speed during cooling requirements and by regulating the heater temperature door and heater/defogger blower speed during heating requirements. If the sensor fails, a fault will be noted in the ACCS and manual temperature control will be required through the control head. The sensor assembly is located behind the co-pilot's knee bolster. If the Garmin G1000 system is installed, the sensor assembly is at the back of the instrument panel behind the MFD or mounted behind the headset holder at the front face of the tower.

(a) Removal of the Cabin Temperature Sensor - To remove the cabin temperature sensor use the following instructions (refer to Figure 213):

1 Remove the right knee bolster per instructions in Chapter 25. If the Garmin G1000 system is installed the sensor may be accessed from underneath the instrument panel or by removing the tower per Chapter 25, as applicable.

2 Disconnect the electrical connection from the cabin temperature sensor.

3 Remove 4 screws attaching the cabin temperature sensor to the top of the right knee bolster, to the sensor mounting plate, or to the headset holder, as applicable.

4 Remove the cabin temperature sensor, spacer, and screen.

(b) Installation of the cabin temperature sensor - installation is the reverse of removal.

(c) Maintenance of the Cabin Temperature Sensor - The cabin temperature sensor is a self-contained unit and maintenance is not required.

(8) Outside Air Temperature (OAT) Sensor - The OAT temperature sensor continuously measures the temperature of the air passing by the outside of the aircraft. The sensor is used as a basic variable for temperature regulation. The ECU also uses the signal from this sensor to calculate the temperature for the OAT indication on the control head display. The temperature sensor is located in the condenser bay on the forward bulkhead. Refer to Figure 214.

(a) Removal of the OAT Sensor - To remove the OAT Sensor use the following instructions (Refer to Figure 214):

1 Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

2 Remove the evaporator/blower assembly cover, the avionics shelf access panel and the air conditioning system bay access cover. Refer to Figure 211.

3 Disconnect the electrical connection to the OAT.

4 Remove the "C" clip attaching the OAT sensor to the bulkhead and remove the sensor.

(b) Installation of the OAT Sensor - Installation is the reverse of removal.

1 Install PR1428 B1/2 or RTV5818 sealant, as required, at all openings between the air conditioning bay and the cabin area (including underlying floors) to completely seal off the cabin area from the air conditioning bay.

2 Install McMaster-Carr foam tape (P/N 8512K24), as required, around the perimeter of the air conditioning system bay.

3 Replace the air conditioning system bay access cover. Seal the air conditioning system bay access cover to the floor using 3M 425 adhesive foil tape to create a leak proof seal.

4 Replace the avionics shelf access panel, and the evaporator/blower assembly cover.

(c) Maintenance - The OAT Sensor is a self-contained unit and maintenance is not required.

(9) Defog/Floor Vent Valve Assembly - During heater operation this valve normally distributes air to the floor vents at a volume of approximately 90 percent, with the remaining 10 percent going to the windscreen defog outlets. When the Defog mode is selected, the valve distributes nearly 100 percent of the air to the windscreen defog outlets. The valve operates in either a "full on" or "full off" position. A minimum of 10 percent of the air distributed by the heater/defog blower is distributed to the defog ducts anytime the heater/defog blower is operating. This valve is attached to the upper windscreen defrost channel/plenum, beneath the center of the instrument panel. Refer to Figure 215.

(a) Removal of the Defog/Floor Vent Valve Assembly - To remove the defog/floor vent valve assembly use the following instructions.

1 Disconnect the electrical connector from the defog/floor vent valve.

2 Remove the hose clamps attaching the defog/floor vent valve to the flexible ducting and defrost channel.

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a On Garmin G1000 aircraft, the defog/floor vent valve may be attached directly to the defrost channel. With this condition, remove the hose clamp and a MS51861-25C screw. There is a length of silicone sheeting between the defog/floor vent valve and the defrost channel.

3 Pull the flexible ducting away from the defog/floor vent valve.

4 Remove the RTV5818 silicone sealant applied between the defog/floor vent valve and the defrost channel and remove the valve.

(b) Installation of the Defog/Floor Vent Valve

1 Defog/floor vent valve attached directly to the defrost channel:

a Fill the gap between the inner diameter of the valve flange and the outer diameter of the defrost channel flange with 0.062 inch or 0.093 inch AMS3320 silicone sheet.

b When installing a new defog/floor vent valve, match drill a 0.166-inch diameter (No. 32) hole in the valve flange.

c Install a MS51861-25C screw through the valve flange, the defrost channel flange, and the silicone sheet. Dry torque hand tight plus 1/4 turn.

d Apply RTV5818 silicone all around the inlet port, install flexible ducting, and secure with a hose clamp.

e Reconnect the electrical connector.

f Turn on the power, and test the defog/floor vent valve for proper operation.

2 Defog/floor vent valve not attached directly to the defrost channel:

a Apply RTV5818 silicone all around the inlet and outlet ports of the defog/floor vent valve.

b Slide the flexible ducting over the inlet and outlet ports of the defog/floor vent valve. Rotate the valve as required to clear the rudder pedal tubes and ECS servo motor actuator screw, and secure with hose clamps.

c Reconnect the electrical connector.

d Turn on the power, and test the defog/floor valve for proper operation.

(c) Maintenance - The defog/floor vent valve assembly is a self-contained unit. Only a periodic check for proper valve operation is required.

(10) ECS Shut-Off Valve Assembly - When the ACCS is operating in the air conditioning mode, this valve remains closed to prevent outside air from entering through the ECS mixing valve and entering the cabin. During the heat mode the valve opens allowing air from the ECS mixing valve to enter the cabin. This valve is attached to the heater blower, beneath the center of the instrument panel. Refer to Figure 216.

(a) Removal of the ECS Shut-Off Valve Assembly - To remove the ECS shut-off valve assembly use the following instructions:

1 Disconnect the electrical connector from the ECS shut-off valve assembly.

2 Remove 3 tangential worm drive hose clamps attaching the ECS shut-off valve assembly to the flexible ducting and to the ECS Fan.

3 Pull the flexible ducting away from the ECS shut-off valve assembly.

4 Remove the RTV5818 silicone sealant applied between the Shut-Off Valve Assembly and the ECS Fan and remove the shut-off valve.

(b) Installation of the ECS shut-off valve assembly - to install the ECS Shut-Off Valve Assembly use the following instructions:

1 Installation is the reverse of removal.

2 Apply RTV5818 silicone sealant at the connection between the defrost plenum and the shut-off valve assembly before securing the clamp.

(c) Maintenance of the ECS Shut-Off Valve Assembly - the ECS shut-off valve assembly is a self-contained unit. Only a periodic check for proper valve operation is required.

(11) Heater Temperature Actuator (Servo Motor) - Heated air temperature in the cabin is adjusted via the heater temperature actuator. Its adjusting range extends from heating final stop (all air channeled from the heat exchanger) to cooling final stop (no air channeled from the heat exchanger). The potentiometer reports the position of the actuator to

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the ECU as a feedback value. The heater temperature actuator is activated by the ECU according to the temperature selected on the control head. The heater temperature actuator is located on the heater/ECS mixing box in the forward cabin. Refer to Figure 216 and Section 21-20-00, Figure 204.

(a) Removal and installation of the Heater Temperature Actuator - Removal and installation of the heater temperature actuator is the same as removal and installation of the servo motor, per Section 21-20-00, Servo Motor.

(b) Maintenance - The Heater Temperature Actuator is a self-contained unit and requires only a periodic check for proper operation.

(12) Electronic Control Unit (ECU) - The ECU is the control unit of the ACCS incorporating all the electronic circuitry and utilizing digital processing logic. The ECU receives signals from the control head and supplies outputs accordingly. The ECU also contains the blower control unit which regulates the evaporator and the 11 heater blower speeds. Voltage at the blowers is reported back to the control head. For S/N 42001 to 42501, the ECU is located on the pilot's side, beneath the evaporator cover, in the baggage compartment. Refer to Figure 212. For S/N 42502 and On, the ECU is mounted on the fuselage skin under the baggage bulkhead access panel.

(a) Removal of the ECU - To remove the ECU use the following instructions.

1 Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

2 For S/N 42001 to 42501, remove the evaporator/blower assembly cover. Refer to Figure 211. For S/N 42502 and On, remove the baggage bulkhead access panel.

3 Disconnect all electrical connections to the ECU.

4 Remove 3 screws (S/N 42001 to 42501) or 4 self-locking nuts and washers (S/N 42502 and On) and remove the ECU.

(b) Installation of the ECU - Installation of the ECU is the reverse of removal.

(c) Maintenance - The ECU is a self-contained unit and maintenance is not required.

(13) Control Head - For the Basic or Avidyne option, the control head is located to the right of the flap panel. For the Garmin G1000 option the control head is located in the tower assembly. The control head receives inputs from the pilot, or copilot, and the ACCS components. It is the user interface for the ACCS. Fault codes generated by the ACCS are displayed on the control head LED screen for troubleshooting. Refer to Figure 217.

(a) Removal of the Control Head - To remove the control head use the following instructions:

1 Turn the power off and disconnect the wiring harness from the back of the control head.

2 From the back side of the instrument panel remove the four nuts and keeper clamp attaching the Control Head to the instrument panel.

3 Remove the Control Head by pulling it forward from the front of the instrument panel.

(b) Installation of the Control Head - Installation of the control head is the reverse of removal.

(c) Maintenance of the Control Head - The control head is a self-contained unit. A periodic check for defog/floor vent assembly, ECS shut-off valve assembly, and heater temperature actuator operation is required. A functional check should be completed at each 100 hour or annual inspection.

1 Evaluate the adequacy of the system in terms of crew and passenger comfort, as well as the defroster's ability to control fogging of the windscreen.

2 Check for proper operation of the fresh air, air conditioning, heat, defog, air volume control, temperature control, and air output control.

3 Evaluate the eyeball vents for proper operation.

F. **Tools and Equipment** - The following tools and equipment are required for leak testing, evacuating, and charging the refrigerant system:

(1) Gaseous dry nitrogen, regulated source (0-500 psig or 0-35 bar)

(2) R-134a refrigerant

(3) Refrigerant oil, Ester RL-500S.

(4) R-134a refrigerant charging manifold with gauges and hoses with quick-disconnect couplers. The coupler has a knob on top which is used to engage and disengage the Schrader valve in the aircraft service ports.

(5) R-134a charging station and recovery/recycling station (SAE standard J2210).

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(6) Air conditioning vacuum pump
(7) Hose adapter, for nitrogen bottle (1/2 in. male acme to 1/4 in. SAE female flare)
(8) Hose adapter for vacuum pump (1/2 in. male acme to 1/4 in. SAE female flare)
(9) Electronic leak detector.
(10) Burroughs tool No. BT-33-73F (or equivalent) belt tension gauge
(11) Torque wrench capable of reading lb-in. or Nm.
(12) Small hand tools and socket set
(13) Thermometer, 0-150° F
(14) Inspection Mirror, adjustable
(15) Service light or flashlight
(16) Shop hand towels
(17) Weight scale, 0-50 lbs (0-25 kg) capable of 0.1 pound (5 gram) precision

G. **Leak Check Procedure** - System leak check is required whenever the air conditioning system is not cooling properly, there is a system component replacement, or a loss of R-134a refrigerant charge.
(1) Determine if the system is fully charged. Refer to Figure 218.
(2) If the system is empty evacuate and charge the system with 0.6 lbs. (296 grams) of R-134a refrigerant.
(3) Position the aircraft in a wind-free work area.
(4) Remove the engine cowling per instructions in Chapter 71.
(5) Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
(6) Remove the evaporator/blower assembly cover, the avionics shelf access panel and the air conditioning system bay panel.
(7) Operate the aircraft engine with the air conditioning on for 5 to 10 minutes until the operating temperature and pressures are achieved based on the existing refrigerant charge.
(8) Aircraft cabin doors and the baggage compartment door must be left open during test operation.
(9) Shut off the aircraft engine. Use an R-134a electronic leak detector and search for leaks. Move the probe slowly along the bottom side of lines and fittings, because R-134a is heavier than air. Fittings, lines, or components that appear to be oily usually are the site of a refrigerant leak.
(10) To inspect for evaporator core leaks, ensure the aircraft engine is NOT running, set the ACCS temperature to 55° F (13° C), select "LO" for the blower speed and check the air conditioning outlet vents. Also inspect the evaporator drain tube outlet for the presence of refrigerant oil.
(11) Tighten or repair joints as required to stop leaks.

CAUTION: Do not overtighten plumbing connections. Stripped threads or cracked flanges may result.

H. **System Evacuation Procedure** - If the air conditioning system has been open to the atmosphere, it must be evacuated before the system can be charged. Moisture and air mixed with refrigerant will raise the compressor head pressure above acceptable operating levels. This will reduce the performance of the air conditioning system and damage the compressor. Moisture will also boil at near room temperature when exposed to vacuum.R-134a refrigerant is a hydrofluorocarbon (HFC) that does not contain chlorine. An R-134a refrigerant recovery/recycling station that meets SAE standard J2210 must be used to evacuate the refrigerant system. Refer to the operating instructions provided with the equipment for proper operation.
(1) Connect the charging manifold to the service ports.
(2) Connect the manifold charging (yellow) hose to the vacuum pump and turn on the pump.
(3) Open both valves of the charging manifold gauge set.
(4) Open both valves of the charging hose quick connect fittings.
(5) Observe the charging manifold gauges to verify vacuum.
(6) Evacuate the system until 25 to 27 in. Hg (635 to 685 mm Hg) or greater vacuum is obtained.
(7) Close all valves and turn off the vacuum pump.
(8) If the system maintains the specified vacuum for 30 minutes, start the vacuum pump, open the suction and discharge

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valves, and evacuate the system for an additional 40 minutes, minimum.

(9) If the specified vacuum is not maintained, locate and repair the refrigeration system leak before continuing.

(10) Close all valves and turn off the vacuum pump.

(11) Disconnect the manifold service hose from the vacuum pump. The system is now ready for refrigerant charging.

CAUTION: Any change in vacuum pressure, or failure to achieve a system pressure of 25 to 27 in. Hg (635 to 685 mm Hg) vacuum indicates the presence of a plumbing leak. Repeat the leak check procedure. Locate and fix all leaks.

l. **System Charging Procedure** - There are various methods of charging refrigerant into the air conditioning system. These include using a refrigerant recovery/recycle unit (refer to the manufacturer's instruction manual). The use of manifold pressure gauges, R-134a cylinder, and a weight scale is outlined in this section. Only R-134a refrigerant is to be used. Other refrigerants will damage the system. Overcharging of the system will result in reduced performance, reduced service life, and/or damage to the system components.

(1) Leak check and evacuate the system per the Leak Check and System Evacuation procedures.

(2) Connect the yellow manifold charging hose to the R-134a cylinder and open the valve.

(3) Crack open the charge hose fitting at the manifold gauge set and vent air from the hose until refrigerant is evidently escaping.

(4) Place the refrigerant bottle on a 0-50 lb. (0-25 kg) scale and record the weight.

(5) Open both manifold gauge set valves. Add refrigerant to the system until pressure stabilizes.

(6) Close the manifold valves and verify that system pressure is 50 psig (3.4 bar) or greater.

NOTE: The system pressure must be above 50 psig (3.4 bar) to close the low-pressure cutoff switch. Otherwise the compressor will not turn on.

(7) If necessary, warm the refrigerant bottle by immersion in warm water.

CAUTION: Do not use open flame to warm the refrigerant bottle. Do not heat the bottle above 125° F (51° C).

(8) Move the aircraft to a run-up area.

(9) Start the aircraft engine. Select "AUTO" on the ACCS control head and set the temperature to 55° F (13° C). Idle the aircraft engine at 1200 to 1800 RPM.

(10) With the system operating, observe the system discharge and suction pressures.

(11) With the R-134a cylinder connected to the charging hose, charging container shutoff valve open and hose purged of air, slowly open the suction (blue) manifold valve. The suction pressure will increase to 60 to 70 psig (4.1 to 4.8 bar).

NOTE: As refrigerant enters the compressor, a slight increase in compressor discharge pressure will be noticed.

(12) Continue to add refrigerant until 24 oz. (680 grams) of refrigerant has been added.

(13) Close the suction manifold valve (blue) and let the system operate for 5 to 10 minutes to evaluate performance. Refer to Table 203 to diagnose any performance issues.

NOTE: Letting the system stabilize is required until the expansion valve can stabilize the system pressure.

NOTE: System charge is approximately 24 oz. (680 grams) of R-134a refrigerant.

CAUTION: Do NOT overcharge the air conditioning system. Overcharging will cause excessive compressor head pressure, loss of cooling, noise and system failure.

(14) With the system fully charged and operating, observe the suction and discharge pressures. Typical values at various ambient temperatures, with hot cabins, are shown below. These values should be used as a reference for troubleshooting, not as a sole source. Refer to Table 204.

Table 204. Typical Suction and Discharge Pressure

Ambient Temperature	21° C(70°F)	27° C(80° F)	32° C(90° F)	38° C(100° F)	43° C (110° F)
Air Temperature At Evaporator Outlet	7-9° C(45-48° F)	8-13° C(46-55° F)	11-16° C(52-61° F)	14-20° C(57-68° F)	17-25° C(63-77° F)

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Low Side Service Port Pressure	18	20	22	25	26
	psi	psi	psi	psi	psi
	30	35	39	43	47
High Side Service Port Pressure	150	160	220	250	280
	psi	psi	psi	psi	psi
	220	235	310	320	340

(15) Allow the system to operate for 10 minutes then shutdown.

NOTE: After shutdown, both suction and discharge pressures will begin to equalize.

(16) Close the refrigerant container shutoff valve. Record the refrigerant container final weight and calculate the system refrigerant charge.

(17) Turn the knob on the suction and discharge charging hose quick coupler to closed position and disconnect the hoses from the service ports.

(18) Remove the yellow charging hose from the refrigerant container and store the manifold gauge set.

(19) Install the service port caps.

J. **System Flushing** - System flushing can be successfully done to systems where there is insignificant or no evidence of debris or FOD in the system. Failure to follow the actions and direction below will void all Seamech warranty considerations.

(1) System flushing ***cannot*** be done to the complete system where a compressor has overheated and failed. The industry term for this type of failure is "Black Death" and is usually the result of low amounts of compressor lubricating oil in the system. This type of failure burns the remaining oil in the system and, combined with the polymers in the compressor, will form a hard, carbonized and impossible to remove buildup primarily in the receiver dryer and condenser. In this case, the hoses between the compressor and evaporator module, as well as the receiver dryer and condenser must be replaced. The expansion valve and evaporator module should be removed and inspected for evidence of black carbonized buildup. If black buildup exists, the expansion valve and/or evaporator module must also be replaced. If no evidence of black carbonized buildup exists, the expansion valve and evaporator module can be flushed using the techniques below.

(2) If a contamination event is the result of an internal compressor mechanical failure or other debris in the system, and there is evidence of metal or debris in the VCS components, then all of the VCS hoses and components must be disconnected and inspected. Today's high-efficiency condenser and evaporator coils are difficult to clean of all debris. These coils are parallel flow devices with deep wells on the manifolds where debris can and flushing material can pool making debris removal by flushing impossible. If there is evidence of debris in these devices, it is recommended that they be replaced to ensure the system can operate debris free when reassembled and serviced. In the event of a compressor mechanical failure, it is likely that the receiver/dryer, condenser and perhaps the evaporator module may require replacement. Hoses and other components of the system should be able to be flushed with success.

(3) If flushing can be done, only DuPont(TM) Mobile AC Flush Fluid is authorized on Seamech systems. More information about the fluid and where it can be obtained is available at http://refrigerants.dupont.com/Suva/en_US/products/suva_mobile_flush.html. High pressure and flow flushing equipment is required. Traditional flush guns, aerosol flushes and pour in flush methods do not produce the velocity necessary to force contaminates out of the affected component.

CAUTION: It is critical that ALL of the flush be removed from the condenser. Any remaining flush in the condenser can cause the compressor to fail prematurely. If there is any doubt as to whether there is flush in the condenser, it is strongly advised that the condenser be replaced. Evidence of flush in a failed compressor will void any compressor warranties.

(4) **VERY IMPORTANT** - Flushing ***must*** take place in the reverse flow direction (low side to high side) to push contaminates back out of the affected components on a component by component basis. This means when an evaporator coil is flushed, the thermal expansion valve (TXV) must be removed and flushing material must enter into the downstream or suction side fitting of the evaporator coil. In a similar way, flushing material must be applied to the bottom or discharge fitting of the condenser. Use gravity to assist in the cleaning.

(5) The thermal expansion valve must be removed from the evaporator module and thoroughly cleaned. Any

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contaminates in the expansion valve may cause the valve to operate improperly.

(6) The receiver dryer can never be flushed because it is not possible to remove all of the flushing compounds. The receiver dryer contains desiccant which will absorb the flushing compound. The receiver dryer should be replaced if there was any type of contamination event.

(7) All system hoses should be disconnected and flushed independently, opposite the direction of normal system refrigerant flow.

(8) Compressors shall never be flushed. Flushing material will harm the compressor and displace any oil in the unit. Do not flush compressors.

(9) Once all of the affected components of the system have been flushed, all components shall be blown dry with clean and very dry filtered shop air or nitrogen. **VERY IMPORTANT** - using air or nitrogen that is not dry will introduce moisture into the system components. This will cause internal corrosion over time.

(10) It is vitally important to remove all of the flushing chemicals from all of the components. This may mean the components have to be rotated, shaken, or tilted to ensure all of the flushing material has been removed. Residual flushing material can absorb the compressor oils and can lead to premature failure of the compressor and degraded cooling performance. Repeat the "blow out" and removal of the flushing compounds until you are confident that all contamination debris and flushing chemicals have been removed.

(11) All hose O-rings shall be replaced and the system reassembled per this maintenance manual.

(12) Follow requirements for servicing the system with oil. Only Ester RL-500S system oil can be used. If a new compressor is used, they usually contain the required amount of oil internal to the compressor. The VCS can then be re-serviced per this maintenance manual.

K. **Pressure and Performance Troubleshooting**

Table 205. Pressure and Performance Diagnosis

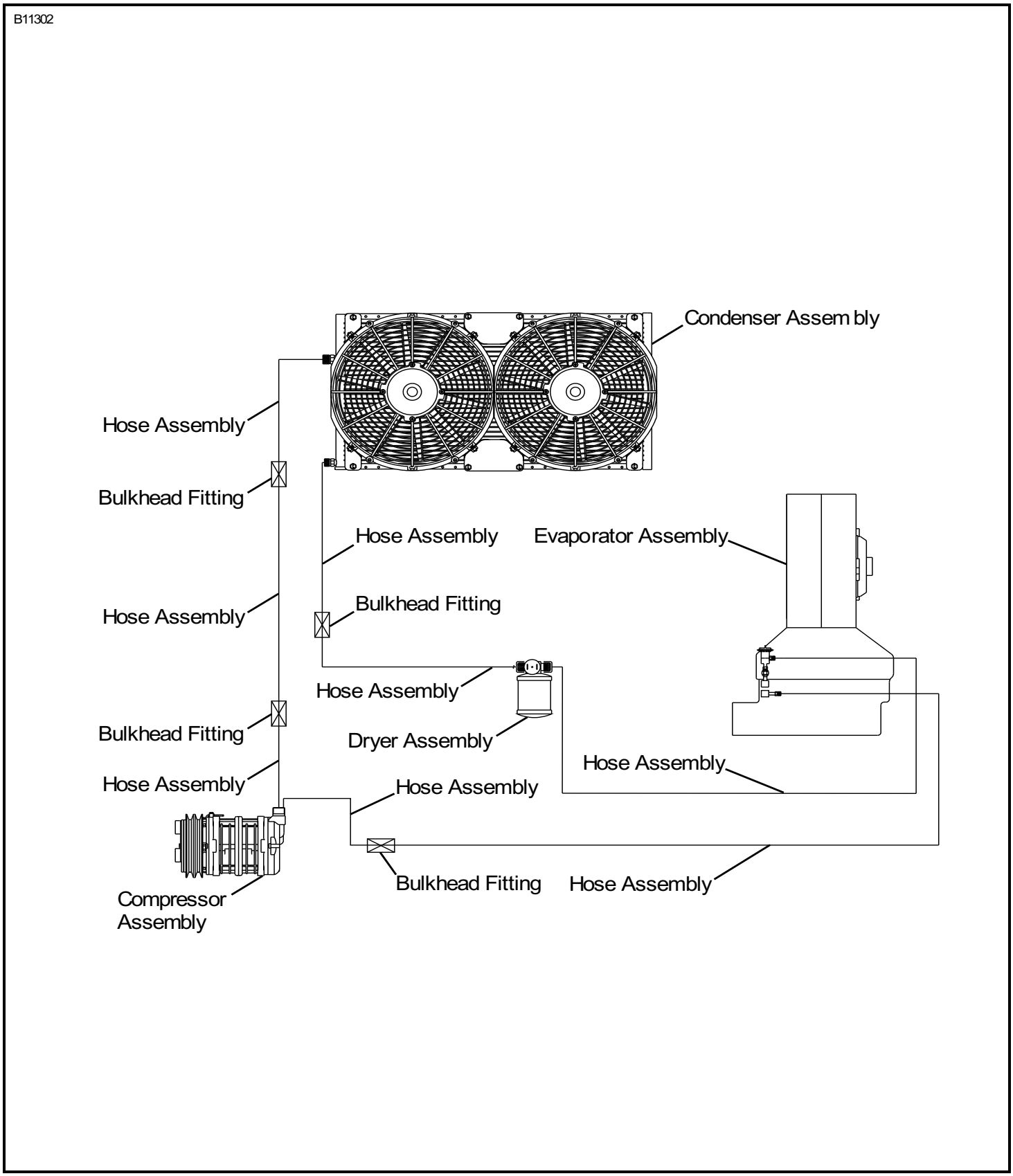
Condition	Possible Cause	Correction
Low side and high side pressure low	System refrigerant is low	Discharge, evacuate, leak test charge system
Low side pressure high and high side pressure low	Internal leak in compressor	Replace compressor.
	Drive belt slipping	Tension or replace belt.
Low side and high side pressure high	Condenser fins obstructed	Clean condenser fins.
	Air in refrigeration system	Evacuate, leak test and charge system.
	Condenser fans inoperative	Troubleshoot condenser fans.
	Refrigerant system overcharged	Recover refrigerant and recharge
Low side pressure low and high side pressure high	Restriction in refrigerant hose	Check hoses for kinks and replace if necessary.
	Restriction in receiver-dryer	Replace receiver-dryer.
	Restriction in condenser	Replace condenser.
	Expansion valve is defective	Replace expansion valve.
Low side and high side pressures normal (inadequate cooling)	Check for correct cooling at evaporator assembly outlet.	If temperature is correct, check for obstruction or leak in cabin ducting.
Compressor noise	Internal compressor damage	Replace compressor.
	Refrigerant system overcharged	Recover refrigerant and recharge.
	Loose compressor mounting	Tighten compressor mounting bolts to correct torque values.
	Loose or worn compressor drive belt	Tension or replace belt.
Excessive vibration	Incorrect belt tension	Tension or replace belt.
	Compressor loose	Tighten compressor mounting bolts to correct torque values.

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	Refrigerant system overcharged	Recover refrigerant and recharge.
	Drive or idler pulley worn	Replace defective pulley.
Condensation leaking inside aircraft	Evaporator drain plugged or kinked	Clean drain hose and check for proper installation.
Frozen evaporator coil	Faulty evap temp sensor	Replace evap temp sensor.
	Obstructed evaporator coil	Removed obstruction.

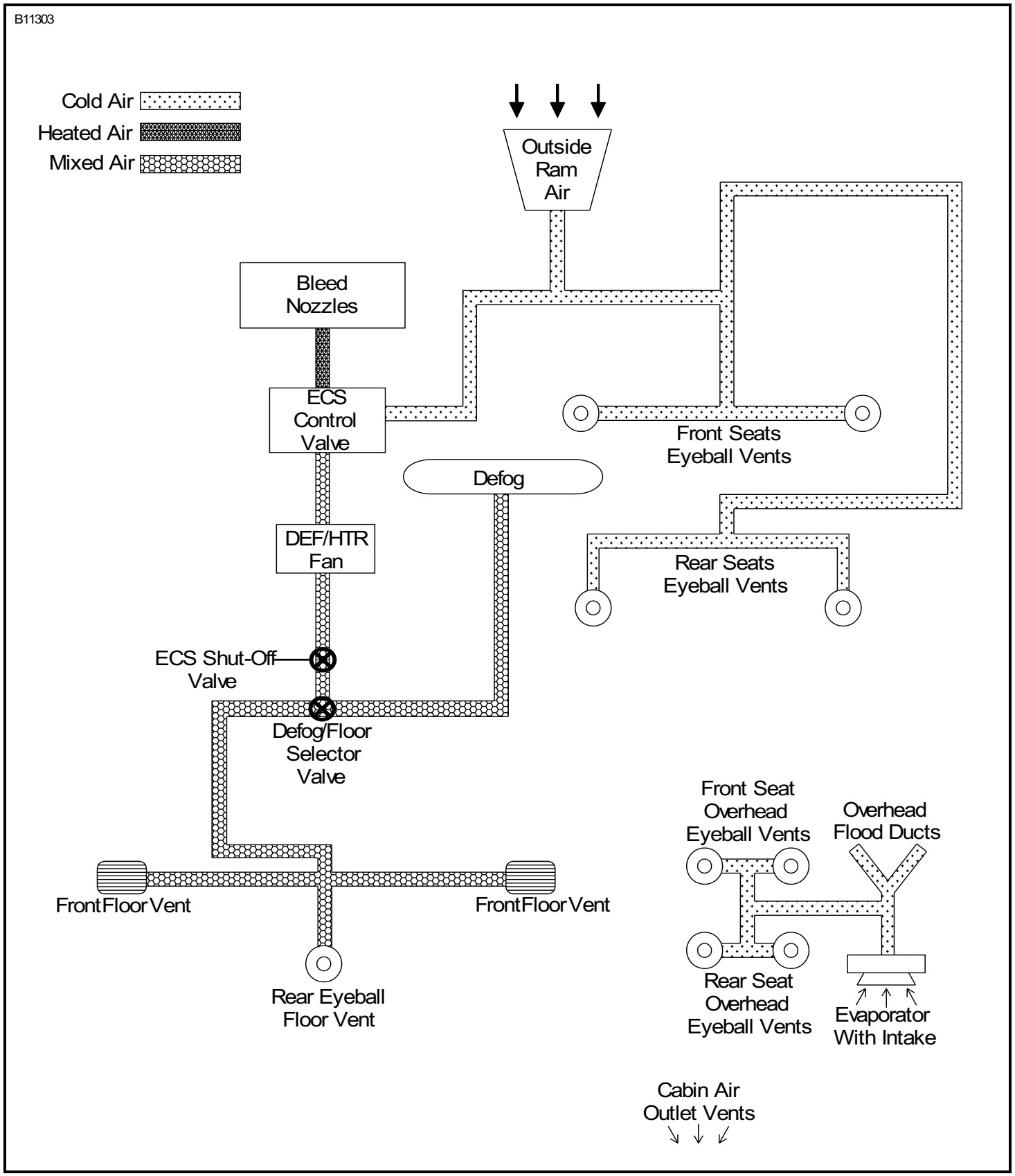
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Figure 201 : Sheet 1 : Vapor Cycle System (VCS) Air Conditioning



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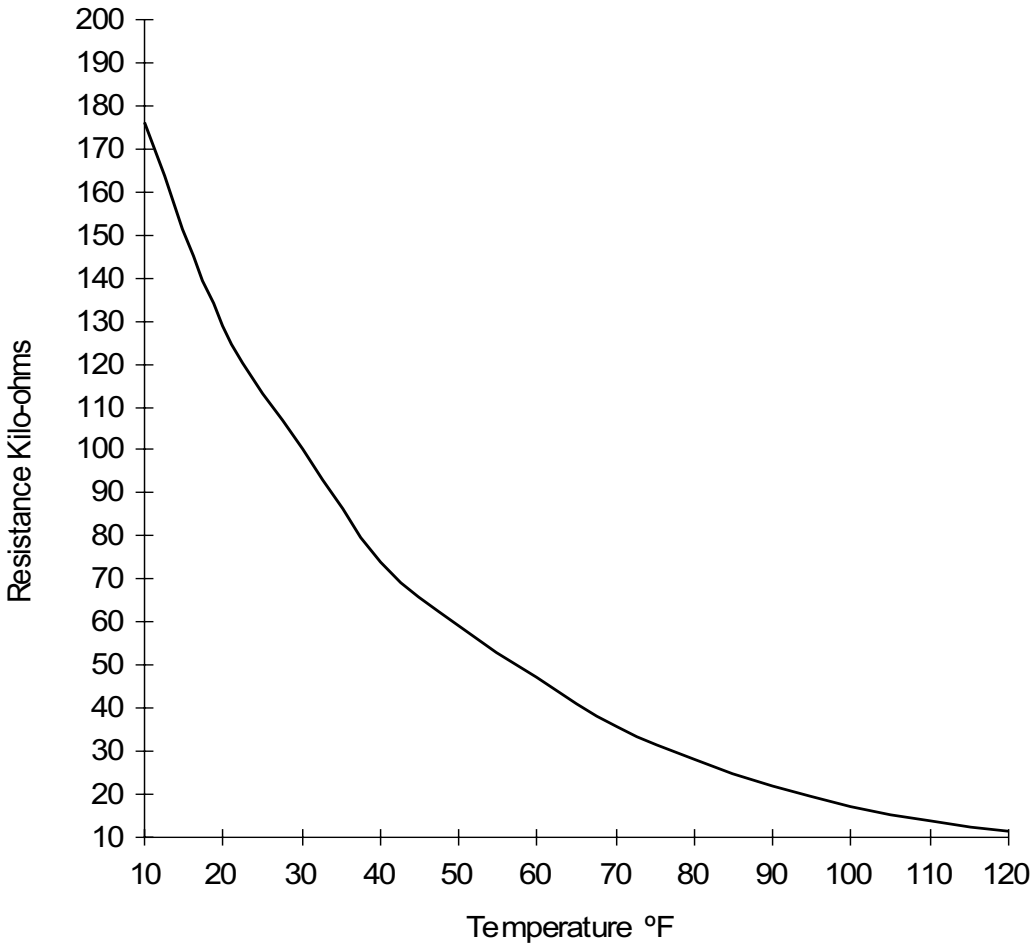
Figure 202 : Sheet 1 : Airflow Diagram



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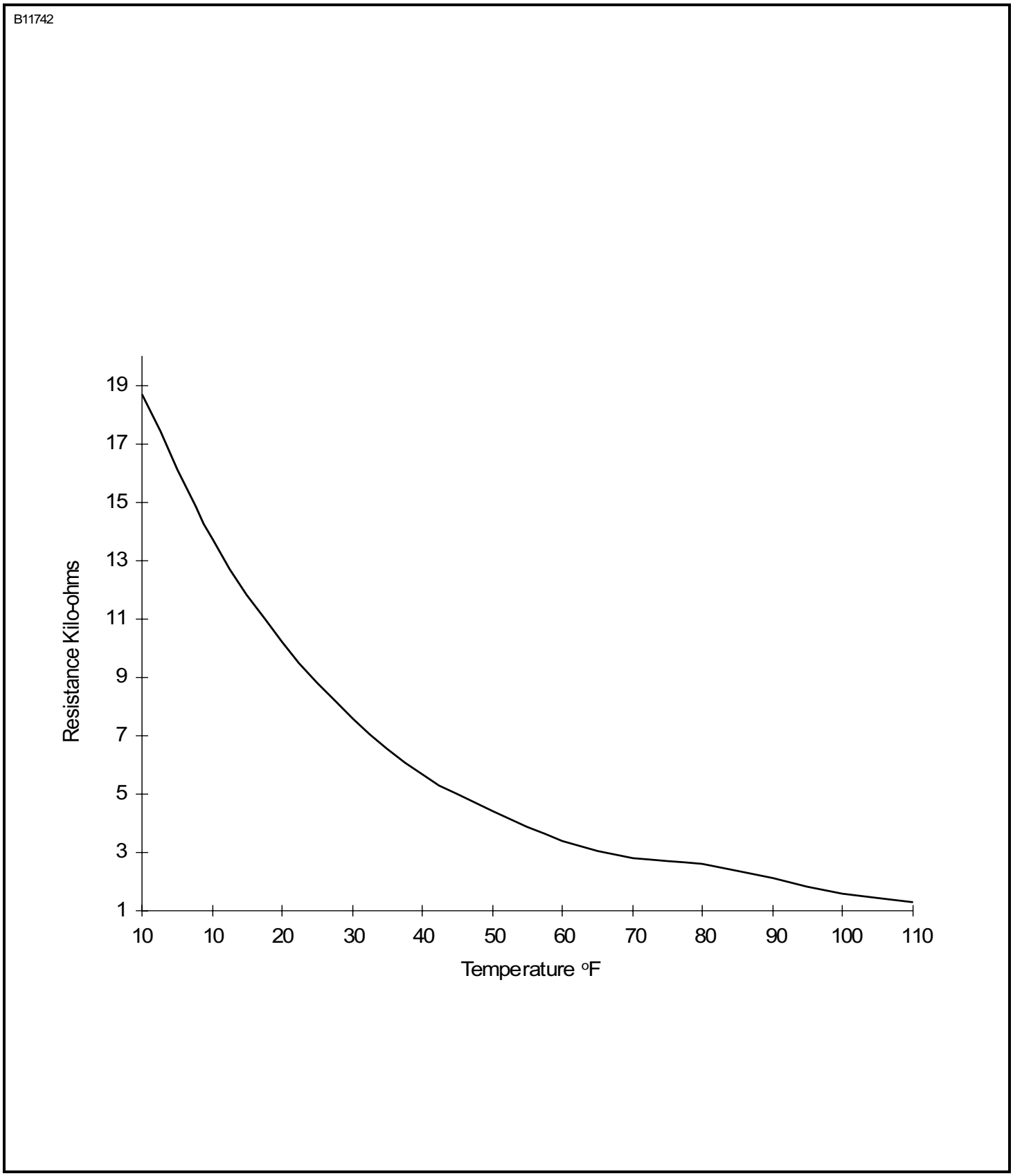
Figure 203 : Sheet 1 : Cabin Temperature Sensor Resistance vs. Temperature Curve

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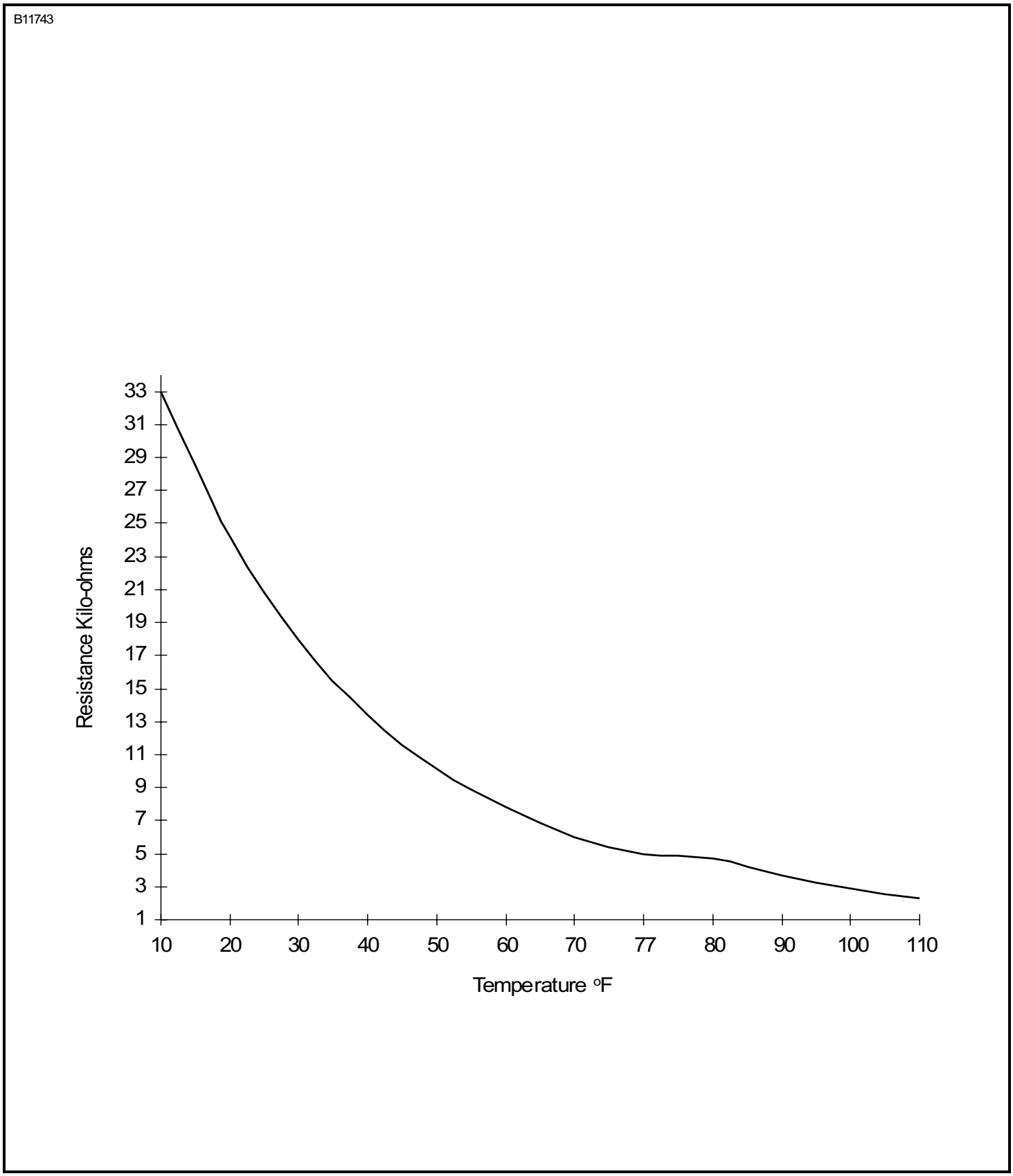
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Figure 204 : Sheet 1 : OAT Sensor Resistance vs. Temperature Curve



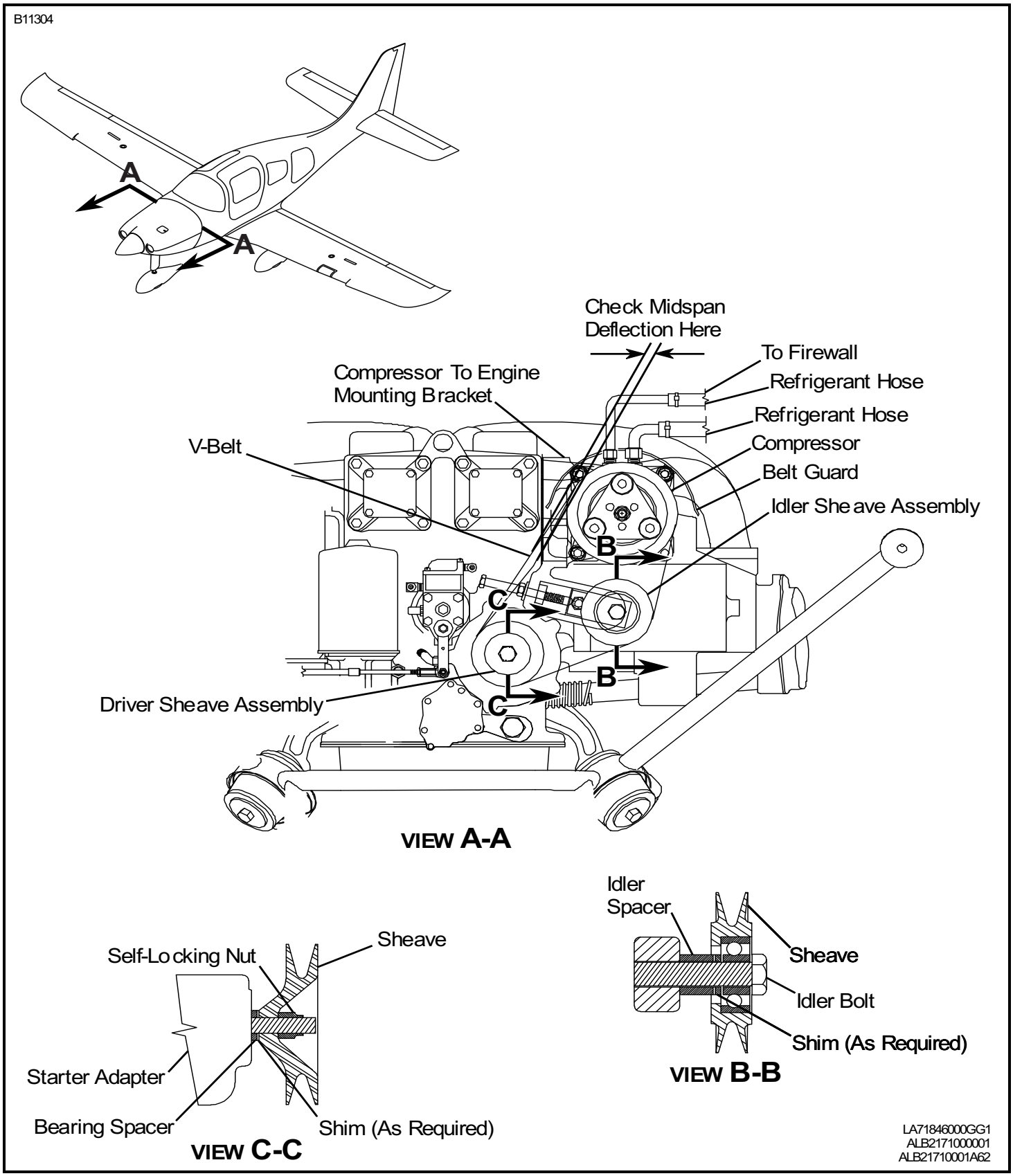
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Figure 205 : Sheet 1 : Evaporator Coil Temperature Sensor Resistance vs. Temperature Curve



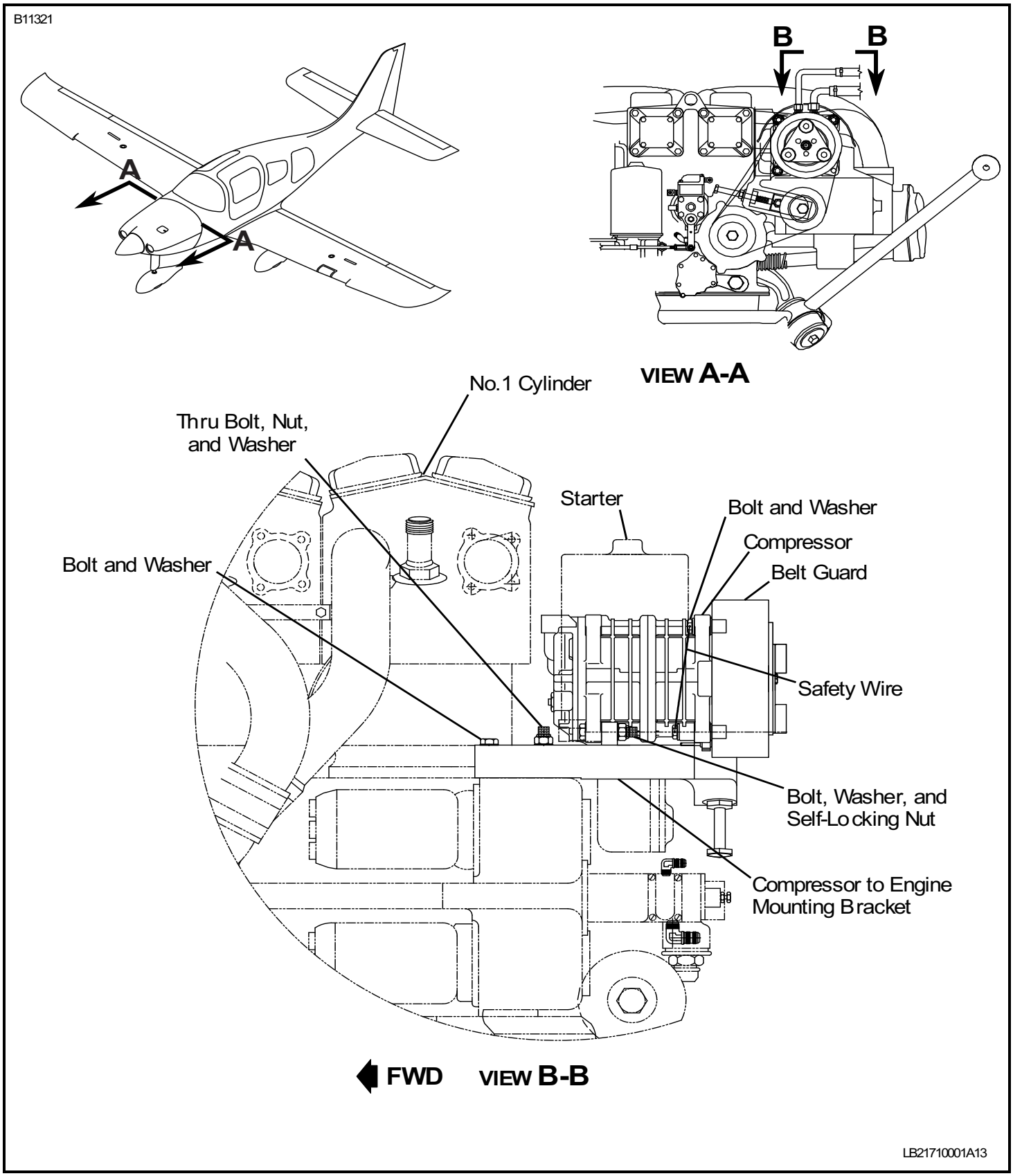
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Figure 206 : Sheet 1 : Engine Driven Compressor



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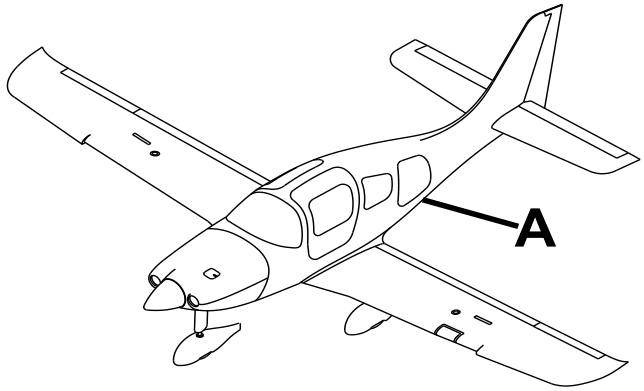
Figure 206 : Sheet 2 : Engine Driven Compressor

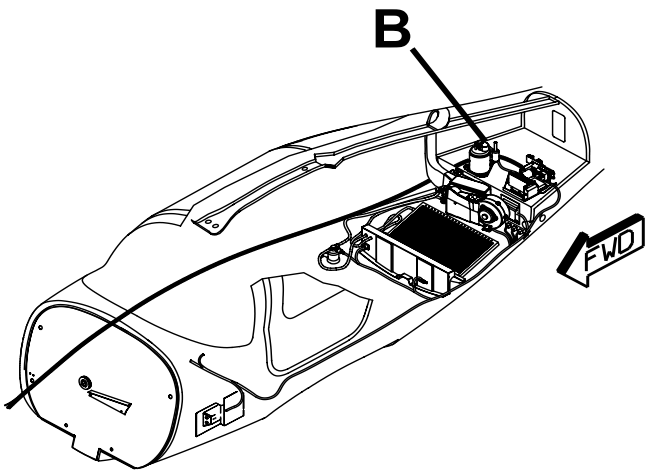


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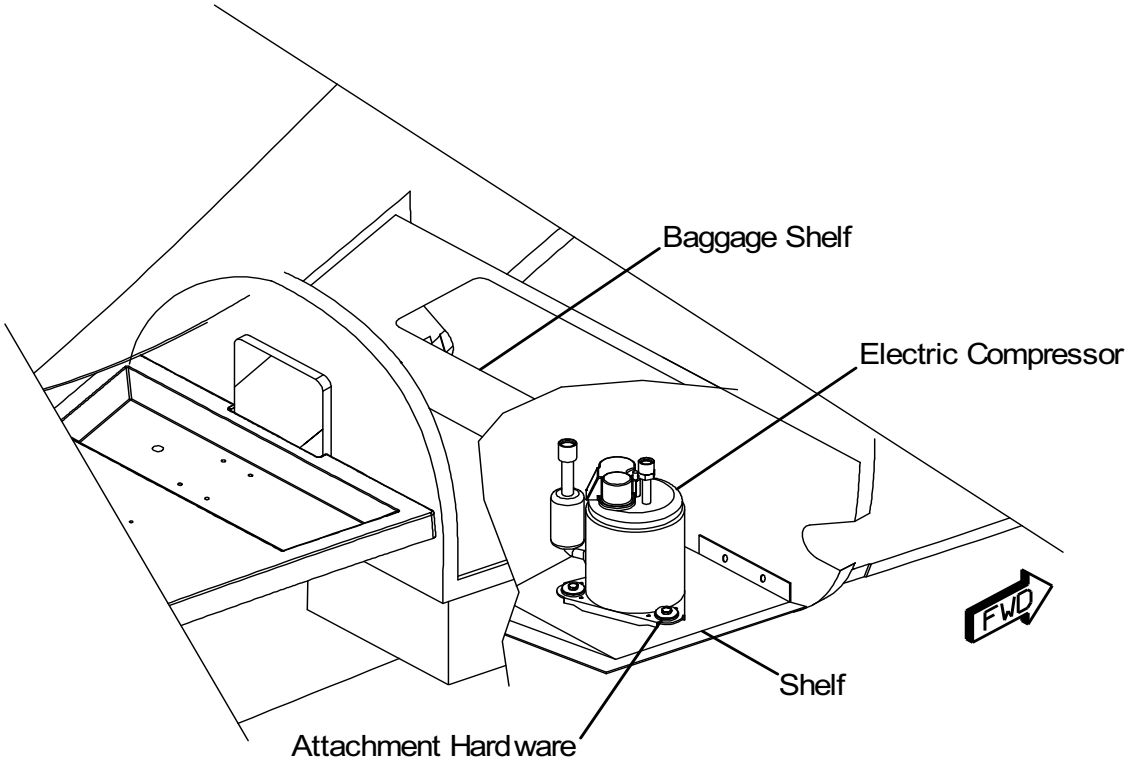
Figure 207 : Sheet 1 : Electrical Compressor

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DETAIL A

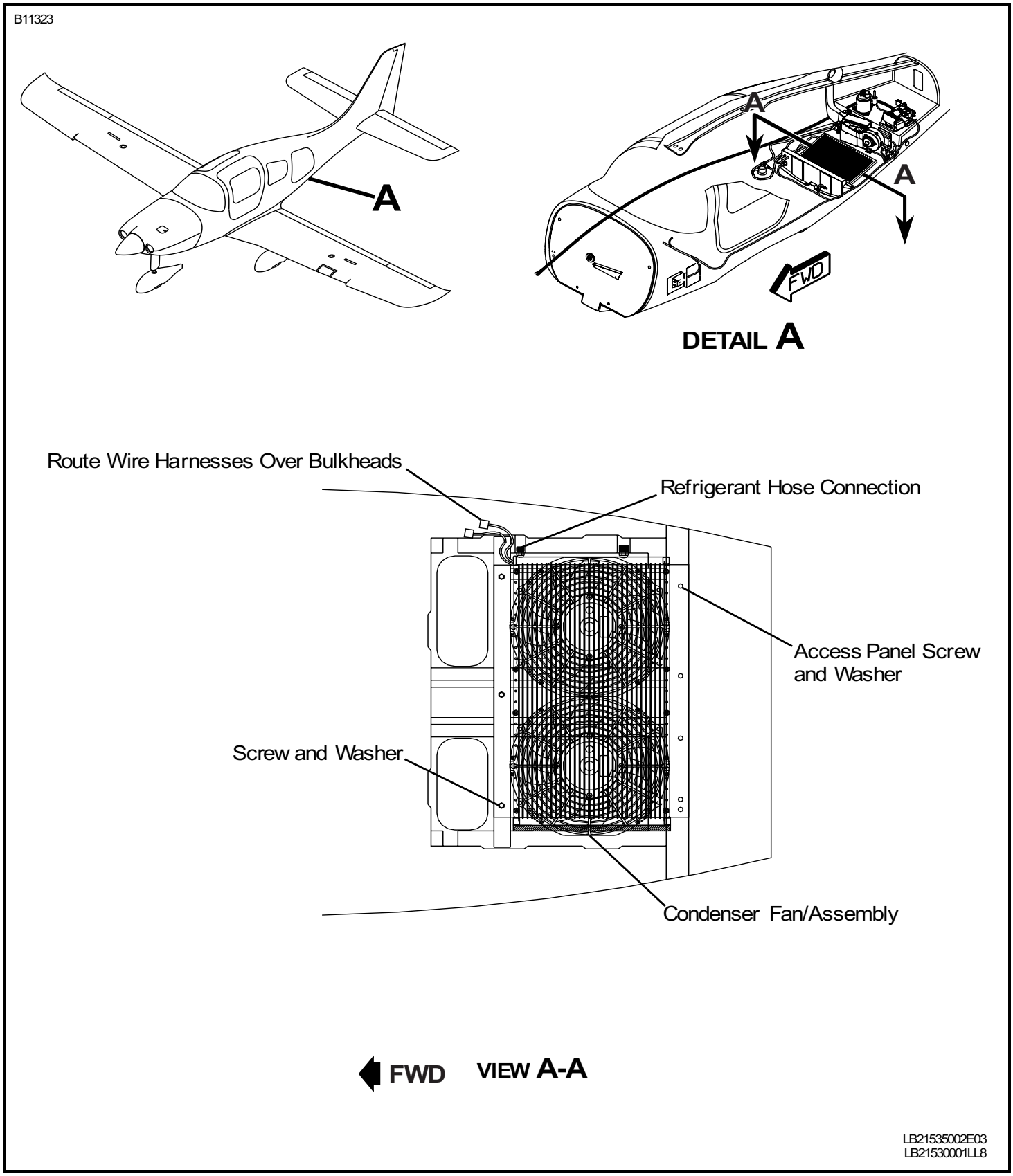


DETAIL B

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LB21530000E109

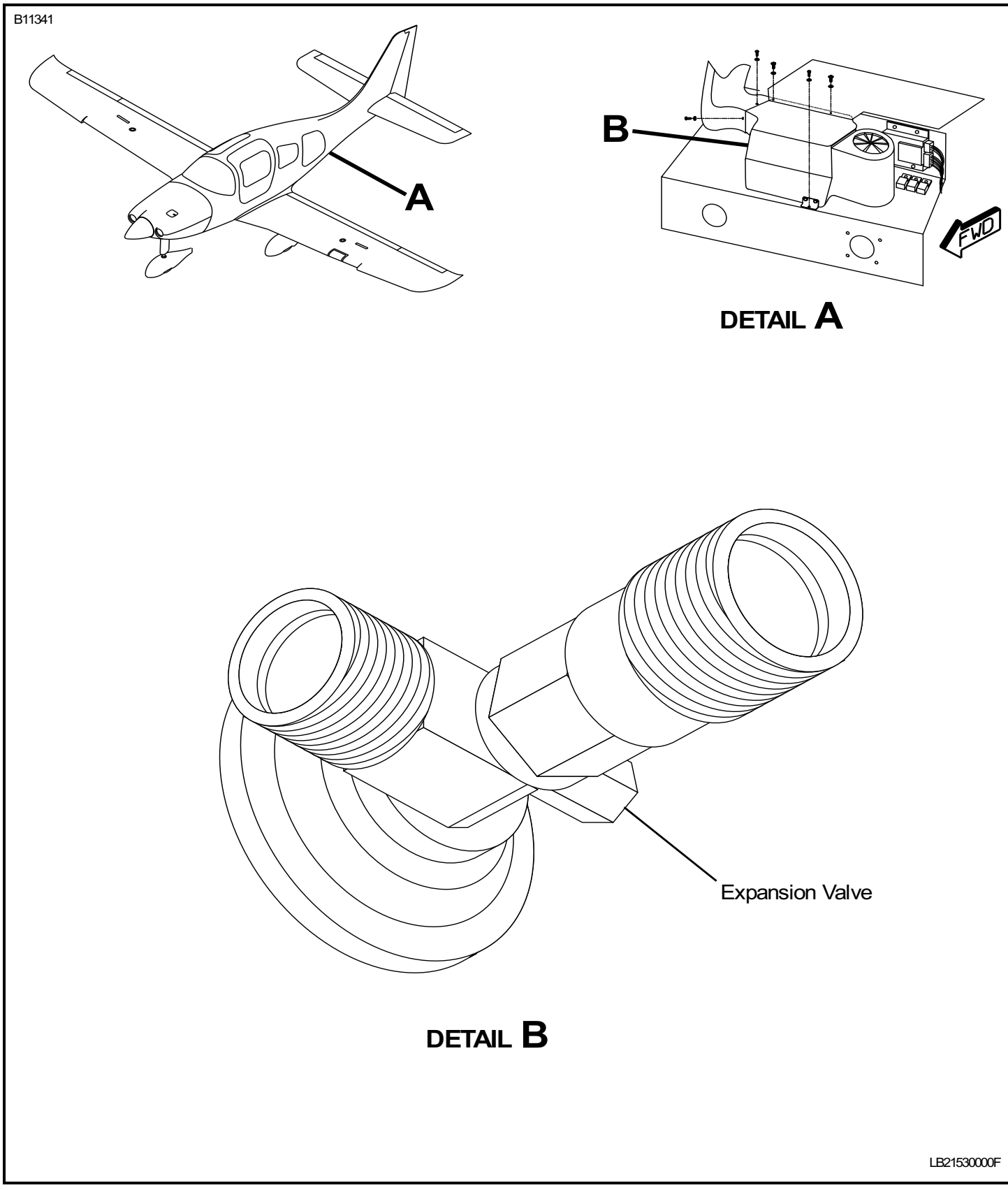
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Figure 208 : Sheet 1 : Condenser/Fan Assembly



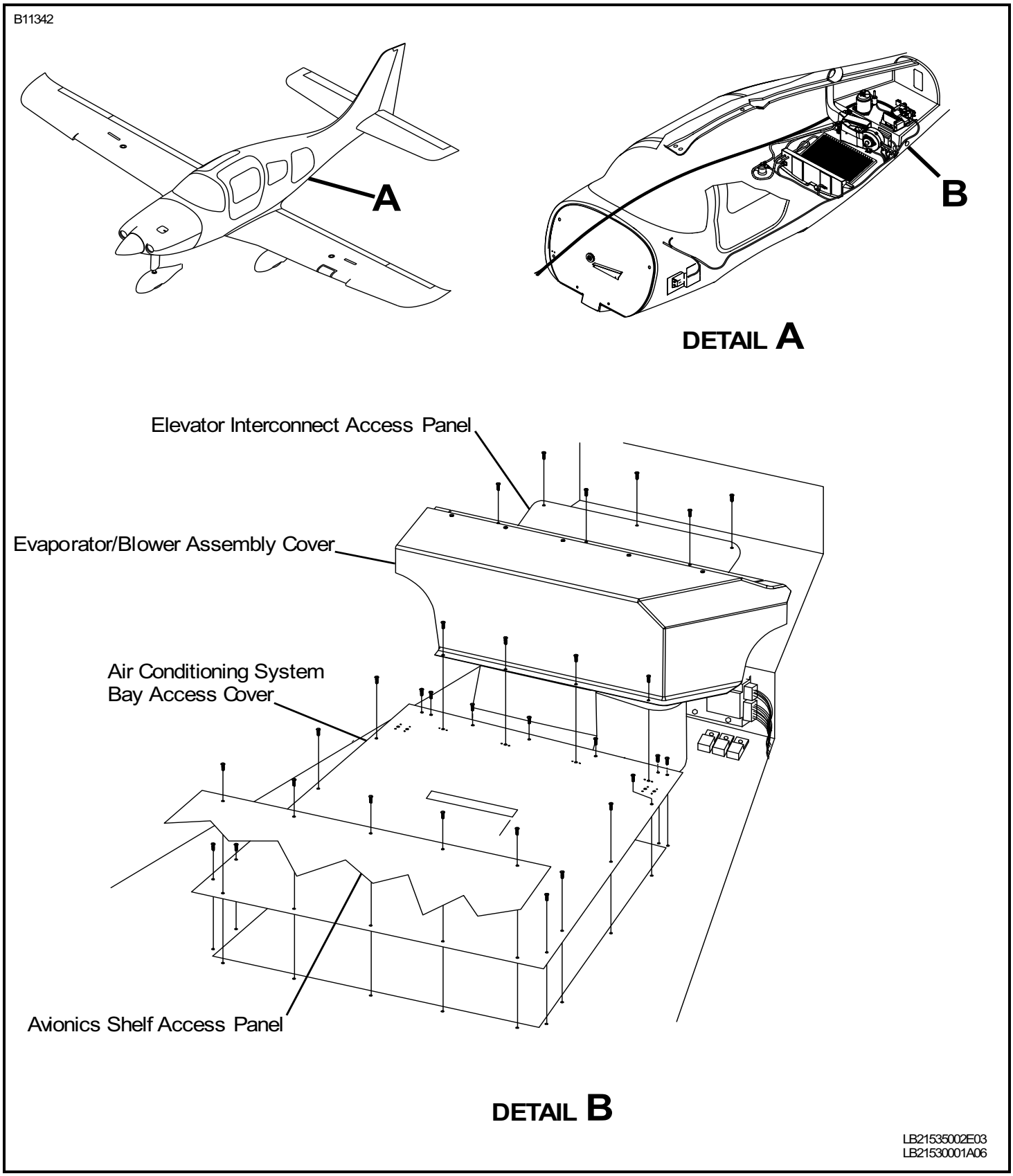
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Figure 209 : Sheet 1 : Expansion Valve



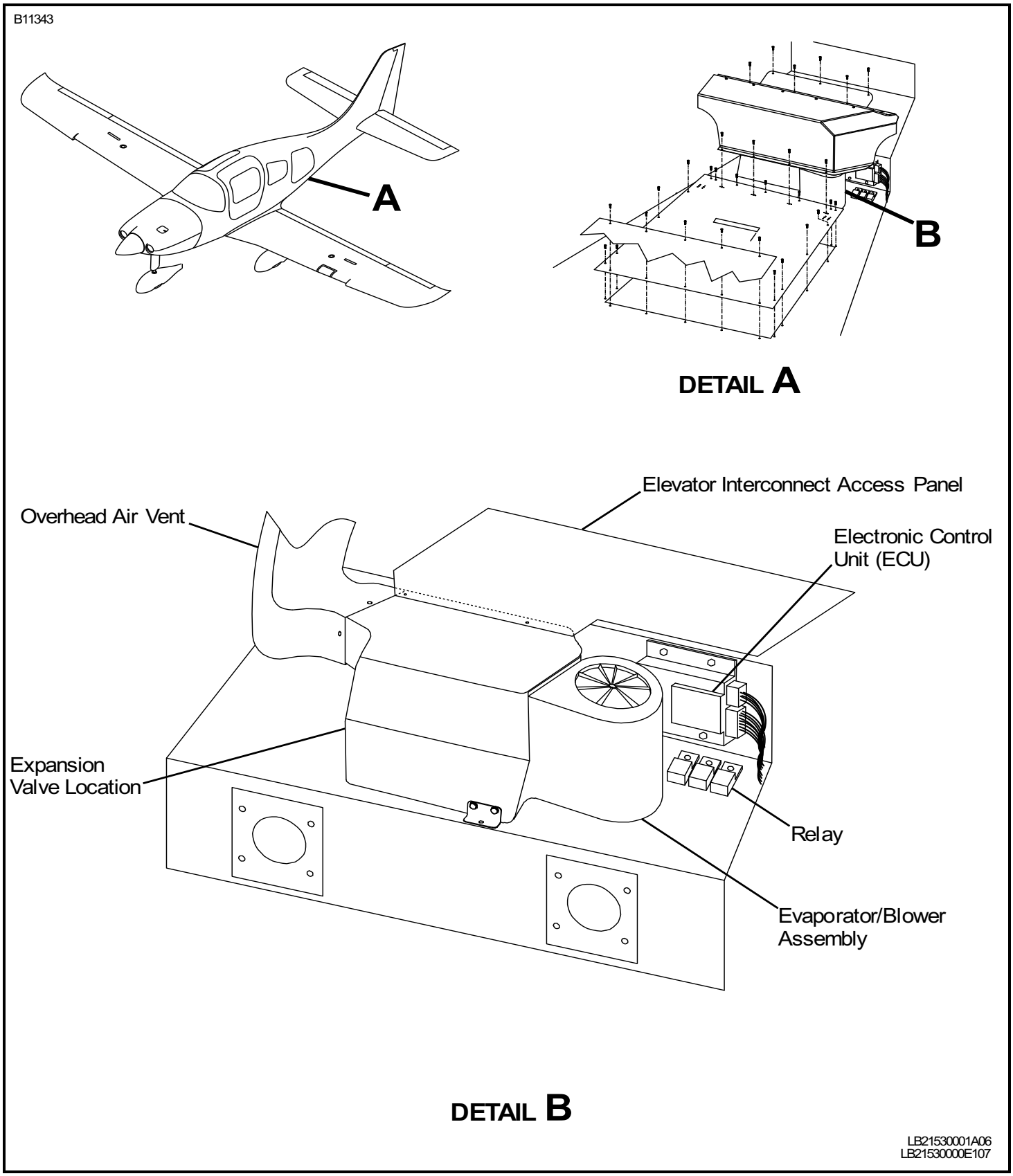
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Figure 210 : Sheet 1 : A/C Access Panels and Covers



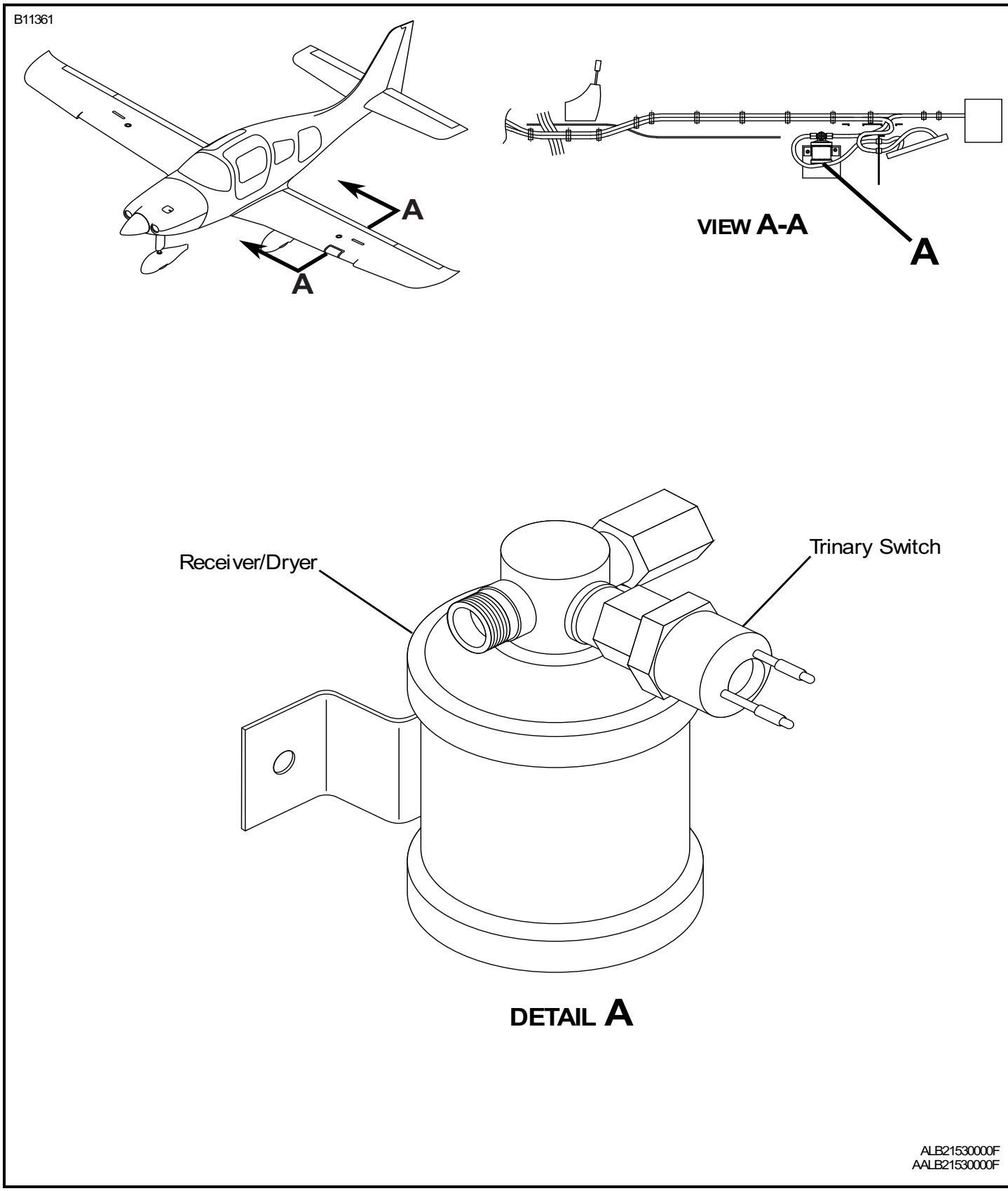
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Figure 211 : Sheet 1 : Evaporator/Blower Assembly



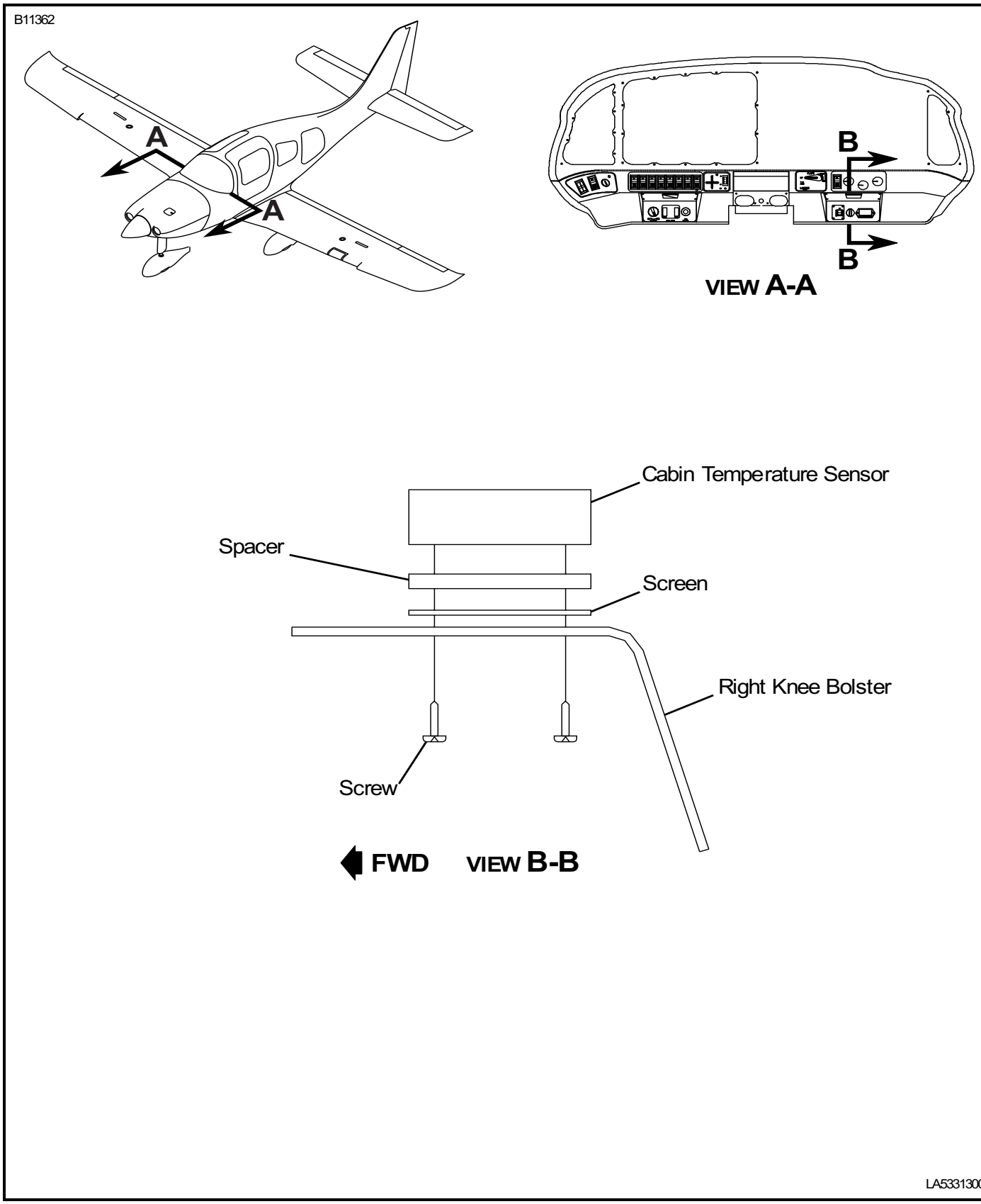
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Figure 212 : Sheet 1 : Receiver/Dryer with Trinary Switch



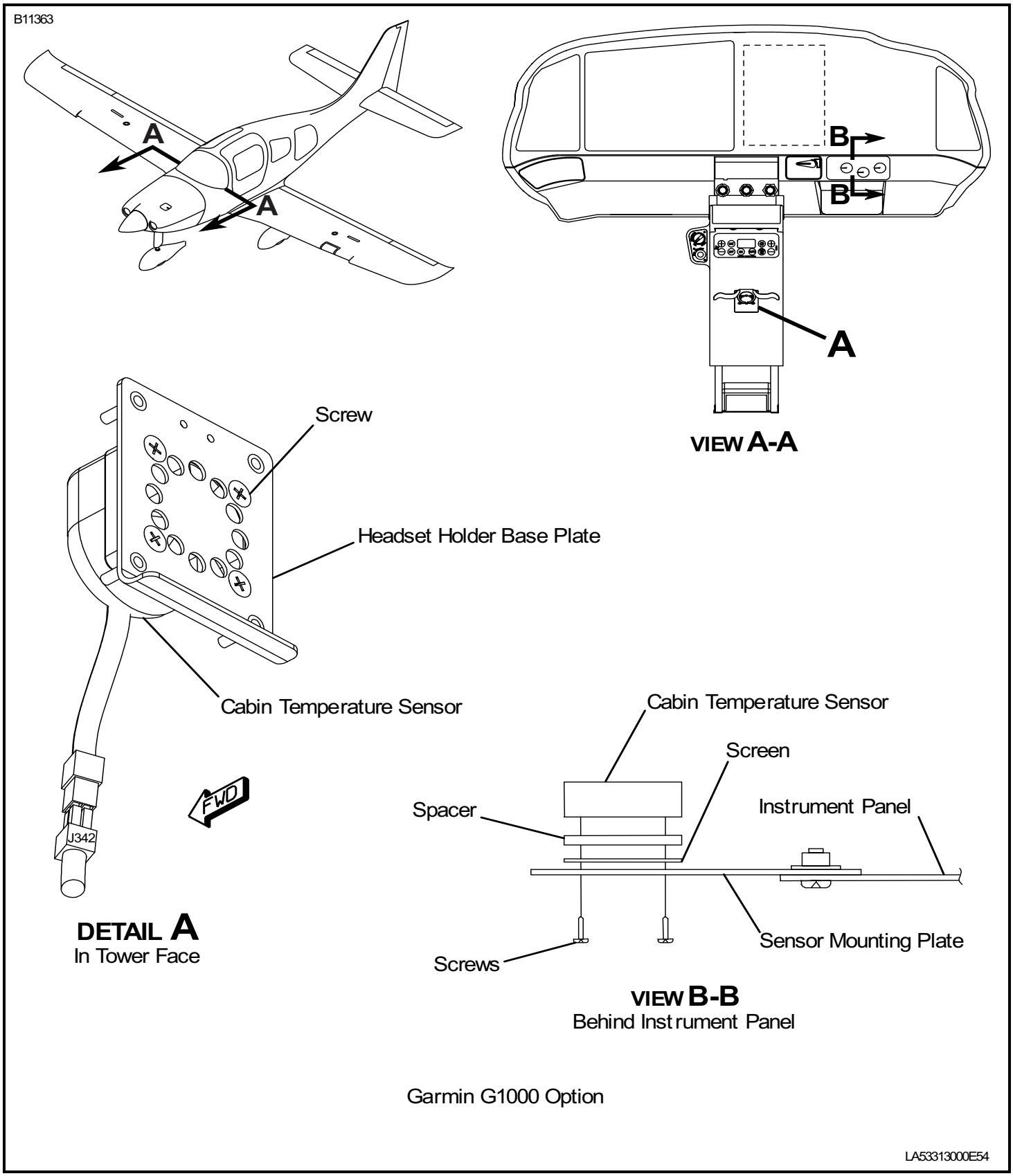
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Figure 213 : Sheet 1 : Cabin Temperature Sensor



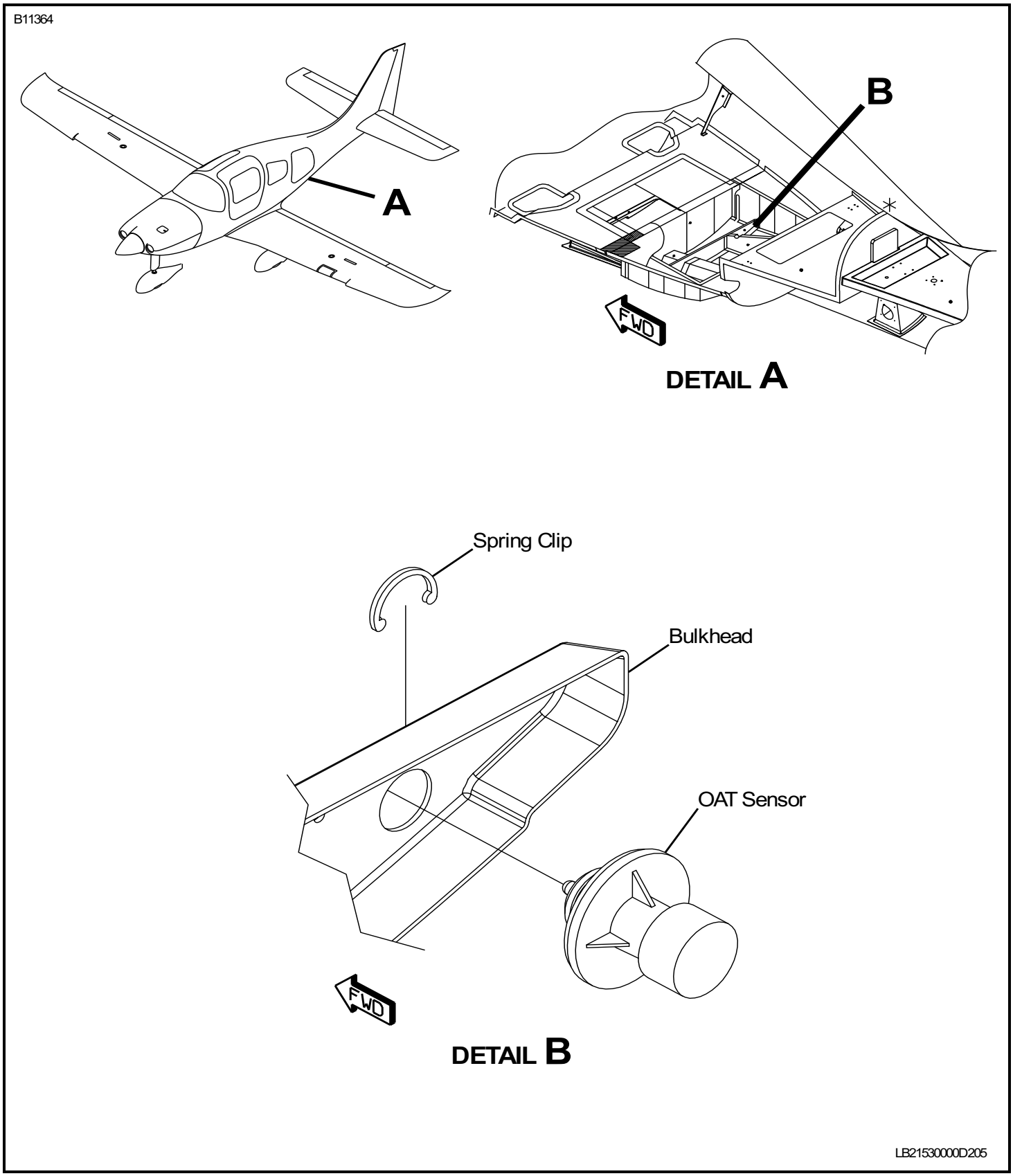
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Figure 213 : Sheet 2 : Cabin Temperature Sensor



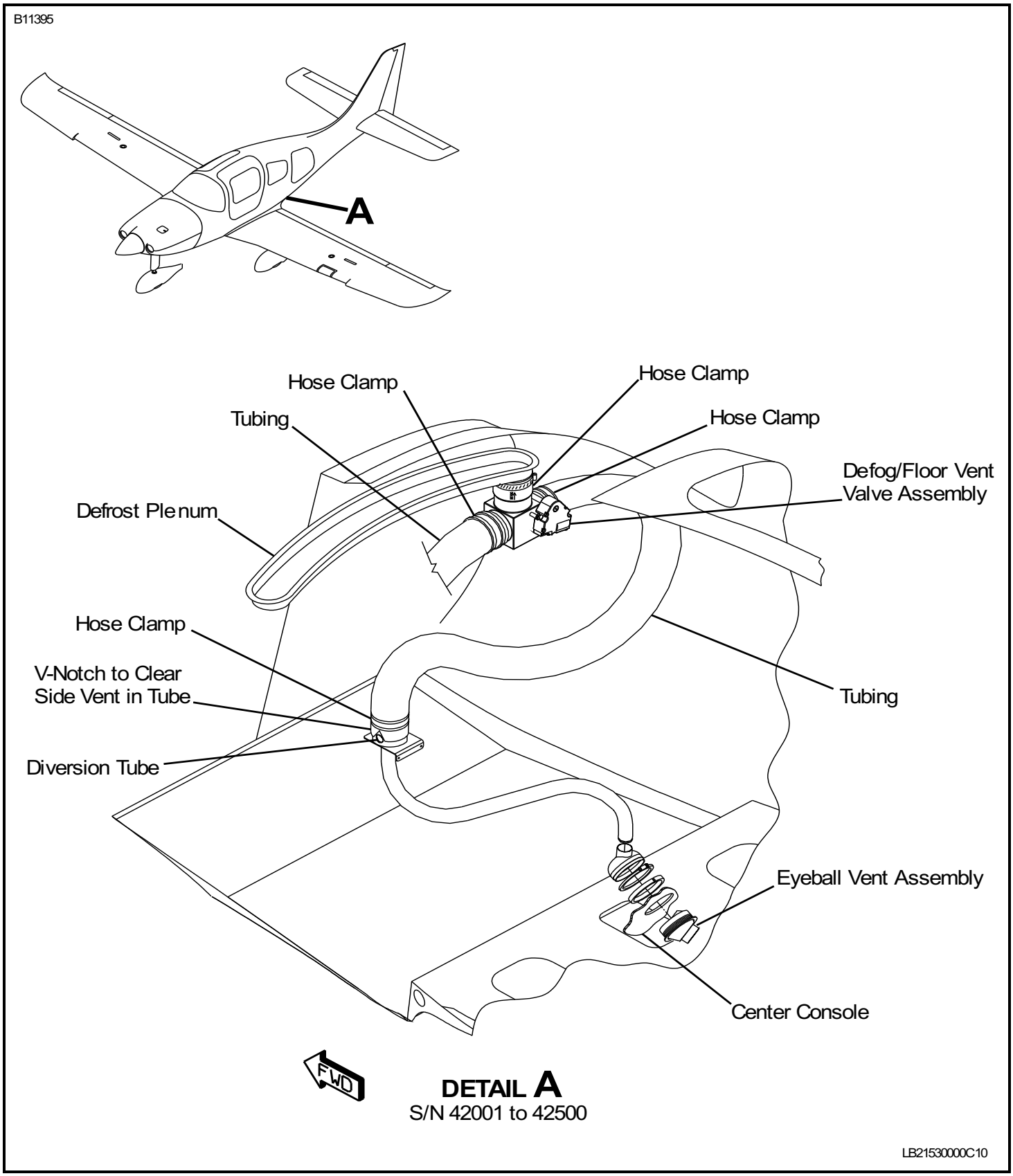
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Figure 214 : Sheet 1 : OAT Sensor



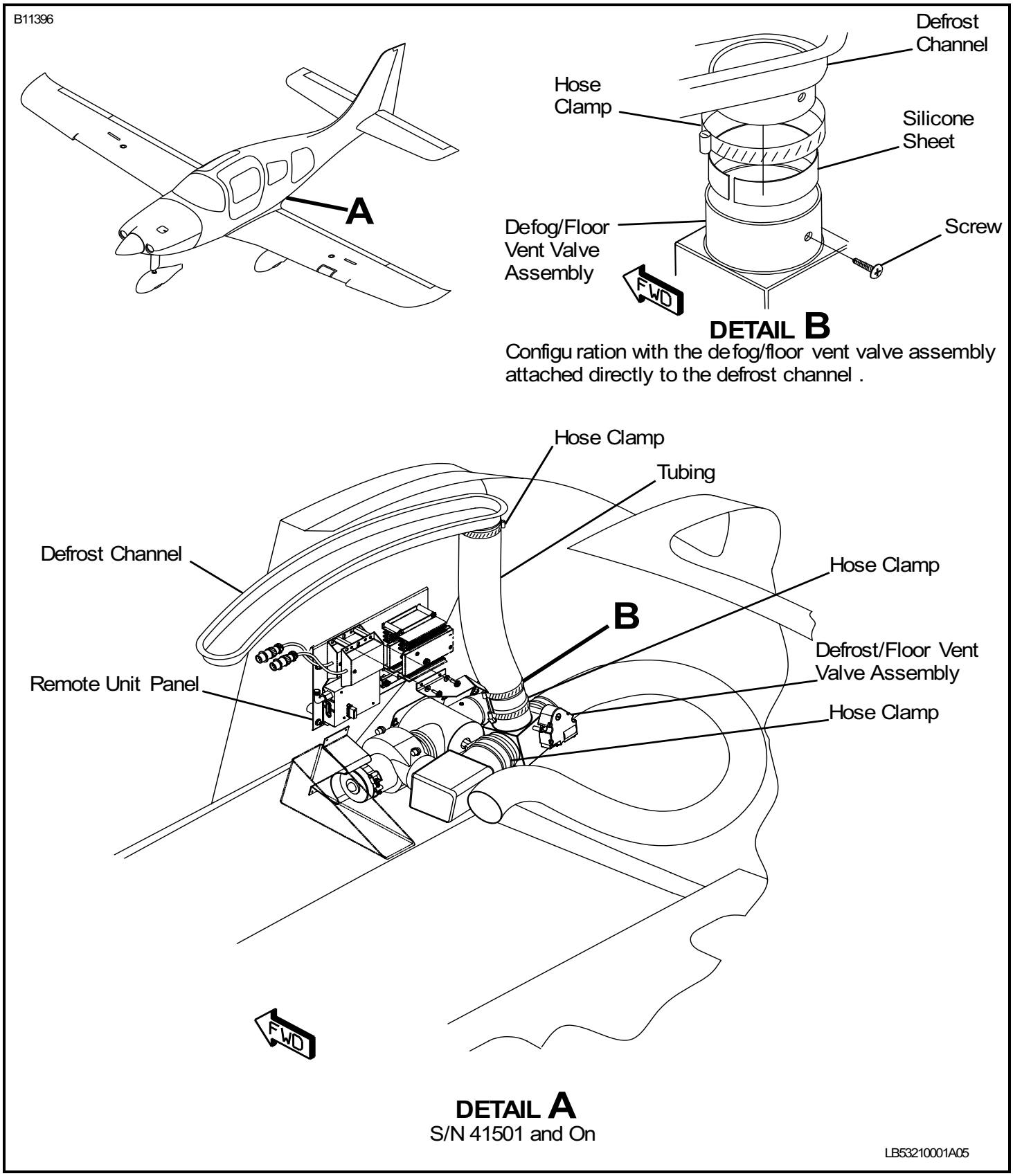
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Figure 215 : Sheet 1 : Defog/Floor Vent Valve Assembly



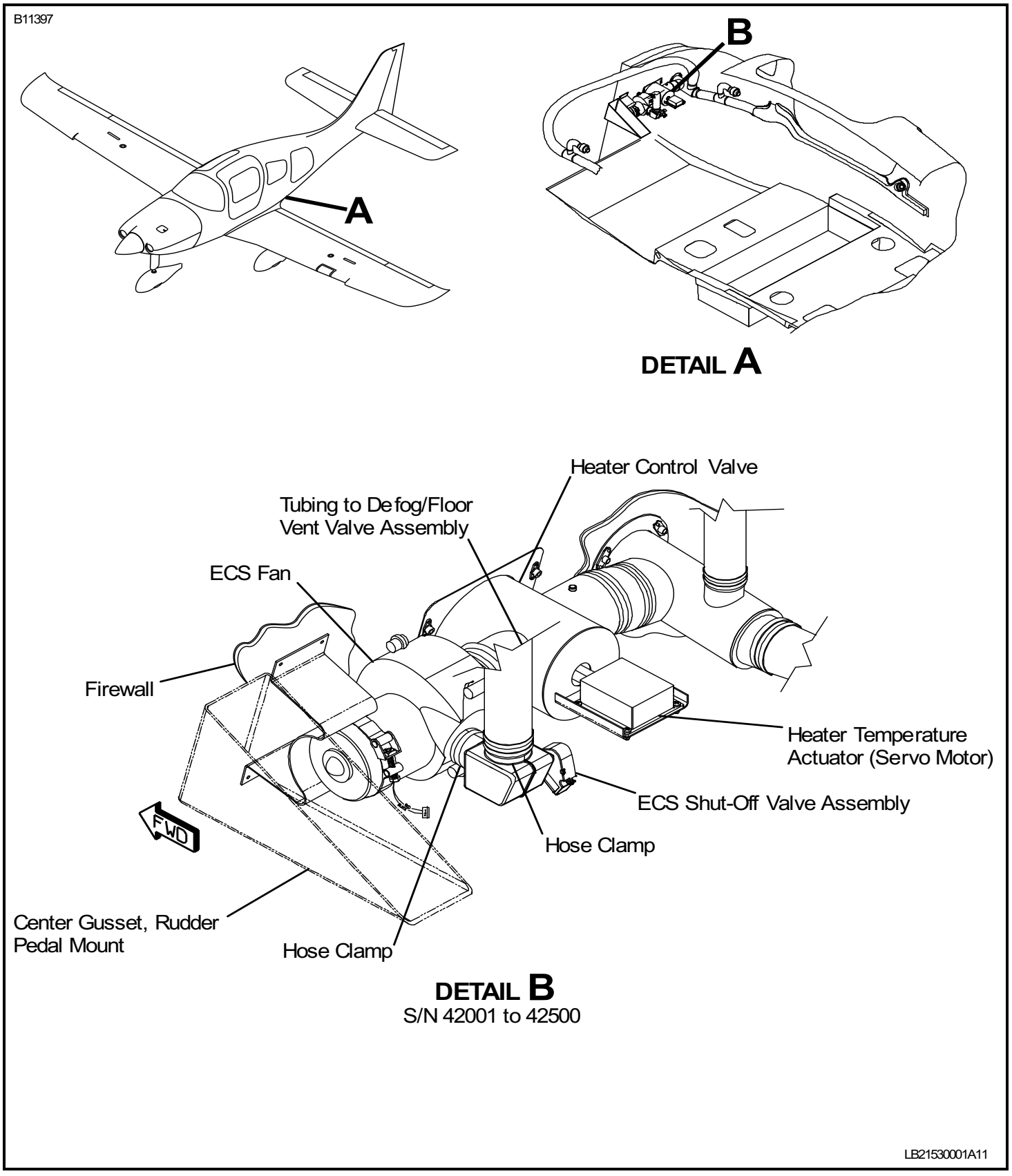
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Figure 215 : Sheet 2 : Defog/Floor Vent Valve Assembly



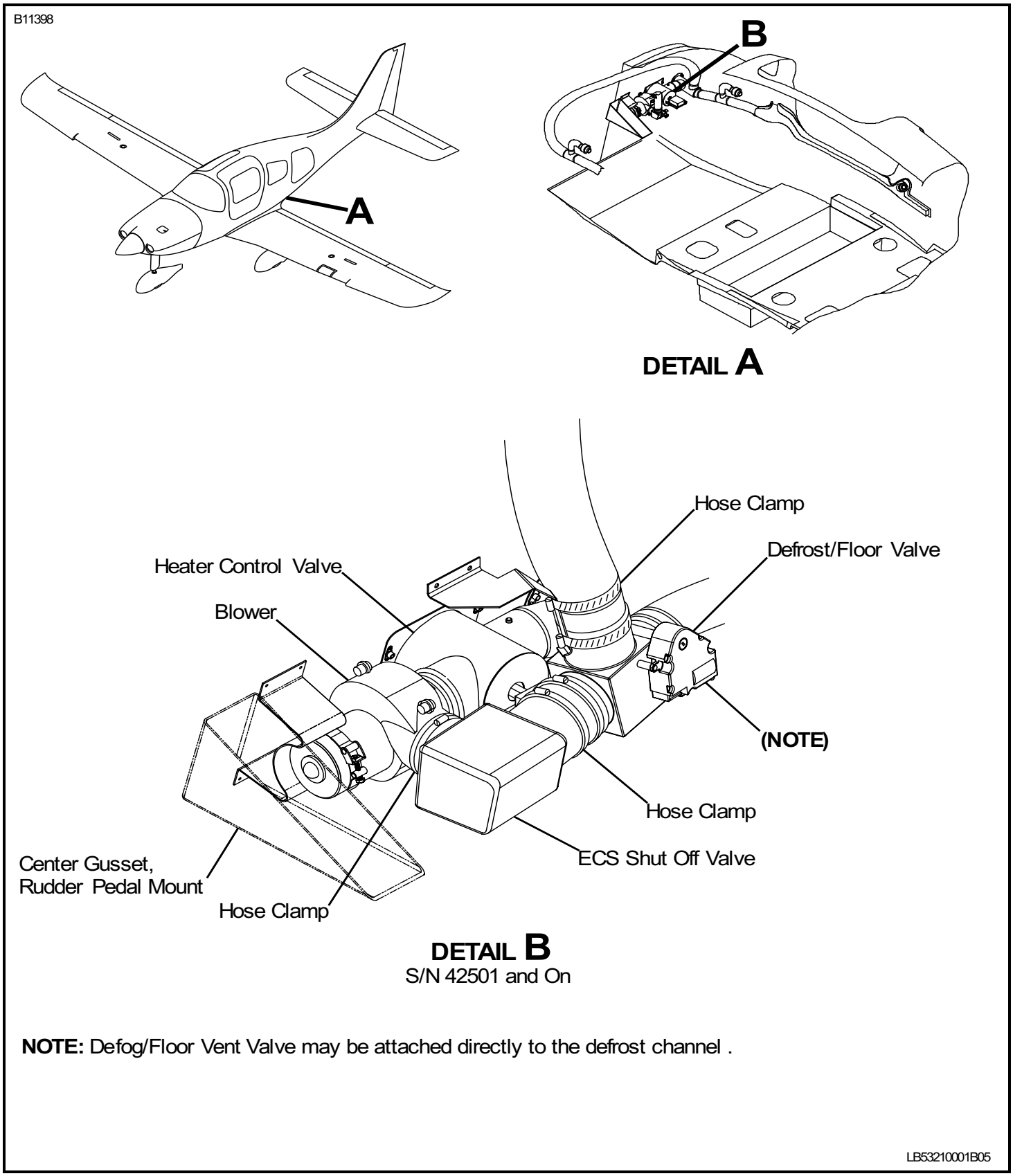
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Figure 216 : Sheet 1 : ECS Shut-Off Valve Assembly



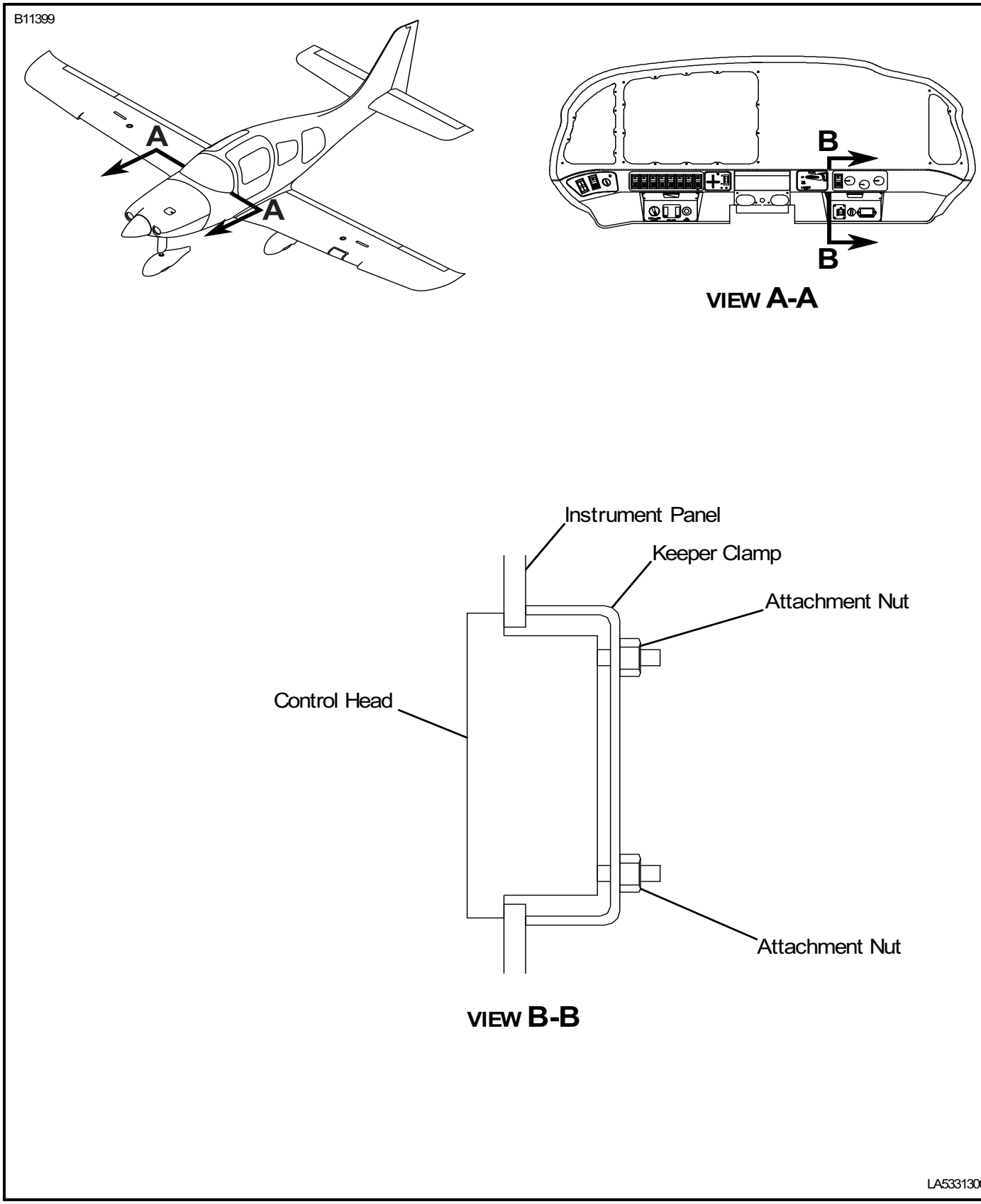
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Figure 216 : Sheet 2 : ECS Shut-Off Valve Assembly



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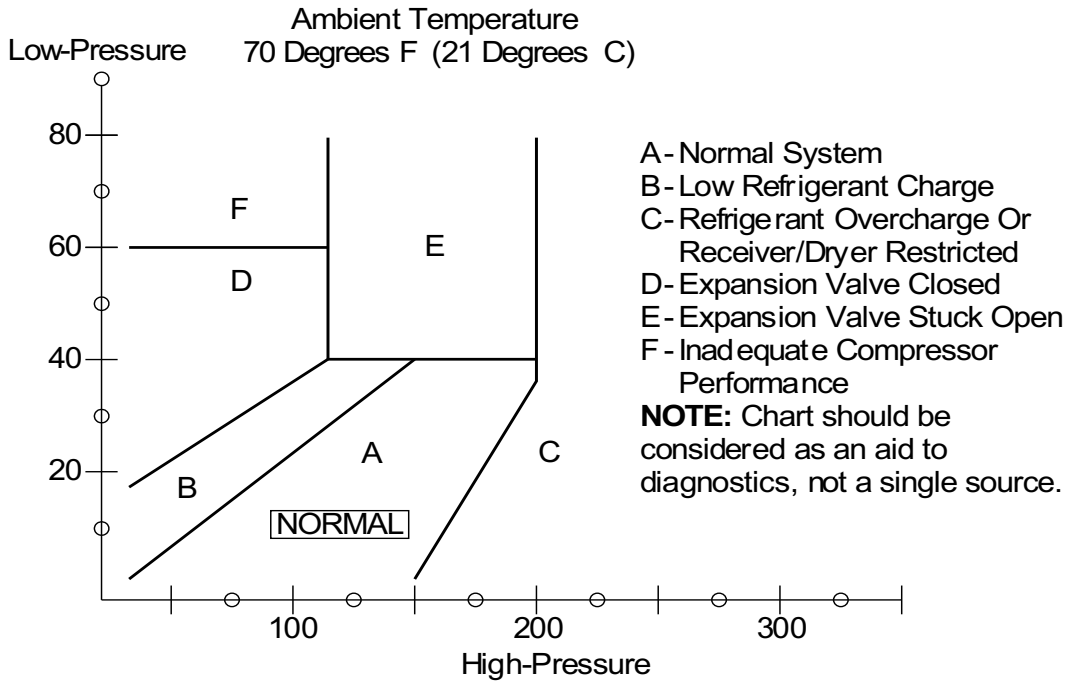
Figure 217 : Sheet 1 : Control Head Attachment



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Figure 218 : Sheet 1 : Pressure and Performance

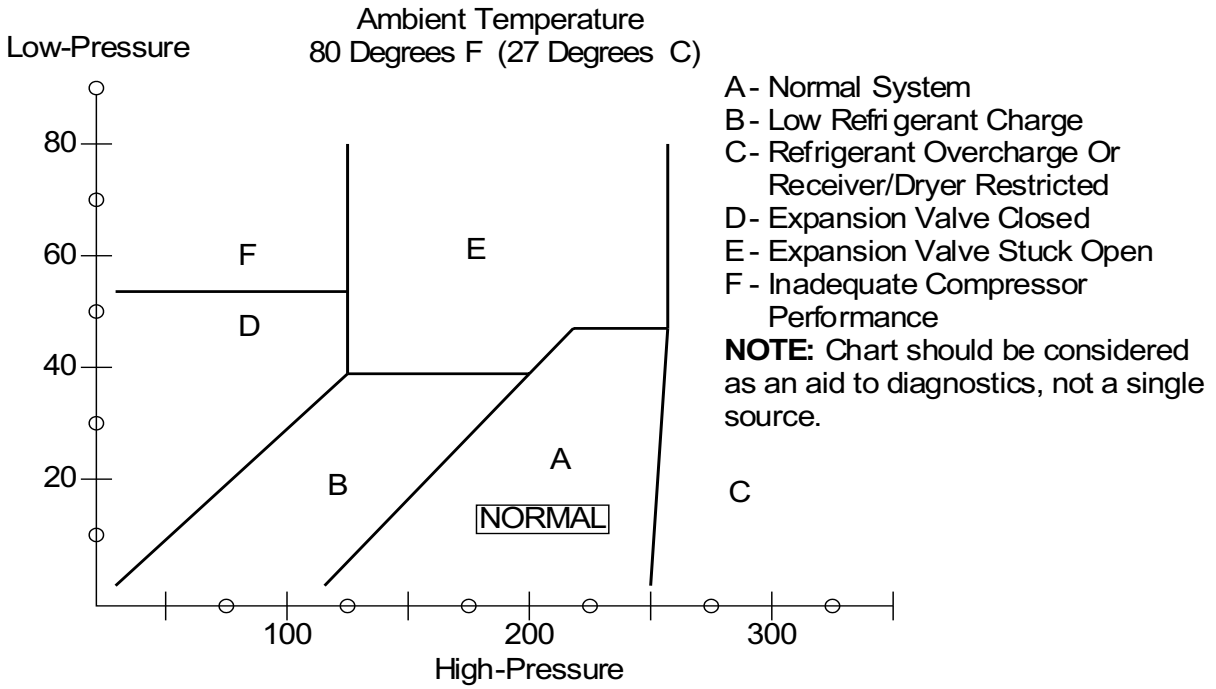
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Figure 218 : Sheet 2 : Pressure and Performance

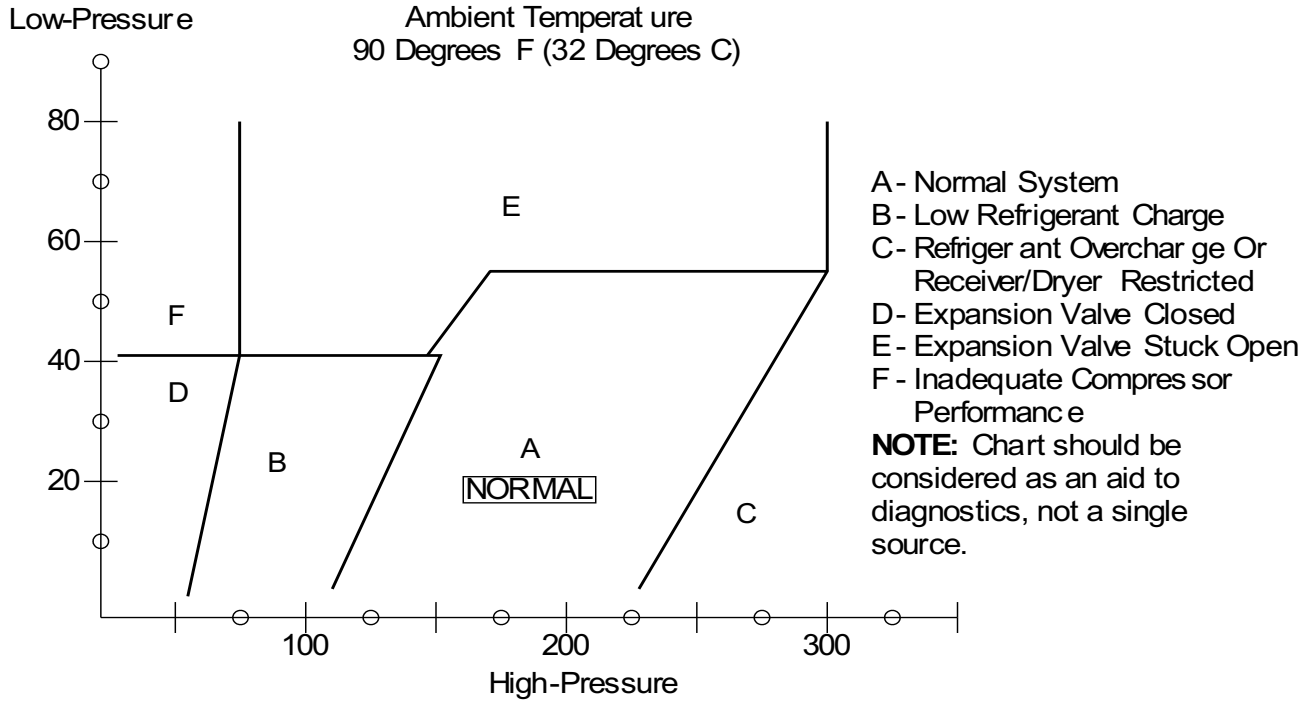
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Figure 218 : Sheet 3 : Pressure and Performance

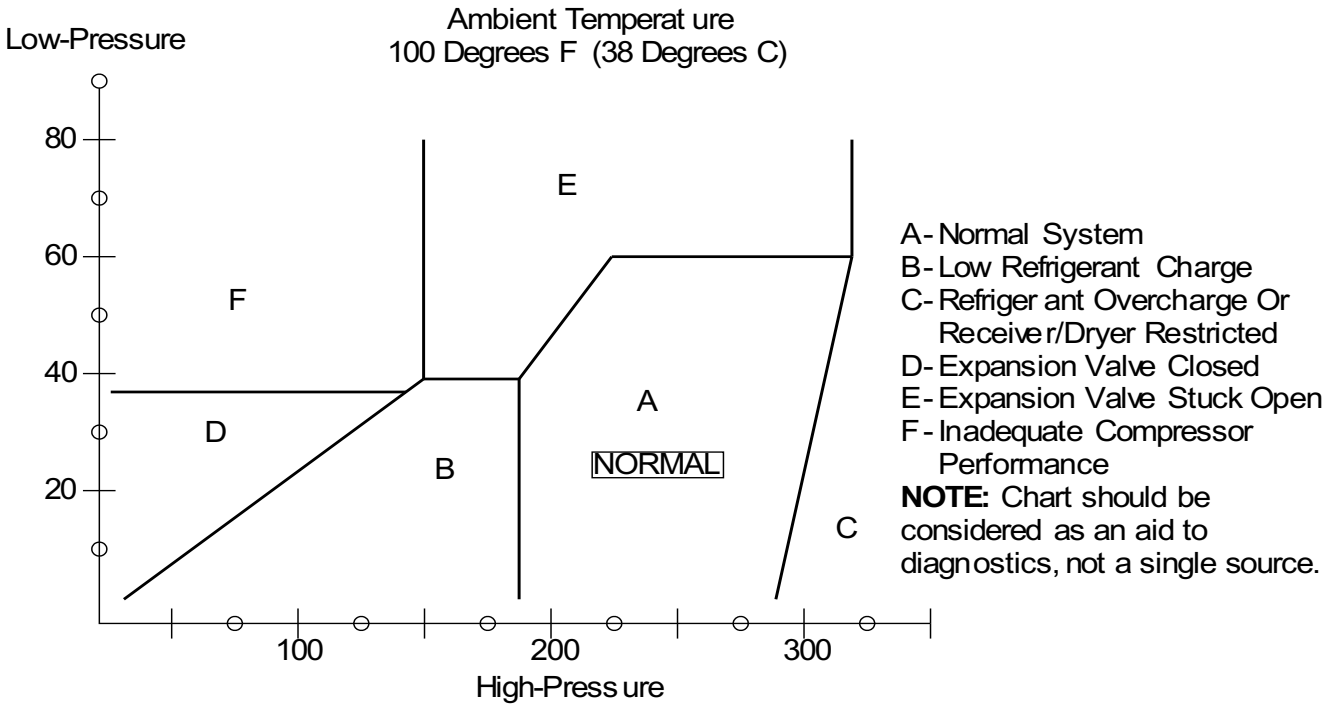
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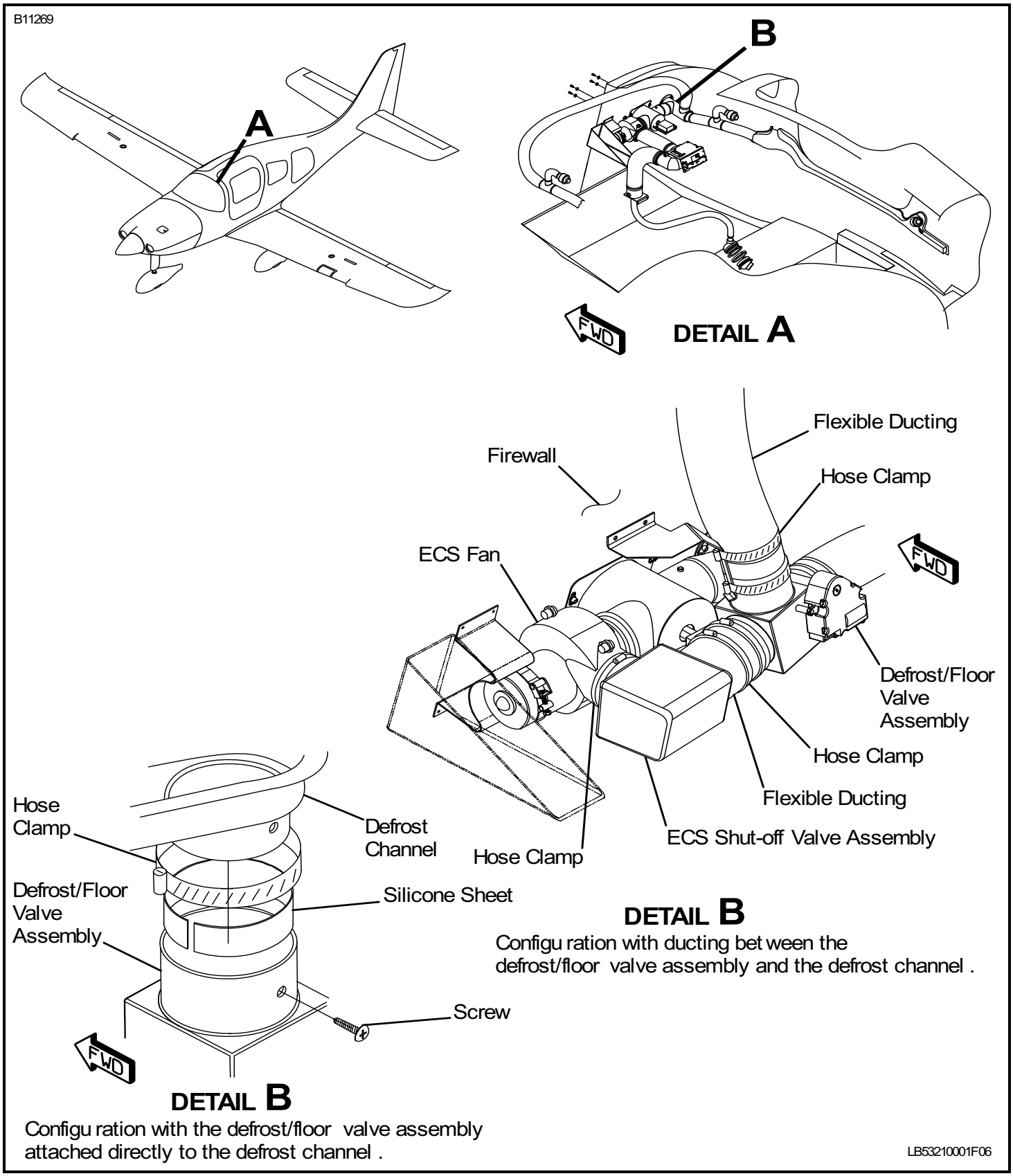
Figure 218 : Sheet 4 : Pressure and Performance

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Figure 204 : Sheet 1 : ECS Shut-Off Valve and Defrost/Floor Valve Assemblies



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FLIGHT CONTROLS - GENERAL

1. General - Scope and Definition

A. **General Description of System** - The Cessna 350 (LC42-550FG) has dual flight controls and may be flown from either the pilot or copilot's seat. Dual pairs of foot pedals actuate the rudder and control the brakes, which are also used for steering (differential braking). Push-pull tubes actuate the ailerons and elevator. The rudder is actuated via foot pedals and control cables. The wing flaps are electrically actuated via a drive motor and push-pull tubes. Rod end bearings are used throughout the control system. These bearings do not require any maintenance. Longitudinal pitch trim is achieved via an electrically actuated trim tab located in the elevator. Lateral trim is achieved via an electrically actuated trim tab on the right aileron. Directional trim is achieved via a fixed trim tab on the rudder.

B. **Tolerances and Play** - Table 1 lists the allowable control and trim tab deflections, free play limits, and tolerances.

Table 1. Control Surface Deflections, Tolerances, and Free Play Limitations

		Deflection	Tolerance	Free Play Limits (NOTE 1)
Wing Flaps	Cruise	0 degrees	±1 degree	---
	Takeoff	12 degrees		
	Landing	40 degrees		
Ailerons	Up	21.6 degrees	±1 degree	---
	Down	17.7 degrees		
Aileron Trim Tab	Up	22.4 degrees	±1 degree	0.063 in. (2.5 percent c)
	Down	19.6 degrees		
Aileron Servo Tab	Up	13 degrees at 17.7 degrees ail. deflection	±2 degrees	0.063 in. (2.5 percent c)
	Down	19 degrees at 21.6 degrees ail. deflection		
Elevator	Up	13 degrees	+0 degrees/-0.5 degrees	---
	Down	12 degrees	±1 degree	
Elevator Trim Tab	Up	21 degrees	±1 degree	0.080 in. (2.5 percent c)
	Down	30 degrees		
Rudder	Left	17 degrees	±1 degree	---
	Right	17 degrees		
Rudder Limiter	Left	11.5 degrees	± 0.5 degrees	---
Rudder Trim Tab	---	±30 degrees	N/A	---

NOTE 1:

The tab free play at the tab trailing edge should be less than 2.5 percent of the tab chord aft of the hinge line, measured at the station where the free play is measured.

C. **Rigging and Flight Controls** - Rigging and adjustment of the control system and surfaces is covered in their respective sections.

2. Trim System - General

A. The Cessna 350 (LC42-550FG) has a two-axis electric trimming system. The elevator trim tab is located on the right side of the elevator, and the aileron trim tab is on the right aileron. Both tabs are electrically controlled by a hat switch on each control stick, and the trim position is annunciated on the trim panel located to the right of the rocker switch panel for the Basic or Avidyne option or the Engine Indication System (EIS) page of the MFD for the Garmin G1000 system. The trim servos are protected by one-amp circuit breakers. Refer to Figure 1 or Figure 2 for an illustration of the system.
The trim surfaces are moved by push rods connected between each tab and a servomotor. The aileron tab has one actuating rod and the elevator tab has two. The second actuating rod on the elevator is a redundant system and is

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provided for the more critical tab in the system. The frictional device installed on the aileron tab should never be lubricated.

B. **Hat Switches** - The trim tabs are controlled through the use of a hat switch on the top portion of the pilot's and copilot's control stick at the three and nine o'clock positions, respectively. Moving the switch forward will correct a tail heavy condition, and moving it back will correct a nose heavy condition. Moving the hat switch left or right will correct right wing heavy and left wing heavy conditions, respectively.

C. **Simultaneous Trim Application** - If both switches (pilot's and copilot's) are moved in the same direction at the same time, the trim will operate in the direction selected. If the switches are simultaneously moved in opposite directions, e.g., pilot's is nose down and copilot's is nose up, the pilot's selection overrides the copilot's, and the trim will move to a nose down position (trim will not move with the Garmin G1000 option in this case). Finally, if the switches are simultaneously selected in different directions, e.g., elevator trim is input by one pilot and aileron trim is input by the other, each trim tab will move in the direction selected.

D. **Trim Position Indicator**

(1) **Basic or Avidyne Option** - The trim position is displayed on two light bars using a series of blue LEDs and two green LEDs that are arranged on the trim panel in the shape of a plus sign. The vertical lights indicate the position of the elevator trim, and the horizontal lights show the position of the aileron trim. The middle green lights in each bar indicate the approximate neutral position. The blue lights are sequentially lit and extinguished as the trim tab moves through its range of travel. If the two green LEDs in the middle of the "+" are lit and no blue lights are illuminated, both tabs are in the approximate neutral position. The LED's level of brightness is controlled by the position lights switch. When the position lights are on, the trim lights are in the dim mode, and when the position lights are off, the trim lights are in the bright mode. Should one or all of the trim indicator LEDs stop functioning, remove the trim panel, and have it serviced by a qualified avionics repair technician.

(2) **Garmin G1000 Option** - The trim position is displayed in the Trim Group on the Engine Indication System (EIS) page of the MFD. The vertical mark indicates the position of the elevator trim and the horizontal mark shows the position of the aileron trim. The blue band for each axis indicates the approved takeoff ranges.

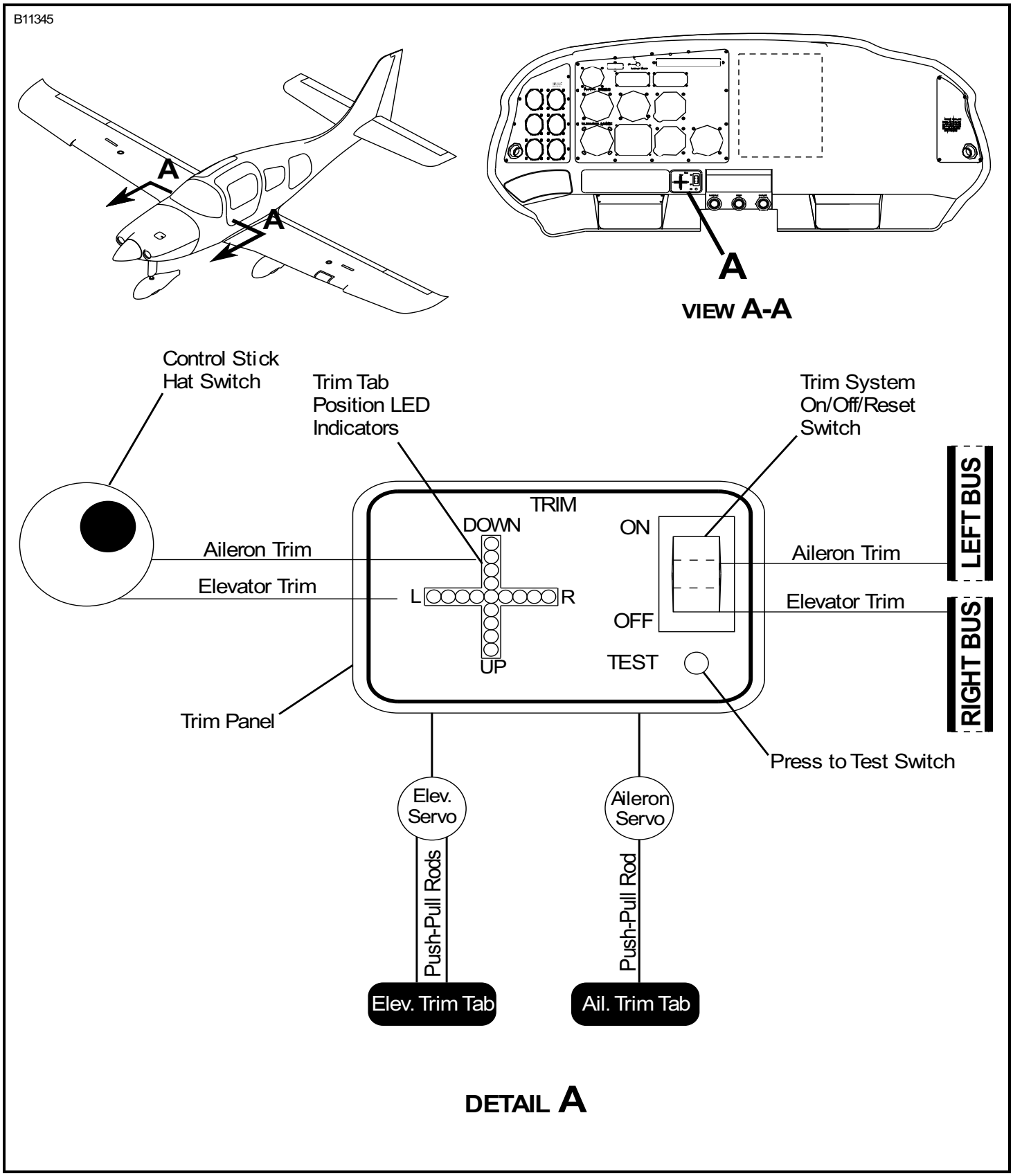
E. **Trim On/Off/Reset Switch**

(1) **Trim On/Off/Reset Switch (Basic or Avidyne Option)** - The trim system on/off/reset switch on the right side of the panel turns off power on all the trim tabs. This switch is used if a runaway trim condition is encountered. Refer to the POH for an expanded discussion of this issue. The press-to-test switch verifies the operation of all the LEDs associated with the trim, flaps, fuel tank position, and annunciator panel. When the test position is selected, all related LEDs illuminate in the bright mode. A light that fails to illuminate should be replaced.

(2) **Autopilot/Trim Master Switch** - The autopilot/trim master switch, to the right of the avionics master switch in the overhead console, turns off power on all the trim tabs. This switch is used if a runaway trim condition is encountered. The switch can be cycled to reset or restore normal trim operations. Refer to the POH for an expanded discussion of this issue. The press-to-test switch is discussed in chapter 7 of the POH under the heading *Flaps, and 5 Pack Switch Annunciation (Press- to-Test)*.

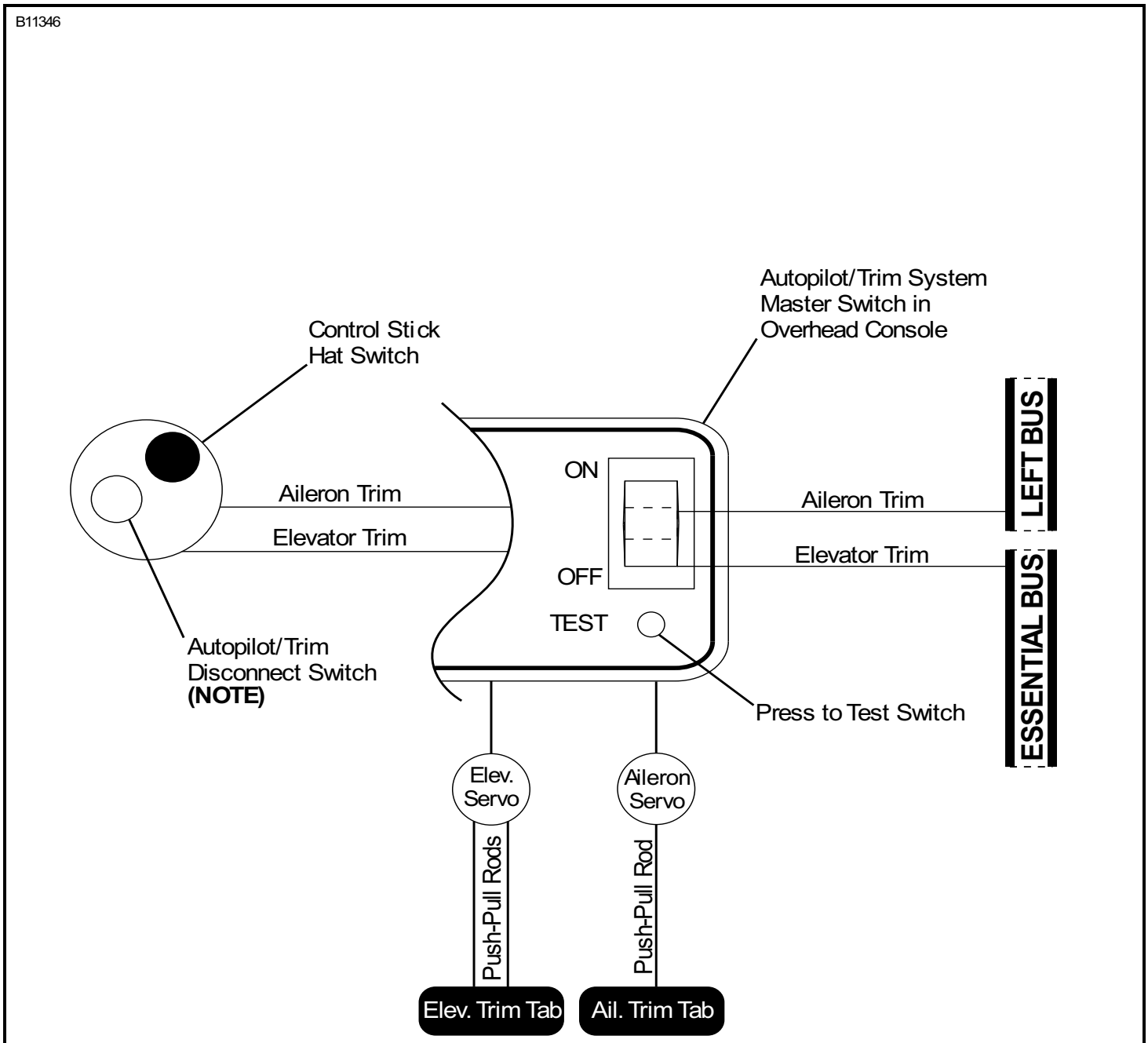
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Figure 1 : Sheet 1 : Trim System Layout (Basic or Avidyne Option)



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Figure 2 : Sheet 1 : Trim System Layout (Garmin G1000 Option)



NOTE: Pushing the Autopilot/Trim Disconnect Switch stops trim.

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CONTROL SURFACE STATIC BALANCING - MAINTENANCE PRACTICES

1. Control Surface Static Balancing - General

A. Control surface balance must be checked when control surfaces are repaired, altered, or repainted. All control surfaces must be repainted in accordance with Chapter 51. The control surface balance limits shown in Table 201 apply to a complete painted control surface only. Complete control surfaces include (as applicable) balance weight, all attachment hinges and hardware, trim tabs, grounding straps, and static wicks.

Table 201. Control Surface Static Balance Limits

		Maximum Total Weight	TE Hinge Moment (NOTE 1)
Aileron	Left	7.40 lbs.	+2.25 to +3.25 in.-lbs.
	Right	7.80 lbs.	
Elevator	---	32.30 lbs.	+37.00 to +39.00 in.-lbs.
Rudder	---	14.35 lbs.	+6.20 to +9.20 in.-lbs.

NOTE 1:

All hinge moments are trailing edge heavy.

NOTE: If specified moments cannot be met, heavier balance weights may be obtained from the manufacturer. This approach should be considered before reworking any repair or restripping and repainting the affected control surface. A lighter weight can be produced by shaving existing balance weight per the instructions outlined in the sections below.

2. Required Equipment For Balancing

A. Obtain or otherwise fabricate two knife-edges (refer to Figure 201) that are approximately 6 - 12 in. in height such that they may be placed on a table and stabilized to prevent tipping. The knife-edge hinge supports must be level and perpendicular to the control surface hinge axis. The knife-edges should have attached a 1.5 to 2.0-inch section of MS20001P5 piano hinge to match the elevator attachment hinge. The knife-edges should also be small enough in height to fit within the rudder at the hinge pin locations.

B. A calibrated scale accurate to ±0.01 lbs.

C. Calculator.

D. Calibrated leveling device (digital inclinometer, bubble level, etc) accurate to ±0.2 degrees.

3. Balancing Procedures - Aileron

A. **Aileron Static Balancing** - Use the following procedure for static balancing the ailerons:

(1) Mount the two knife-edges horizontally such that a line drawn through the hinge line support points is level and perpendicular to the supporting knife-edges.

(2) Set the aileron trim tab to the neutral position.

(3) Remove the ailerons per the instructions in Section 27-10-00, Aileron Removal and Installation.

(4) On the left aileron, tape the servo tab in the neutral position with a small piece of masking tape. Tape the two drive rods to the servo tab hinge. Include all attachment hardware for the control rods.

(5) Install the attachment hardware into each of the three hinges.

(6) Place either aileron on the knife-edges such that the inboard and outboard attachment bolts are resting on the knife-edges. Swing the aileron up and down to ensure that the knife-edges do not interfere with the aileron and hinges.

(7) Position the aileron such that the chord line is level, and measure the trailing edge weight at the inboard end. To check that the chord line is level, measure the angle of the top surface of the aileron at the inboard end, and confirm that it is 10.9±0.2 degrees from horizontal. Refer to Figure 202.

(8) Calculate the moment by multiplying the control surface load by 6.72 inches (moment arm). Compare this value with that in Table 201.

(9) If the moment is too light, material may be removed from the balancing weight to achieve the correct moment. Remove material from the aft face of the balancing weight first. If further weight reduction is needed, remove material from between the two fastener holes. The cross-hatched areas in Figure 203 indicate the areas where material may be removed from the aileron balancing weights. The width may be adjusted as long as a minimum of 0.125-inch gap is maintained between the counterweight and the aileron cove in the wing.

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(10) If the moment is too heavy, a heavier balancing weight may be obtained from the manufacturer.

WARNING: Do not remove material from the forward or lower edge of the balancing weights. It is important that the contour of the forward and lower edge and the width of the balancing weights be maintained.

4. Balancing Procedures - Elevator

A. **Elevator Static Balancing** - Use the following procedure for static balancing the elevator.

(1) Mount the two knife-edges horizontally such that a line drawn through the hinge line support points is level and perpendicular to the supporting knife-edges.

(2) Set the elevator trim tab to the neutral position.

(3) Remove the rudder per the instructions in the Rudder Removal and Installation procedures.

(4) Remove the elevator per the instructions in the Elevator Removal and Installation procedures.

(5) Remove the forward half of the elevator attachment hinge by cutting the safety wire and removing the hinge wire.

(6) Turn the elevator upside down, and align the hinge with the hinge sections located on the knife-edges. Reinsert the hinge wire into the piano hinge on the elevator.

NOTE: Position the knife-edges such that they are equal distant from each end of the elevator.

(7) With the elevator chord line level, accurately measure the control surface load at the trailing edge a distance of 20.00±0.125 inches from either end. The elevator chord line is level when the angle of the upper or lower surface is 6.0±0.2 degrees, measured perpendicular to the hinge line. Refer to Figure 204.

(8) Calculate the moment by multiplying the control surface load by 10.00 inches (moment arm). Compare this value with that in Table 201.

(9) If the moment is too light, material may be removed from the balancing weight to achieve the correct moment. Determine which weight is installed from Figure 205; cross-hatched areas indicate where material can be removed. Remove material from the aft mounting hole first, followed by the center hole, and the forward mounting hole last. When removing material from the balancing weights, remove equal amounts of material from the balancing weight on each end of the elevator.

WARNING: Do not remove material from the forward or lower edge of the balancing weights and do not reduce the overall width of the balancing weights. It is important that the contour of the forward and lower edge and the width of the balancing weights be maintained.

(10) Use new MS20365-428A self-locking nuts at final installation of the elevator balancing weights to the elevator. Torque the nuts 45 to 50 in.-lbs.

5. Balancing Procedures - Rudder

A. **Rudder Static Balancing** - Use the following procedure for static balancing the rudder.

(1) Remove the rudder per the instructions in the Rudder section.

(2) Mount both knife-edges horizontally and parallel to each other.

(3) Place the rudder on the knife-edges such that there is one knife-edge on the top and lower hinge pins. A line drawn through the hinge line support points must be level and perpendicular to the supporting knife-edges. Move the rudder up and down to ensure that the knife-edges do not interfere with the rudder and cause friction that could affect the surface load.

(4) With the rudder chord line level, accurately measure the control surface load at the trailing edge at the lower edge of the trim tab. Refer to Figure 206. The rudder has a symmetrical cross section and thus the chord line is level when the center of the leading edge is at the same height as the center of the trailing edge.

(5) Calculate the moment by multiplying the control surface load by 10.00 in. (moment arm). Compare this value with that in Table 201.

(6) If the moment is too light, the balancing weight may be shaved down to achieve the correct moment. When shaving the balancing weight, maintain a minimum thickness of 0.375 in. from the edge of the bolt hole to the top edge of the part. Refer to Figure 207.

6. Rudder Limiter System

A. **Description of Rudder Limiter System**

(1) **Operation** - The rudder limiter is an integral part of the stall system on the Cessna 350 (LC42-550FG). The system is

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a part of the safety-of-flight design of the Cessna 350 and is an important factor in spin resistance. It is designed to limit full left rudder deflection from a normal (approximately) 17.0±1.0 degrees; to approximately 11±0.5 degrees during certain aircraft angle of attack and engine conditions. The system is activated when two conditions simultaneously exist: (1) the airplane's stall warning is active and (2) the engine manifold pressure is more than 12 in. of Hg. When the system is activated, a solenoid near the left rudder pedal moves a cam that physically limits the travel of the left rudder pedal. There is a time delay of approximately one second before the system is activated. This delay feature prevents inadvertent activation of the rudder limiter in turbulent air.

(2) **Fail-Safe Feature** - The rudder limiter comes with a fail-safe feature. The system is armed when the airplane's electrical power is turned on; however, all electronic and electrical switching is in the relaxed position. When the stall warning is active and manifold pressure is more than 12 inches of Hg, the system activates from this "relaxed armed" position. If either of the two inputs to the system should fail, the rudder limiter will still operate. For example, if the manifold pressure transducer becomes inoperative, the rudder limiter will be activated by the sole input from the stall warning. Conversely, if the stall warning fails, the rudder limiter will be functional, provided the stall warning detector is operative, i.e., freely moves up and down. The operating condition of this fail-safe system can be verified from time to time through the use of a special ground testing procedure. The rudder limiter operates only when the stall warning is active and the throttle is advanced to a setting that produces more than 12 inches of Hg manifold pressure. Deflection of the right rudder pedal is unaffected by the rudder limiter system.

B. **Rudder Limiter Assembly Removal**

(1) The rudder limiter assembly is located on the pilot's side of the cabin, on the left side of the rudder crossover tube behind the rudder pedals. To easily access the rudder limiter assembly, you must first remove the pedals, the knee bolster, and the pilot's seat. Refer to Chapters 25 and 27 for details on removal of these items.

(2) Unfasten the rudder limiter assembly by unscrewing the cap screws as shown in Figure 208.

(3) Disconnect the wires on the solenoid and proximity switch.

(4) Remove the rudder limiter assembly.

C. **Rudder Limiter Assembly - Installation**

(1) Perform installation in the reverse sequence of the removal procedure. See Figure 208.

(2) Adjust the firewall stop by loosening the stop nut and screwing out the stop until it firmly touches the firewall. Retighten the stop nut.

(3) Re-connect the wires to the proximity switch and solenoid.

(4) Test function of the rudder limiter assembly by following the steps listed in the Rudder Limiter Test section.

D. **Rudder Limiter Solenoid Replacement** - It is not necessary to remove the entire rudder limiter assembly when replacing the solenoid, shown in Figure 209.

(1) Disconnect the solenoid.

(2) Screw the nut off the solenoid, and slide the solenoid off the bracket.

NOTE: Care must be taken to prevent the spring that is located on the other side of the nut holding the solenoid from coming off.

(3) Installation is the opposite of removal. Torque the nut 100 to 110 in.-lbs.

(4) Test function of the rudder limiter assembly by following the steps listed in the Rudder Limiter Test section.

E. **Rudder Limiter Sensor Replacement** - It is necessary to remove the entire rudder limiter assembly when replacing the sensor, shown in Figure 209.

(1) Remove the rudder limiter assembly.

(2) Remove the retaining nut from the sensor, and slide the sensor off of the rudder limiter bracket.

(3) Installation is the opposite of removal.

(4) Ensure the gap between the end of the sensor and limiter cam is 0.15±0.07 inches.

(5) Ensure proper on and off operation by checking continuity. With the cam nearest the sensor, the circuit should close. With the cam against the stud the circuit should be open.

F. **Rudder Limiter Test** - The purpose of this test is to verify operation of the manifold pressure transducer, the solenoid and rudder pedal cam, and the general integrity of the rudder limiter system.

(1) While sitting in the pilot's seat with the master switch on and the engine off, depress the test button on the trim panel.

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See Figure 210 for Basic or Avidyne option only. With the Garmin G1000 option the rudder limiter test button is in the overhead console beneath the A/P trim switch.

(2) When the test button is depressed, the pilot will hear (after a one-second delay) and feel the solenoid near the left rudder pedal engage, at which time the RUDR LMTR annunciator will illuminate for the Basic or Avidyne option or display in the annunciation window of the PFD for the Garmin G1000 option, and the left rudder travel will be restricted.

(3) When the operation is verified, release the test switch. The rudder limiter test switch is also used to test the integrity of the trim, flap, fuel tank LEDs (Basic or Avidyne option), and 5 pack switches (Garmin G1000 option). The pilot should remember that anytime these lights are tested, the rudder limiter will engage.

(4) With the master switch on and the engine off, pull the stall warning circuit breaker and have someone move the stall vane to the up position.

(5) The rudder limiter should engage even though there is no aural stall warning.

(6) Repeat the procedure with the engine instrument's circuit breaker pulled and the stall warning breaker reset.

(7) This time the rudder limiter will engage with an aural stall warning, even though there is no manifold pressure indication.

WARNING: The rudder limiter is an integral part of the Cessna 350 aircraft. It is mandatory that the rudder limiter be in full working order for any flight. Should problems exist that are not described in this manual, contact the factory for assistance.

NOTE: The rudder limiter assembly is factory-set and should be replaced as a unit instead of field modification of components.

Table 202. Troubleshooting the Rudder Limiter Cam System

SYMPTOM	PROBABLE CAUSE	ACTION
Rudder limiter fails to activate.	a) Solenoid does not activate	1. Check the rudder limiter cam system circuit breaker. 2. Press the test button, and listen for the solenoid to activate. 3. Push the stall vane up. After about a second, a "click" near the pedals should be heard as the rudder limiter engages.
	b) Bad wiring or connectors (refer to wiring manual for further details)	1. Check for continuity and current at the solenoid. Replace bad connectors or wiring. 2. Check the circuit breaker, and replace breaker if needed. 3. If wiring is acceptable, then replace the rudder limiter assembly.
	c) Mechanical blockage	1. Check that any foreign material such as carpet is not blocking the cam. 2. Check that solenoid shaft moves freely when activated. If it does not move freely, replace the rudder limiter assembly.
Rudder limiter annunciator fails to light with rudder	a) Light bulb burned out	1. Test the annunciator light using the annunciator test switch. If light does not operate, replace LED. (See Chapter 33)

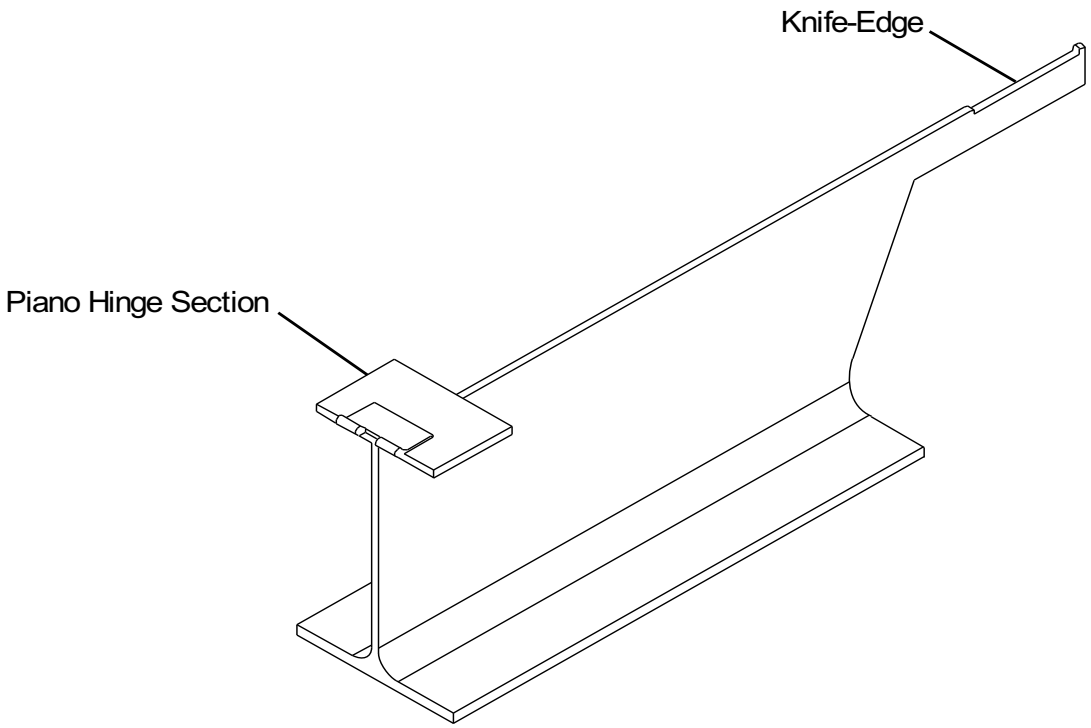
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limiter engaged.	b) Proximity switch failure	1. With electrical power on, place magnetic screwdriver between proximity switch and rudder limiter cam. 2. If the annunciator fails to light, check the wiring and connections. 3. If they are correct, replace the rudder limiter assembly.

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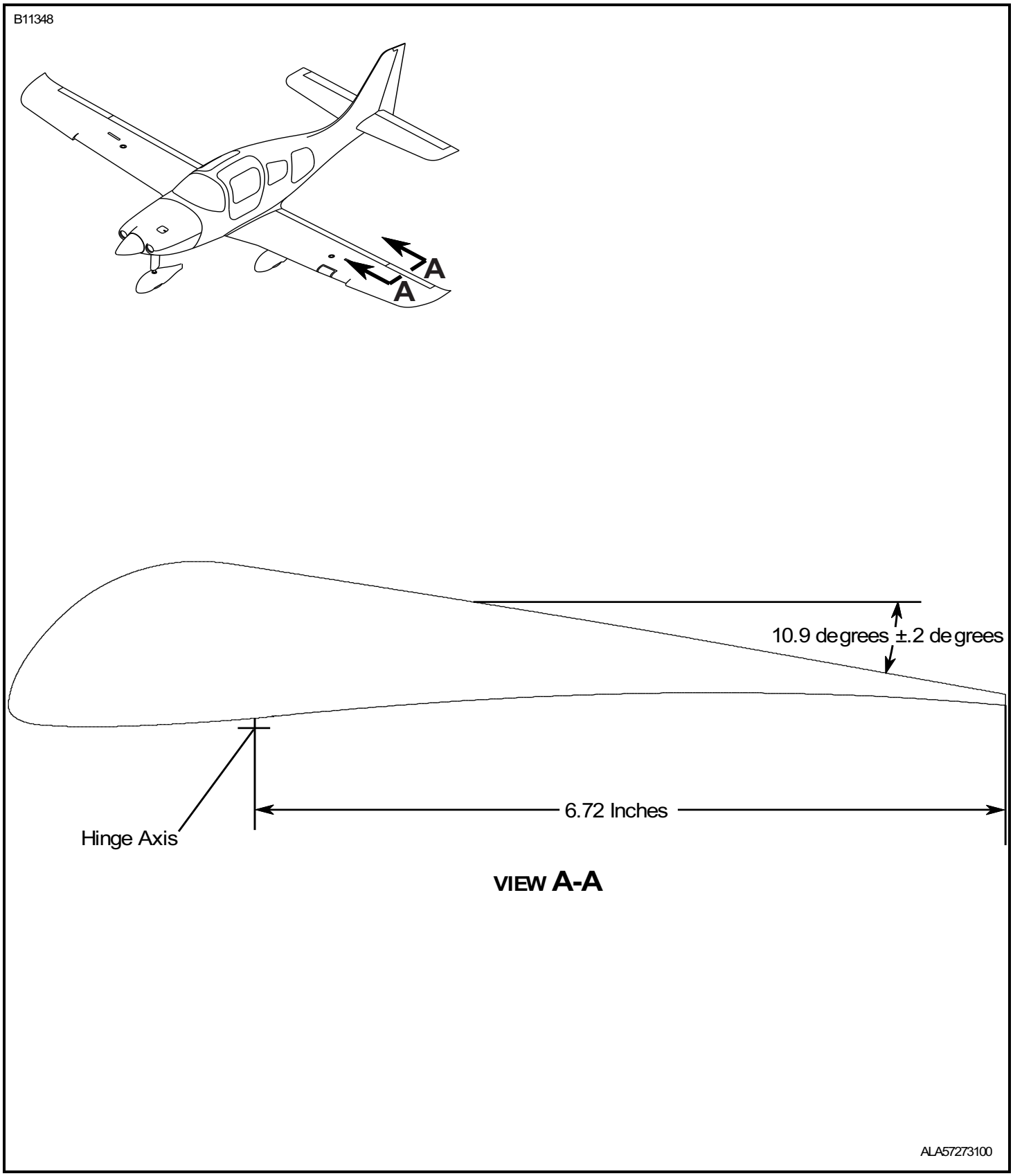
Figure 201 : Sheet 1 : Typical Knife-Edge

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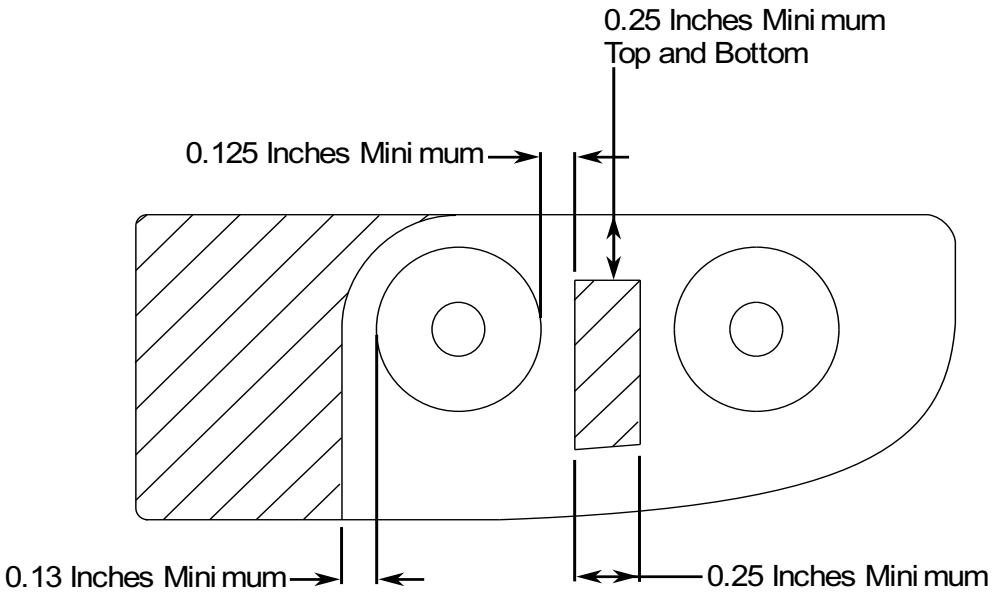
Figure 202 : Sheet 1 : Level Aileron Chord Line and Moment Arm



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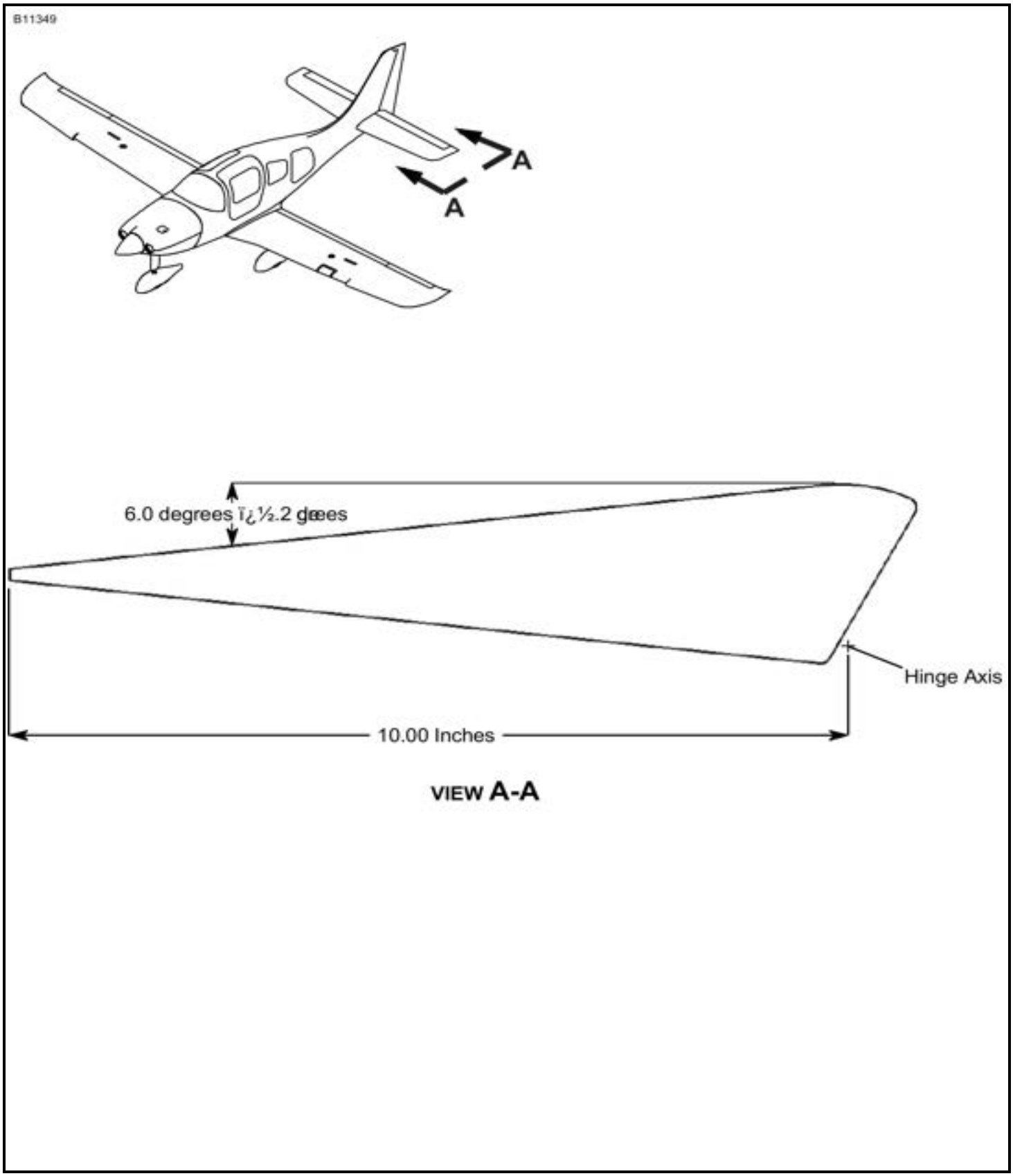
Figure 203 : Sheet 1 : Allowable Areas of Material Removal, Aileron Balancing Weight

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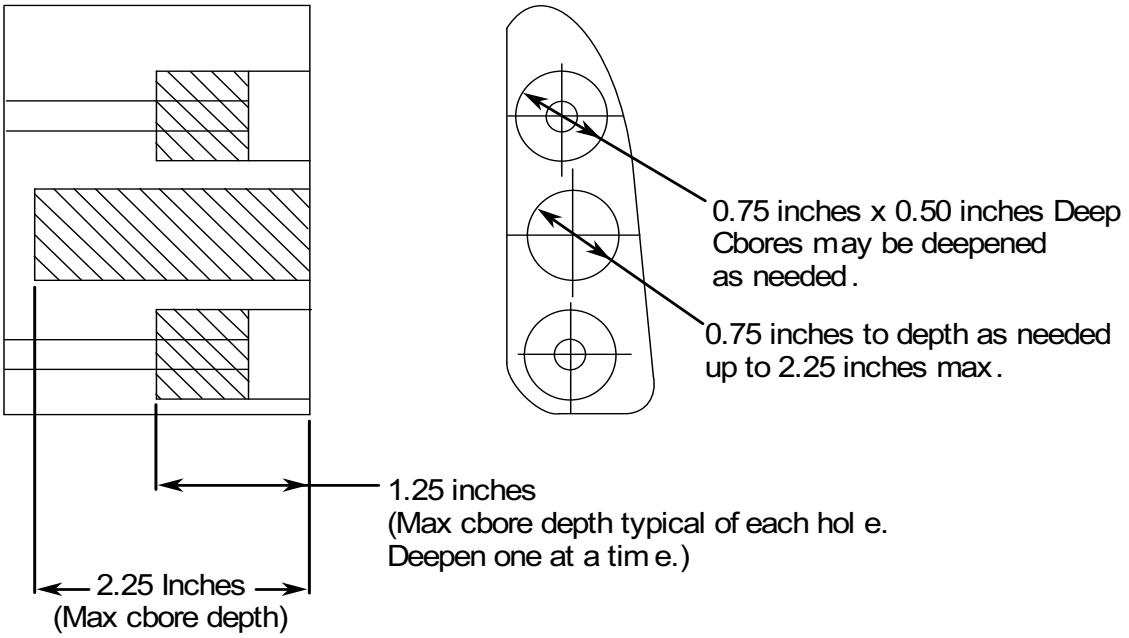
Figure 204 : Sheet 1 : Level Elevator Chord Line and Moment Arm



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Figure 205 : Sheet 1 : Allowable Areas of Material Removal, Elevator Balancing Weight

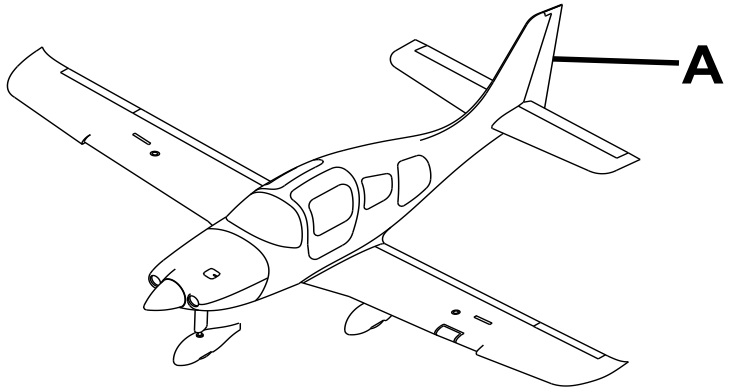
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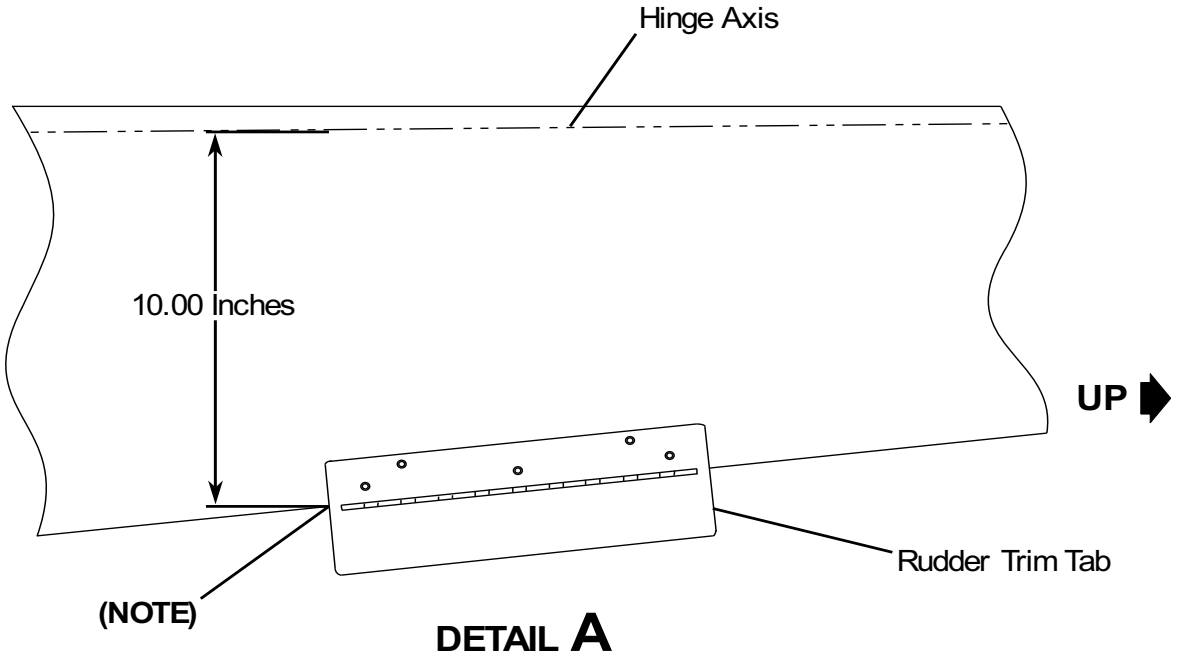


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Figure 206 : Sheet 1 : Location for Measuring the Rudder Surface Load

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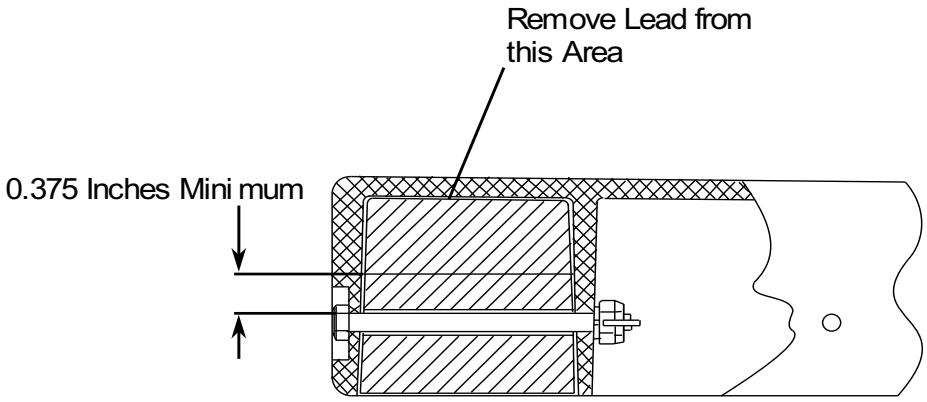
DETAIL A

NOTE: Measure control surface load at this location.

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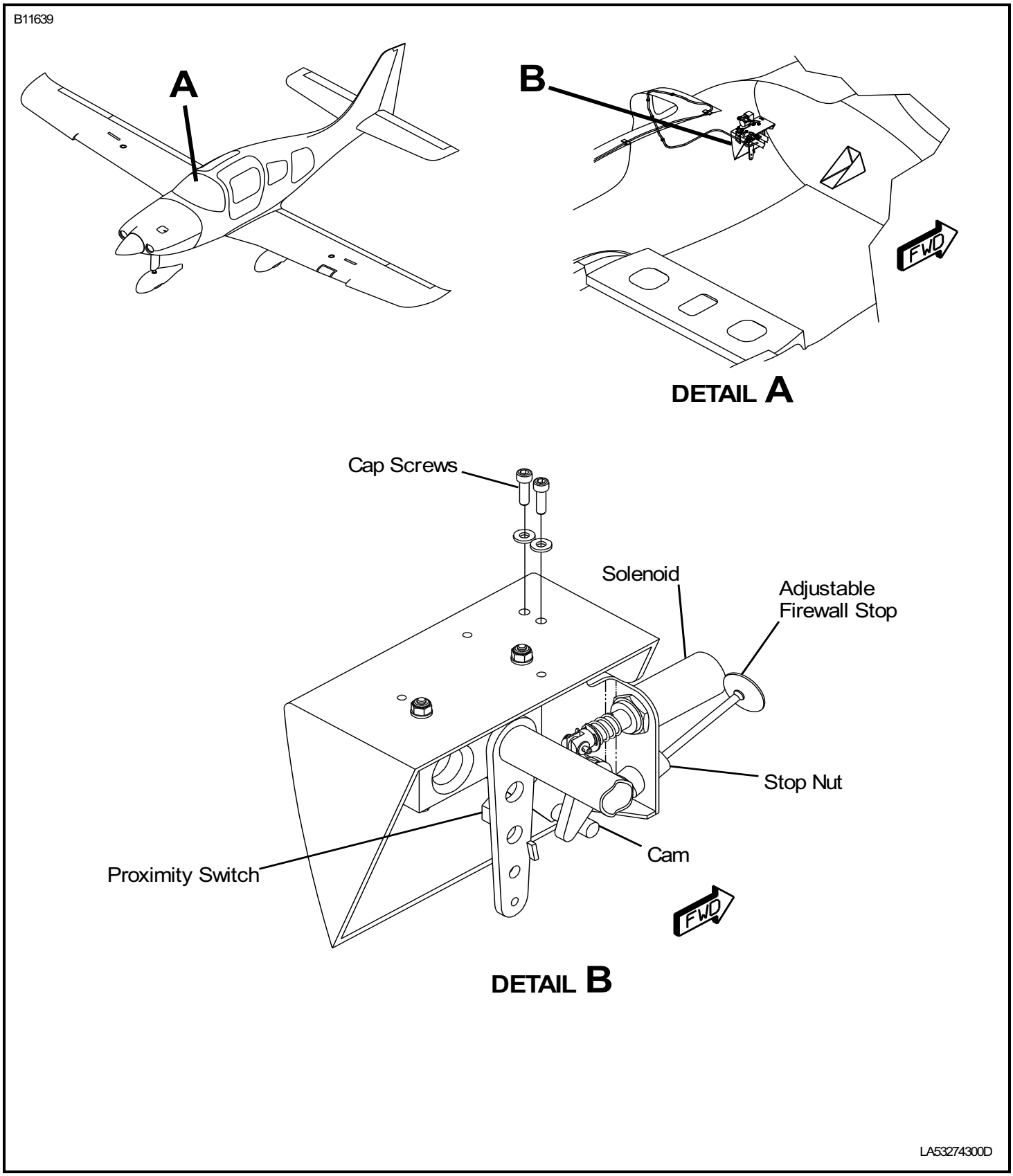
Figure 207 : Sheet 1 : Allowable Area of Material Removal, Rudder Balancing Weight

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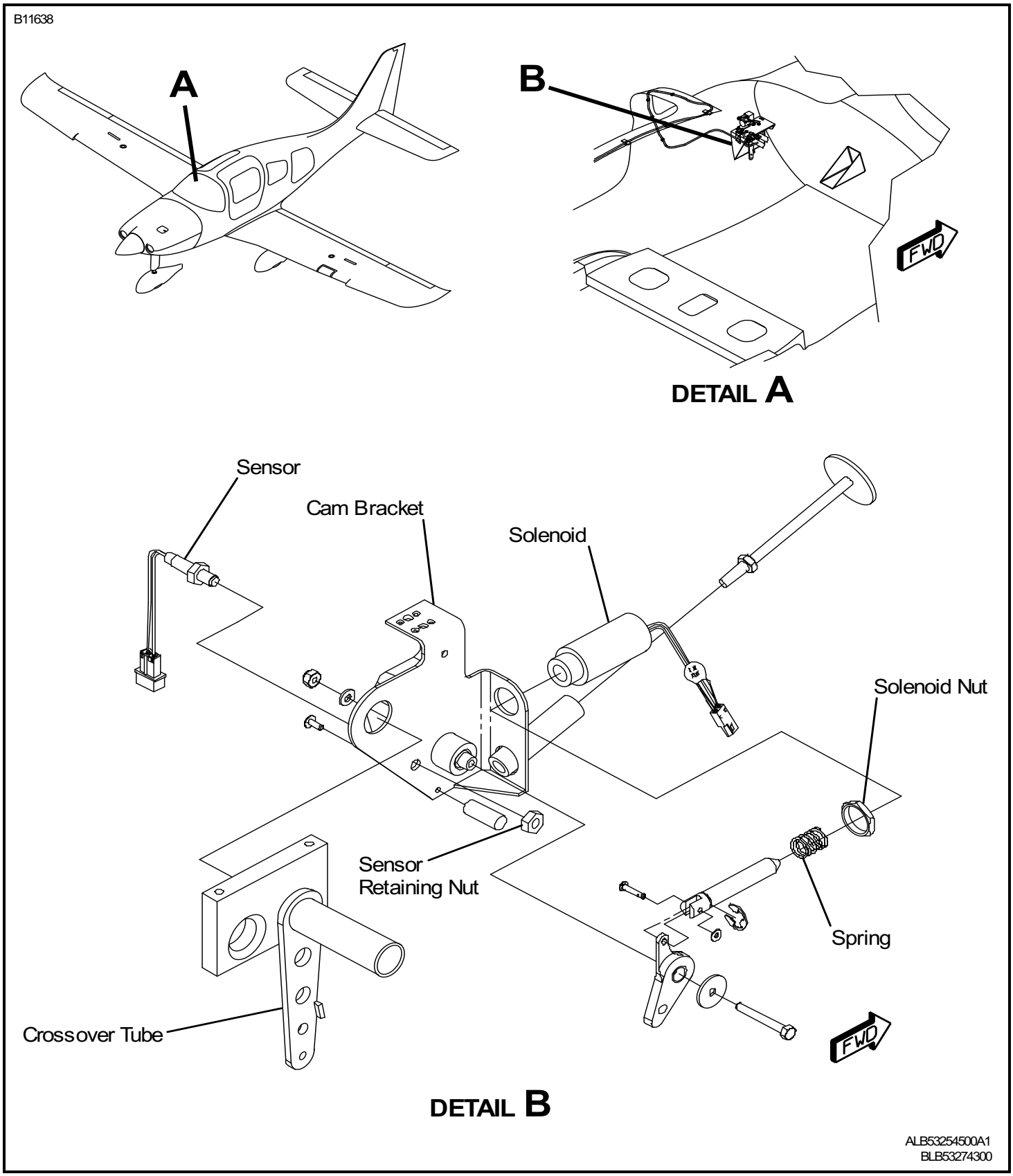
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Figure 208 : Sheet 1 : Rudder Limiter Assembly



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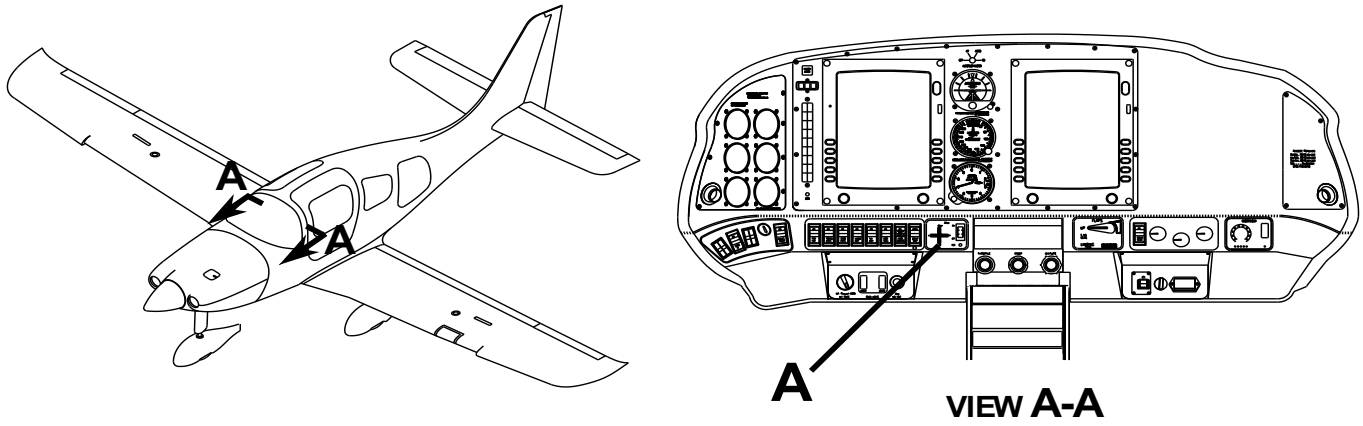
Figure 209 : Sheet 1 : Rudder Limiter Assembly

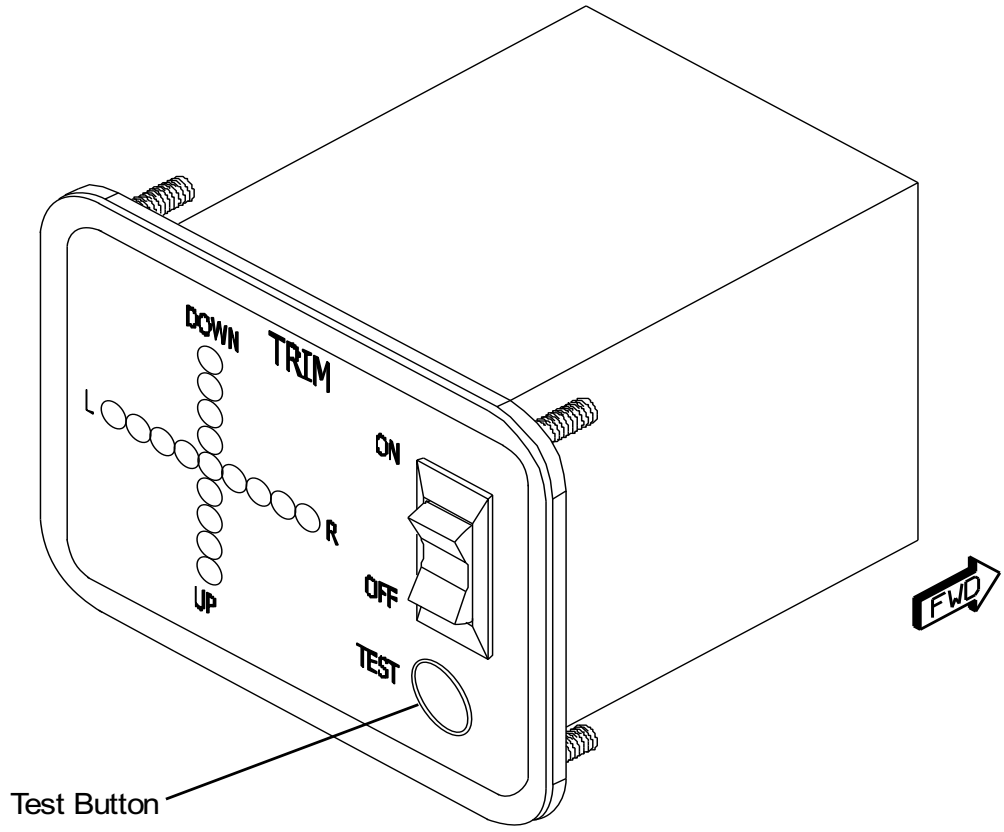


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Figure 210 : Sheet 1 : Rudder Limiter Test Button (Basic or Avidyne Option Only)

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DETAIL A

ALB53311800A
AALC53310000C

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AILERON CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Aileron System - General

A. The ailerons are of a carbon fiber composite construction. The left aileron has a servo tab for reducing pilot forces during flight. The right aileron has an electrically actuated trim tab that is used for lateral trim during flight. Lead counterweights static balance the ailerons.

2. Ailerons

A. Aileron Removal and Installation

(1) Disconnect the aileron final drive from the aileron drive horn by removing the cotter pin, castellated nut, and bolt.

(2) Disconnect the grounding strap at the middle aileron hinge from the aileron by removing the AN526-832R8 screw.

(3) On the right wing, disconnect the aileron trim servo wire, accessed through the outer wing-bay access panel.

(4) On the left aileron, disconnect the servo tab drive rods from the wing aileron hinge by removing the cotter pins and clevis pins.

(5) Remove the aileron from the wing aileron hinges by removing the cotter pin, castellated nut, and bolt at each of the three hinge locations.

(6) Slide the aileron straight aft. Take care not to move the right aileron too far aft and break the wiring connector for the trim system.

(7) Gently pull the wiring harness for the aileron servo from out of the wing, and disconnect the wiring connector.

(8) Reinstall the ailerons in the reverse order.

(9) After installing the ailerons, check that the deflections per Section 27-00-00, Table 1, are correct.

B. **Aileron Rigging and Adjustment** - The aileron control system is rigged at the factory and should not need any further adjustments. For new ailerons, or ailerons requiring significant readjustment in order to achieve the correct deflection limits, use the following procedure:

NOTE: Keep the flaps in the fully-retracted position (0 degrees) during the rigging process.

(1) With the aileron final drives disconnected, verify smooth operation throughout the full range of travel for each aileron.

(2) Place the flap/aileron deflection template, part number TA57270000, on the upper surface of the wing between the outboard end of the flaps and the inboard end of the aileron.

(3) The downward deflection of an aileron should be approximately 17.7 degrees. If the deflection of an aileron is not enough, carefully sand material from its down-stop to achieve this deflection. If the deflection is too great, contact the manufacturer for new aileron down-stops.

(4) Level the aircraft about the roll axis before proceeding.

(5) Disconnect the aileron crossover control rod in the center section of the wing from the aileron torque tubes. Set both control sticks in the cockpit to a position of 88±2 degrees as shown in Figure 201. Adjust the rod ends on the crossover tube so the assembly may be installed between the torque tubes and tighten the jam nuts. Reinstall the torque tube.

(6) With the aileron control rods disconnected, lock both the left-hand and right-hand aileron bellcranks in the neutral position within the bellcrank brackets by placing a 1/4 inch- diameter pin through the rigging hole. An AN4 bolt works well for this.

(7) With the control sticks still at 88±2 degrees, adjust the rod ends on the aileron control rods as required to allow them to fit between the aileron torque tubes and the bellcranks. Finger-tighten the control rod connections.

(8) Clamp the ailerons in the neutral position (0 degrees deflection on the rigging template). Adjust the length of the final drive to fit between the bellcrank and the aileron hinge and finger-tighten the connections.

(9) Remove the pins from the bellcranks and release the control sticks. Unclamp the ailerons and rotate them until they reach the up-stop (which is controlled by the down-stop on the opposite aileron). The up-stop on one aileron and the down-stop on the opposite aileron should make contact at the same time while ensuring the trailing edge is inside the template up-travel window. Both right-hand and left-hand sides are set in this manner. If more travel is needed, a combination of up-stop adjustment and/or careful sanding of material from the down-stop on the opposite aileron may be required. Continue this procedure until both ailerons meet all of the deflection requirements of Section 27-00-00, Table 1.

(10) Ensure that the gap at the outboard edge of the aileron between the wing and the aileron is 0.123 to 0.309 inches

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(11) Ensure that the gap at the inboard edge of the aileron between the flap and the aileron (when the flaps are retracted) is 0.125 to 0.375 inches. Ensure minimum 0.125 inch clearance when the flaps are extended.

(12) While the aileron is in neutral position ensure that the gap between the wing trailing edge and the aileron is 0.150 to 0.250 inches along the entire span of the aileron.

(13) Verify there is sufficient thread engagement on all rod ends by checking that the threads are visible within the witness hole on each.

(14) Once the correct aileron deflections and proper gaps are achieved, retighten all jam nuts and torque 12 to 15 ft.-lbs.

(15) Ensure the aileron bellcrank, final drive, and all associated hardware has a minimum of 0.040 inches clearance from the wing skins through full range of travel after the aileron is properly rigged.

C. **Aileron Rigging and Adjustment (Minor Adjustment Only)** - For ailerons requiring only minor adjustment, use the following procedure:

NOTE: Keep the flaps in the fully-retracted position (0 degrees) during the rigging process.

(1) With the aileron final drives disconnected, verify smooth operation throughout the full range of travel for each aileron.

(2) Place the flap/aileron deflection template, part number TA57270000, on the upper surface of the wing between the outboard end of the flaps and the inboard end of the aileron.

(3) The downward deflection of an aileron should be approximately 17.7 degrees. If the deflection of an aileron is not enough, carefully sand material from its down-stop to achieve this deflection. If the deflection is too great, contact the manufacturer for new aileron down-stops.

(4) Level the aircraft about the roll axis before proceeding.

(5) Remove the pins from the bellcranks and release the control sticks. Unclamp the ailerons and rotate them until they reach the up-stop (which is controlled by the down-stop on the opposite aileron). The up-stop on one aileron and the down-stop on the opposite aileron should make contact at the same time while ensuring the trailing edge is inside the template up-travel window. Both right-hand and left-hand sides are set in this manner. If more travel is needed, a combination of up-stop adjustment and/or careful sanding of material from the down-stop on the opposite aileron may be required. Continue this procedure until both ailerons meet all of the deflection requirements of Section 27-00-00, Table 1.

(6) Ensure that the gap at the outboard edge of the aileron between the wing and the aileron is 0.123 to 0.309 inches

(7) Ensure that the gap at the inboard edge of the aileron between the flap and the aileron (when the flaps are retracted) is 0.125 to 0.375 inches. Ensure minimum 0.125 inch clearance when the flaps are extended.

(8) While the aileron is in neutral position ensure that the gap between the wing trailing edge and the aileron is 0.150 to 0.250 inches along the entire span of the aileron.

(9) Verify there is sufficient thread engagement on all rod ends by checking that the threads are visible within the witness hole on each.

(10) Once the correct aileron deflections and proper gaps are achieved, retighten all jam nuts and torque 12 to 15 ft.-lbs.

(11) Ensure the aileron bellcrank, final drive, and all associated hardware has a minimum of 0.040 inches clearance from the wing skins through full range of travel after the aileron is properly rigged.

D. **Aileron Service and Maintenance** The ailerons do not require any service or maintenance. The ailerons should be visually inspected annually for cracking at all hinge attachment locations. A tap test should be conducted on the aileron at the outboard end cap to aid in detection of any latent damage, delaminations, or disbonds. The rod end bearings used throughout the aileron control system do not require maintenance other than cleaning during routine inspections to remove any accumulation of dust or grime. Liquid lubricants should not be used on the rod end bearings as this may cause the Teflon liner to swell, creating excess friction in the bearing. Should the rod end need to be replaced, please contact the manufacturer. The aileron linear bearing should be inspected every 100 hours or annually. Inspect each side of the bearing for foreign object debris. Ensure there is no debris or adhesive attached to the control rod, bearing housing, bearing race, ball bearings, or rollers, as applicable. Move the aileron controls through their full range of motion and ensure there is no scarring or damage to the control rod, especially in the vicinity of the linear bearing. Normal wear at the linear bearing is indicated by a slightly darker color on the control rod. If damage is noted contact the manufacturer.The aileron control rods should be visually inspected every 500 hours or biennially. To remove the push-pull tubes for inspection:

(1) Remove the aileron bellcrank access panel located on the lower surface of the wing directly in front of the inboard aileron hinge.

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(2) Disconnect the aileron push-pull tube from the aileron bellcrank.

(3) Remove the aft wing belly access panel to gain access to the aileron crossover tube.

(4) Disconnect the aileron push-pull tube from the aileron torque tube bellcrank.

(5) Slide the push-pull tube out and down to clear the wing belly.

NOTE: It may be necessary to loosen or reposition the roll servo mount to give enough clearance to prevent damage to the rod during removal and installation. Refer to Chapter 22, Garmin GFC 700 Autopilot - Maintenance Practices for additional information on the installation of the roll servo mount.

(6) Reinstall the push-pull tubes in the reverse order.

(7) Reinstall all access panels and check the aileron controls for binding and friction.

CAUTION: Never lubricate the bearings that support the aileron torque tube.

3. Aileron Trim Tab

A. **Aileron Trim Tab Removal and Installation**

(1) Remove the four MS24693-C28 countersunk screws from the aileron trim servo access panel located on the lower surface of the right aileron.

(2) Disconnect the aileron trim tab drive rod and friction device clevis from the trim tab drive horn by removing the cotter pins and the clevis pins.

(3) Remove the MS24665-151 cotter pins on either end of the piano hinge, and remove the hinge wire. Once the hinge wire is removed, the aileron trim tab may be removed.

(4) Remove the four MS24693-C28 countersunk screws from the top surface of the right aileron.

(5) Carefully remove the trim tab servo from within the aileron.

(6) Once the trim motor is clear of the aileron, disconnect the wiring harness.

(7) Reinstall the trim motor and trim tab in the reverse order. Torque all screws 10 to 12 in.-lbs.

B. **Aileron Trim Tab Rigging and Adjustment** The aileron trim tab and servo are preset at the factory, and further adjustment should not be necessary. Should the trim tab servo motor need to be replaced, contact the manufacturer for a replacement part and for detailed rigging instructions. The trim tab deflections should be checked periodically to ensure that the trim system is functioning properly. Use the following procedure for checking the trim tab deflections.

(1) Set the aileron to the neutral (0 degrees) position.

(2) Using a digital inclinometer or other calibrated angle-measuring device, check the deflections shown in Figure 202.

C. **Aileron Trim Tab Service and Maintenance** The aileron trim tab is a flat aluminum plate. The trim tab does not require any maintenance. The trim tab hinge should be inspected annually for wear and corrosion. The hinge may be lubricated with MIL-L-7870 general purpose lubricating oil to help reduce wear and friction. The aileron trim tab friction device should be inspected every 500 hours, or biennially, and adjusted as necessary to 13.4 to 19.0 lbs. sliding friction. The friction device should be free from oils, grease, dirt, etc. If excessive wear is noted or smooth operation is not possible, the friction device should be replaced. A replacement, part number LA57273240, may be obtained from the manufacturer. The attachment bracket should be inspected every 500 hours or biennially for wear and replaced as necessary. The trim tab motor is a sealed unit and does not require service or maintenance.

4. Aileron Trim Tab Servo

A. **Aileron Trim Tab Removal and Installation** (Refer to Figure 203.)

(1) Remove the four MS24693-C28 countersunk screws from the aileron trim servo access panel.

(2) Disconnect the aileron trim tab drive rod and friction device clevis from the trim tab drive horn by removing the cotter pins and clevis pins.

(3) Remove the four MS24693-C28 countersunk screws from the upper surface of the right aileron.

(4) Carefully remove the trim tab motor from within the aileron.

(5) Once the trim motor is clear of the aileron, disconnect the wiring harness.

(6) Reinstall the trim tab motor in the reverse order. Torque all screws 10 to 12 in.-lbs.

B. **Aileron Trim Tab Rigging and Adjustment** The trim tab motor is preset at the factory and should not need further adjustment. Should adjustment be necessary, use the following procedure (refer to Figure 203):

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(1) Disconnect the friction device drive rod from the trim tab drive horn.

(2) Extend the trim motor drive arm all the way out.

(3) Set the trim tab at the up deflection limit, and adjust the drive rod as necessary to line up with the trim tab drive horn.

(4) Retract the trim motor drive arm, and check that the trim tab deflection angle is correct. Adjust the clevis on the drive rod as necessary to achieve the correct deflection. Maintain proper thread engagement (threads must be visible through the inspection hole located in the clevis).

(5) With the trim tab connected to the motor, retract the tab to the neutral position.

(6) Adjust the friction device drive rod, and connect it to the trim tab drive horn.

(7) Reinstall all clevis pins and cotter pins.

(8) Recheck deflection angles, and repeat above procedure if necessary.

C. **Aileron Trim Tab Service and Maintenance** The aileron trim tab servo is a sealed unit and does not require service or maintenance. Should the motor begin to slip or stop functioning properly, contact the manufacturer for a replacement.

5. Aileron Servo Tab

A. **Aileron Servo Tab Removal and Installation** (Refer to Figure 203.)

(1) Remove the cotter pins and clevis pins that attach the threaded drive rods to the servo tab bracket.

(2) Remove the MS24665-151 cotter pins from both ends of the piano hinge.

(3) Remove the hinge wire from the hinge, and remove the servo tab from the aileron.

(4) Reinstall the servo tab in the reverse order. Apply zinc oxide to the threaded push rod where the clevis attaches. Re-check deflections.

B. **Aileron Servo Tab Rigging and Adjustment** The aileron servo tab is preset at the factory and further adjustment should not be necessary. The servo tab deflections should be checked periodically to ensure that the system is functioning properly. Use the following procedure for checking the servo tab deflections.

(1) Place the aileron/flap deflection template, part number TA57270000, on the upper wing surface between the outboard end of the flap and the inboard end of the aileron.

(2) Set the aileron to the neutral (0 degrees) position.

(3) Set the servo tab to the neutral, 2 degrees trailing edge down, position by adjusting the drive rod clevises as necessary. Refer to Figure 204. Maintain proper thread engagement (threads must be visible through the inspection hole located in the clevis).

(4) Cycle the aileron to the up and down deflection limits, and check the servo tab deflection angles as shown in Figure 204.

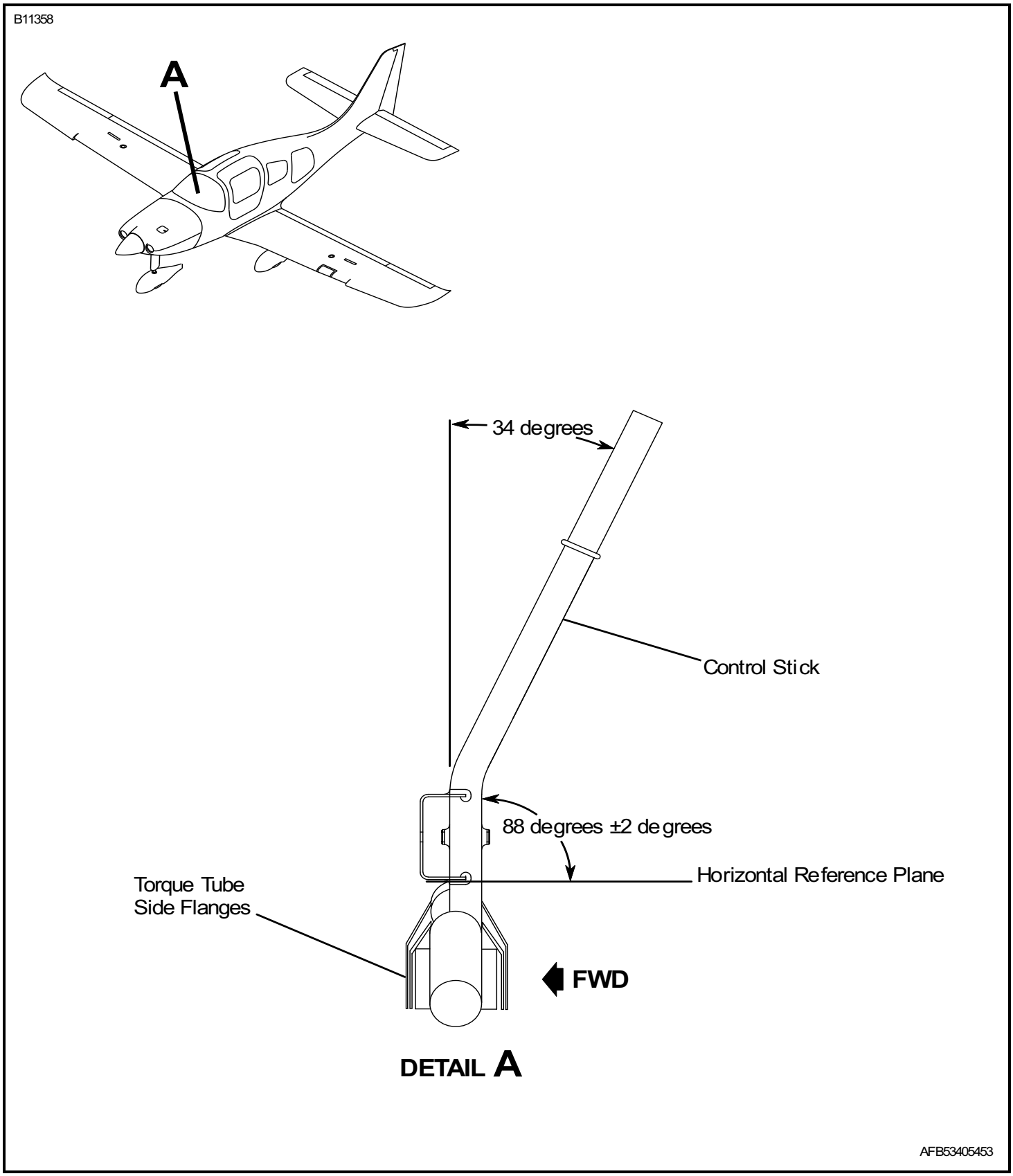
(5) Adjust the drive rods as necessary to achieve the correct deflections. Maintain proper thread engagement (threads must be visible through the inspection hole located in the clevis). Torque all jam nuts 30 to 36 in.-lbs.

NOTE: Aileron servo tab deflection angles are measured with respect to a nominal horizontal reference plane.

C. **Aileron Servo Tab Servicing and Maintenance** The aileron servo tab does not require any service or maintenance. The attachment hinge should be inspected every 500 hours or biennially for wear and replaced if necessary. The hinge may be lubricated with MIL-L-7870 general purpose lubricating oil to help reduce friction and wear.

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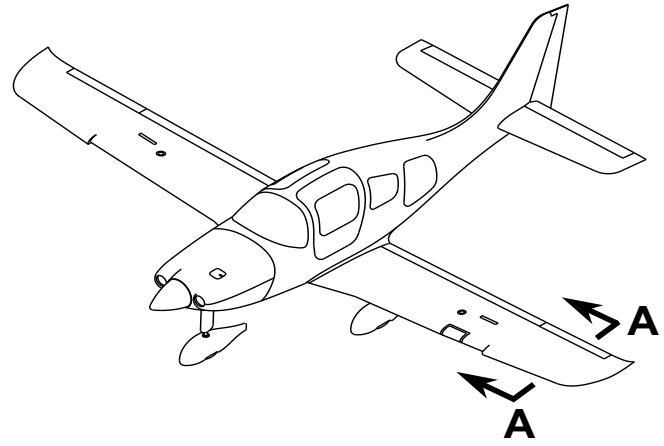
Figure 201 : Sheet 1 : Control Stick Neutral Position for Aileron Rigging

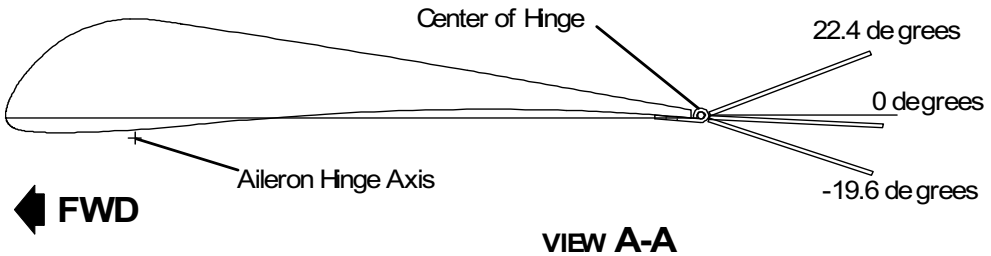


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Figure 202 : Sheet 1 : Aileron Trim Tab Deflection Angels

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VIEW A-A

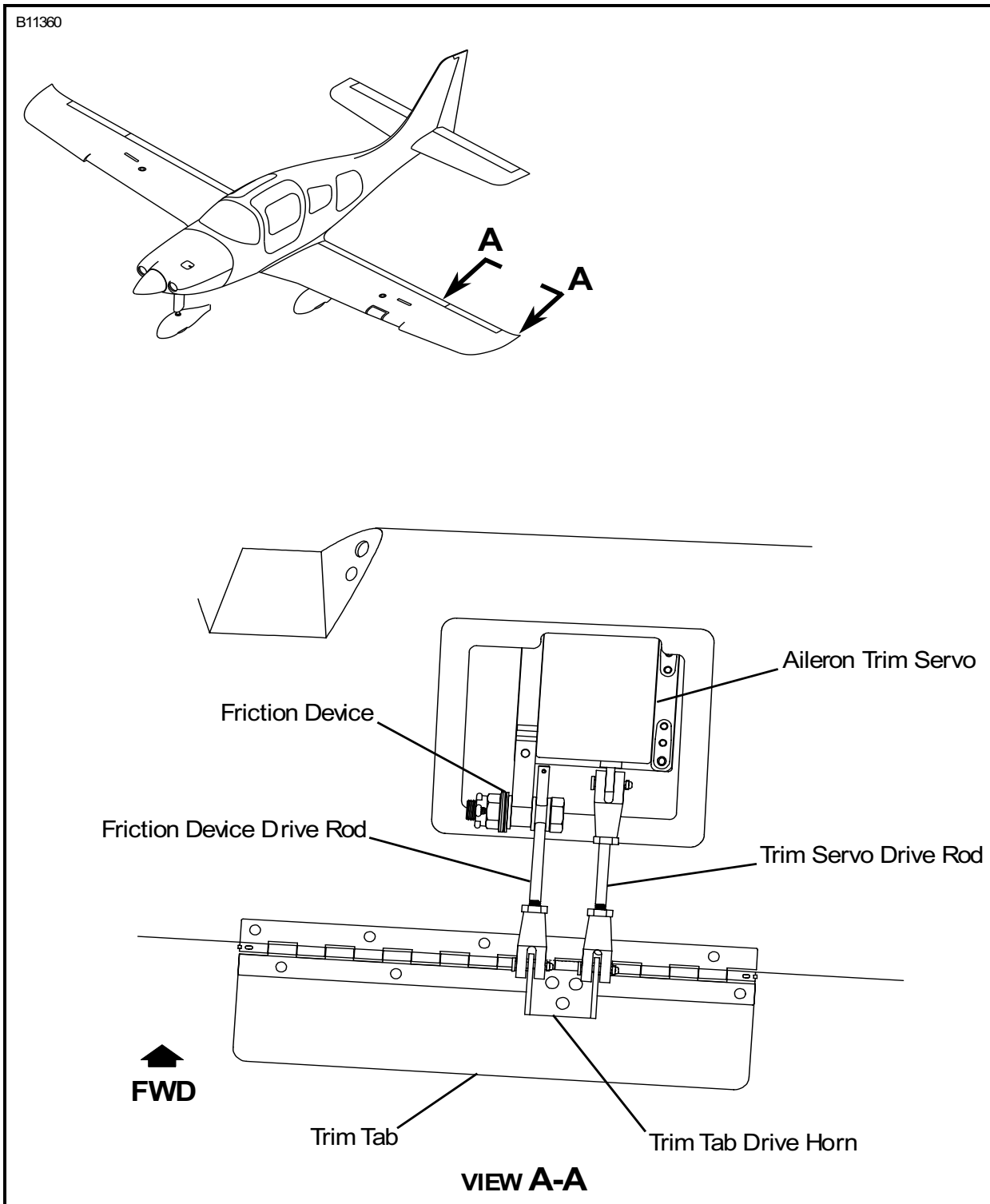
Tab 22.4 degrees up ± 1 degree with aileron at 0 degrees.

Tab 3 degrees down ± 1 degree with aileron at 0 degrees (NEUTRAL REF)

Tab 19.6 degrees down ± 1 degree with aileron at 0 degrees.

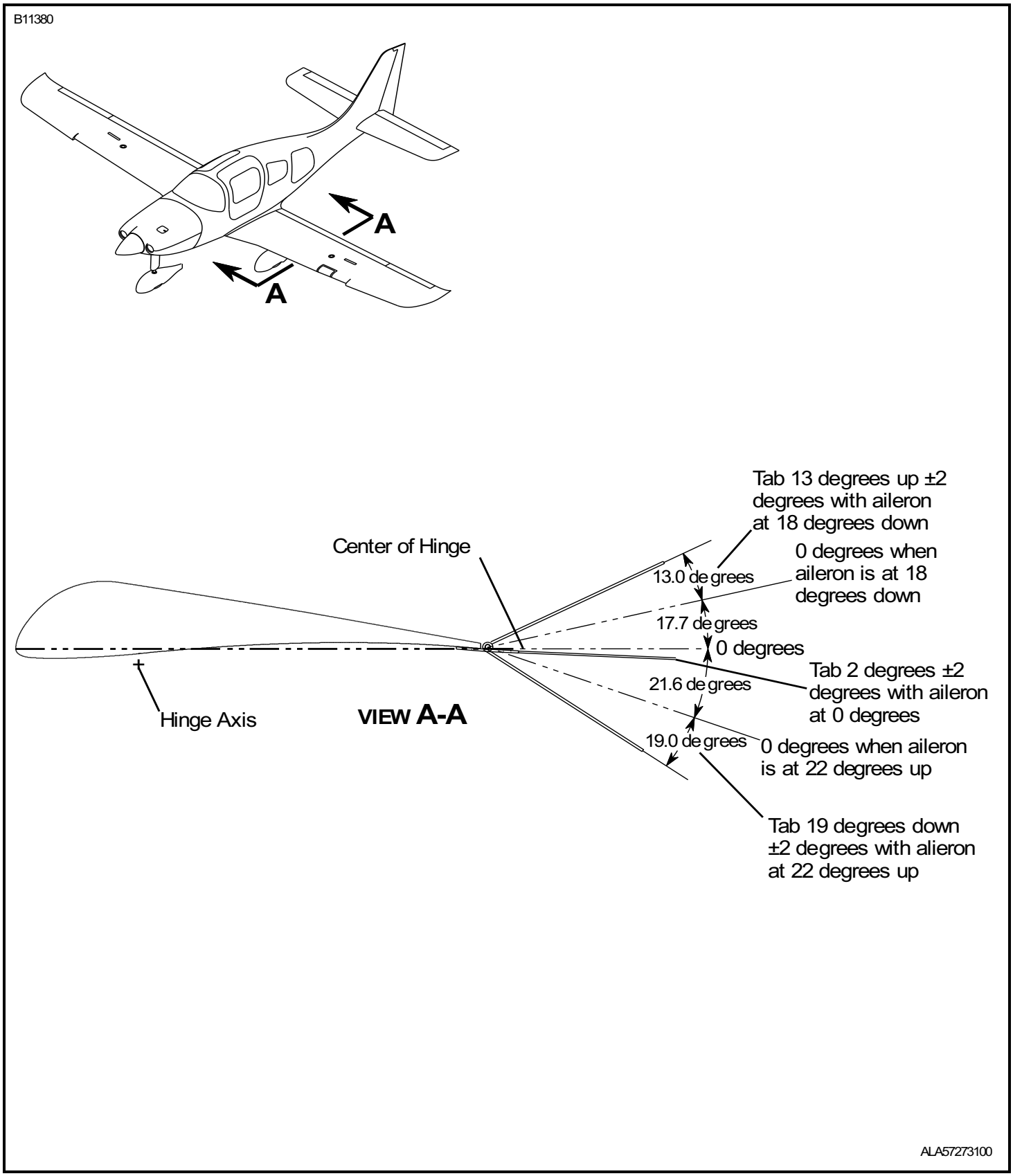
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Figure 203 : Sheet 1 : Aileron Trim System



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Figure 204 : Sheet 1 : Aileron Servo Tab Deflection Angles



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RUDDER CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Rudder System - General

A. The rudder attaches to the aft vertical closeout rib at three hinge points. Rudder cables attached to the rudder drive horn link the rudder to the rudder pedals. A fixed rudder trim tab is attached to the rudder at approximately mid span. The trim tab is ground adjustable for pilot comfort during cruise or takeoff power settings. A lead counterweight is used to static balance the rudder.

2. Rudder

A. Rudder Removal and Installation

(1) Detach the two rudder cables from the drive horn by removing the cotter pin from each nut, and then removing the bolt.

(2) Remove and discard the cotter pin from the lower hinge, and remove the castellated nut and washer.

NOTE: If the washer between the nut and the lower bracket is an NAS1149F0463P (approximately 0.500 inch diameter) then you must discard it and replace it with an MS15795-853 (approximately 0.688 inch diameter) washer.

(3) Remove the rudder by lifting it up and aft. Take care not to pull too far aft and break the metal braid that is still attached to the rudder leading edge.

(4) Once the rudder is removed, detach the metal braid by removing the attachment screw.

(5) Install the rudder in the reverse order. A maximum of one NAS1149F0432P washer may be used to ensure there is a minimum clearance of 0.062 to 0.125 inches between the rudder actuator and the rudder attachment assembly (refer to Figure 201).

(6) Install the MS15795-853 washer, castellated nut and cotter pin.

(7) Recheck all attachments for security and safety.

B. Upper Rudder Hinge Removal and Installation

NOTE: The Instructions are the same for both the middle and upper hinge.

(1) Remove the Upper Rudder Hinges. Refer to Figure 201.

(a) Remove the rudder. Refer to Rudder Removal and Installation.

(b) Remove the bolts and washers that attach the hinge to the vertical stabilizer.

(c) Remove the hinge from the airplane.

(2) Install the Upper Rudder Hinges. Refer to Figure 201.

(a) Put the hinge in place on the airplane.

(b) Attach the hinge with bolts and washers.

(c) Torque the bolts to 33 inch-pounds, +3 or -3 inch-pounds (3.73 N-m, +0.34 or -0.34 N-m).

(d) Install the rudder. Refer to Rudder Removal and Installation.

C. Lower Rudder Hinge Removal and Installation

(1) Remove the Lower Rudder Hinge. Refer to Figure 201.

(a) Remove the rudder. Refer to Rudder Removal and Installation.

(b) Remove the screws and bolts that attach the hinge to the vertical stabilizer.

(c) Remove the hinge from the airplane.

(2) Install the Lower Rudder Hinge. Refer to Figure 201.

(a) Put the hinge in place on the airplane.

(b) Attach the hinge with bolts, screws, and washers.

(c) Torque the bolts and screws to 33 inch-pounds, +3 or -3 inch-pounds (3.73 N-m, +0.34 or -0.34 N-m).

(d) Install the rudder. Refer to Rudder Removal and Installation.

D. **Rigging and Adjustment** The rudder deflections are preset at the factory and should not need any further adjustment. If the rudder cable is replaced, the deflections will need to be rechecked and adjusted as necessary. Should this need to be done, use the rigging procedure as outlined below:

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NOTE: Rigging of the rudder will be easier with at least two individuals: one to hold the template in the proper position and one to make the required adjustments.

CAUTION: If the inner rudder cable tube is removed, make sure that when the tube is installed again the barbed insert is installed in the forward end of the tube. If the inner and outer tube clamp bolt and nut were loosened, the nut must be tightened to a torque of 20-25 inch pounds. Refer to Figure 202.

(1) Remove the locking clips from the left and right turnbuckles at the forward end of the rudder cable near the rudder pedals. Remove the locking clips from the rudder pedal interconnect turnbuckle (located behind the instrument panel near the firewall), and loosen the interconnect cable.

(2) Ensure that the rudder cables are not attached to the rudder pedal crossover arms.

(3) Thread the left rudder cable, right rudder cable and rudder pedal interconnect turnbuckles so that the clevis threads are just visible. Apply three turns to each clevis to put the threads inside the turnbuckle.

(4) Clamp the elevator in the neutral position such that the trailing edge is in alignment with the trailing edge of the horizontal stabilizer.

(5) Adjust the rudder to the neutral position by moving it left and right until it is centered about the vertical tail.

(6) Position the rudder deflection template, part number TA53550000, on the rudder such that the radius portion is located at the bottom edge of the rudder trim tab and the middle portion is aligned with the leading edge of the rudder. This action will position the template perpendicular to the hinge axis of the rudder.

(7) Move the rudder to the left and right until it contacts the rudder stop.

(8) Confirm that the deflection limits are 17±1 degrees. If the deflection limits are greater than the allowed tolerance, shim the rudder stop with thin washers as necessary to achieve the correct deflection. If the limits are less than the allowed tolerance, grind on the angled face of the stop until the desired angle is achieved.

NOTE: If grinding on the rudder stop is required to achieve the required rudder deflection, apply zinc oxide primer or a similar corrosion protecting treatment to the exposed metal.

(9) Attach the rudder cables to the rudder pedal crossover rods.

(10) Align the rudder to the 17±1 degree position on the right and push the pilot's side right rudder pedal to its full forward position.

(11) Two methods, as follows, may be used for attaching the rudder cables to the rudder drive horn. The first method employs a cable thimble, compression sleeve, and shackle. The second method utilizes a fork end cable terminal. Whichever method is used, contact Cessna for parts and instructions.

(a) Cable Thimble, Compression Sleeve and Shackle (Refer to Figure 203)

1 At the aft of the plane pull the left side rudder cable taut and mark the cable, with an indelible marker, where it intersects with the forward edge of the connection hole in the rudder drive horn.

2 Using a cable thimble, clevis, and temporary clamp form a loop at the end of the rudder cable with the indelible mark at the apex of the loop. DO NOT remove excess cable.

3 Align the rudder to the 17±1 degree position on the left and push the pilot's side left rudder pedal to its full forward position.

4 At the aft of the plane pull the right side rudder cable taut and mark the cable, with an indelible marker, where it intersects with the forward edge of the connection hole in the rudder drive horn.

5 Using a cable thimble, clevis, and temporary clamp form a loop at the end of the rudder cable with the indelible mark at the apex of the loop. DO NOT remove excess cable.

6 Temporarily connect the rudder drive cables to the rudder drive horn. Insert excess cable into the fuselage. Ensure that the temporary clamp does not catch on the fuselage as the rudder deflections are adjusted.

7 Align the rudder to the 11.5±0.5 degree position on the template.

8 On the pilot's side, pull the left rudder pedal to its full aft position

9 Engage the rudder limiter, and depress the left rudder pedal until the connecting arm contacts the rudder limiter cam.

10 Place an adjustable rod or similar device between the left rudder pedal and the vertical face of the floor directly beneath the pilot's seat. Adjust the rod so as to maintain enough pressure on the left rudder pedal to keep the rudder limiter cam in the engaged position.

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11 Adjust the left rudder cable turnbuckle until the rudder cable is tight.

12 Unblock the rudder pedal and the rudder. Set the rudder to 0 degrees deflection.

13 Adjust the right side rudder cable turnbuckle until the right rudder pedal is aligned with the left rudder pedal.

14 With the rudder at 0 degrees deflection, adjust the rudder pedal interconnect turnbuckle until it takes a weight of 5 lbs. to deflect the cable 1.00±0.25 inches. Refer to Section 22-01-00 for cable tension if autopilot is installed.

15 Recheck all deflections, and repeat the rigging procedure as necessary until the proper deflections are achieved.

16 Without disturbing the turnbuckles, disconnect the rudder cables from the rudder pedal crossover rods.

17 Disconnect the rudder cables from the rudder drive horn.

18 Using an indelible marker, make three marks on the rudder cable and cable thimble to facilitate alignment of the cable onto the thimble after the temporary clamp has been removed.

19 Remove the temporary clamp.

20 Insert the cable end into a compression fitting and form a loop.

21 Align the cable onto the thimble using the three indelible marks as a guide.

22 Snug the compression fitting up against the thimble and compress the fitting.

23 Reattach the rudder cables to the rudder horn and to the pedal crossover rods and recheck all deflections. Repeat the rigging procedure as necessary until the proper deflections are achieved.

24 Cut off excess rudder cable at the compression fitting.

(b) Fork End Cable Terminal (Refer to Figure 203)

1 Temporarily attach fork end cable terminals to the left and right side of the rudder drive horn.

2 At the aft of the plane insert the rudder cable into the left fork end cable terminal and pull it taut. Mark the cable, with an indelible marker, where it intersects with the inner edge of the terminal sleeve just behind the fork of the terminal.

3 Align the rudder to the 17±1 degree position on the left and push the pilot's side left rudder pedal to its full forward position.

4 At the aft of the plane insert the rudder cable into the right fork end cable terminal and pull it taut. Mark the cable, with an indelible marker, where it intersects with the inner edge of the terminal sleeve just behind the fork of the terminal.

5 Disconnect the rudder cables from the pedal crossover rods.

6 Remove the terminals and cut the rudder cables 1/8 inch behind the mark toward the fore of the aircraft.

7 Insert the rudder cable into the left cable terminal so that the cable is 1/8 inch inside the terminal sleeve at the fork end.

8 Swage the fork end cable terminal.

NOTE: The swaging tool required to swage the fork end cable terminal is a special tool not available in most repair shops. Contact Cessna for instructions.

9 Insert the rudder cable into the right cable terminal so that the cable is 1/8 inch inside the terminal sleeve at the fork end.

10 Swage the fork end cable terminal.

11 Reattach the rudder cables to the rudder horn and to the pedal crossover rods.

12 Align the rudder to the 11.5±0.5 degree position on the template.

13 On the pilot's side, pull the left rudder pedal to its full aft position.

14 Engage the rudder limiter, and depress the left rudder pedal until the connecting arm contacts the rudder limiter cam.

15 Place an adjustable rod or similar device between the left rudder pedal and the vertical face of the floor directly beneath the pilot's seat. Adjust the rod so as to maintain enough pressure on the left rudder pedal to keep the rudder limiter cam in the engaged position.

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16 Adjust the left rudder cable turnbuckle until the rudder cable is tight.

17 Unblock the rudder pedal and the rudder. Set the rudder to 0 degrees deflection.

18 Adjust the right side rudder cable turnbuckle until the right rudder pedal is aligned with the left rudder pedal.

19 With the rudder at 0 degrees deflection, adjust the rudder pedal interconnect turnbuckle until it takes a weight of 5 lbs. to deflect the cable 1.0±0.25 inches. Refer to Section 22-01-00 for cable tension if autopilot is installed.

20 Recheck all deflections, and repeat the rigging procedure as necessary until the proper deflections are achieved.

21 Reinstall the locking clips into all turnbuckles.

22 Cycle the rudder from left to right to ensure that there is no binding or excessive friction in the system.

(c) Service and Maintenance

1 The rudder does not require any service or maintenance. However, due to its composite construction, the rudder should be visually inspected annually for cracking at all hinge attachment locations and the trim tab attachment location. A tap test should be conducted on the rudder at the drive horn location to aid in detection of any latent damage or disbonds along the lower rudder rib. The rudder does not have any access panels, therefore, the counterweight needs to be removed in order to inspect the internal structure. To remove the rudder counterweight, remove the attachment screws, and slide the counterweight straight off. The rudder may also be inspected via the holes at the middle and top rudder hinge pin locations. Common inspection tools that may be used or needed are mirrors, bore scopes, etc. The rudder should be removed and inspected for wear or corrosion of the hinge pins. If wear or corrosion results in a pin diameter of less than 0.238 inch the hinge pin must be replaced. You can apply lubricant or a corrosion preventative compound to the hinge pins on assembly to help minimize corrosion but the spherical bearings used in the attachment brackets on the vertical closeout rib do not require any maintenance and should not be lubricated. The hinge brackets should be inspected for corrosion or other damage and replaced if necessary.

(d) **Rudder Cable Removal** The rudder cable should be inspected thoroughly and replaced if any excessive wear or damage is observed. To remove the rudder cable use the following steps.

1 Contact the manufacturer for a replacement.

2 Detach the rudder cable from the rudder drive horn per the rudder installation and removal instructions in this section.

3 Cut the rudder cable to remove the turnbuckle, and slide the cable aft through the aft end of the fuselage.

4 Install the new rudder cable by inserting the end without turnbuckle into the fore end of the rudder cable tube and sliding it aft to exit at the rudder drive horn.

5 Follow the procedures in the Adjustment section for connection, rigging, and adjustment. Ensure that the requirements listed in Table 1 are met.

3. Rudder Trim Tab

A. **Removal and Installation** Should the trim tab need to be replaced, carefully drill out the five CCR264CS-4-02 attachment rivets to remove the trim tab. Obtain a new trim tab, part number LA53550001, from the manufacturer and reinstall with five new rivets of the same type and size. Take care when drilling out rivets so as not to enlarge holes.

CAUTION: Due to the carbon fiber composite construction of the rudder, do not use aluminum rivets. The new rivets must be of a corrosion resistant material (i.e. stainless steel) in order to prevent galvanic corrosion.

B. **Adjustment** - The rudder trim tab is a flat aluminum plate that is riveted to the trailing edge of the rudder at approximately mid-span. The trim tab is attached with five CCR264CS-4-02 countersunk rivets. The trim tab may be adjusted for pilot comfort during cruise or takeoff power settings.The trim tab is not adjustable during flight and must be adjusted while on the ground. To adjust the rudder trim tab use the following instructions.

(1) Clamp two pieces of wood or other rigid material to either side of the trim tab. The wood blocks should be equal in length to the trim tab.

(2) Align the edges of the wood block with the slots in the trim tab.

(3) Once the wood blocks are positioned correctly, simply bend the trim tab in the desired direction. Do not to exceed the

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deflection angles described in Section 27-00-00, Table 1.

C. **Service and Maintenance** The rudder trim tab should be inspected for wear and cracking of the slots that run the length of the tab (bend line). If excessive wear or cracking is observed, the trim tab should be replaced.

4. Rudder Pedal Bushing

A. Rudder Pedal Bushing Removal.

(1) Remove the forward seat bottom cushions. Refer to the Chapter 25, Airplane Seats - Maintenance Practices.

(2) Remove the turnbuckle clips and disconnect the rudder interconnect cable at the turnbuckle.

NOTE: The rudder interconnect cable is located down and forward of the instrument panel.

(3) Remove the left and right pulley holder from the firewall bracket by removing the attaching bolt.

(4) Remove the cotter pins, nuts and bushings from the AN3-26 bolts.

NOTE: Discard the cotter pins.

B. Rudder Pedal Bushing Installation.

(1) Put one bushing into each end of each pulley holder (four total)

NOTE: For Airplane LC42-550FG -42061 and 42062 incorporating SB-04-016A, install bushing P/N 03FB04.

(2) Install the pulley holder to the firewall bracket.

(3) Install the nuts and new cotter pins P/N MS24655-151 on the bolts.

(4) Connect the rudder interconnect cable.

(5) Put the rudder at 0° deflection.

(6) Adjust the rudder interconnect turnbuckle until it takes a weight of 5.00 lbs. (2.27 kg) to deflect the cable 1.00 inch + 0.25 or - 0.25 inch (25.40 mm +6.35 or - 6.35 mm).

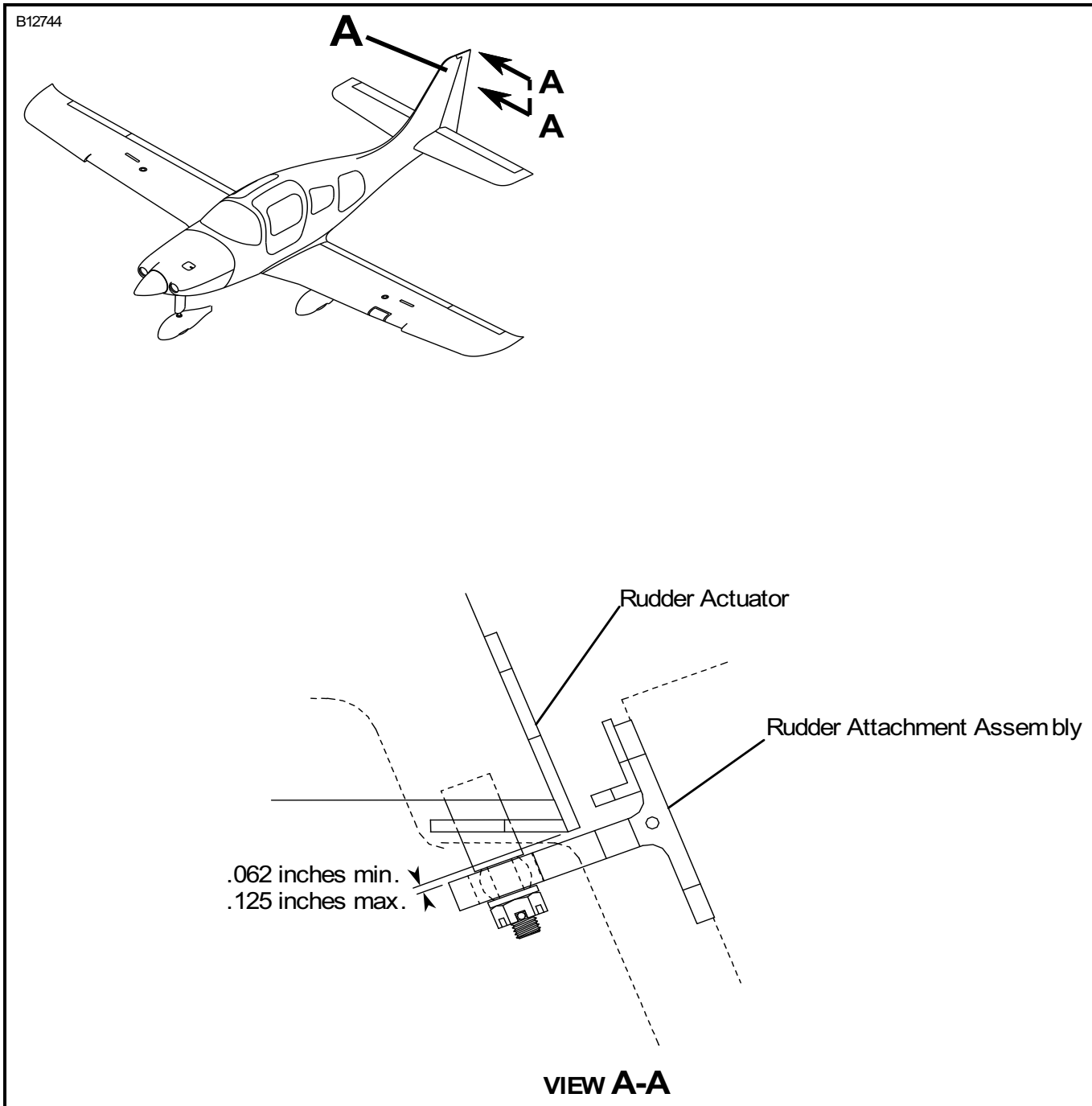
(7) After adjusting the turnbuckle, install the turnbuckle clips.

(8) Install the seat bottom cushions.

(9) Do an operational check of the rudder controls to make sure there is freedom of movement and full rudder deflection.

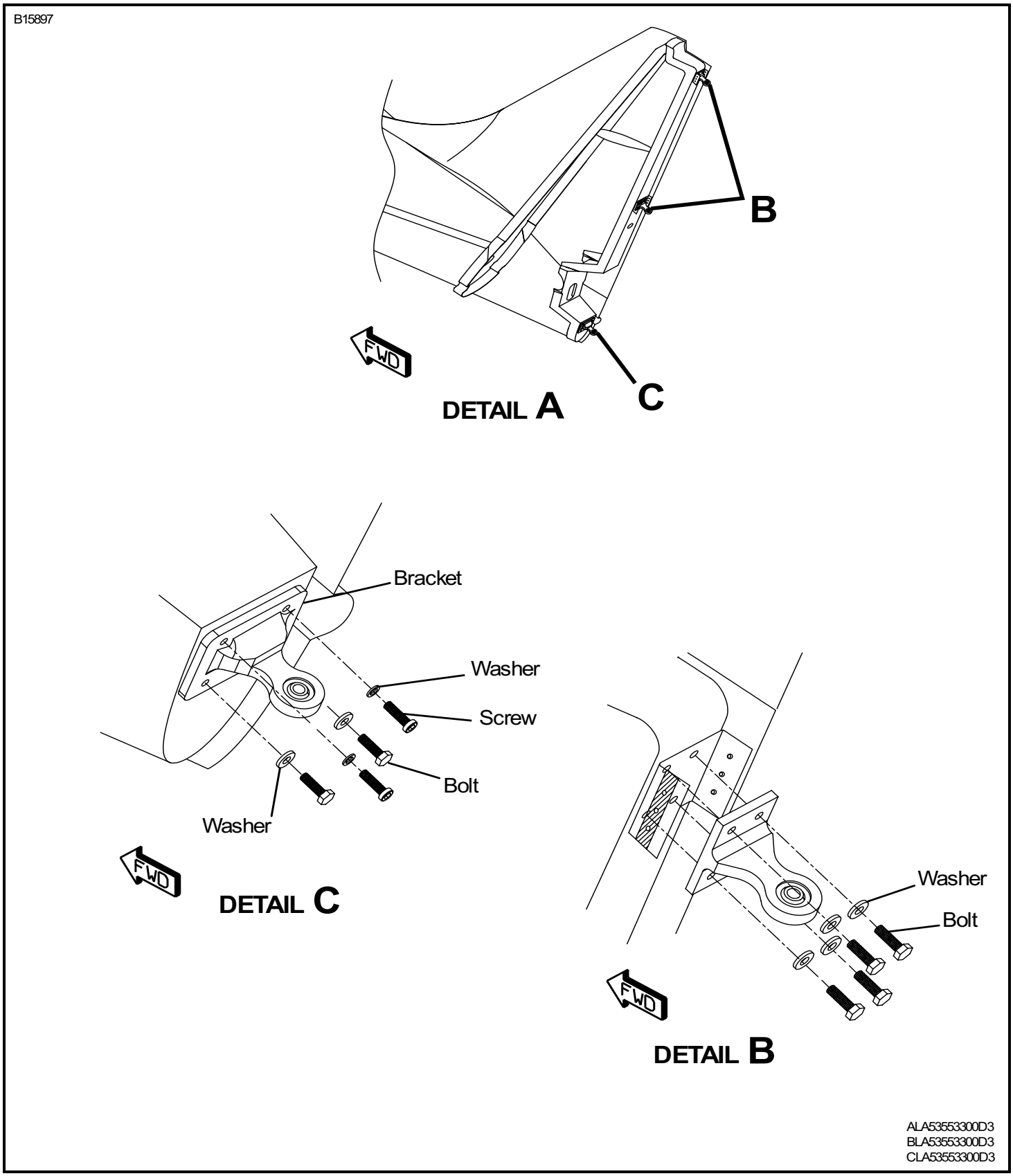
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Figure 201 : Sheet 1 : Rudder Installation.



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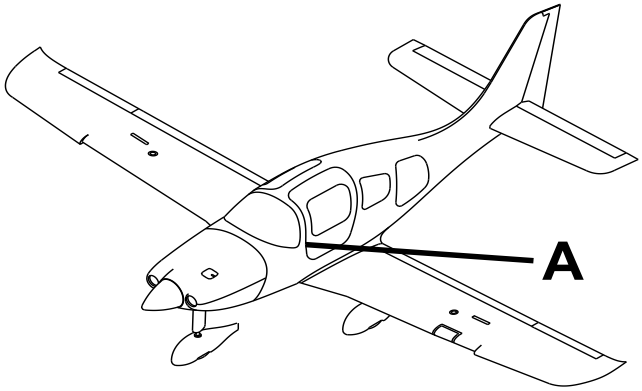
Figure 201 : Sheet 2 : Rudder Installation.

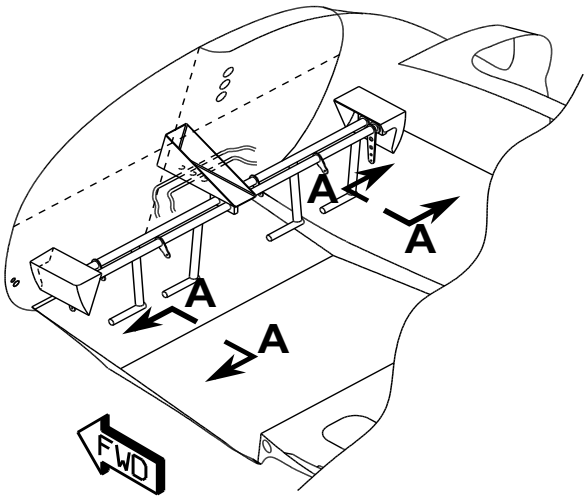


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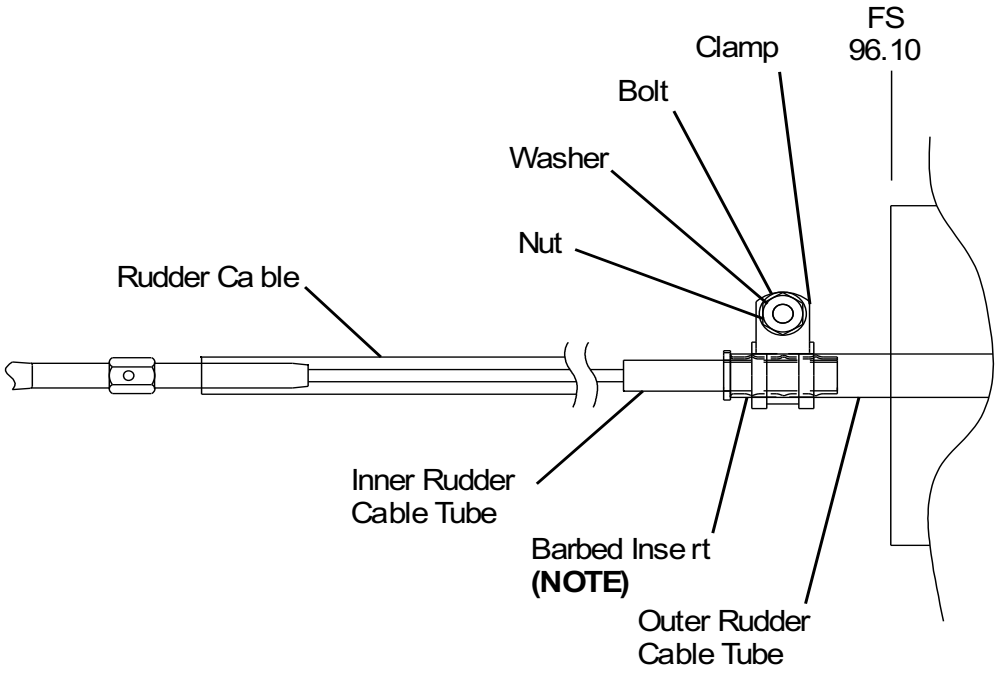
Figure 202 : Sheet 1 : Forward Rudder Cable Installation

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DETAIL A



VIEW A-A
Right Side Shown, Left Side Opposite

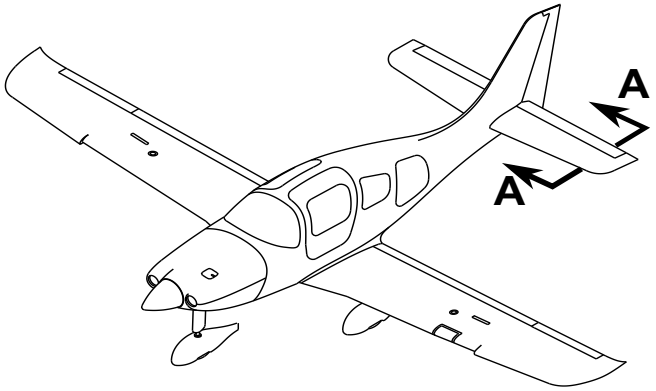
NOTE: Airplanes Incorporating SB10-27-03

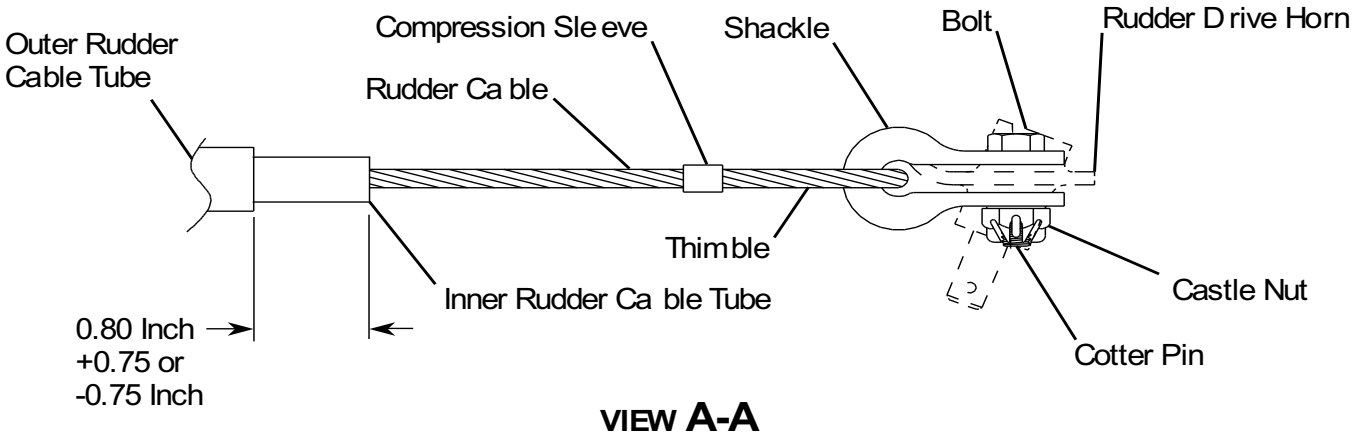
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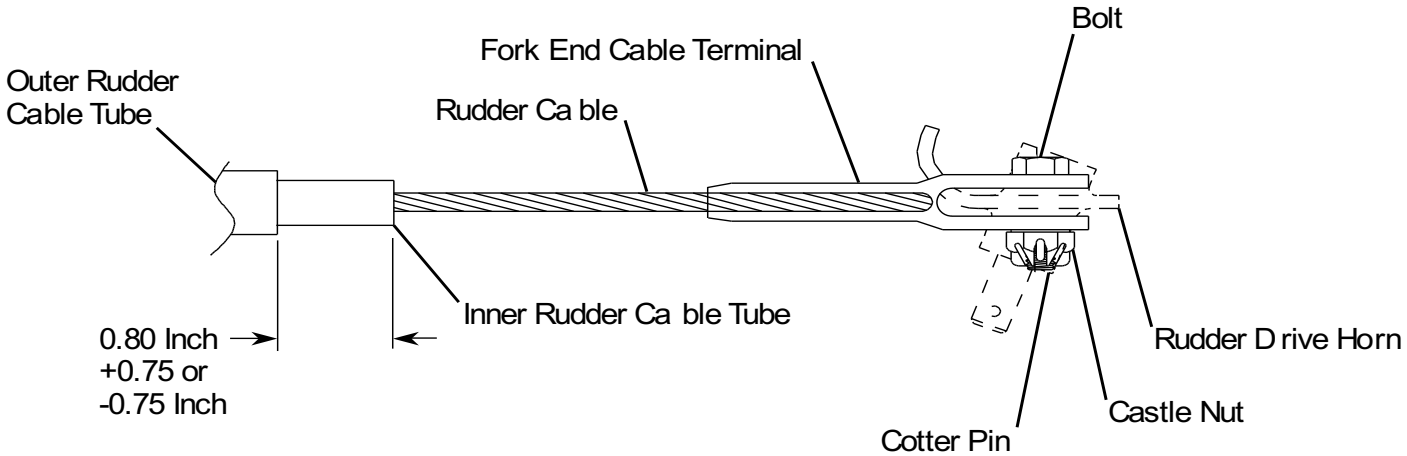
Figure 203 : Sheet 1 : Aft Rudder Cable Installation

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VIEW A-A
Cable Thimble, Compression Sleeve and Shackle
Left Side Shown, Right Side Opposite



VIEW A-A
Fork End Cable Terminal
Left Side Shown, Right Side Opposite

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ELEVATOR CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Elevator System - General

A. The elevator is of a carbon fiber composite construction and is attached to the horizontal stabilizer with two piano hinges. A series of push-pull tubes connects the elevator to the left and right control sticks. Simple rod end bearings are used throughout the elevator control system. Lead counterweights balance the elevator.

CAUTION: Due to its size and weight, removal of the elevator may require more than one individual.

2. Elevator

A. Elevator Removal and Installation

(1) Remove the rudder per the instructions in Section 27-20-00, Rudder Removal and Installation.

(2) Remove the cotter pin and retaining hardware from the aft elevator push rod and the elevator drive horn.

(3) Remove the 36 MS24694-C50 screws that attach the elevator hinge to the horizontal stabilizer.

(4) Remove the elevator by sliding it straight aft. Take care not to move it too far aft and break the wiring connector.

(5) Disconnect the wiring connector for the elevator trim servo and the grounding strap at the elevator horn, and remove the elevator completely from the aircraft.

(6) Install the elevator in the reverse order. Recheck all attachment hardware for security and safety. Torque all attachment screws 30 to 36 in.-lbs.

B. Elevator Rigging and Adjustment

(1) Remove the following interior panels per the instructions in Section 25-30-00: FLSPL, FLSPR, MLSPL, MLSPR, right and left door tread.

(2) Remove the baggage bulkhead access panel per the instructions in Section 53-30-00, Baggage Bulkhead Access Panel.

(3) With the elevator securely fastened to the horizontal stabilizer, clamp the elevator such that its trailing edge is aligned with the trailing edge of the horizontal stabilizer.

(4) Set the control stick to the vertical (0 degrees) position as shown in Figure 201.

(5) Loosen the jam nuts on the rod end bearings on all elevator push-pull tubes as shown in Figure 202.

(6) Place the elevator deflection template, part number TA55270000, on the horizontal stabilizer 76.5 inches inboard from the tip of the horizontal stabilizer and perpendicular to the hinge axis. This template has three slots that define the neutral, upper, and lower elevator deflection locations. The middle slot defines the neutral position, while the upper and lower slots define the upper and lower deflections limits, respectively. (Refer to Figure 203.)

(7) With the elevator in the neutral position (trailing edge in alignment with the middle slot on the template) rotate the elevator upwards until it just contacts the stop. Check that the trailing edge of the elevator is between the upper and lower edges of the uppermost slot on the template.

CAUTION: Do not press too hard on the elevator against the elevator up stop, as this will cause a false indication of the elevator deflection.

(8) Rotate the elevator to the down position, and allow the elevator to rest freely against the down stops. Check that the trailing edge is between the upper and lower edges of the lower slot on the template.

(9) Adjust the elevator up and down stops as necessary to align the trailing edge of the elevator between the upper and lower edges of the top and bottom slots in the template.

(a) To adjust the elevator up stop, loosen the jam nut, turn the socket head cap screw in the desired direction until the proper up deflection is met, then re-tighten the jam nut and torque 100 in.-lbs.

(b) To adjust the elevator down stops and increase the amount of down deflection, sand on the surface of the stop located in the counterweight pockets until the proper down deflection is achieved. If the down deflection is too much, contact the manufacturer for new elevator stops.

(10) Adjust the rod end bearings in the elevator push-pull tubes such that the following criteria are met:

(a) The elevator deflections per Section 27-00-00, Table 1, are achieved.

(b) The control stick is in a vertical (0 degrees) position when the elevator is at neutral (0 degrees) deflection. (Refer to Figure 201.)

(c) The elevator interconnect weldment has adequate clearance (0.125 inches minimum) with the adjacent aircraft

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structure.

(d) Control stick is free from interference with adjacent side interior panels and the instrument panel.

(e) All rod ends have a minimum 0.5-inch thread engagement (threads should be visible through inspection hole in ends of push-pull tubes).

(11) Ensure that the gap between the outboard end of the elevator and the horizontal stabilizer is 0.125 to 0.250 inches.

(12) Torque all AN316-6 jam nuts at control rod ends to 95 to 110 in.-lbs. and all AN316- 5 jam nuts to 70 to 85 in.-lbs.

(13) Cycle the control stick fore and aft to ensure that there is no binding in the system.

C. **Elevator Servicing and Maintenance** The elevator is attached to the horizontal stabilizer with two piano hinges. Each hinge should be inspected annually for excessive wear and cracks. The hinge may be lubricated with MIL- 7870 general purpose lubricating oil to help reduce wear and friction. The rod end bearings used throughout the elevator control system do not require maintenance other than cleaning during routine inspections to remove any accumulation of dust or grime. Liquid lubricants should not be used on the rod end bearings as this may cause the Teflon liner to swell creating excess friction in the bearing. Should the rod end need to be replaced, contact the manufacturer.

3. Elevator Trim Tab

A. **Elevator Trim Tab Removal and Installation**

(1) Gain access to the elevator trim motor by removing the access panel located on the lower surface of the elevator.

(2) Disconnect the trim motor drive rods from the trim tab drive horn by removing the cotter pins and clevis pins.

(3) Rotate the trim tab upwards to gain access to the screws attaching the trim tab hinge to the elevator.

(4) Remove the 11 AN526C632R screws attaching the hinge to the elevator.

(5) Reinstall the trim tab in the reverse order, and torque all screws 10 to 12 in.-lbs.

(6) Run the trim tab to the up and down deflection positions to ensure that there is no binding of the system.

B. **Elevator Trim Tab Rigging and Adjustment** The elevator trim tab deflections are preset at the factory and should not require any further adjustment. The trim tab deflections should be checked periodically to ensure that the system is functioning properly. If the trim tab deflections are checked and found to be outside the acceptable limits per Section 27-00-00, Table 1, contact the manufacturer for further instructions.

C. **Elevator Trim Tab Servicing and Maintenance** The trim tab is attached to the elevator with one piano hinge. The hinge should be visually inspected annually for excess wear and cracking and replaced as necessary. The hinge should be replaced if the hinge becomes worn to the point that the trim tab free play limits per Section 27-00-00, Table 1, are exceeded. The hinge may be lubricated with MIL-L-7870 general purpose lubricating oil. Should the nutplates located on the nutplate assembly, part number LA55273818, inside the elevator become worn or stripped, contact the manufacturer for a replacement.

4. Elevator Trim Tab Servo

A. **Elevator Trim Tab Servo Removal and Installation**

(1) Gain access to the elevator trim motor by removing the access panel located on the lower surface of the elevator.

(2) Disconnect the trim motor drive rods from the trim tab drive horn by removing the cotter pins and clevis pins.

(3) Remove the six AN526C632R10 screws that attach the trim motor hinge to the elevator hinge.

(4) Carefully remove the trim motor from within the elevator.

CAUTION: Take care when removing the trim motor so as not to damage the wiring and connectors located on the outside of the trim motor housing. Should any of the wiring or connectors become damaged or break, contact the manufacturer for a replacement.

(5) After removing the trim motor, disconnect the wiring harness.

(6) Reinstall the trim motor in the reverse order. Torque all screws 10 to 12 in.-lbs.

B. **Elevator Trim Tab Servo Rigging and Adjustment** The elevator trim motor is an integral part of the elevator trim tab. The trim tab deflections are preset at the factory, therefore, the trim tab and motor should not require any further adjustment unless it needs to be removed and replaced. If the trim tab deflections are checked and found to be outside the acceptable limits per Section 27-00-00, Table 1, the following procedure should be used.

(1) Using elevator trim template TA55273800 Rev B, measure out from the right side of the vertical 23.875 in. to BL 26.175. Carefully place the template on the elevator and over the trim tab. Have an assistant drive the elevator trim

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tab to the full down position (trim tab up) as indicated by the template.

NOTE: Actuation of the elevator trim system may have to be stopped before the 'DOWN' microswitch is activated. If the microswitch is activated before full down travel is achieved, back-off the screw on the potentiometer drive bracket so that the full throw of the trim tab can be accomplished.

(2) With the trim tab in the full down position (trim tab up), adjust the down screw on the potentiometer drive bracket (refer to Figure 204) so that the screw just closes the microswitch, deactivating the motor. Tighten the jam nut on the adjusting screw.

(3) Using trim tab template TA55273800 Rev B as a reference, have an assistant drive the elevator trim system to the full up position (trim tab down).

NOTE: Actuation of the elevator trim system may have to be stopped before the 'UP' microswitch is activated. If the microswitch is activated before full up travel is achieved, back-off the up-screw on the potentiometer drive bracket so that the trim tab can be moved to match the required template markings.

(4) With the trim tab in the full up position (trim tab down) adjust the up screw on the potentiometer drive bracket so that the screw just closes the microswitch deactivating the motor. Tighten the jam nut on the adjusting screw.

(5) Drive the elevator trim system to the full down position using the control stick trim button while an assistant watches the actuator and trim tab linkage for clearances and smooth operation.

CAUTION: If the microswitch is severely misadjusted, damage to the aircraft and/or components may occur while trimming to either stop. Watch carefully during this phase of operation for clearances and any indication of binding.

(6) Using template TA55273800 Rev B, check position of the trim tab. Re-adjust the down screw on the potentiometer drive bracket above and re-tighten the jam nut on the adjusting screw, if necessary. Check trim panel position indications.

(7) Drive the elevator trim system to the full up position while an assistant checks actuator and trim tab linkage for clearances and smooth operation.

CAUTION: If the microswitch is severely misadjusted, damage to the aircraft and/or components may occur while trimming to either stop. Watch carefully during this phase of operation for clearances and any indication of binding.

(8) Using template TA55273800 Rev B, check position of trim tab. Re-adjust the down screw on the potentiometer drive bracket above and re-tighten the jam nut on the adjusting screw, if necessary. Check trim panel position indication.

(9) Cycle the elevator trim system through full deflections, checking for security, proper travel, and indications.

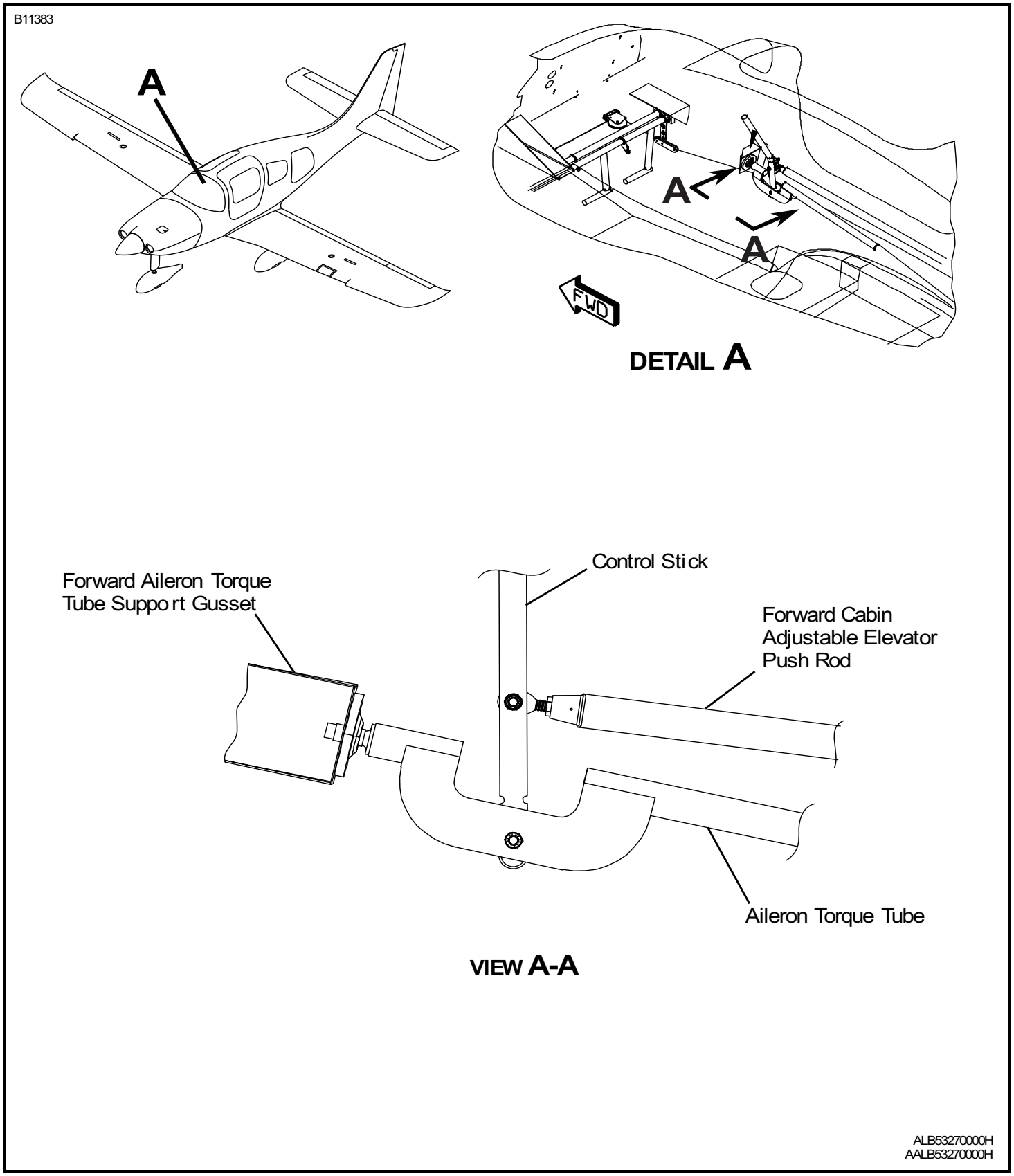
NOTE: When the trim tab is in the neutral position (faired evenly with elevator), the trim panel should indicate one green light above center.

(10) Install the elevator trim actuator cover panel.

C. **Elevator Trim Tab Servo Servicing and Maintenance** The elevator trim motor should be removed for inspection and service every 3000 hours. The trim motor drive rod gears should be inspected for wear and lubricated with heavy-duty gear grease. The worm gears attached to the drive motor shaft do not need to be lubricated but should be inspected for wear. If any of the gears are worn, they should be replaced. Replacements may be obtained from the manufacturer.

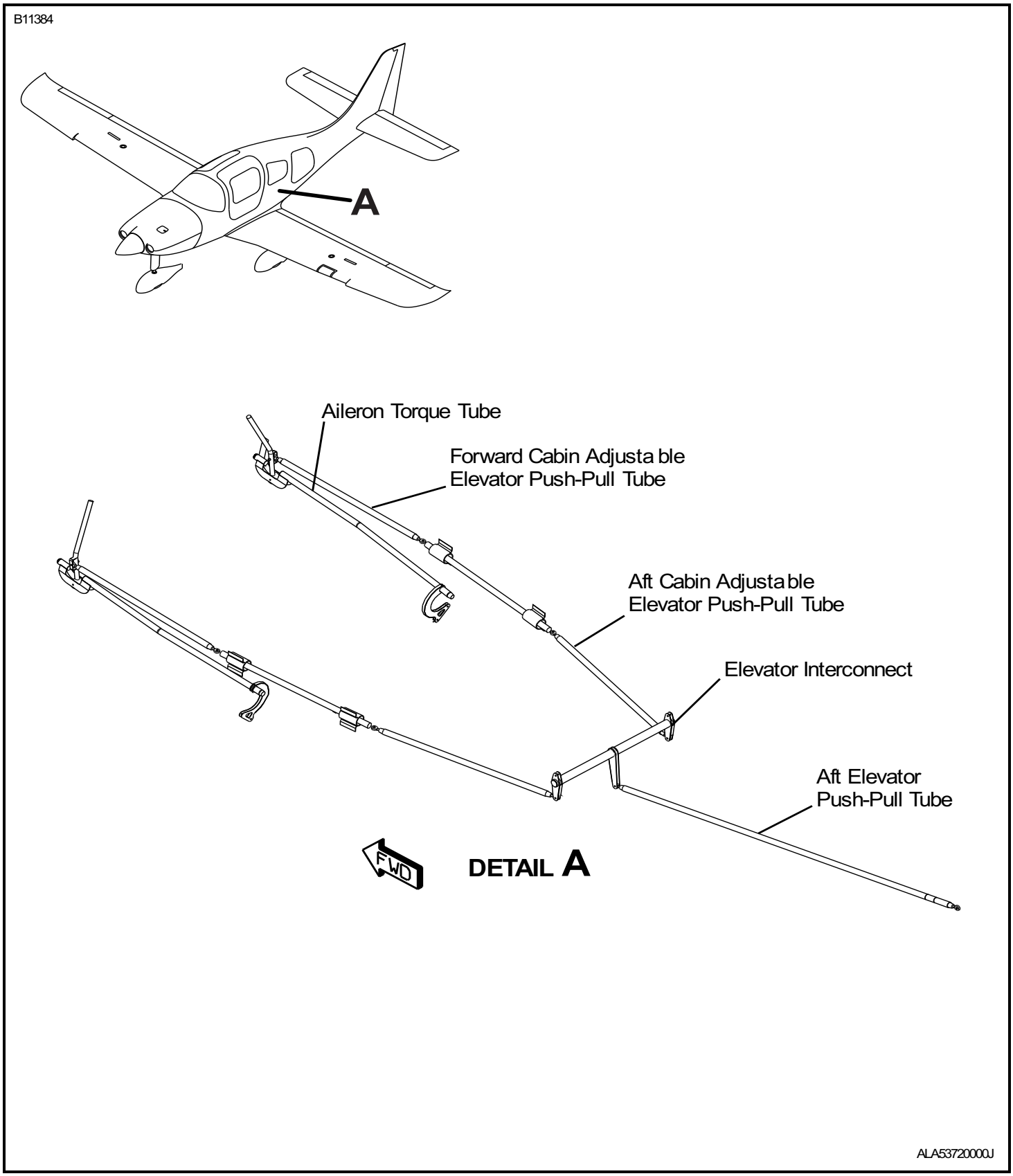
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Figure 201 : Sheet 1 : Control Stick Neutral Position



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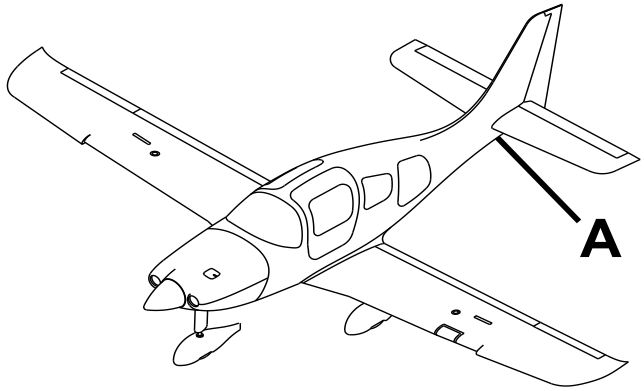
Figure 202 : Sheet 1 : Fuselage Control System Layout

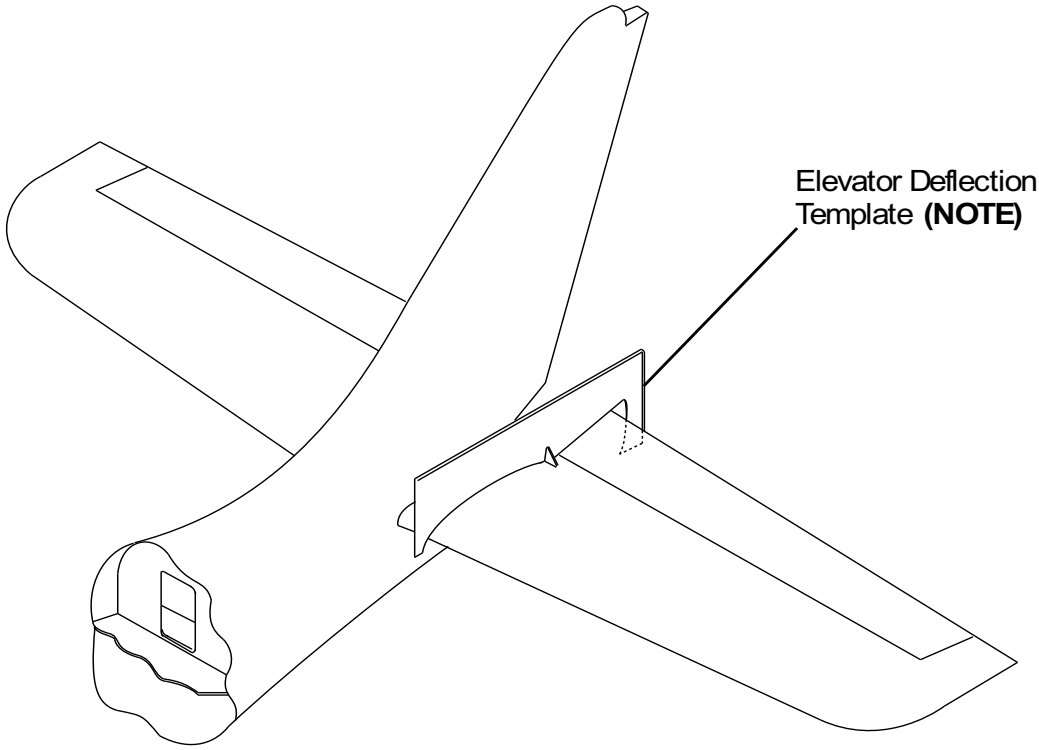


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Figure 203 : Sheet 1 : Elevator Deflection Template

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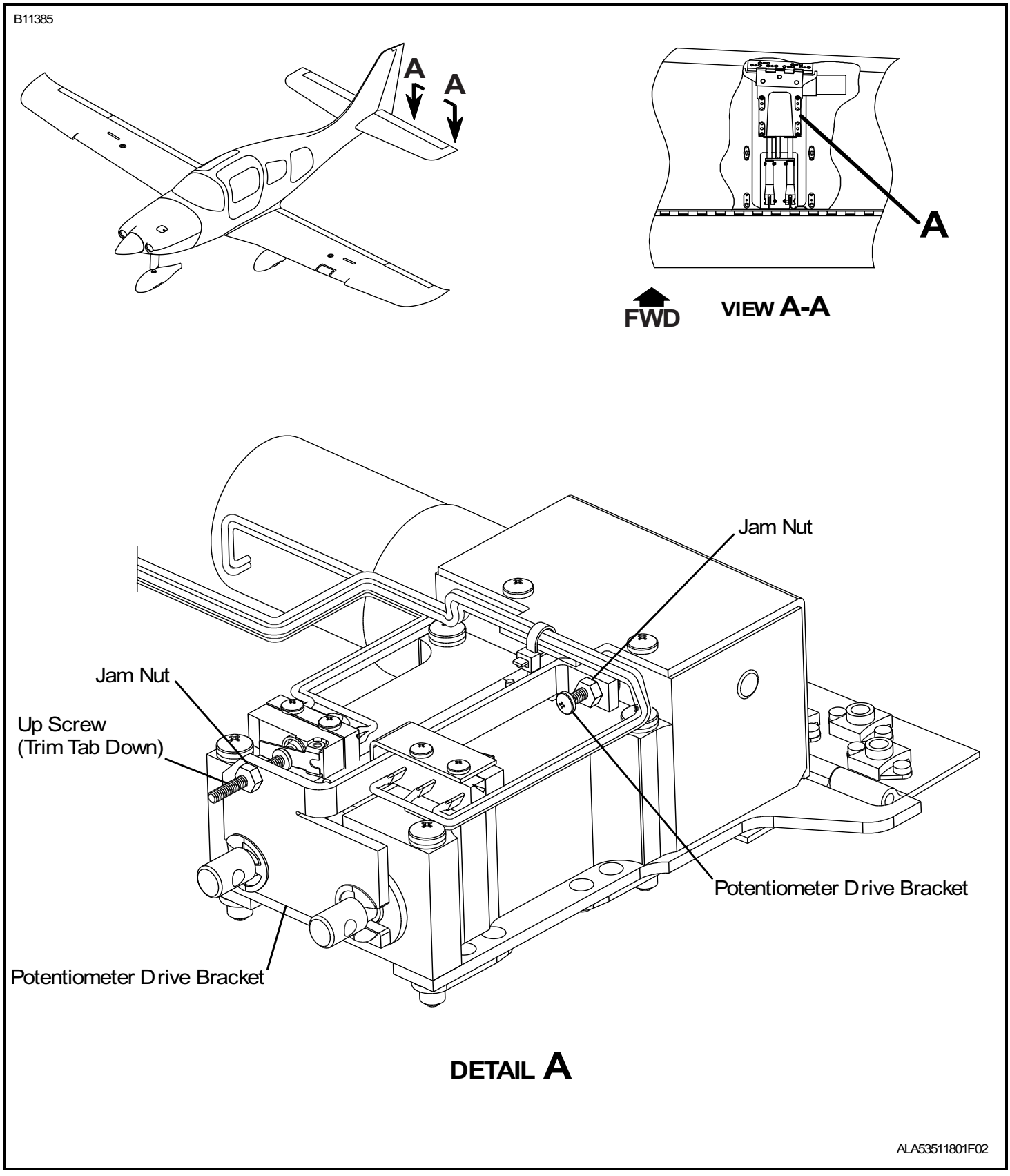


DETAIL A

NOTE: Located at 76.5 inches from the outboard ti p.

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Figure 204 : Sheet 1 : Elevator Trim Actuator



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FLAP CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Flap System - General

A. The flap system is located in the wing, aft of the rear spar, with the exception of the electrical system linking the flap switch to the electric flap actuator. The system uses four control rods with rod ends mounted on each threaded control rod. The rod ends use a Teflon liner which requires no lubrication. Refer to Figure 201.

B. Primary flap stops are electric provided by the limit switch assembly. The flap system secondary stops are mechanical with up stops provided by the flap hinges and the flap central bellcrank assembly. Down stops are provided by the flap actuator. Refer to Chapter 33 for additional information regarding the electrical connections and flap switch.

2. Flaps

A. Flap Removal and Installation

(1) Lower flaps to the landing position.

(2) Remove the cotter pin, castellated nut, bolt, and washers from the flap drive control rod to the flap. Note that small washers may be installed between the flap drive hinge and the control rod end. These washers are used to increase the misalignment capability of the rod ends, and each aircraft may have a different number of washers, or none at all.

(3) Remove the cotter pins, castellated nuts, washers, and bolts from the flap hinge pivots.

(4) Remove the flap by pulling the flap aft and away from the wing.

(5) Reinstall in the reverse order. When installing a new flap or reinstalling an existing flap, ensure positive clearance between the flap upper surface and the flap cove. Also ensure that all four flap hinges align with no binding. Flap position may be adjusted, if necessary, by removing or installing shims to the flap hinges as follows:

(a) No more than 3 shims (P/N LA57265125) are allowed per hinge element of each hinge.

(b) When removing the hinge, note the location and type of all fasteners and washers.

(c) Due to the addition or removal of shims, replace any bolts or screws as required so that their length maintains two threads minimum projection (including the chamfer on the end of the bolt) through the nut, nutplate, or rivnut. The bolt must also have sufficient threads to prevent bottoming out before the part is secured.

(d) Dry torque 10-32 fasteners 28-34 in-lbs.

(e) Dry torque 6-32 fasteners 6-11 in-lbs.

(6) Torque all castellated nuts 72 to 84 in.-lbs., and install cotter pins.

B. **Flap Rigging and Adjustment** Once the flap system is properly rigged, only minor adjustments are required unless components such as control rods, actuator, or limit switches are replaced. It is important to start the rigging with the flap actuator and the flap central bellcrank. Refer to Figure 202. To rig the flap system, use the following steps:

(1) With the flap control rods disconnected from the outboard bellcranks, extend the actuator (flap switch up/cruise position).

(2) By adjusting the limit switch rod (refer to Figure 204) and the actuator rod end if necessary, adjust the flap up position so a 0.063-inch spacer fits between the flap central bellcrank and the web on the central bellcrank bracket as shown in Figure 203. Adjustments can easily be made with the actuator retracted (landing position). Torque the limit switch jam nuts 72 to 84 in.-lbs. and the flap actuator rod end jam nut 30 to 36 ft.-lbs. A minimum of 1-1/2 diameter thread engagement is required on the actuator rod end.

(3) Pin the outboard bellcranks with a 0.25 in. diameter bolt or equivalent, and adjust the flap control rods with the actuator set in the up position (Refer to Figure 204.)

(4) Adjust the left and right control rod assemblies to length between the flap central bellcrank in the up position and the pinned outboard bellcranks. After adjusting the control rod assemblies, torque the jam nuts 140 to 180 in.-lbs., and verify that the minimum thread engagement is achieved by checking that the threads extend past the inspection hole in the control rods. Torque bolts through the bellcrank and control rods 72 to 84 in.-lbs., and install the cotter pin in the castellated nut. (Refer to Figure 204.) Note the use of washers between the bellcrank and both sides of the rod end. If replacement of these washers is necessary, they may be replaced with either AN960-416 washers.

(5) After adjusting the left and right control rod assemblies, adjust the final drive length to achieve a 0±1-degree deflection with the flaps in the up position. After adjusting the correct length, torque the jam nuts 140 to 180 in.-lbs., and verify that the minimum thread engagement is achieved by checking that the threads extend past the inspection hole in the control rods (Refer to Figure 205). Torque bolts through the bellcrank and control rods 72 to 84 in.-lbs., and install the cotter pin in the castellated nut. Note the use of washers between the bellcrank and both sides of the rod

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end. If replacement of these washers is necessary, they may be replaced with AN960-416 washers.

(6) Using the flap/aileron template, part number TA57270000, lower the flaps to the takeoff position, and adjust the takeoff micro-switches to achieve 12±1-degree deflection on both flaps. With the extension micro-switch adjusted, move the retraction micro-switch as close as possible to the extension switch without interference. Note when adjusting the switches, always check deflection when the flaps are extended from the up position to the takeoff position.

(7) Lower the flaps to the landing position, and adjust the landing micro-switch to achieve 40±1-degree deflection on both flaps using the rigging template.

(8) Ensure that the gap at the inboard edge of the aileron between the flap and the aileron (when the flaps are retracted) is 0.125 to 0.375 in. Ensure positive clearance when the flaps are extended.

(9) Actuate the flap system, and check for binding or rubbing of components with the flap system. If problems are noted, re-check the rigging.

C. **Flap Servicing and Maintenance** The flaps do not require any service or maintenance. The flaps should be visually inspected annually for cracking at all hinge attachment locations. A tap test should be conducted on the flap at the inboard and outboard end caps to aid in detection of latent damage or disbonds. Liquid lubricants should not be used on the rod end bearings as this may cause the Teflon liner to swell creating excess friction in the bearing. Should the rod end need to be replaced, contact the manufacturer.

3. Flap Actuator

A. The flap system is actuated by an electric linear actuator mounted to a bracket on the aft spar in the belly-section of the aircraft wing. This actuator assembly includes a motor, clutch, ballscrew, and limit switch assembly. The limit switch assembly provides the electronic flap stops for the three positions: cruise, takeoff, and landing. Refer to Figure 206.

(1) **Flap Actuator Removal and Installation**

(a) Flaps must be in the cruise or landing position to remove the actuator. It is recommended to place the flaps in the cruise (up) position.

(b) Access the flap actuator through the flap and aileron pushrod access panel.

(c) Disconnect the power wire bundle to the actuator assembly. (Refer to Figure 207.)

(d) Tape or block the flaps to prevent them from moving when the flap actuator is disconnected from the central flap bellcrank.

(e) Through the holes in the flap central bellcrank bracket, disconnect the flap actuator from the bellcrank as shown in Figure 207.

(f) Remove the bolt holding the actuator to the bracket. Note the spacers and washers as shown in Figure 208 and Figure 209.

(g) Remove the actuator from the airplane.

(h) Reinstall in the reverse order.

(i) If the limit switches have been moved, a recheck of the flap rigging is required and adjustments required as necessary.

(j) Torque all 0.250 in. diameter bolts 72 to 84 in.-lbs. and all 0.375 in. diameter bolts 240 to 260 in.-lbs., and install the cotter pin.

(2) **Flap Actuator Maintenance**

(a) During normal inspections and maintenance, there should be checks for binding, fretting, chafed wires, and slop in the actuator mounting. Check the groove for the limit switch assembly and the limit switch rod. (Refer to Figure 204.)

(b) If there is excess play in the actuator mounting, inspection and possible replacement of the spacers is required. To replace the spacers, follow the instructions for actuator removal and installation. Install new spacers, bolts, and washers as necessary.

(c) Reinstall in the reverse order.

(d) If the limit switches have not been moved, a recheck of the flap rigging is required and adjustments required as necessary.

(e) Torque all 0.250 in. diameter bolts 72 to 84 in.-lbs. and all 0.375 in. diameter bolts 240 to 260 in.-lbs., and install

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the cotter pin.

(f) If problems are found with the micro-switches, replacement is required. Contact the manufacturer for replacement micro-switches. Replace the micro-switch by removing wires and unscrewing the micro-switch. Replacement is a direct reversal of the removal. R-rigging is required for the replaced micro-switch.

(g) If the internal clutch requires replacement, contact the manufacturer for a replacement actuator.

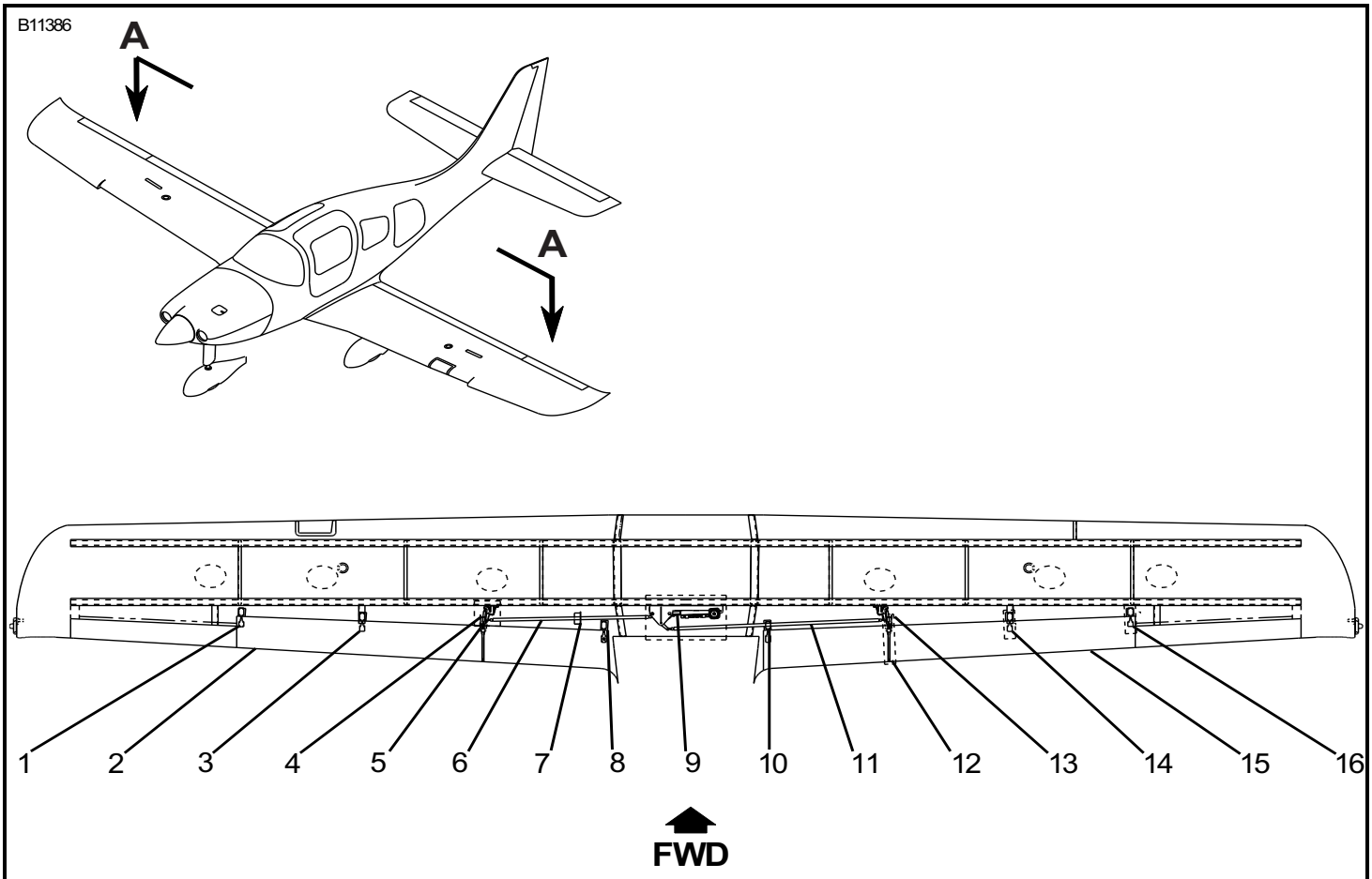
(h) All the bearings and rod ends used in the flap system are Teflon ball bearings that require little maintenance other than cleaning during routine inspections. Do not use liquid lubricants as they cause the Teflon to swell creating excess friction in the bearing. If bearings or rod ends require replacement, contact the manufacturer for replacement parts.

(i) During normal inspections, check control rods, bellcranks, hinges, and brackets for excessive wear, cracks, scratches, and corrosion. Replace worn or damaged components as necessary.

(j) Refer to Chapter 33 of this manual for detailed maintenance procedures of the flap electrical system.

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Figure 201 : Sheet 1 : Flap System Overview



VIEW A-A

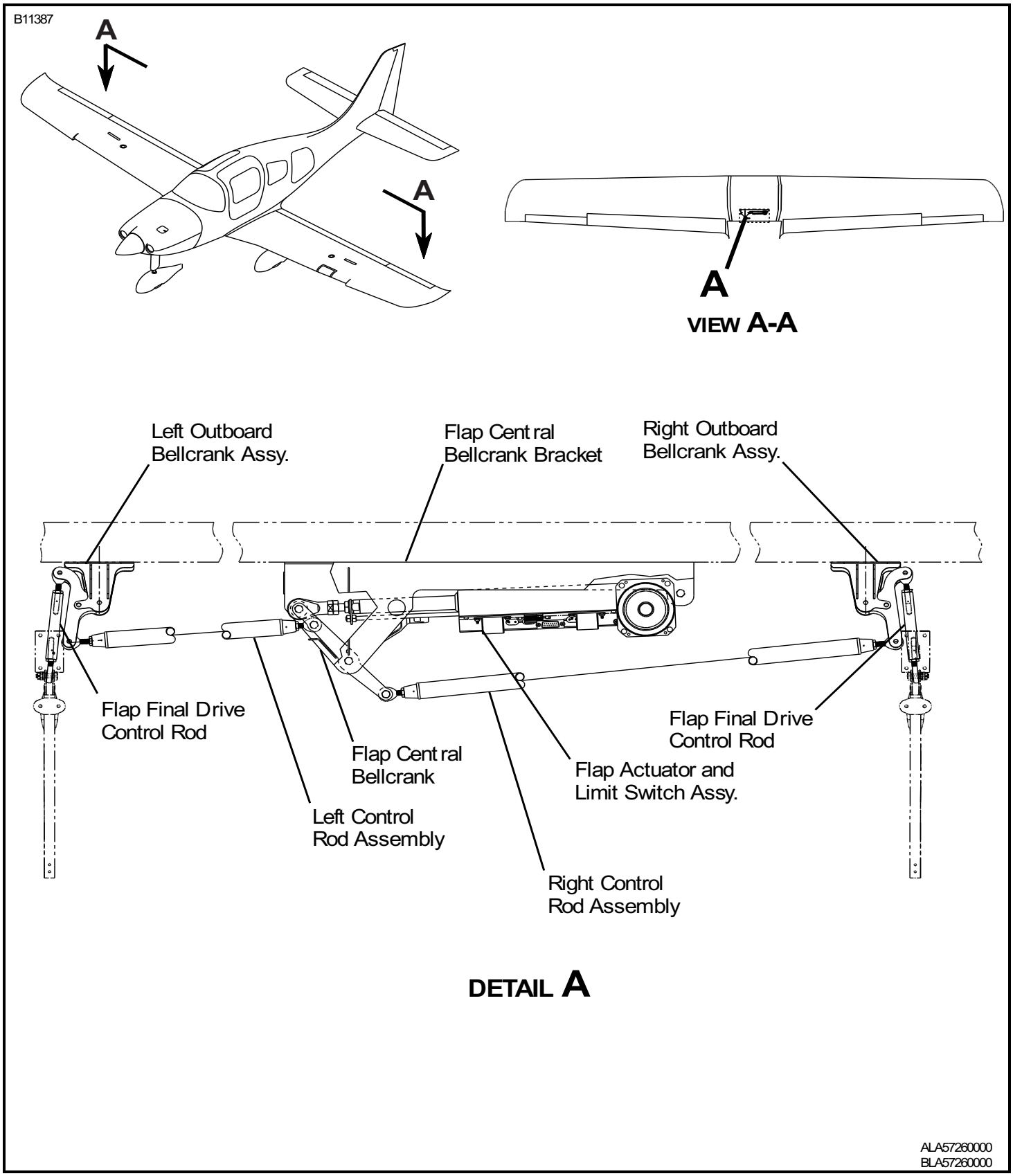
#	ITEM	#	ITEM
1.	WS 143.5 flap hinge assembly	9.	Flap central bellcrank and actuator assembly
2.	Left flap control surface	10.	WS 26.5 flap hinge assembly
3.	WS 104.5 flap hinge assembly	11.	Right flap control rod
4.	Flap drive bellcrank and final drive - left wing	12.	WS 65.5 flap hinge assembly (drive hinge)
5.	WS 65.5 flap hinge assembly (drive hinge)	13.	Flap drive bellcrank and final drive - right wing
6.	Left flap control rod	14.	WS 104.5 flap hinge assembly
7.	Left flap control rod guide*	15.	Right flap control surface
8.	WS 26.5 flap hinge assembly	16.	WS 144 flap hinge assembly

* The control rod guide is not present on all aircraft

ALA57260000

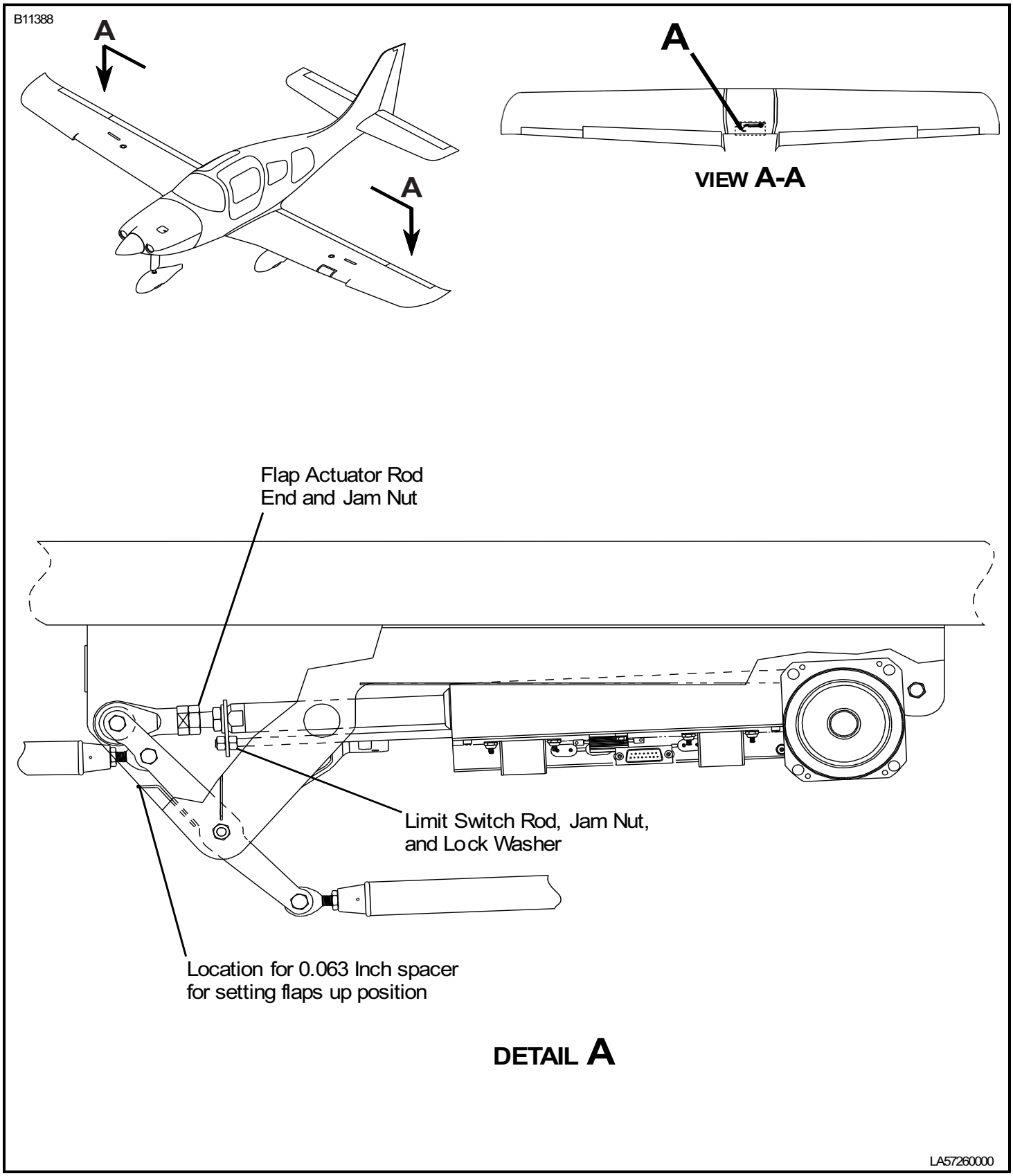
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Figure 202 : Sheet 1 : Flap Rigging Overview



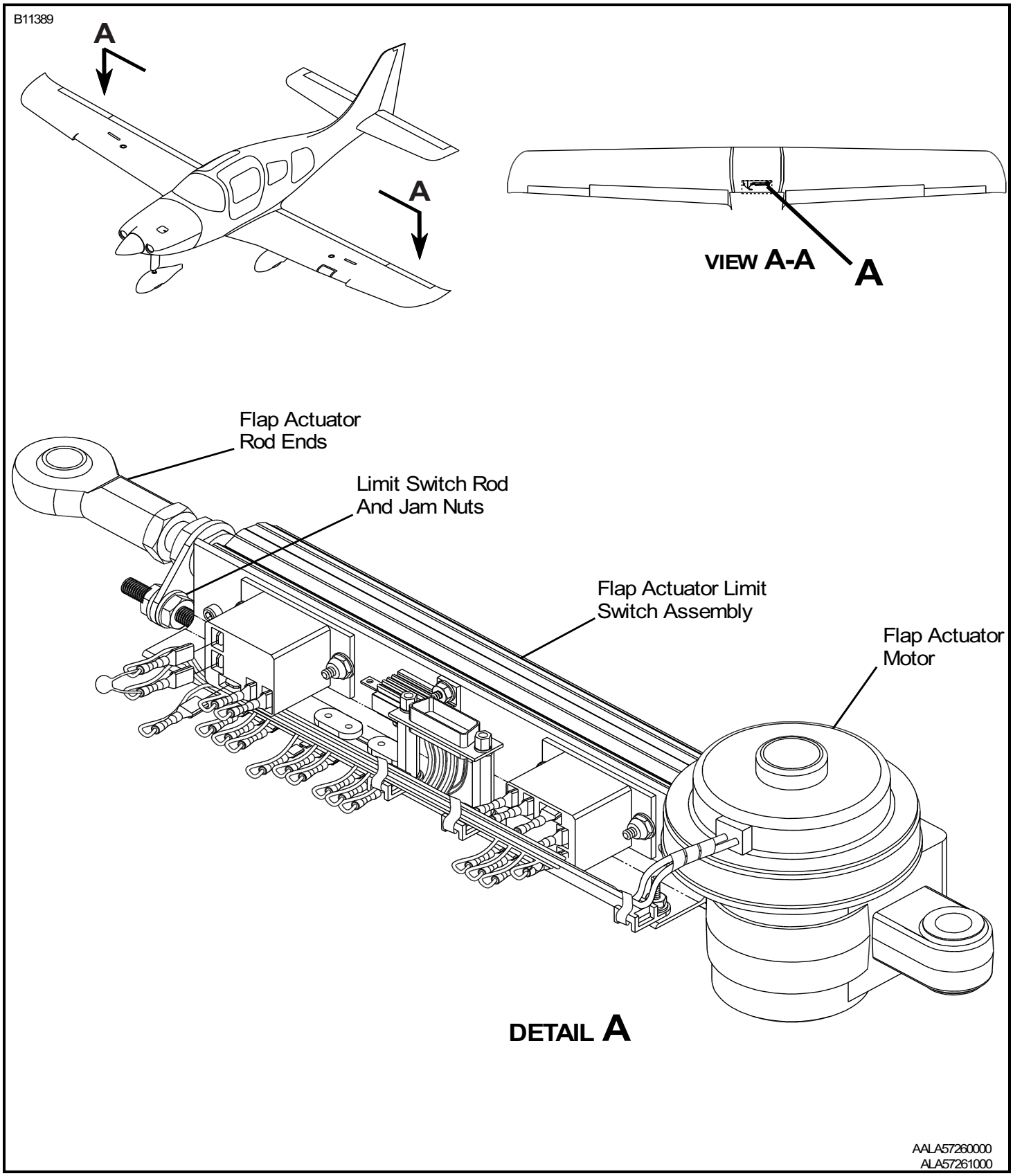
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Figure 203 : Sheet 1 : Flap Central Bellcrank and Flap Actuator



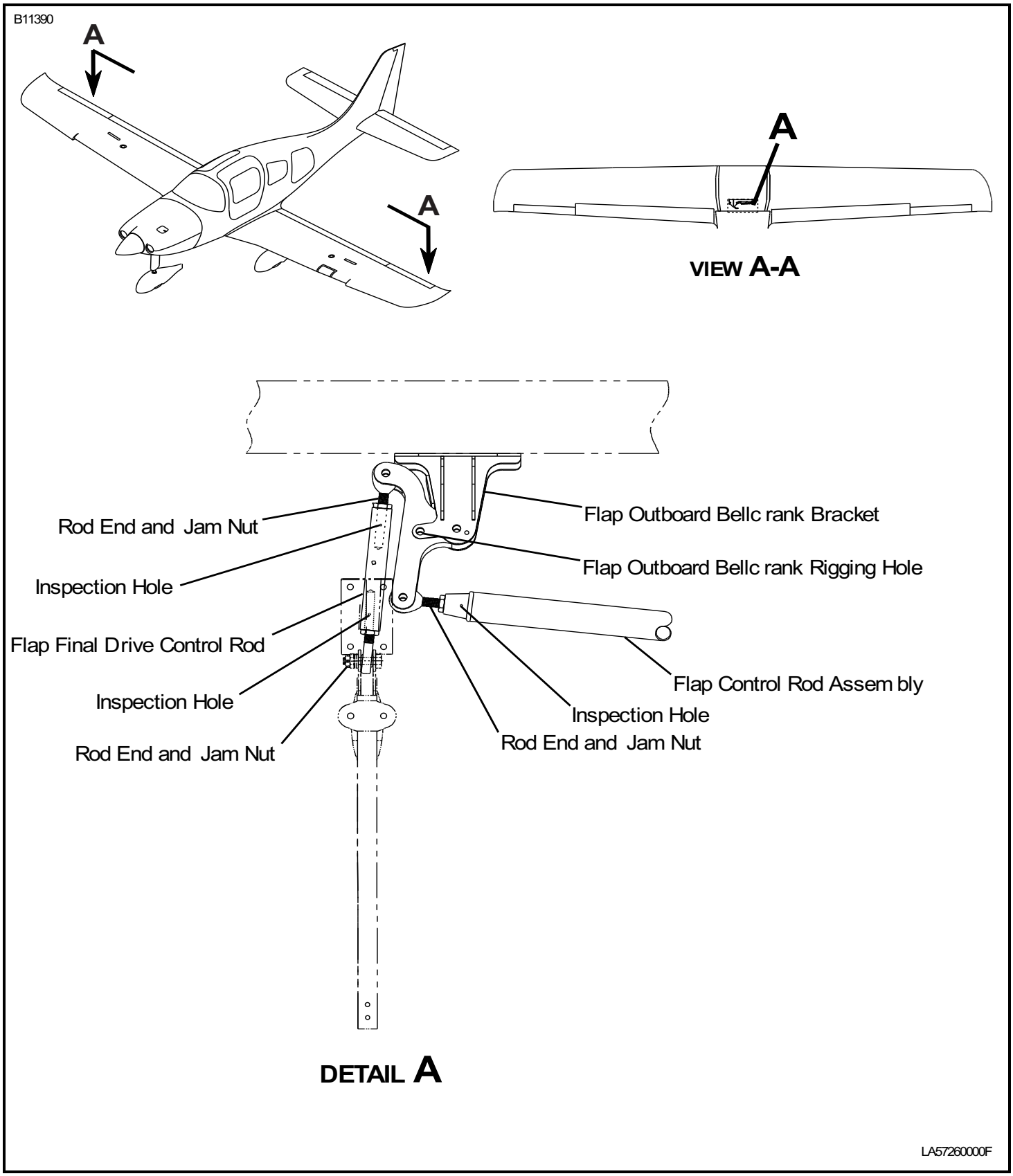
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Figure 204 : Sheet 1 : Flap Actuator Detail - Top View



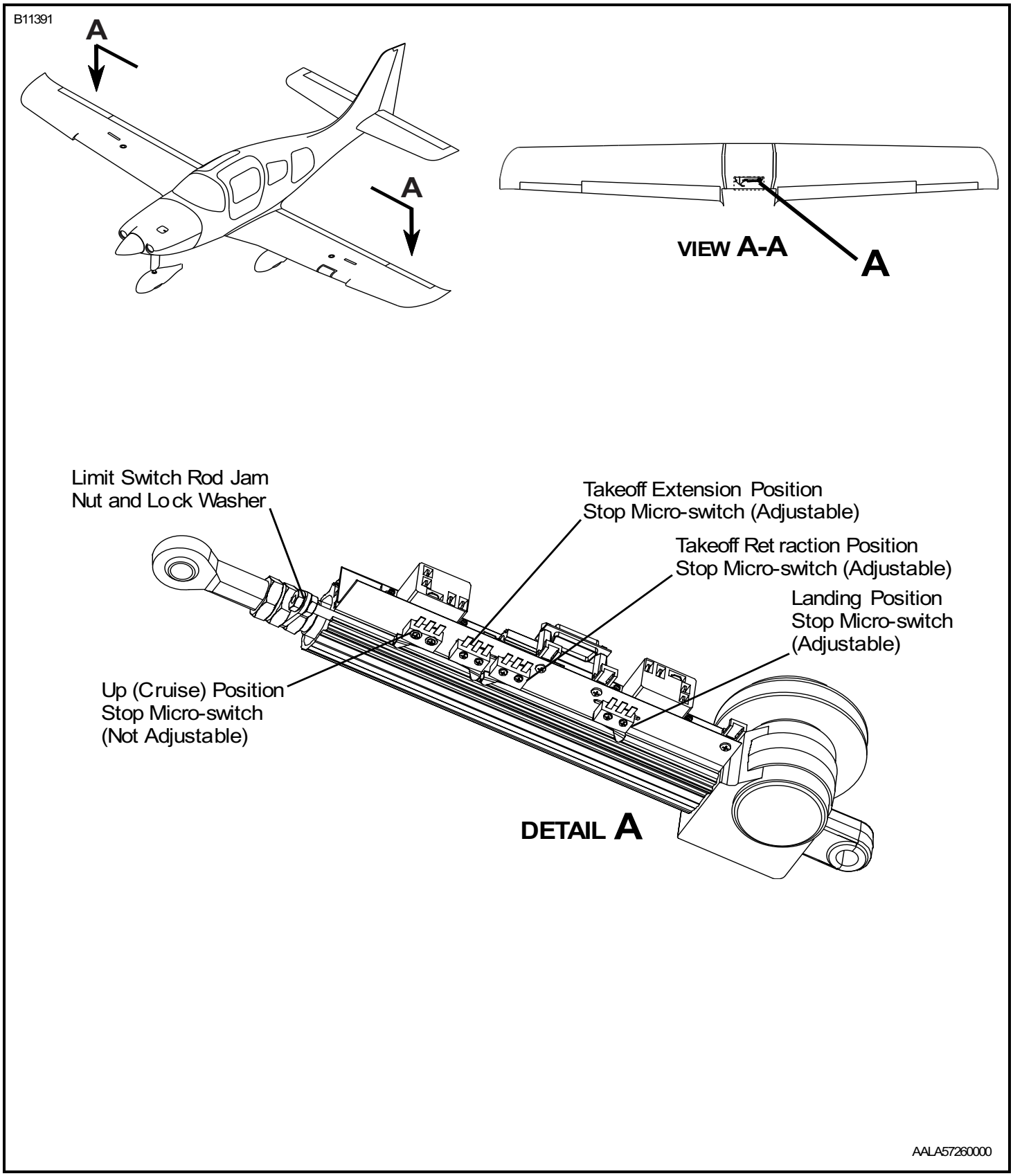
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Figure 205 : Sheet 1 : Flap Outboard Bellcrank and Final Drive



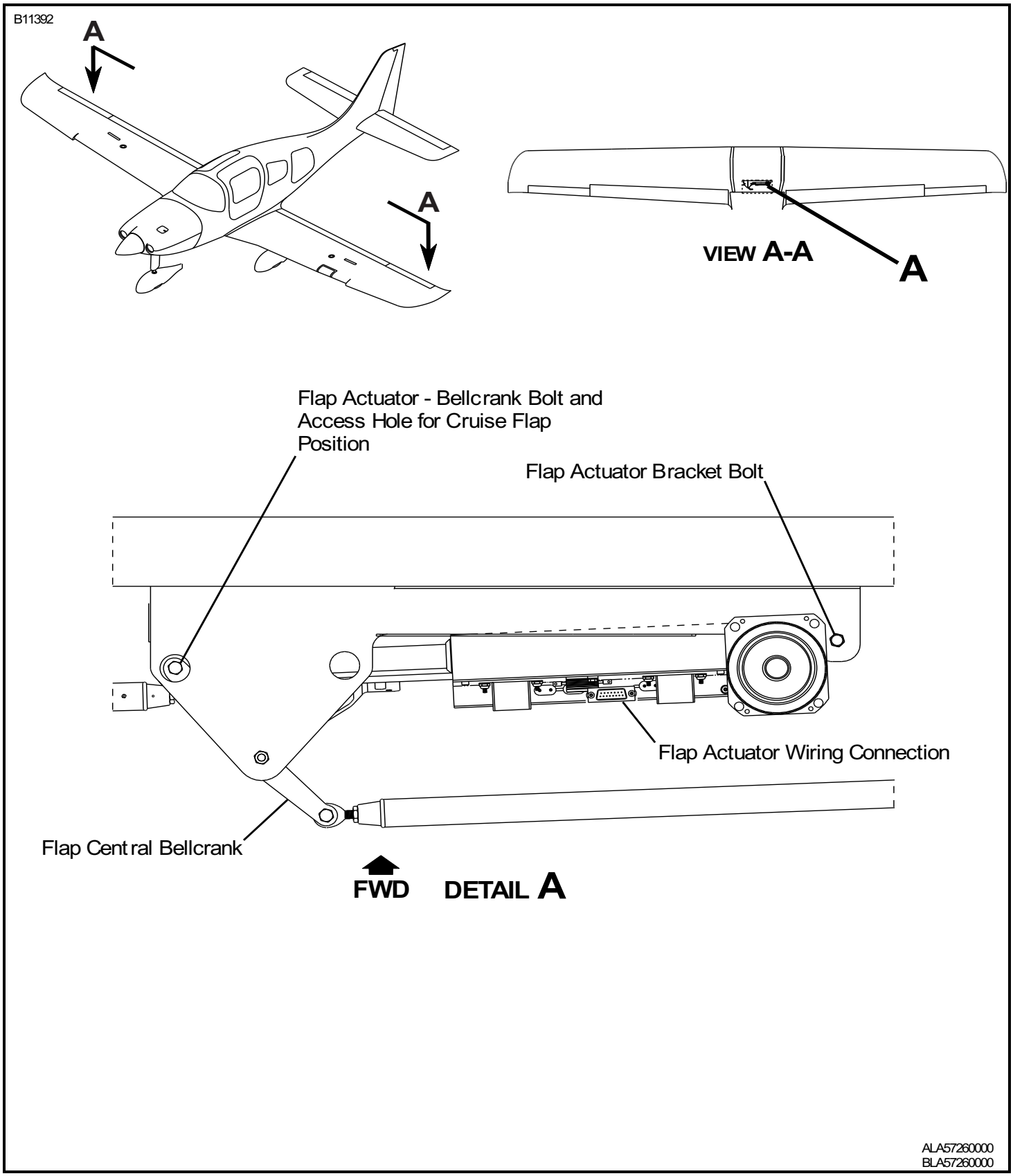
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Figure 206 : Sheet 1 : Flap Actuator and Limit Switch Detail



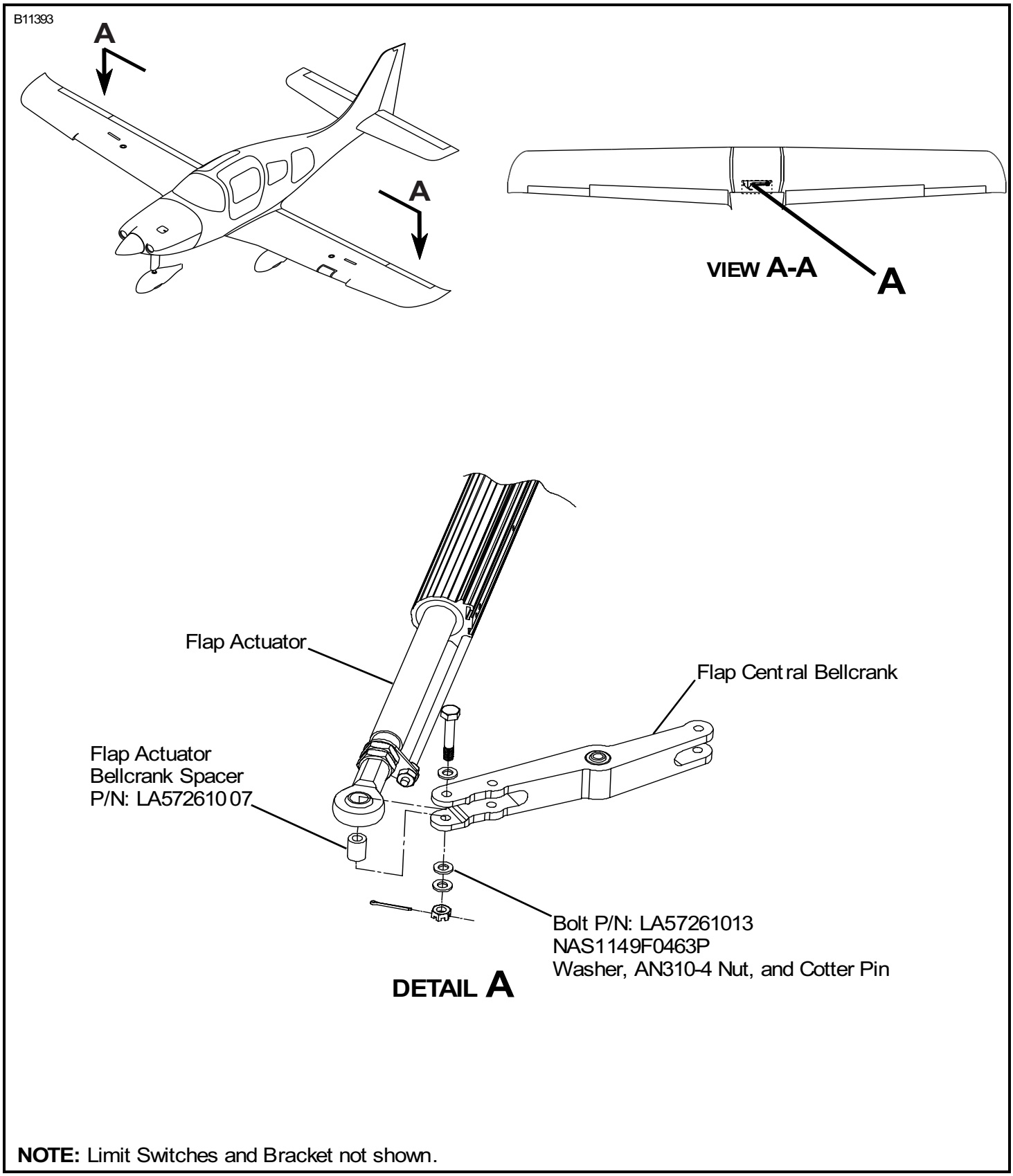
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Figure 207 : Sheet 1 : Flap Central Bellcrank and Actuator Assembly



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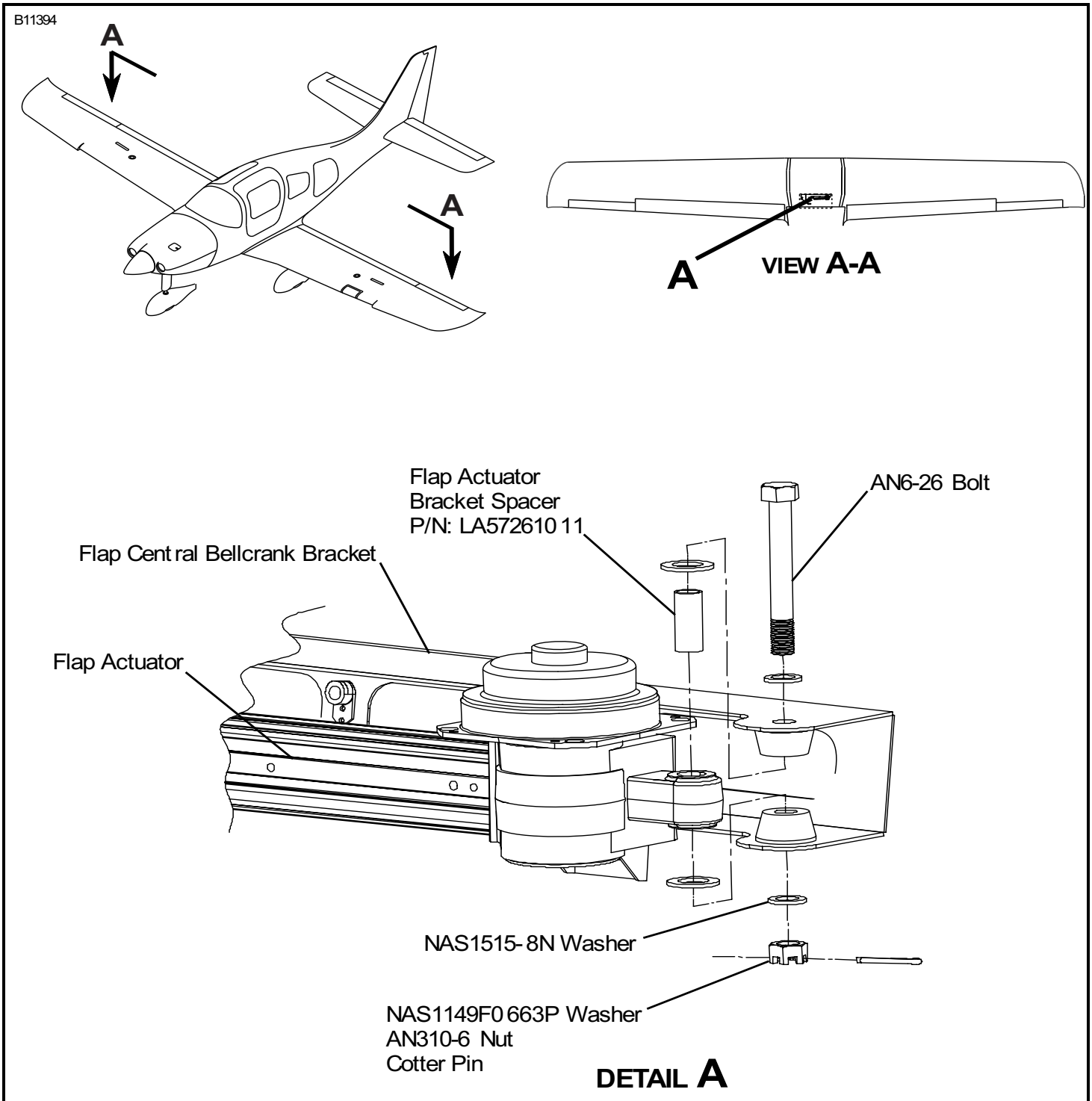
Figure 208 : Sheet 1 : Flap Actuator to Bellcrank Bolt and Spacer Detail



NOTE: Limit Switches and Bracket not shown.

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Figure 209 : Sheet 1 : Flap Actuator to Bracket Bolt and Spacer Detail



NOTE: Limit Switches not shown.

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SPEEDBRAKE(TM) SYSTEM - MAINTENANCE PRACTICES

1. SpeedBrake™ System (Optional) - General

A. **SpeedBrake™ System** - The Precise Flight SpeedBrake™ 2000 System provides expedited descents at low cruise power, glide path control on final approach, airspeed reduction, and an aid to the prevention of excessive engine cooling in descent. Extended SpeedBrakes™ will stow immediately upon application of the rudder limiter and require the SpeedBrake™ rocker switch be cycled OFF and then ON to re-extend the SpeedBrakes™. The SpeedBrakes™ will not automatically re-extend and must be recycled after the following conditions:

(1) Circuit Breaker Pull

(2) Automatic Stowage Due to Asymmetric Deployment or Low Voltage

(3) Rudder Limiter Solenoid Engagement

B. **SpeedBrake™ Components** - The Series 2000 SpeedBrake™ consists of wing mounted electric SpeedBrake™ Cartridges. A central logic-switching unit interconnects each SpeedBrake™ cartridge electronically and a panel mounted SpeedBrake™ actuator switch controls SpeedBrake™ deployment. The SpeedBrake™ cartridges receive electrical power from the aircraft electrical bus through a disconnect type circuit breaker. (Refer to Figure 201)

(1) Rocker Switch - The SpeedBrake™ rocker switch is located next to the throttle in the center of the instrument panel. The switch is positioned UP/ON to fully deploy and is positioned DOWN/OFF to retract the SpeedBrakes(TM).

(2) Annunciator - The SpeedBrake™ Annunciator is located above and to the right of the airspeed indicator on the pilot instrument panel. The Annunciator will fully light after the SpeedBrake™ Switch is toggled ON and both brakes are in the up position. If one or both lights in the annunciator fails to light and both brakes do not extend after the switch is toggled on, it indicates a failure of one or both SpeedBrake™ cartridge(s) and the SpeedBrake™ switch should be toggled off. When the SpeedBrake™ Switch is toggled OFF, the annunciator will extinguish when both brakes are fully stowed in the wing.

NOTE: The SpeedBrake™ Cartridges and/or the Asymmetric Logic Control Unit must be returned to the manufacturer should repair or replacement be required. Even though only one cartridge may require repair or replacement, both cartridges must be returned to the manufacturer.

2. SpeedBrake™

A. **SpeedBrake™ Cartridge Removal** - The SpeedBrake™ cartridges are located within the upper side of the wing and approximately 1/3 wing length from the tip of the wing. One SpeedBrake™ cartridge in each wing. (Refer to Figure 202 and Figure 203.)

(1) Remove the access panel on the underside of the left or right outboard wing.

(2) Disconnect the electrical connection.

(3) Remove the two 8-32 screws and remove the cartridge. Note the screw length and the hole from which it was removed. The screws may be of differing lengths and must be reinstalled in the holes from which they were removed.

(4) A bottom mount bar is attached to the bottom of the SpeedBrake™ cartridge by two 8-32 round head screws. The bottom mount bar may be removed by removal of these screws.

B. **SpeedBrake™ Cartridge Installation** (Refer to Figure 202 and Figure 203).

(1) Attach the bottom mount bar to the bottom of the SpeedBrake™ cartridge with two 8-32 round head screws. Torque the screws 6 to 8 in.-lb.

(2) Insert the speed brake cartridge into the wing.

(3) Connect the electrical connection.

(4) Install the 8-32 screws. Screws may be of differing lengths and must be reinstalled in the holes from which they were removed. Torque 15 to 18 in-lbs. Installed SpeedBrake™ cartridge should be flush with the wing surface within ±0.020 inches. Bottom wing skin should have no more than 0.003-inch distortion after torquing.

(5) Replace the access panel.

C. **Asymmetric Logic Control Unit Removal**- The asymmetric logic control unit is located in the avionics shelf under the baggage compartment floor. (Refer to Figure 201.)

(1) Remove the carpet and access panel from the baggage compartment floor per instructions in Chapter 25.

(2) Loosen the mounting screws that hold the unit to the avionics shelf.

(3) Disconnect all connectors to the unit.

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(4) Remove the unit.

D. **Asymmetric Logic Control Unit Installation** (Refer to Figure 201).

(1) Connect the connectors.

(2) Place the unit in its place on the avionics shelf.

(3) Install screws to connect the unit to the shelf and torque screws 15 in.-lbs.

(4) Install the access panel.

(5) Replace the carpet.

E. **Servicing and Maintenance**

(1) Check SpeedBrake™ Cartridge cap strip cover screws and top attachment screws for security. If loose, remove screws and apply Loctite 242 and re-torque to 8 in-lbs.

(2) Annually remove the SpeedBrake™ Cartridge cover plate, clean and inspect for damage, corrosion, looseness and proper operation. Lubricate worm and worm gear with LUBRIPLATE.

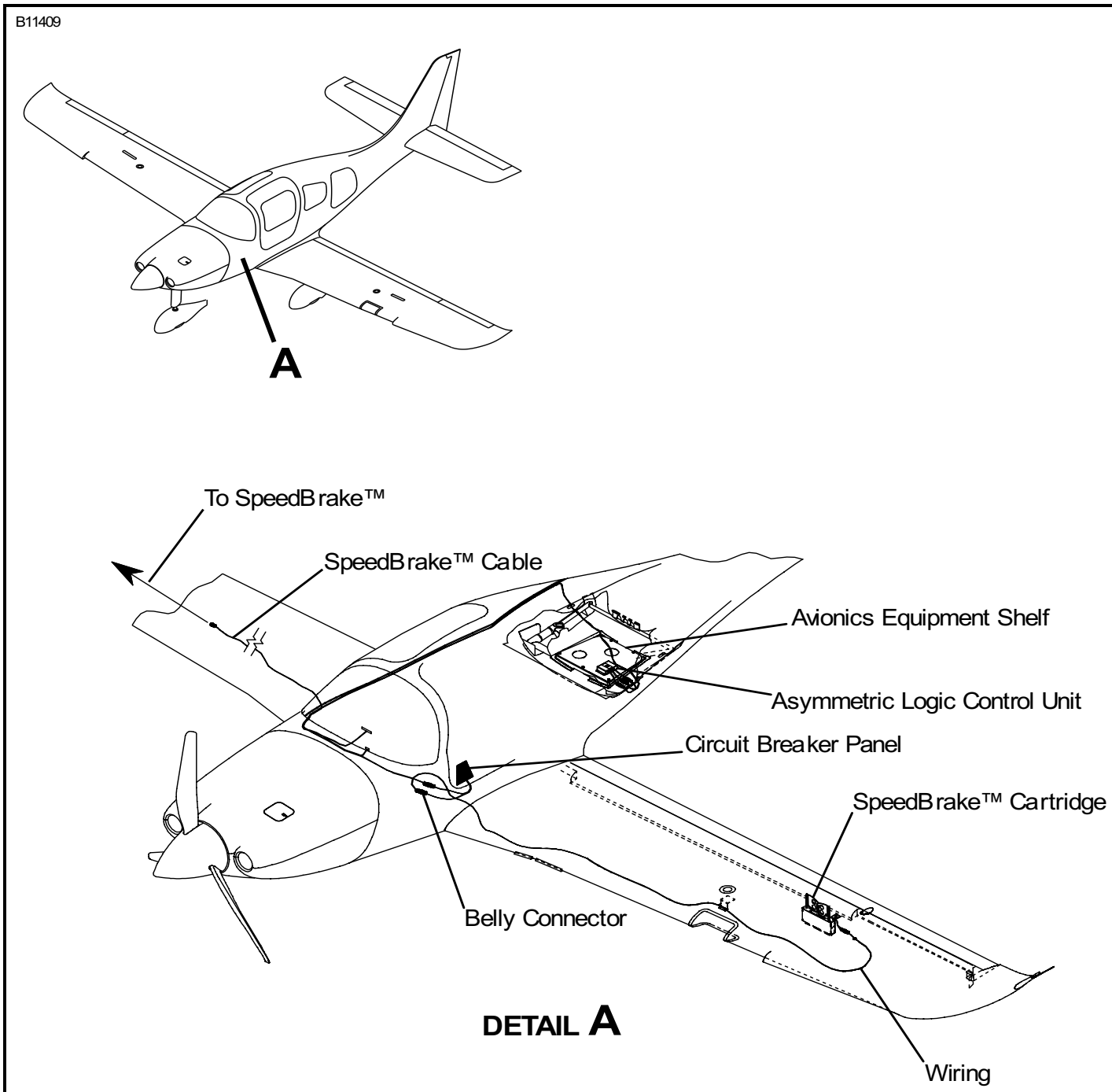
(3) Reinstall the cover plate by applying Loctite 242 to the screw threads and torque to 8 in-lbs.

(4) Return the SpeedBrake™ Cartridge to the manufacturer every 1000 hours for clutch lubrication and spring replacement.

(5) Return the SpeedBrake™ Cartridge to the manufacturer every 5000 hours for drive assembly replacement.

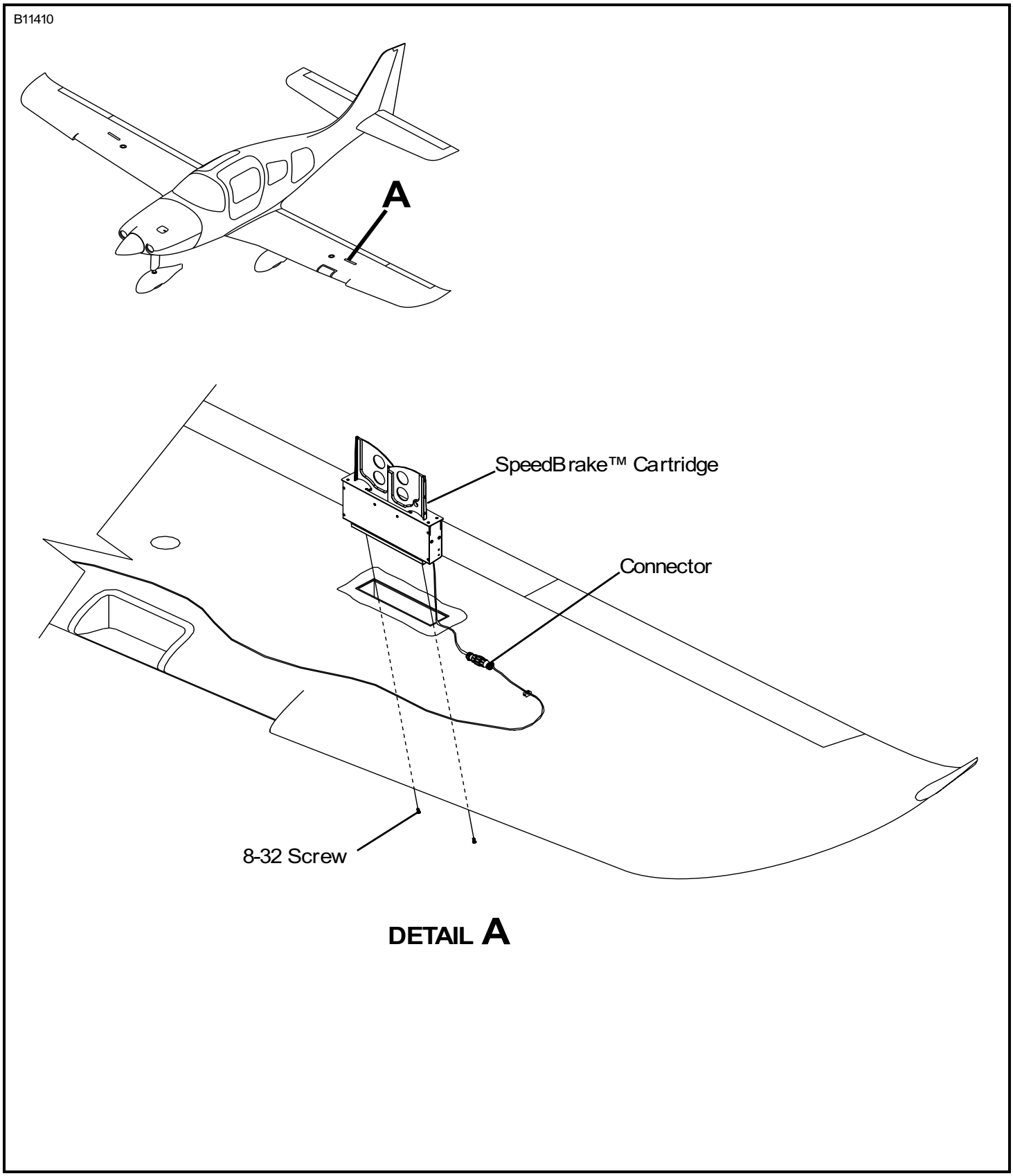
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Figure 201 : Sheet 1 : SpeedBrake™ System



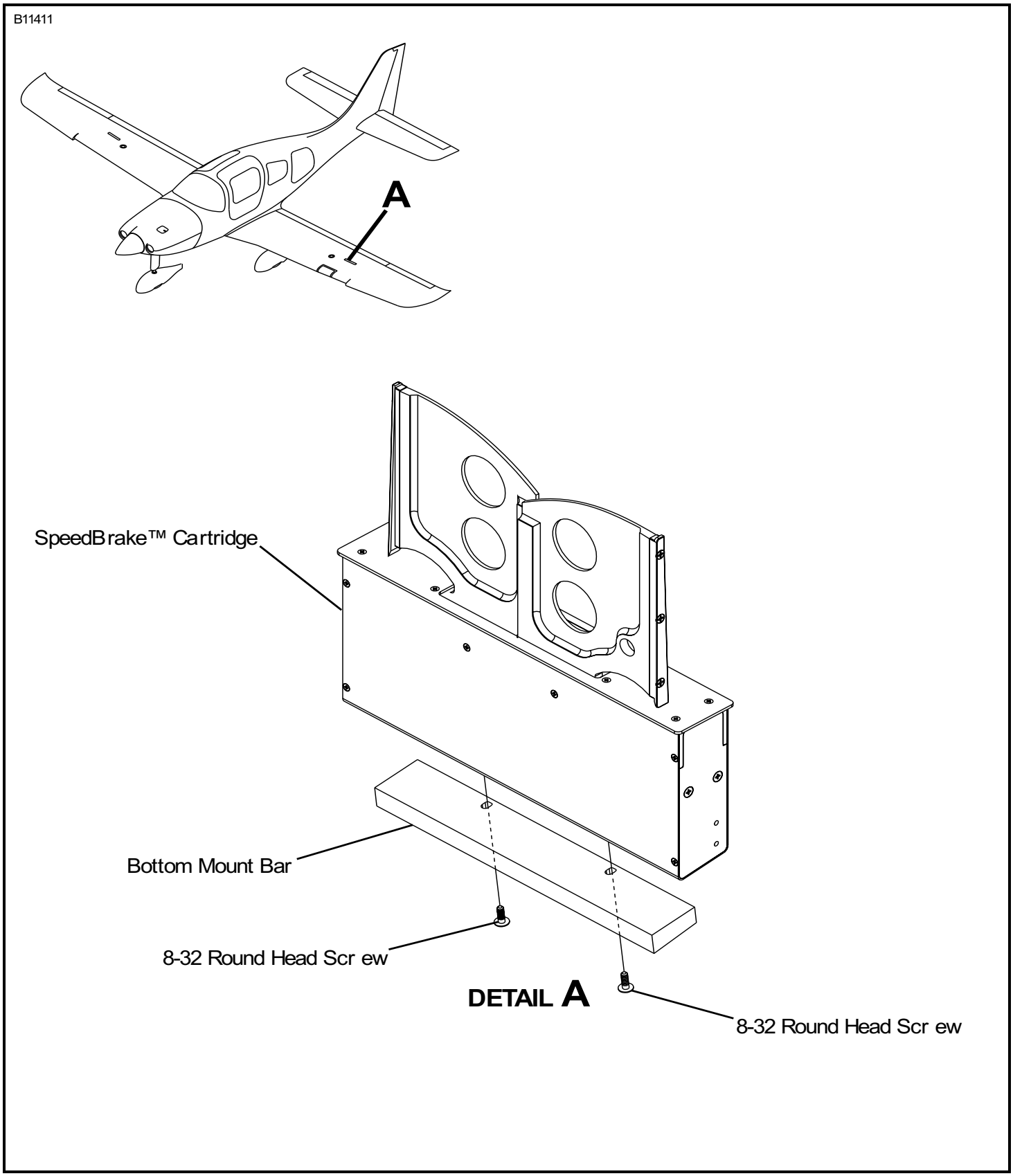
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Figure 202 : Sheet 1 : SpeedBrake™ Cartridge



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Figure 203 : Sheet 1 : SpeedBrake™ Cartridge Detail



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STANDARD PRACTICES, AIRFRAME - GENERAL

1. General

A. This chapter describes the standard maintenance practices for the Cessna 350 (LC42-550FG) aircraft. The items covered are typical and apply to many areas or systems of the aircraft. Unique maintenance procedures are covered in the appropriate chapter in the maintenance manual.

NOTE: Most of this chapter is derived from FAA Advisory Circular (AC) 43.13-1B, which should be used (latest revision) as the resource for this information.

2. Material and Tool Cautions - General

A. **Mercury**

(1) Mercury can corrode and embrittle metallic structures by the amalgamation process. Complete destruction of the load carrying capability of the structure will result.

(2) No thermometers or equipment containing mercury can be used on the aircraft.

B. **Epoxy Resins**

(1) Epoxy resins, particularly the hardeners, are chemically active and can cause chemical burns, allergic reactions, and strong sensitization.

(2) Particular care should be taken to avoid contact with, or breathing the vapors from epoxy resins.

C. **Solvents**

(1) General use solvents may include the following:

(a) Gasoline

(b) Oil

(c) Alcohol

(2) Many of these solvents are flammable.

(3) These solvents can cause injury if contact with the skin or eyes is made.

(4) Many of these solvents are poisonous.

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TORQUE DATA - MAINTENANCE PRACTICES

1. Torque Data - Maintenance Practices

WARNING: The torque values outlined in this chapter and other chapters must be used during installation and repair of components. Improper torque values may result in overstressing of components. Bolted joints in composite structures are sensitive to overload if improper over-torque is used.

A. **Torque Values** - Torque tables are listed in this section. If a component requires specific torque values, those values will be given in the applicable chapter and section of the maintenance manual. Refer to the following figures and tables for specific types of torque values:

(1) Figure 202 / Table 202 - Values for Tapped Holes and Clearance Holes

(2) Figure 203 / Table 203 - Values for Nutplates

(3) Figure 204 / Table 204 - Values for Hose and Tube Fittings

(4) Figure 205 / Table 205 - Values for Rivnuts

(5) Table 206 - Recommended Torque and Clamp Load Specifications (Applicable to Fasteners with a Minimum Tensile Strength Greater Than 74 ksi)

(6) Table 207 - Recommended Jam Nut Torque and Clamp Load Specifications (Applicable to Fasteners With a Minimum Tensile Strength Greater Than 74 ksi)

B. **Free Running Torque** - The torque values listed do not include free running torque. The free running torque is the torque value to rotate the nut on the threaded shaft before contact, pull-up, and tightening. Self-locking nuts are used in applications where they will not be removed often. Repeated removal and installation will cause the self-locking nut to lose its locking feature. They should be replaced when they are no longer capable of maintaining the minimum prevailing torque.

C. **General Torquing Notes**

(1) The values given do not apply to threaded adjustment devices such as rod ends or turnbuckles.

(2) Appropriate torque wrenches should be used. Adjustments should be made for the readings if adapters are required. Refer to specific instructions supplied with the torque wrench.

D. **Torque Adapters** - Using one or more extensions with a torque wrench can put large side loads on torqued fasteners or anything they are attached to (i.e., hoses or tubes); therefore, the operator must support the fastener assembly appropriately when needed.
When using an extended-length adapter on a torque wrench (i.e. crow's foot), apply the following equation:

Table 201. Torque Adapter Equation

$T_E = (T \times L) / (L + E)$ where,

T = Specified torque value to achieveL = Wrench length from grip center to turn center. Refer to Figure 201.E = Length, in axial direction of wrench, from wrench turn center to extension turn center. Refer to Figure 201.T_E = Torque wrench setting when using adapter

Table 202. Values for Tapped Holes and Clearance Holes

GENERAL DESIGN DATA FOR TAPPED HOLES, CLEARANCE HOLES, AND TORQUE AND LOAD VALUES										
NOTE: **Refer to diagram in Figure 202 when consulting this table.**										
		Thread Size								
Thread Size		6-32	8-32	10-32	1/4-20	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20
Max. Diameter		.138	.164	.189	.249	.249	.312	.374	.437	.499
Bolt Clearance Hole Diameter and Tolerance	.005 Tol	.148	.174	.200	.260	.260	.323	.385	.448	.510
	.010Tol	.158	.184	.210	.270	.270	.333	.395	.458	.520
Tap Drill Diameter	Min.	.1040	.1300	.1560	.1960	.2110	.2670	.3300	.3830	.4460
	Max.	.1140	.1389	.1641	.2067	.2190	.2754	.3372	.3918	.4537

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Tap Drill Point Max.		.0329	.0401	.0474	.0597	.0632	.0795	.0973	.1130	.1310
2 Thread Depth		.0625	.0625	.0625	.1000	.0714	.0833	.0833	.1000	.1000
Chamfer Dia.	Min.	.138	.164	.190	.250	.250	.313	.375	.438	.500
	Max.	.158	.184	.210	.270	.270	.333	.395	.458	.520
Tensile StressArea		.00909	.0140	.0200	.0318	.0364	.0580	.0878	.1187	.1599
Non-Lubricated (Dry)Recommended Torque		10 to 12 lb.-in.	20 to 25 lb.-in.	30 to 36 lb.-in.	72 to 84 lb.-in.	72 to 84 lb.-in.	140 to 180 lb.-in.	240 to 280 lb.-in.	55 to 60 lb.-in.	80 to 85 lb.-ft.
LubircatedRecommended Torque		6 to 8 lb.-in.	12 to 15 lb.-in.	20 to 25 lb.-in.	50 to 58 lb.-in.	50 to 58 lb.-in.	95 to 125 lb.-in.	165 to 195 lb.-in.	38 to 42 lb.-in.	55 to 60 lb.-ft.
Thread Length in Aluminum	6061-T6	.145	.225	.320	.440	.440	.550	.690	.790	.930
	2024-T4	.109	.170	.240	.330	.330	.420	.530	.600	.710
Approx. Load in Pounds (Metal)		540	840	1200	2100	2100	3400	5200	7000	9600
100 DegreeChamfer	Dia. +.010/-.000	.279	.334	.387	.510	.510	.638	.755	.894	1.022
	Depth +.005/-.000	.057	.068	.080	.106	.106	.133	.159	.186	.213

Table 203. Values for Nutplates

GENERAL DESIGN DATA FOR BLIND INSTALLATION OF NUTPLATES WITH MINIMUM 2 THREAD PROJECTION PER MS33588

NOTE:

Refer to the diagram in Figure 203 when consulting this table.

For maximum possible thread use:Pre-install bolt without mounting part(s) until lightly bottomed.

CAUTION:

Do not damage threads.

Measure distance between bolt head and structure. Distance should be no more than .9 x thickness of mounting part(s). For floating-type nutplates, add .010 to bolt clearance holes.Do not use aluminum rivets through carbon materials. Use stainless pop- or pull-type rivets when passing through carbon materials.REF: Acceptable Methods, Techniques And Practices, Aircraft Inspection And Repair, AC43.13-1B

		Thread Size							
Thread Size		6-32	8-32	10-32	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20
Max. Diameter of Thread		.138	.164	.189	.249	.312	.374	.437	.499
Bolt Clearance Hole Diameter and Tolerance	.005 Tol	.148	.174	.200	.260	.323	.385	.448	.510
	.010Tol	.158	.184	.210	.270	.333	.395	.458	.520
2 Thread Minimum Projection		.0625	.0625	.0625	.1000	.0714	.1111	.1250	.520
Thread Length of Nutplate Ref.		.171	.188	.188	.219	.266	.282	.328	.547
2 Thread Projection + Nutplate		.234	.251	.251	.291	.349	.365	.428	.647
Min. Number of Turns		7.5	8.0	8.0	8.25	8.5	8.75	8.75	13.0
Ref. Tensile Stress Area		.00909	.0140	.0200	.0384	.0580	.0878	.1187	.1599

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Non-Lubricated (Dry)Recommended Torque(Metal)	10 to 12 lb.-in.	20 to 25 lb.-in.	30 to 36 lb.-in.	72 to 84 lb.-in.	140 to 180 lb.-in.	240 to 280 lb.-in.	55 to 60 lb.-in.	80 to 85 lb.-ft.
LubircatedRecommended Torque (Metal)	6 to 8 lb.-in.	12 to 15 lb.-in.	20 to 25 lb.-in.	50 to 58 lb.-in.	95 to 125 lb.-in.	165 to 195 lb.-in.	38 to 42 lb.-in.	55 to 60 lb.-ft.
Approx. Load in Pounds (Metal)	540	840	1200	2100	3400	5200	7000	9600

Table 204. Values for Hose and Tube Fittings

NOTE: **Refer to the diagram in Figure 204 when consulting this table.**				
Size/Thread		Material and Torque Specification(Dry torque values shown, except as noted in General Notes.)		
		Hose and/or Fitting Combination(Refer to Note 1)	Inch-Pounds	Foot-Pounds
-2	5/16-24	Steel to Steel	75-120	6.3 - 10
		Brass/Aluminum/Steel	50-80	4.2 - 6.7
-3	3/8-24	Steel to Steel	95-140	7.9 - 11.7
		Brass/Aluminum/Steel	70-105	5.8 - 8.8
-4	7/16-20	Steel to Steel	135-190	11.3 - 15.8
		Brass/Aluminum/Steel	100-140	8.3 - 11.7
-5	1/2-20	Steel to Steel	170-240	14.2 - 20
		Brass/Aluminum/Steel	130-180	10.8 - 15
-6	9/16-18	Steel to Steel	215-280	17.9 - 23.3
		Brass/Aluminum/Steel	150-195	12.5 - 16.3
-8	3/4-16	Steel to Steel	470-550	39.2 - 45.8
		Brass/Aluminum/Steel	270-350	22.5 - 29.2
-10	7/8-14	Steel to Steel	620-745	51.7 - 62.1
		Brass/Aluminum/Steel	360-430	30 - 35.8
-12	1/16-12	Steel to Steel	855-1055	71.3 - 87.9
		Brass/Aluminum/Steel	460-550	38.3 - 45.8

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General Notes: Pre-Install hose onto fitting finger tight. If parts do not assemble finger tight, check for foreign material on threads and/or damaged threads. Do not force assemblies together.Maintain cleanliness of hose and fittings. Protect all fitting threads from damage and dirt contamination. When attaching stainless steel to stainless steel or stainless steel to aluminum, use MIL-T-5544 Anti-seize/lubricant on fitting threads. To prevent contamination, do not apply Anti-seize to tapered seat. Use lower torque value when Anti-seize is used.

NOTE 1:

Brass/Aluminum/Steel means any combination of brass to aluminum, brass to steel, or aluminum to steel.

Table 205. Values for Rivnuts

GENERAL DESIGN DATA FOR BLIND INSTALLATION OF RIVNUTS WITH MINIMUM 2 THREAD PROJECTION PER MS33588								
NOTE: **Refer to the diagram in Figure 205 when consulting this table.**								
For maximum possible thread use:Pre-install bolt without mounting part(s) until lightly bottomed. **CAUTION:** **Do not damage threads.** Measure distance between bolt head and structure. Distance should be no more than .9 x thickness of mounting part(s). Rivnuts are not self-locking. Do not use aluminum rivnuts through carbon materials. Use stainless rivnuts through carbon materials.REF: Acceptable Methods, Techniques And Practices, Aircraft Inspection And Repair, AC43.13-1B								
		Thread Size						
Thread Size		6-32	8-32	10-32	1/4-20	1/4-28	5/16-18	3/8-16
Max. Diameter of Thread		.138	.164	.189	.249	.249	.312	.374
Bolt Clearance Hole and Diameter	.005 Tol	.148	.174	.200	.260	.260	.323	.385
	.010 Tol	.158	.184	.210	.270	.270	.333	.395
Recommended Rivnut Hole Dia.	Min.	.266	.266	.297	.391	.391	.531	.531
	Max.	.272	.272	.303	.397	.397	.538	.538
2 Thread Projection		.0625	.0625	.0625	.1000	.0714	.1111	.1250
Thread Length of Rivnut		.219	.188	.203			.278	
Min. Thread Projection		.281	.250	.266			.389	
Min. Number of Turns		9.0	7.0	8.5			7.0	
Head Height		0.30	.030	.030	.030	.030	.030	.030
Head Diameter Max.		.390	.390	.415	.515	.515	.685	.685
Non-Lubricated (Dry) Recommended Torque		10 to 12 lb.-in.	18 to 22 lb.-in.	30 to 36 lb.-in.	65 to 75 lb.-in.	70 to 80 lb.-in.	140 to 155 lb.-in.	230 to 270 lb.-in.
Lubricated Recommended Torque		6 to 8 lb.-in.	12 to 15 lb.-in.	20 to 25 lb.-in.	42 to 50 lb.-in.	50 to 58 lb.-in.	85 to 110 lb.-in.	160 to 190 lb.-in.
Approx. Load in Pounds		540	840	1200	2000	2100	3200	4900

Table 206. Recommended Torque and Clamp Load Specifications (Applicable to Fasteners with a Minimum Tensile Strength Greater Than 74 ksi)

Fastener Size	Nominal Diameter	Clamp Load	Torque Dry	Torque Lubed
			(Maximum Error of ±10 percent)	
	[in]	[lbs]	[in-lbs,**ft-lbs**]	[in-lbs,**ft-lbs**]
2-56	.0860	150	2.5	2.0
4-40	.1120	250	5.5	4.0
6-32	.1380	370	10	8.0

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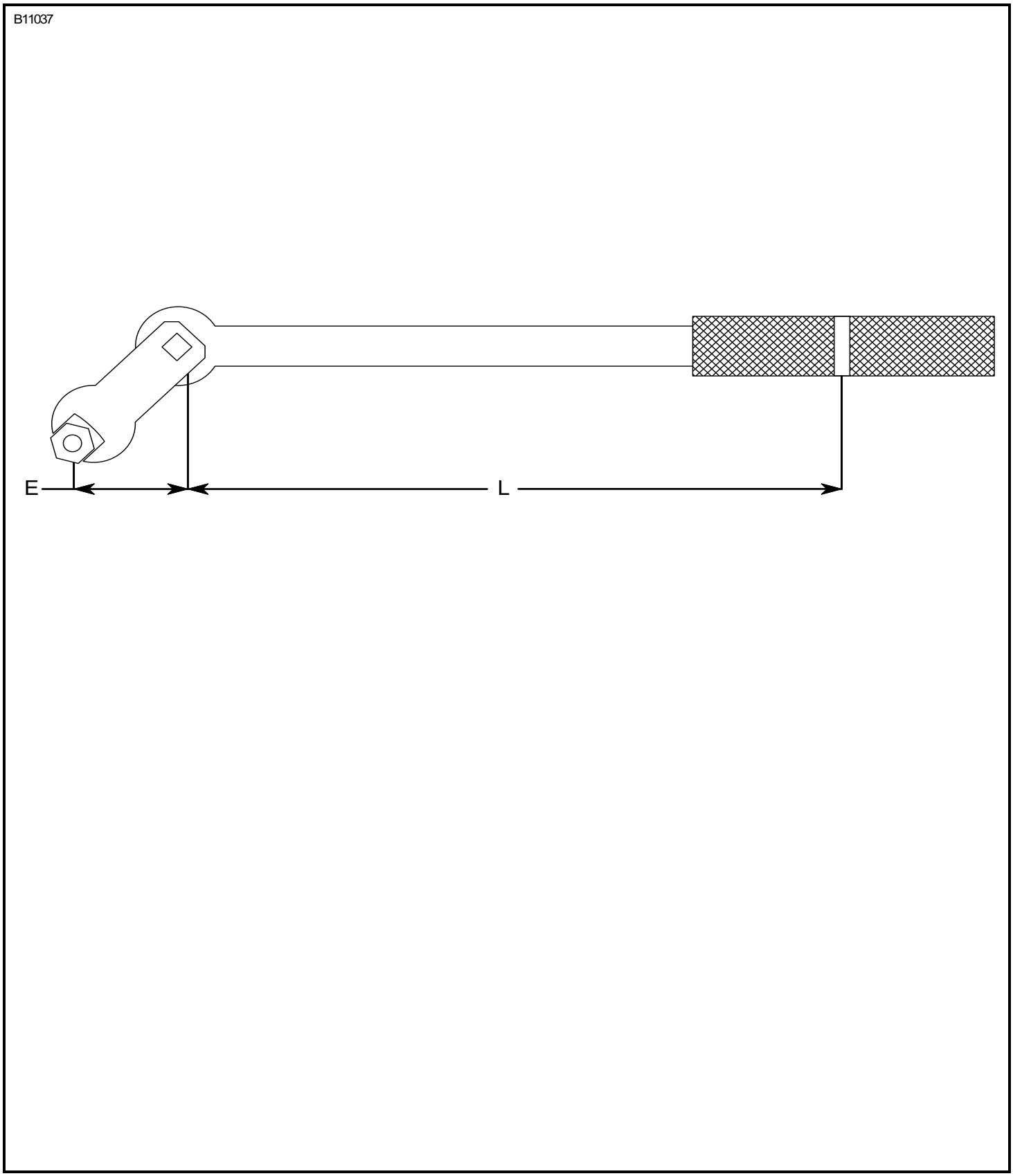
8-32	.1640	580	19	14
10-32	.1900	830	31	24
1/4-20	.2500	1310	66	49
1/4-28	.2500	1500	75	56
5/16-18	.3125	2160	140	100
5/16-24	.3125	2390	150	110
3/8-16	.3750	3200	**20**	**15**
3/8-24	.3750	3620	**23**	**17**
7/16-20	.4375	4900	**36**	**27**
1/2-20	.5000	6600	**55**	**41**
NOTE: **Applicable for standard or self-locking types of nuts**				

Table 207. Recommended Jam Nut Torque and Clamp Load Specifications (Applicable to Fasteners with a Minimum Tensile Strength Greater Than 74 ksi)

Fastener Size	**Nominal Diameter**	**Clamp Load**	**Torque Dry**	**Torque Lubed**
			(Maximum Error of ±10 percent)	
	[in]	[lbs]	[in-lbs,**ft-lbs**]	[in-lbs,**ft-lbs**]
2-56	.0860	62	1.0	0.8
4-40	.1120	100	2.5	1.5
6-32	.1380	150	4.0	3.0
8-32	.1640	250	8.0	6.0
10-32	.1900	360	14	10
1/4-20	.2500	550	27	20
1/4-28	.2500	730	36	27
5/16-18	.3125	800	50	37
5/16-24	.3125	1240	77	58
3/8-16	.3750	1330	100	75
3/8-24	.3750	1330	100	75
7/16-20	.4375	3120	**23**	**17**
1/2-20	.5000	3730	**31**	**23**
NOTE: **AN316 or equivalent**				

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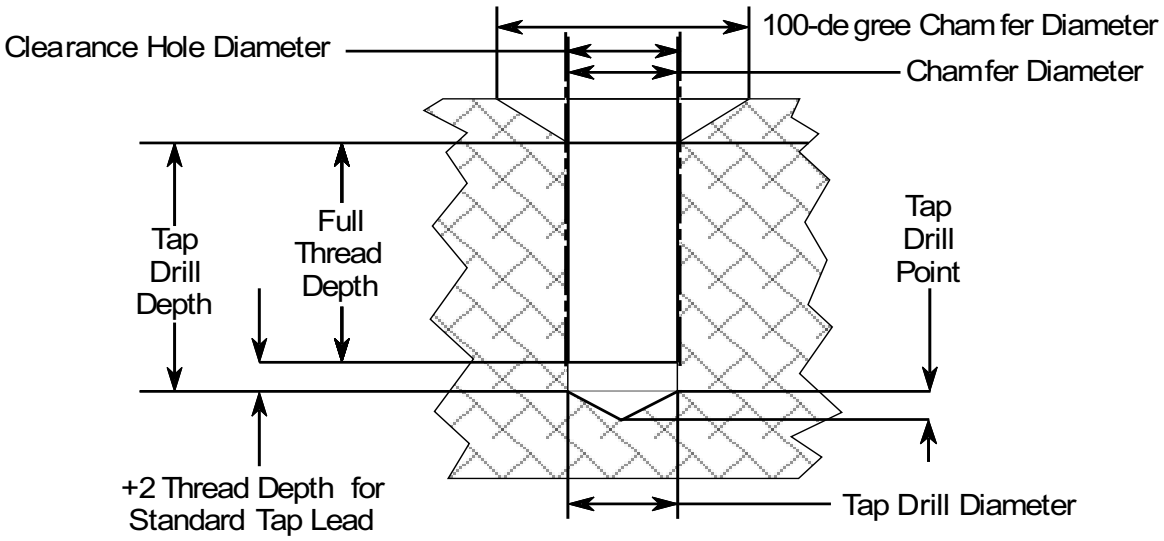
Figure 201 : Sheet 1 : Length Variable for Torque Adapter Equation



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Figure 202 : Sheet 1 : Values for Tapped Holes and Clearance Holes (Refer to Table 202)

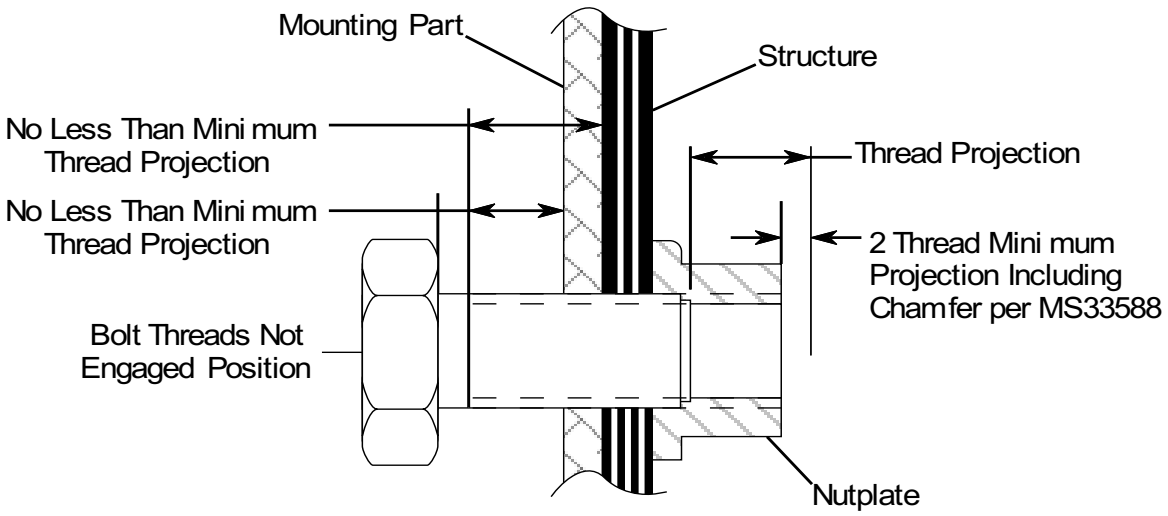
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Figure 203 : Sheet 1 : Values for Nutplates (Refer to Table 203)

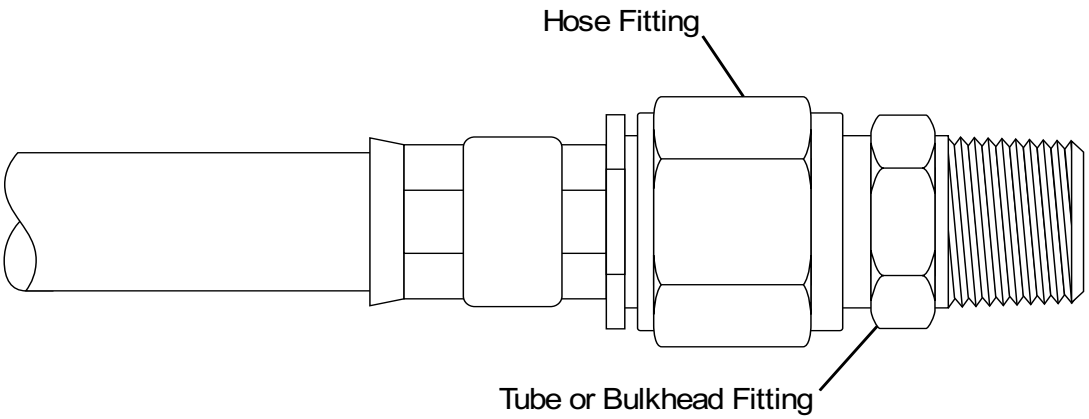
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Figure 204 : Sheet 1 : Values for Hose and Tube Fittings (Refer to Table 204)

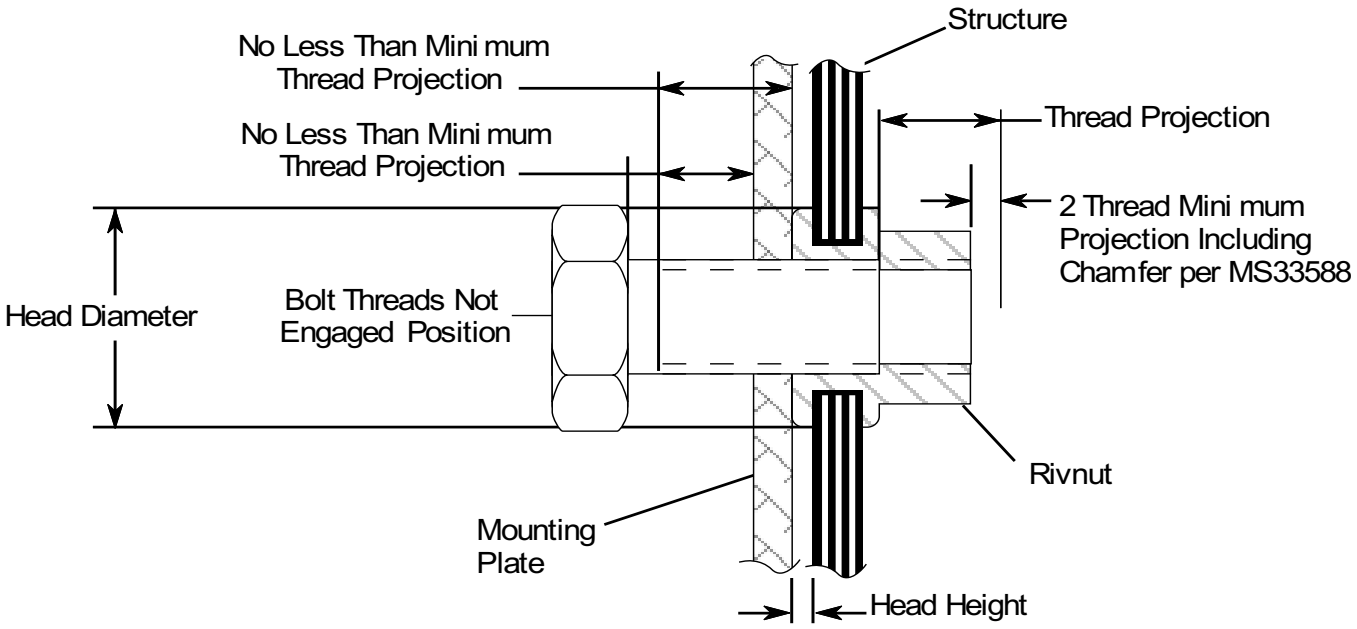
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Figure 205 : Sheet 1 : Values for Rivnuts (Refer to Table 205)

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SAFETYING - MAINTENANCE PRACTICES

1. Safetying - Maintenance Practices

A. **General** - The word *safetying* is a term universally used in the aircraft industry. Briefly, safetying is defined as, "securing by various means any nut, bolt, turnbuckle etc., on the aircraft so that vibration will not cause it to loosen during operation." These practices are not a means of obtaining or maintaining torque; rather, a safety device to prevent the disengagement of screws, nuts, bolts, snap rings, oil caps, drain cocks, valves, and parts. Three basic methods are used in safetying: safety-wire, cotter pins, and self-locking nuts. Retainer washers and pal nuts are also sometimes used.

(1) Wire, either soft brass or steel, is used on cylinder studs, control cable turnbuckles, and engine accessory attaching bolts.

(2) Cotter pins are used on aircraft and engine controls, landing gear, or any other point where a turning or actuating movement takes place.

B. **Safety Wire**

WARNING: Do not use stainless steel, monel, carbon steel, or aluminum alloy safety wire to secure emergency mechanisms such as switch handles, guard covering handles used on exits, fire extinguishers, emergency gear releases, or other emergency equipment. Some existing structural equipment or safety-of-flight emergency devices require copper or brass safety wire (0.020-inch diameter only). Where successful emergency operation of this equipment is dependent on shearing or breaking of the safety wire, particular care should be used to ensure that safetying does not prevent emergency operation.

(1) There are two methods of safety wiring; the double-twist method that is most commonly used, and the single-wire method used on screws, bolts, and/or nuts in a closely-spaced or closed-geometrical pattern such as a triangle, square, rectangle, or circle. The single-wire method may also be used on parts in electrical systems and in places that are difficult to reach. Refer to Figure 201.

(2) When using the double-twist method of safety wiring, 0.032 inches minimum diameter wire should be used on parts that have a hole diameter larger than 0.045 in. Safety wire of 0.020 in. diameter (double strand) may be used on parts having a nominal hole diameter between 0.045- and 0.062-inch with a spacing between parts of less than 2 inches. When using the single-wire method, the largest size wire that the hole will accommodate should be used. Copper wire (0.020 inches diameter), aluminum wire (0.031 inches diameter), or other similar wire called for in specific technical orders should be used as seals on equipment such as first-aid kits, portable fire extinguishers, emergency valves, or oxygen regulators. Refer to Figure 202.

CAUTION: Care should be taken not to confuse steel with aluminum wire.

(3) A secure seal indicates that the component has not been opened. Some emergency devices require installation of brass or soft copper shear safety wire. Particular care should be exercised to ensure that the use of safety wire will not prevent emergency operation of the devices.

2. Safety Wiring Procedures

A. There are many combinations of safety wiring with certain basic rules common to all applications. These rules are as follows:

(1) When bolts, screws, or other parts are closely grouped, it is more convenient to safety wire them in series. The number of bolts, nuts, screws, etc., that may be wired together depends on the application.

(2) Drilled bolt heads and screws need not be safety wired if installed with self-locking nuts.

(3) To prevent failure due to rubbing or vibration, safety wire must be tight after installation.

(4) Safety wire must be installed in a manner that will prevent the tendency of the part to loosen.

(5) Safety wire must never be over-stressed. Safety wire will break under vibrations if twisted too tightly. Safety wire must be pulled taut when being twisted and maintain a light tension when secured.

(6) Safety-wire ends must be bent under and inward toward the part to avoid sharp or projecting ends, which might present a safety hazard.

(7) Safety wire inside a duct or tube must not cross over or obstruct a flow passage when an alternate routing can be used.

B. Check the units to be safety wired to make sure that they have been correctly torqued and that the wiring holes are properly aligned to each other. When there are two or more units, it is desirable that the holes in the units be aligned to each other.

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Never over-torque or loosen to obtain proper alignment of the holes. It should be possible to align the wiring holes when the bolts are torqued within the specified limits. Washers may be used to establish proper alignment. However, if it is impossible to obtain a proper alignment of the holes without under-torquing or over-torquing, try another bolt that will permit proper alignment within the specified torque limits.

C. To prevent mutilation of the twisted section of wire when using pliers, grasp the wires at the ends. Safety wire must not be nicked, kinked, or mutilated. Never twist the wire ends off with pliers. When cutting off ends, leave at least four to six complete turns (1/2 to 5/8 inch long) after the loop. When removing safety wire, never twist the wire off with pliers. Cut the safety wire close to the hole, exercising caution.

D. Install safety wire where practicable with the wire positioned around the head of the bolt, screw, or nut, and twisted in such a manner that the loop of the wire fits closely to the contour of the unit being safety wired.

3. Twisting With Special Tools

A. Twist the wire with a wire twister as follows:

CAUTION: When using wire twisters and the wire extends 3 in. beyond the jaws of the twisters, loosely wrap the wire around the pliers to prevent whipping and possible personal injury. Excessive twisting of the wire will weaken the wire.

B. Grip the wire in the jaws of the wire twister, and slide the outer sleeve down with your thumb to lock the handles of the spring-loaded pin.

C. Pull the knob to make the spiral rod spin and twist the wire.

D. Squeeze handles together to release the wire.

4. Securing Oil Caps, Drain Cocks, And Valves

A. When securing oil caps and drain cocks, the safety wire should be anchored to an adjacent fillister-head screw. Refer to Figure 203. This method of safety wiring is applied to wing nuts, filler plugs, single-drilled head bolts, fillister-head screws, etc., that are safety wired individually. When securing valve handles in the vertical position, the wire is looped around the threads of the pipe leading into one side of the valve, double-twisted around the valve handle, and anchored around the threads of the pipe leading into the opposite side of the valve. When castle nuts are to be secured with safety wire, tighten the nut to the low side of the selected torque range, unless otherwise specified. If necessary, continue tightening until a slot lines with the hole. In blind tapped hole applications of bolts or castle nuts on studs, the safety wiring should be in accordance with the general instructions of this chapter. Hollow-head bolts are safetyed in the manner prescribed for regular bolts.

NOTE: Do not loosen or tighten properly tightened nuts to align safety-wire holes.

NOTE: Although there are numerous safety wiring techniques used to secure aircraft hardware, practically all are derived from the basic examples shown in this text.

5. Proper Securing Of Fittings

A. There are several examples on how to secure fittings listed in this section.

(1) Examples 1, 2, 3, and 4 apply to all types of bolts, fillister-head screws, square-head plugs, and other similar parts, which are wired so that tightening of the other part counteracts the loosening tendency of either part. Refer to Figure 204. The direction of twist from the second to the third unit is counterclockwise in Examples 1, 3, and 4 to keep the loop in position against the head of the bolt. The direction of twist from the second to the third unit in Example 2 is clockwise to keep the wire in position around the second unit. The wire entering the hole in the third unit will be the lower wire, except as in Example 2. By making a counterclockwise twist after it leaves the hole, the loop will be secured in place around the head of that bolt.

(2) Examples 5, 6, 7, and 8 show methods for wiring various standard items. Refer to Figure 205. Note that wire may be wrapped over the unit rather than around it when wiring castle nuts or on other items when there is a clearance problem.

(3) Example 9 shows the method for wiring bolts in different planes. Note that wire should always be applied so that tension is in the tightening direction. Hollow-head plugs shall be wired as shown with the tab bent inside the hole to avoid snags and possible injury to personnel working on the engine. Refer to Figure 206.

(4) Examples 12 and 13 show methods for attaching lead seal to protect critical adjustments. Refer to Figure 207.

(5) Example 14 shows bolt wired to a right-angle bracket with the wire wrapped around the bracket. Refer to Figure 208.

(6) Example 15 shows the correct method for wiring adjustable connecting rods. Refer to Figure 208.

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(7) Example 16 shows the correct method for wiring the coupling nut on flexible line to the straight connector brazed on rigid tube. Refer to Figure 208.

B. Fittings incorporating wire lugs shall be wired as shown in Examples 17 and 18. Where no lock-wire lug is provided, wire should be applied as shown in Examples 19 and 20 with caution being exerted to ensure that the wire is wrapped tightly around the fitting. Refer to Figure 209.

C. Small size coupling nuts shall be wired by wrapping the wire around the nut and inserting it through the holes as shown.

D. Coupling nuts attached to straight connectors shall be wired as shown when the hex is an integral part of the connector. Coupling nuts on a tee shall be wired as shown so that tension is always in the tightening direction.

6. Securing With Cotter Pins

A. Cotter pins are used to secure such items as bolts, screws, pins, and shafts. The use of cotter pins is favored because they can be removed and installed quickly. The diameter of the cotter pins selected for any application should be the largest size that will fit consistent with the diameter of the cotter pin hole and/or the slots in the nut. Cotter pins should not be re-used on aircraft.

B. To prevent injury during and after pin installation, the end of the cotter pin can be rolled and tucked, as shown in Figure 210 and Figure 211.

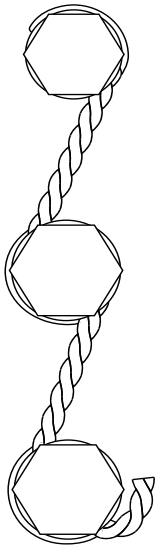
C. When tightening castellated nuts on bolts, begin by applying the minimum torque specified. If the cotter pin hole does not align with the slots in the nut, continue to tighten the nut until the next set of slots is aligned with the hole. In special cases where there is risk of the threads stripping or the bolt shearing, the maximum torque value specified may not be exceeded. For most installations, the nut may be over tightened to permit the slots to align with the hole.

NOTE: In using the method of cotter pin safetying ensure the prong, bent over the bolt, is seated firmly against the bolt shank and does not exceed bolt diameter. Also, when the prong is bent over the nut, ensure the bent prong is down and firmly flat against the nut and does not contact the surface of the washer.

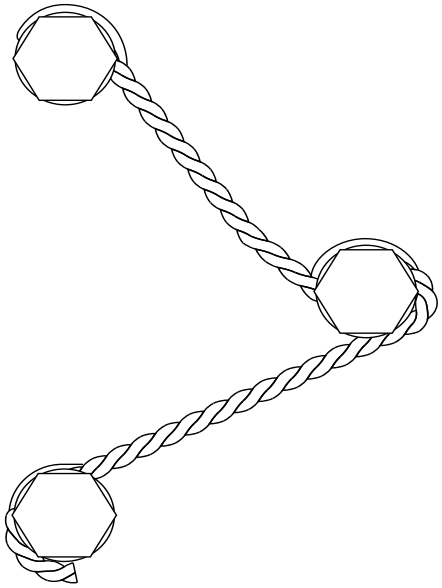
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Figure 201 : Sheet 1 : Examples of Safety Wiring Safetying

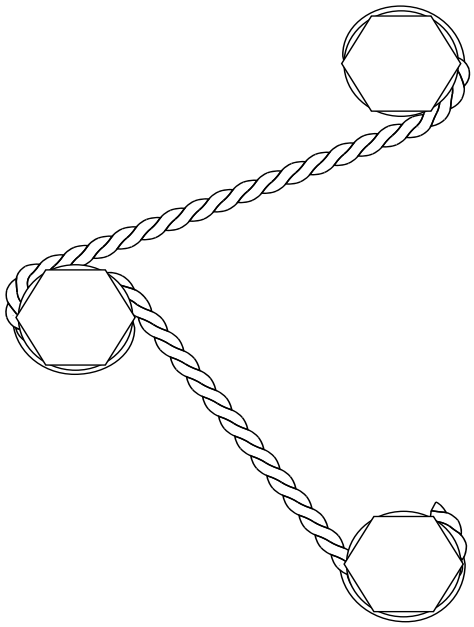
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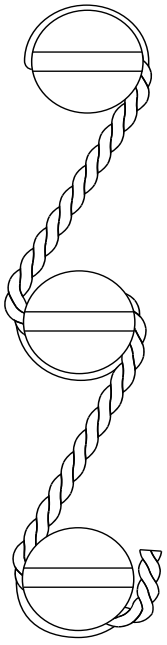
Example 1



Example 2



Example 3

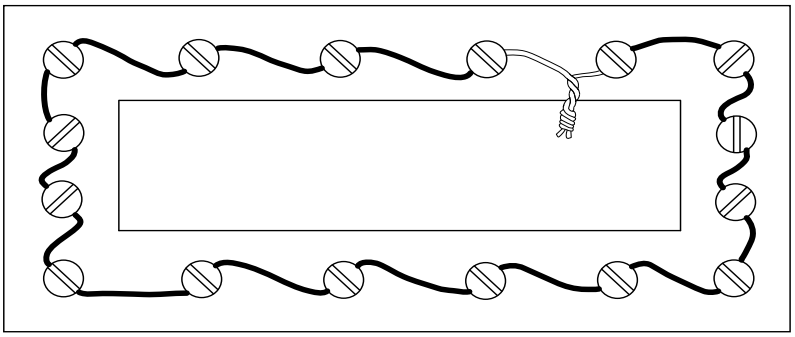


Example 4

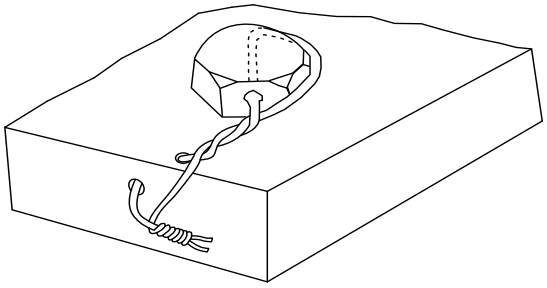
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Figure 202 : Sheet 1 : Securing Screws, Nuts, Bolts, and Snap Rings

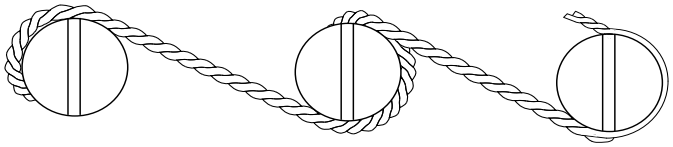
B11720



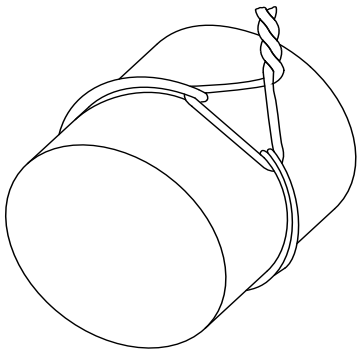
Small screws in closely spaced
Closed geometrical pattern
Single-wire method



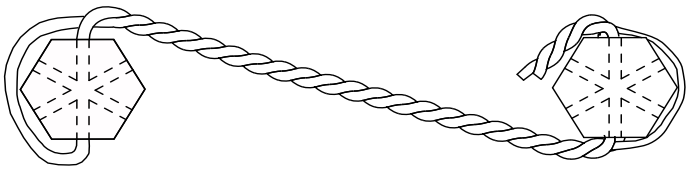
Single-Fastener Application
Double-Twist Method



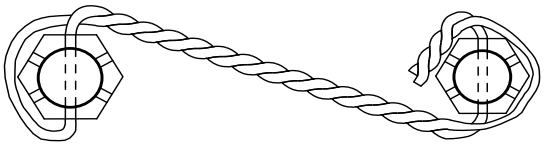
Screwheads
Double-Twist Method



External Snapring
Single-Wire Method



Boltheads



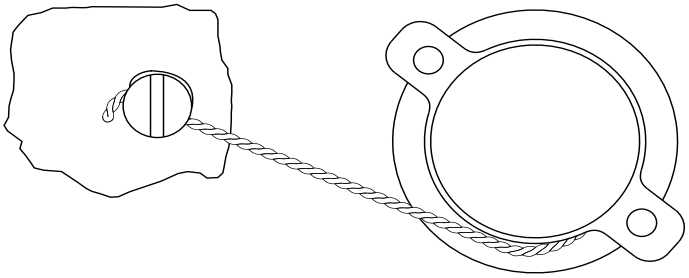
Castle Nuts

NOTE: The safety wire is shown installed for right-hand threads. The safety wire is routed in the opposite direction for left-hand threads.

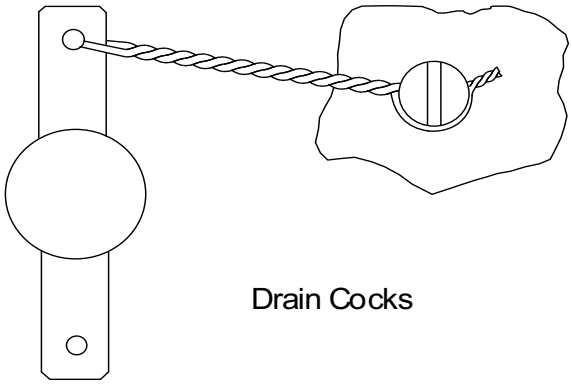
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Figure 203 : Sheet 1 : Securing Oil Caps, Drain Cocks, and Valves

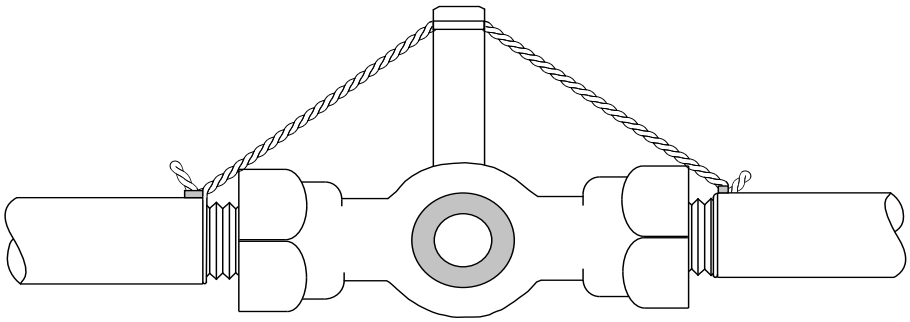
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Oil Caps



Drain Cocks



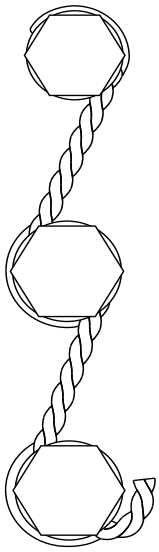
Valves

NOTE: The safety wire is shown installed for right-hand thread s. The safety wire is routed in the opposite direction for left-hand thread s.

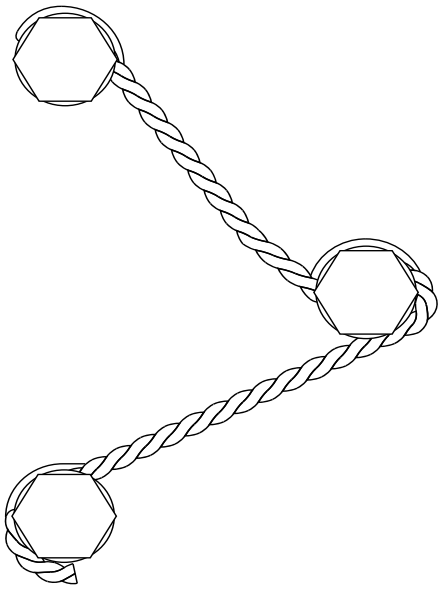
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Figure 204 : Sheet 1 : Securing Bolts, Screws, Plugs, etc.

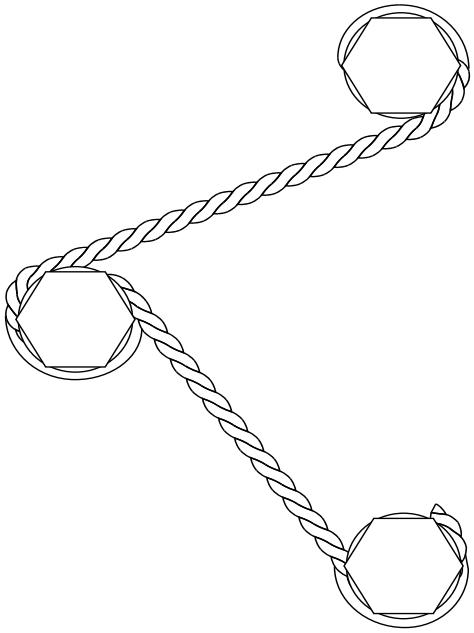
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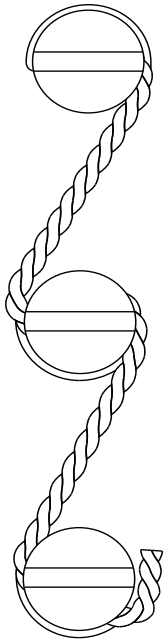
Example 1



Example 2



Example 3

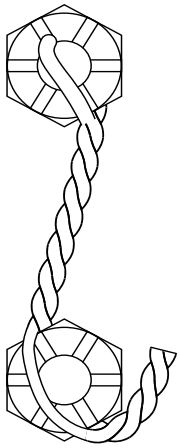


Example 4

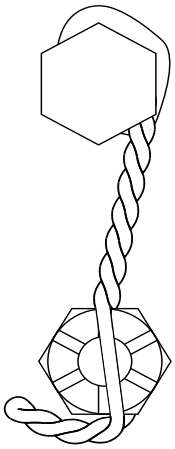
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Figure 205 : Sheet 1 : Securing Various Standard Items

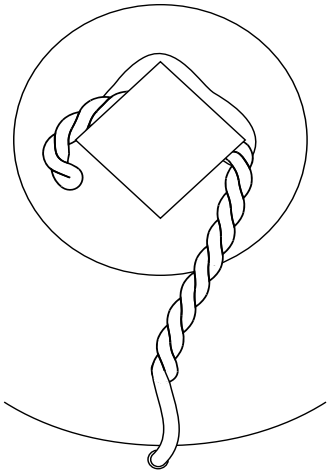
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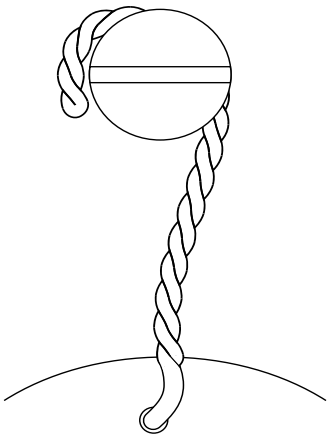
Example 5



Example 6



Example 7

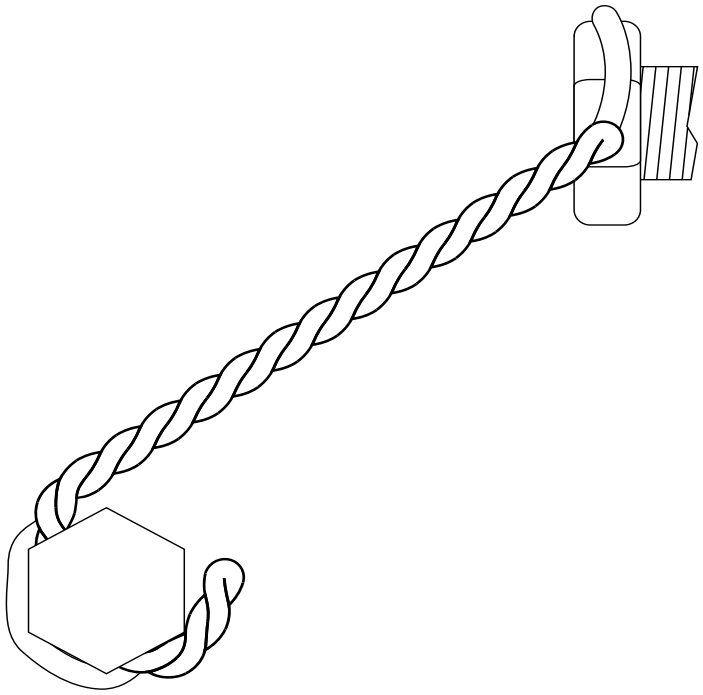


Example 8

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Figure 206 : Sheet 1 : Wiring Bolts in Different Planes

B11724

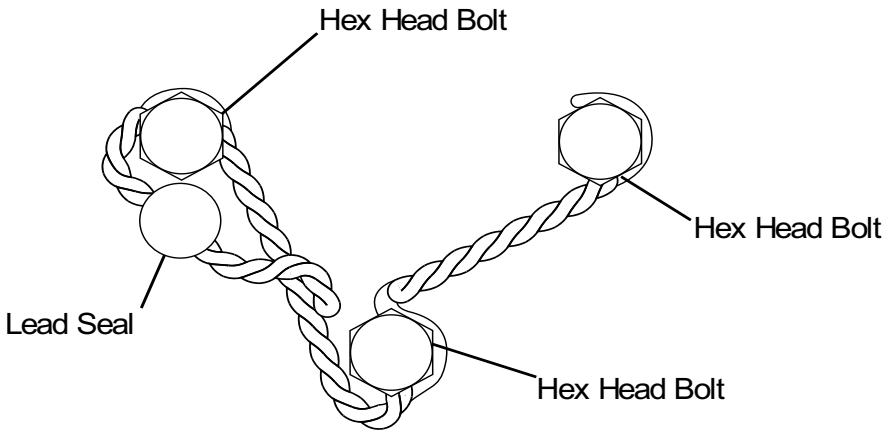


Example 9

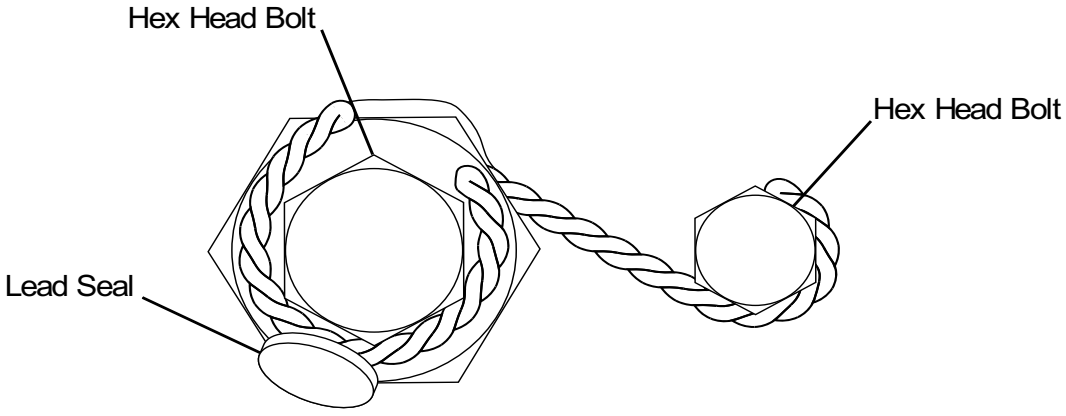
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Figure 207 : Sheet 1 : Attaching Lead Seal

B11745



Example 12

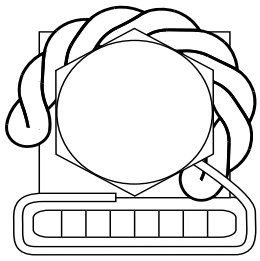


Example 13

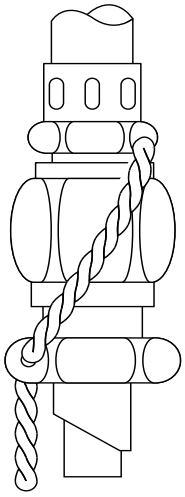
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Figure 208 : Sheet 1 : Wiring Brackets, Connecting Rods, Rigid Tubes

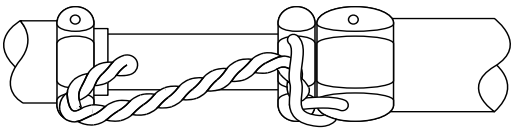
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Example 14



Example 16

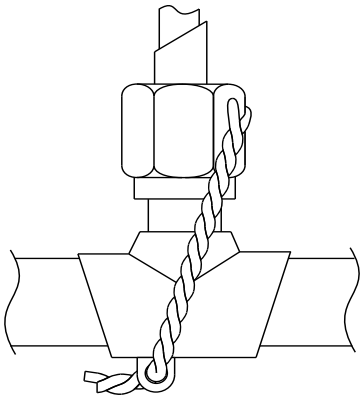


Example 15

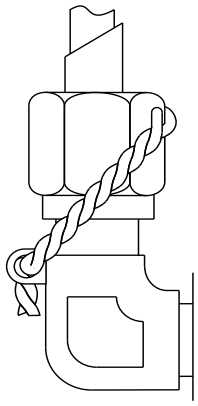
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Figure 209 : Sheet 1 : Fittings Incorporating Wire Lugs

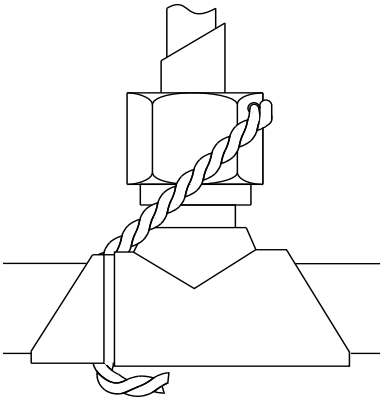
B11747



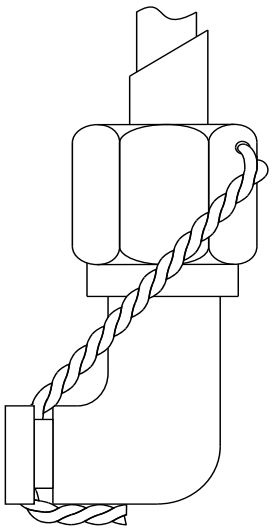
Example 17



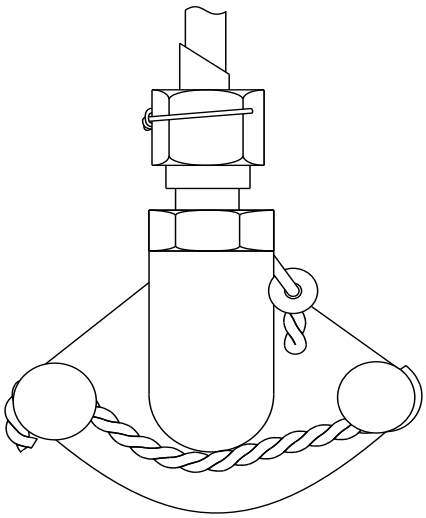
Example 18



Example 19



Example 20

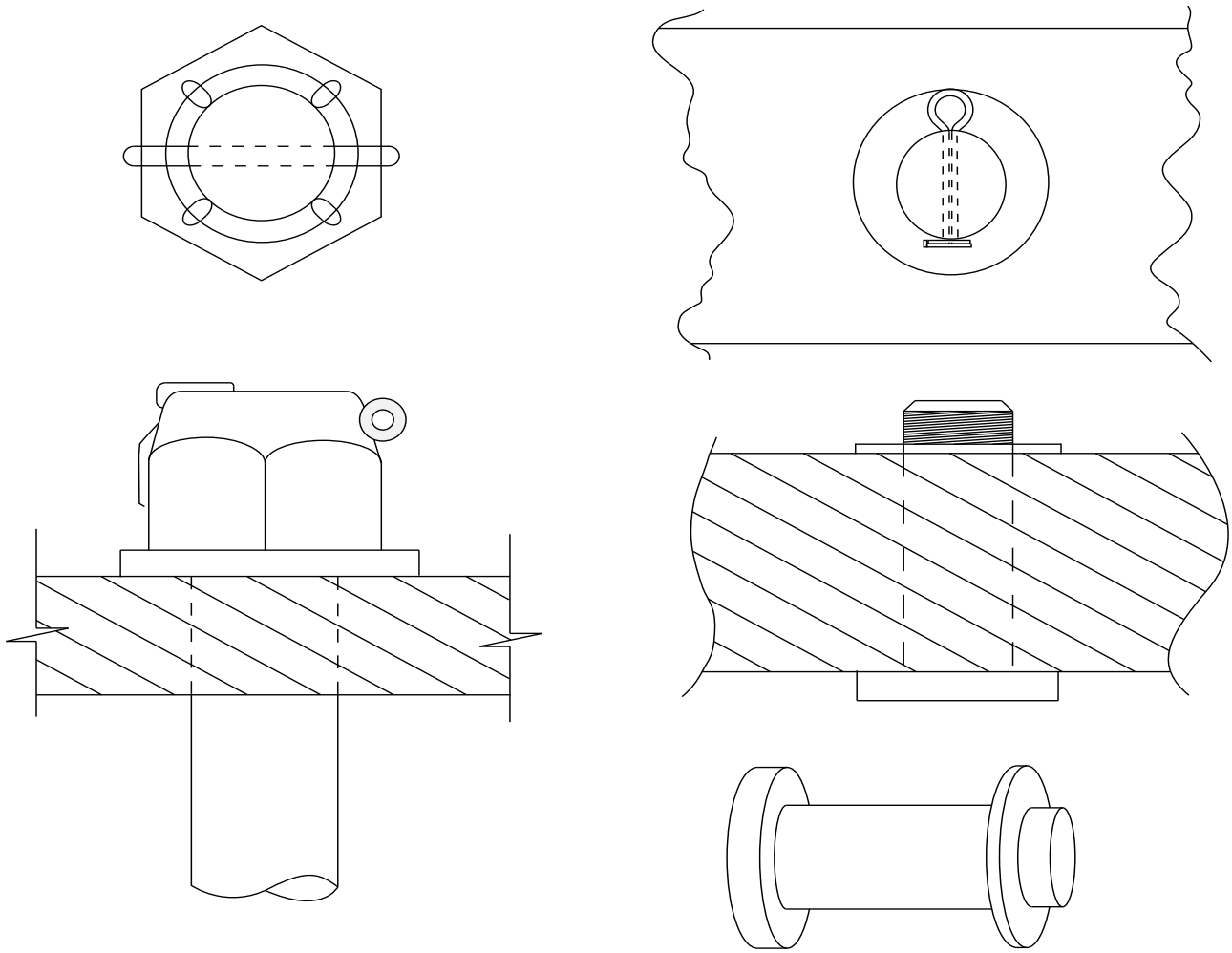


Example 21

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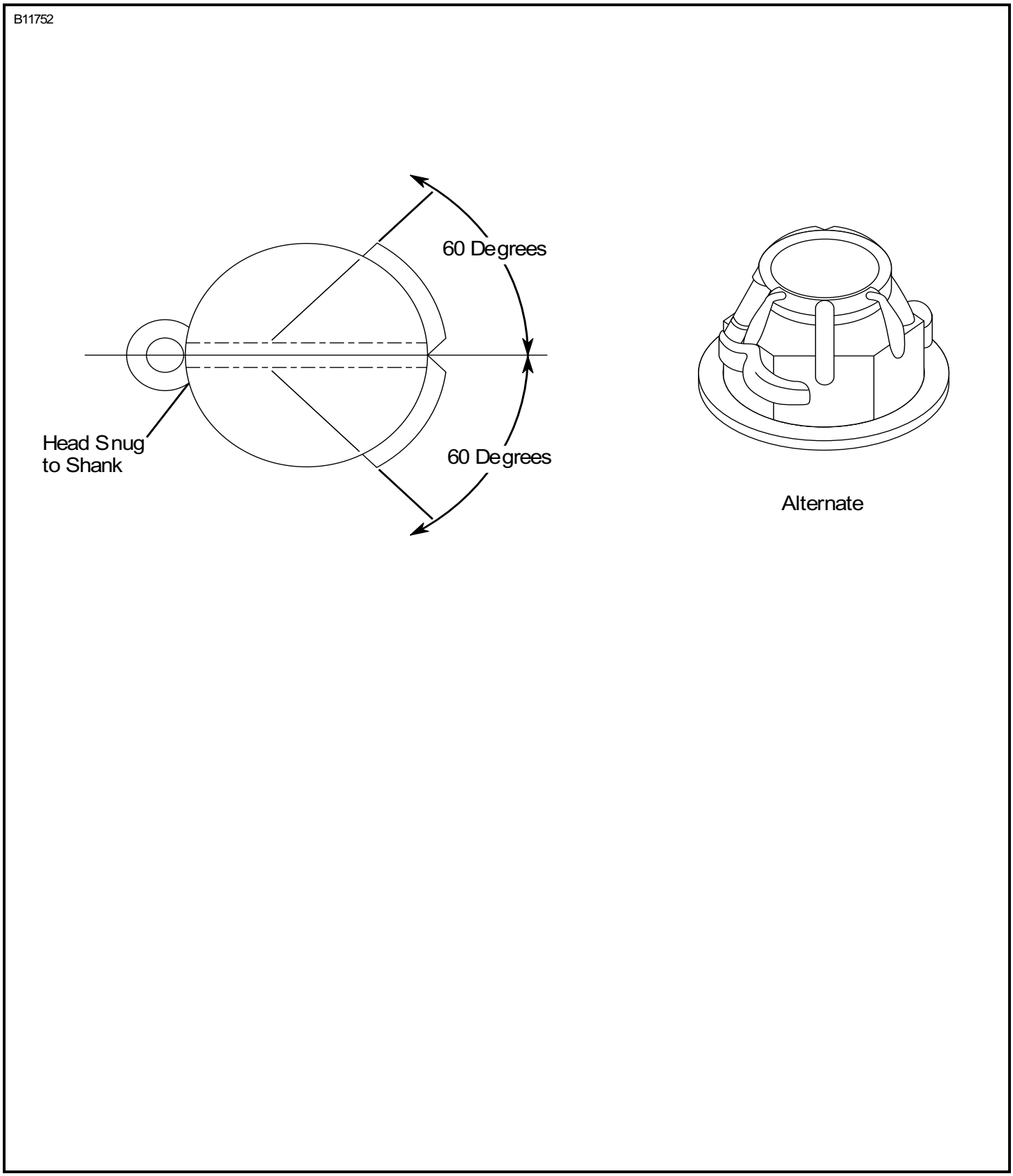
Figure 210 : Sheet 1 : Securing with Cotter Pins

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Figure 211 : Sheet 1 : Alternate Method for Securing with Cotter Pins



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CONTROL CABLES AND PULLEYS - MAINTENANCE PRACTICES

1. Control Cable Wire Breakage and Corrosion Limitations

A. Aircraft control cables in the Cessna 350 (LC42-550FG) are fabricated from corrosion-resistant steel wire of flexible-type construction. For example, 1/16 7 X 7 MIL-W-83420/2 and 1/8 in. 7 X 19 MIL-W-83420 cables are used in the rudder control circuit.

B. **Cable Definitions** - The following cable components are defined in accordance with Military Specifications MIL-W-83420, MIL-C-18375, and MIL-W-87161.

(1) Wire Center - The center of all strands shall be an individual wire and shall be designated as a wire center.

(2) Strand Center or Core - A strand center is a single, straight strand made of preformed wires, similar to the other strands comprising the cable in arrangement and number of wires.

(3) Independent Wire Rope Center (IWRC) 7 by 7 - A cable or wire rope of six strands of seven wires each, twisted or laid around a strand center or core consisting of seven wires.

C. **Flexible Cables** - Flexible, preformed, corrosion-resistant, Type I, composition B cables, MIL-W-87161, MIL-W-83420, and MIL-C-18375 are manufactured from steel made by the electric-furnace process. These cables are of the 3 by 7, 7 by 7, 7 by 19, or 6 by 19 IWRC construction, according to the diameter. The 3 by 7 cable consists of three strands of seven wires each. There is no core in this construction. The 7 by 7 cable consists of six strands, of seven wires each, laid around a center strand of seven wires. The wires are laid so as to develop a cable that has the greatest bending and wearing properties. The 7 by 7 cable has a length of lay of not more than eight times or less than six times the cable diameter. The 7 by 19 cable consists of six strands laid around a center strand in a clockwise direction. The wires composing the seven individual strands are laid around a center wire in two layers. The center core strand consists of a lay of six wires laid around the central wire in a clockwise direction and a layer of 12 wires laid around this in a clockwise direction. The six outer strands of the cable consist of a layer of six wires laid around the center wire in a counterclockwise direction and a layer of 12 wires laid around this in a counterclockwise direction.

D. **Cable Replacement** - Replace control cables when they become worn, distorted, corroded, or otherwise damaged. If spare cables are not available, prepare exact duplicates of the damaged cable. Use materials of the same size and quality as the original. Standard swaged cable terminals develop the full cable strength and may be substituted for the original terminals wherever practical.

E. **Mechanically Fabricated Cable Assemblies**

(1) **Swage-Type Terminals** - Swage-type terminals, manufactured in accordance with AN, are suitable for use in civil aircraft up to, and including, maximum cable loads. When swaging tools are used, it is important that all the manufacturers' instructions, including "go and no-go" dimensions, be followed in detail to avoid defective and inferior swaging. Observance of all instructions should result in a terminal developing the full-rated strength of the cable. Critical dimensions, both before and after swaging, must be checked.

(2) When swaging terminals onto cable ends, observe the following procedures:

(a) Cut the cable to the proper length allowing for growth during swaging. Apply a preservative compound to the cable ends before insertion into the terminal barrel.

NOTE: Never solder cable ends to prevent fraying because the presence of the solder will greatly increase the tendency of the cable to pull out of the terminal.

(b) Insert the cable into the terminal approximately 1 inch and bend toward the terminal. Push the cable end entirely into the terminal barrel. The bending actions puts a kink or bend in the cable end and provides enough friction to hold the terminal in place until the swaging operation can be performed. Bending also tends to separate the strands inside the barrel, thereby reducing the strain on them.

Table 201. Straight-Shank Terminal Dimensions

(Cross reference AN to MS: AN-666 to MS 21259, AN-667 to MS 20667, AN-668 to MS 20668, AN-669 to MS 21260.)							
Key: A = Cable size (inches) B = Wire strands C = Outside diameter before swaging D = Bore diameter before swaging E = Bore length before swaging F = Swaging length before swaging G = Minimum breaking strength (pounds) after swaging H = Shank diameter after swaging - Note: Use gauges in kit for checking diameters.							
A	B	C	D	E	F	G	H

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1/16	7X7	0.160	0.078	1.042	0.969	480	0.138
3/32	7X7	0.218	0.109	1.261	1.188	920	0.190
1/8	7X19	0.250	0.141	1.511	1.438	2,000	0.219
5/32	7X19	0.297	0.172	1.761	1.688	2,800	0.250
3/16	7X19	0.359	0.203	2.011	1.938	4,200	0.313
7/32	7X19	0.427	0.234	2.261	2.188	5,600	0.375
1/4	7X19	0.494	0.265	2.511	2.438	7,000	0.438
9/32	7X19	0.563	0.297	2.761	2.688	8,000	0.500
5/16	7X19	0.635	0.328	3.011	2.938	9,800	0.563
3/8	7X19	0.703	0.390	3.510	3.438	14,400	0.625

(c) Accomplish the swaging operation in accordance with the instructions furnished by the manufacturer of the swaging equipment.

(d) Inspect the terminal after swaging to determine that it is free from die marks and splits and is not out-of-round. Check for cable slippage in the terminal and for cut or broken wire strands.

(e) Using a go no-go gauge or a micrometer, check the terminal shank diameter.

(3) **Nicopress Process** - A patented process using copper sleeves may be used up to the full rated strength of the cable when the cable is looped around a thimble.

(a) Before undertaking a Nicopress splice, determine the proper tool and sleeve for the cable to be used. Refer to proper references for details on sleeves, tools, and the number of presses required for the various sizes of aircraft cable. The tool must be in good working condition and properly adjusted to ensure a satisfactory splice.

(b) To compress a sleeve, have it well-centered in the tool groove with the major axis of the sleeve at right angles to the tool. If the sleeve appears to be out of line after the press is started, open the tool, re-center the sleeve, and complete the press.

(4) **Lap Splice** - Lap or running splices may also be made with copper oval sleeves. When making such splices, it is usually necessary to use two sleeves to develop the full strength of the cable. The sleeves should be positioned as shown in Figure 201, and the compressions made in the order shown. As in the case of eye splices (refer to Figure 202), it is desirable to have the cable ends extend beyond the sleeves sufficiently to allow for the increased length of the compressed sleeves.

F. **Terminal Gauge** - To make a satisfactory copper sleeve installation, it is important that the amount of sleeve pressure be kept uniform. The completed sleeves should be checked periodically with the proper gauge. Hold the gauge so that it contacts the major axis of the sleeve. The compressed portion at the center of the sleeve should enter the gauge opening with very little clearance, as shown in Figure 203. If it does not, the tool must be adjusted accordingly.

2. Cable System Inspection

A. Aircraft cable systems are subject to a variety of environmental conditions and deterioration. Wire or strand breakage is easy to visually recognize. Other kinds of deterioration such as wear, corrosion, and/or distortion are not easily seen; therefore, control cables should be removed periodically for a more detailed inspection.

B. At each annual or 100 hour inspection, all control cables must be inspected for broken wire strands. Any cable assembly that has one broken wire strand located in a critical fatigue area must be replaced.

C. A critical fatigue area is defined as the working length of a cable where the cable runs over, under, or around a pulley, sleeve, or through a fair-lead; or any section where the cable is flexed, rubbed, or worked in any manner; or any point within 1 ft. of a swaged-on fitting.

D. A swaged-on fitting can be an eye, fork, ball, ball and shank, ball and double shank, threaded stud, threaded stud and turnbuckle, compression sleeve, or any hardware used as a termination or end fitting on the cable. These fittings may be attached by various swaging methods such as rotary swaging, roll swaging, hydraulic pressing, and hand swaging tools. (See MIL-T-781.) The pressures exerted on the fittings during the swaging process sometimes pinch the small wires in the cable. This can cause premature failure of the pinched wires, resulting in broken wires.

E. Close inspection in these critical fatigue areas must be made by passing a cloth over the area that will snag on broken wires. This will clean the cable for a visual inspection, and detect broken wires if the cloth snags on the cable. Also, a very

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careful visual inspection must be made since a broken wire will not always protrude or stick out, but may lie in the strand and remain in the position of the helix as it was manufactured. Broken wires of this type may show up as a hairline crack in the wire. If a broken wire of this type is suspected, further inspection with a magnifying glass of 7 power or greater, is recommended. Figure 204 shows a cable with broken wires that was not detected by wiping, but was found during a visual inspection. The damage became readily apparent when the cable was removed and bent as shown in the figure below.

F. Kinking of wire cable can be avoided if properly handled and installed. A cable taking a spiral shape as the result of unnatural twist causes kinking. One of the most common causes for this twist is improper unreeling and uncoiling. In a kinked cable, strands and wires are out of position, which creates unequal tension and brings excessive wear at this part of the cable. Even though the kink may be straightened so that the damage appears to be slight, the relative adjustment between the strands has been disturbed so that the cable cannot give maximum service and should be replaced. Inspect cables for a popped core or loose strands. Replace any cable that has a popped core or loose strands regardless of wear or broken wires.

G. External wear patterns will extend along the cable equal to the distance the cable moves at that location and may occur on one side of the cable or on its entire circumference. Replace flexible and non-flexible cables when the individual wires in each strand appear to blend together (outer wires worn 40 to 50 percent).

H. As wear is taking place on the exterior surface of a cable, the same condition is taking place internally, particularly in the sections of the cable which pass over pulleys and quadrants. This condition is not easily detected unless the strands of the cable are separated. This type of wear is a result of the relative motion between inner wire surfaces. Under certain conditions, the rate of this type of wear can be greater than that occurring on the surface.

I. Carefully examine any cable for corrosion, when it has a broken wire in a section that is not in contact with a wear-producing airframe component, such as a pulley, fair-lead, etc. If the surface of the cable is corroded, relieve cable tension and carefully force the cable open by reverse twisting, and visually inspect the interior. Corrosion on the interior strands of the cable constitutes failure, and the cable must be replaced. If no internal corrosion is detected, remove loose external rust and corrosion with a clean, dry, coarse-weave rag, or fiber brush. Do not use metallic wool or solvents to clean installed cables. Use of metallic wool will embed dissimilar metal particles in the cables and create further corrosion problems. Solvents will remove internal cable lubricant allowing cable strands to abrade and further corrode. After thorough cleaning, sparingly apply specification MIL-C-16173, grade 4, corrosion-preventive compound to the cable. Do not apply the material so thick that it will interfere with the operation of cables at fair-leads, pulleys, or grooved bellcrank areas.

J. Examine cable runs for incorrect routing, fraying, twisting, or wear at fair-leads, pulleys, anti-abrasion strips, and guards. Look for interference with adjacent structure, equipment, wiring, plumbing, and other controls. Inspect cable systems for binding, full travel, and security of attaching hardware. Check for slack in the cable system. Actuate the controls, and check for friction or hard movement that are indications that excessive cable tension exists.

NOTE: If the control movement is stiff after maintenance has been performed on the control surfaces, check for parallel cables twisted around each other or cables connected in reverse.

K. Check swaged terminal reference marks for an indication of cable slippage within the fitting. Inspect the fitting assembly for distortion and/or broken strands at the terminal. Ensure that all bearings and swivel fittings (bolted or pinned) pivot freely to prevent binding and subsequent failure. Check turnbuckles for proper thread exposure and broken or missing safety wires/clips.

L. Inspect pulleys for roughness, sharp edges, and presence of foreign material embedded in the grooves. Examine pulley bearings to ensure proper lubrication, smooth rotation, and freedom from flat spots, dirt, and paint spray. During the inspection, rotate the pulleys, which only turn through a small arc, to provide a new bearing surface for the cable. Maintain pulley alignment to prevent the cable from riding on the flanges and chafing against guards, covers, or adjacent structure. Check all pulley brackets and guards for damage, alignment, and security.

M. Various cable system malfunctions may be detected by analyzing pulley conditions. These include such discrepancies as too much tension, misalignment, pulley bearing problems, and size mismatches between cables and pulleys. Examples of these conditions are shown in Figure 205.

N. Inspect fair-leads for wear, breakage, alignment, cleanliness, and security. Examine cable routing at fair-leads to ensure that deflection angles are no greater than 3 degrees maximum. Determine that all guides and anti-abrasion strips are secure and in good condition.

O. Examine pressure seals for wear and material deterioration. Seal guards should be positioned to prevent jamming of a pulley in case the pressure seal fails and pieces slide along the cable.

3. Corrosion And Rust Prevention

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A. To ensure a satisfactory service life for aircraft control cables, use a cable lubricant to reduce internal friction and prevent corrosion.

B. Coat the cable with rust-preventive oil, and wipe off any excess. Lubrication and corrosion preventive treatment of cables may be effected simultaneously by application of compound MIL-C-16173, grade 4. MIL-C-16173 compound should be brushed, sprayed, or wiped on the cable to the extent it penetrates into the strands and adequately covers the cable surfaces. It will dry tack-free in 24 hours at 77° Fahrenheit.

4. Cable Maintenance

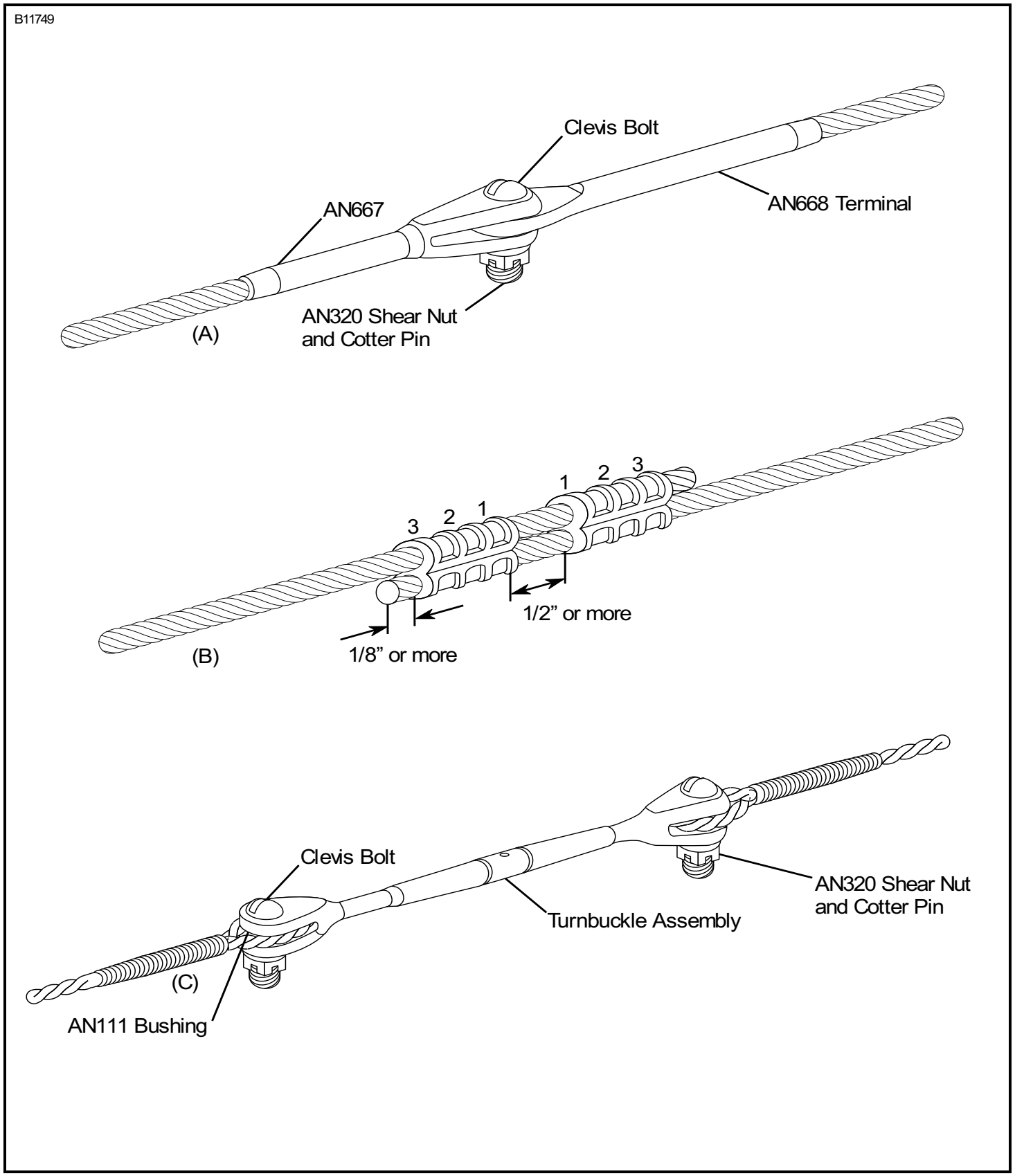
A. Frequent inspections and preservation measures such as rust-prevention treatments, will help to extend cable service life. Where cables pass through fair-leads or over pulleys, remove any accumulated heavy coatings of corrosion-prevention compound. Provide corrosion protection for these cable sections by lubricating with a light coat of grease or general-purpose, low-temperature oil.

5. Cable Tenison Adjustment

A. Carefully adjust the control cable tension in accordance with the airframe requirements. If necessary, compensate for extreme surface temperature variations that may be encountered if the aircraft is operated primarily in unusual geographic or climatic conditions such as arctic, arid, or tropical locations. Use rigging pins and gust locks, as necessary, to ensure satisfactory results. At the completion of rigging operations, check turnbuckle adjustment and safetying in accordance with standard practices and information in this chapter.

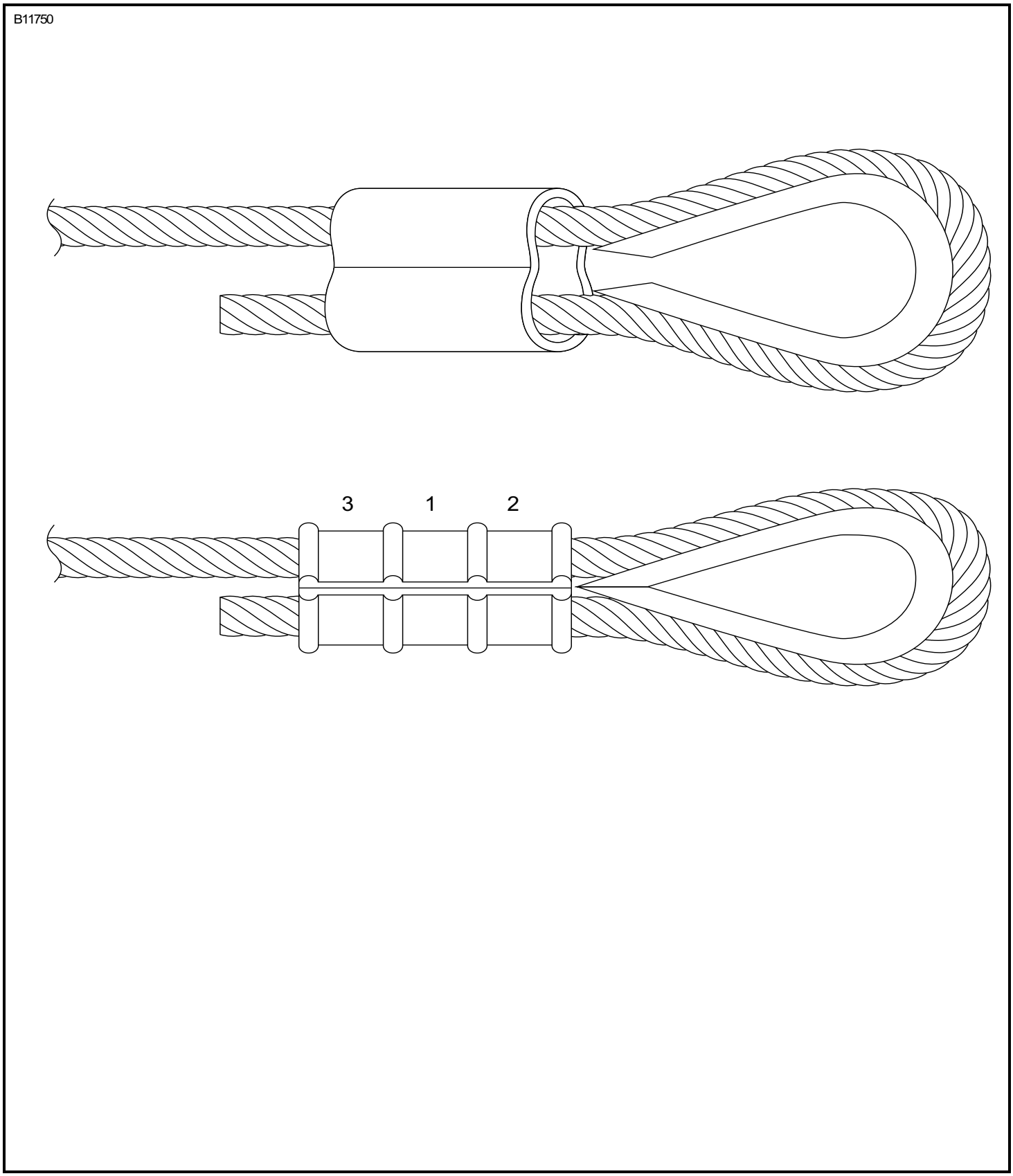
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Figure 201 : Sheet 1 : Lap Splices



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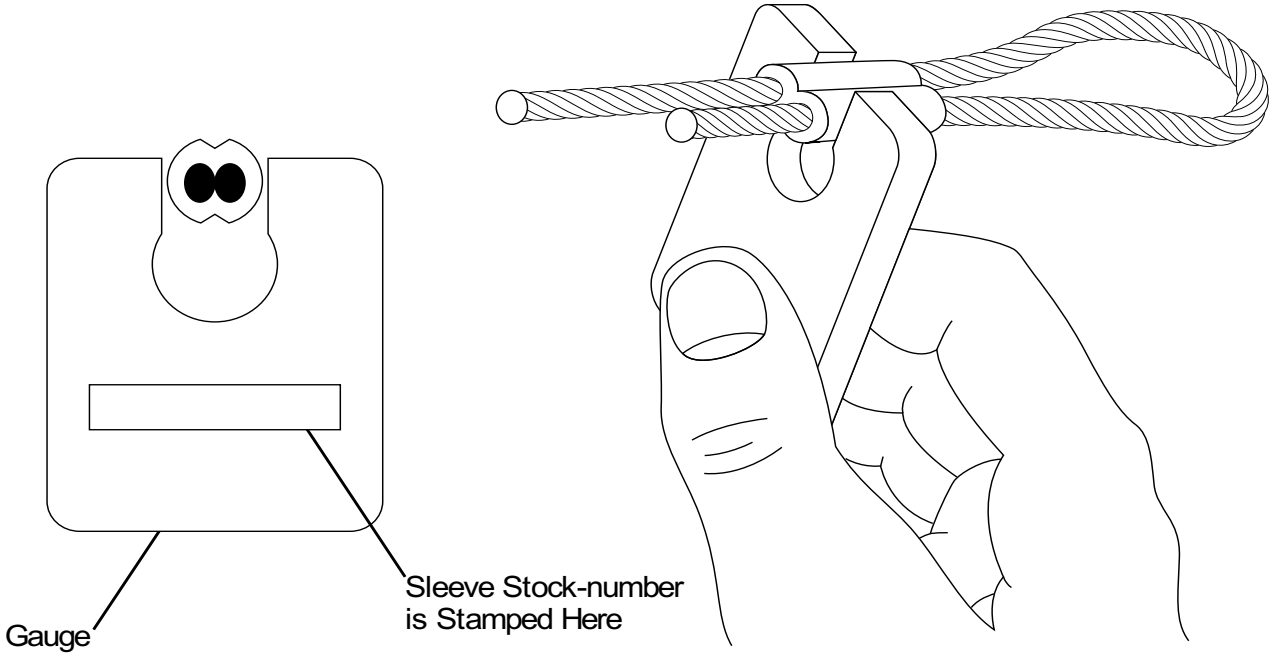
Figure 202 : Sheet 1 : Typical Thimble-Eye Splice



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Figure 203 : Sheet 1 : Typical Terminal Gauge

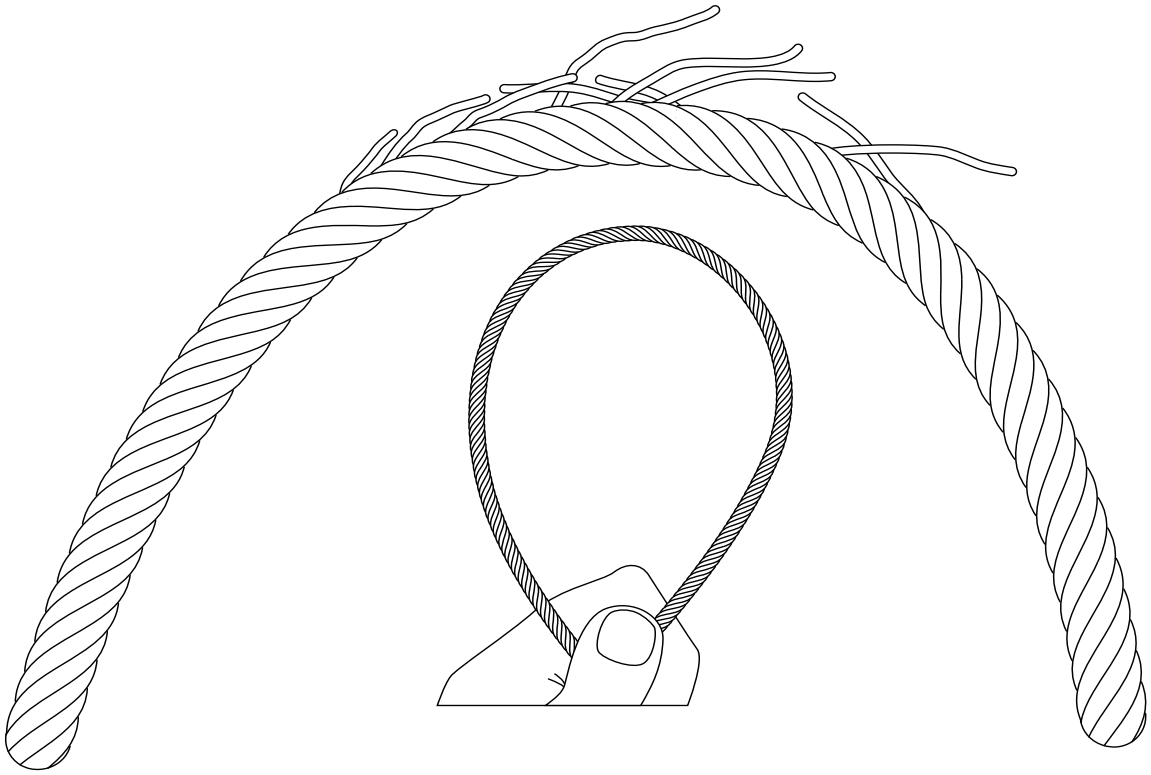
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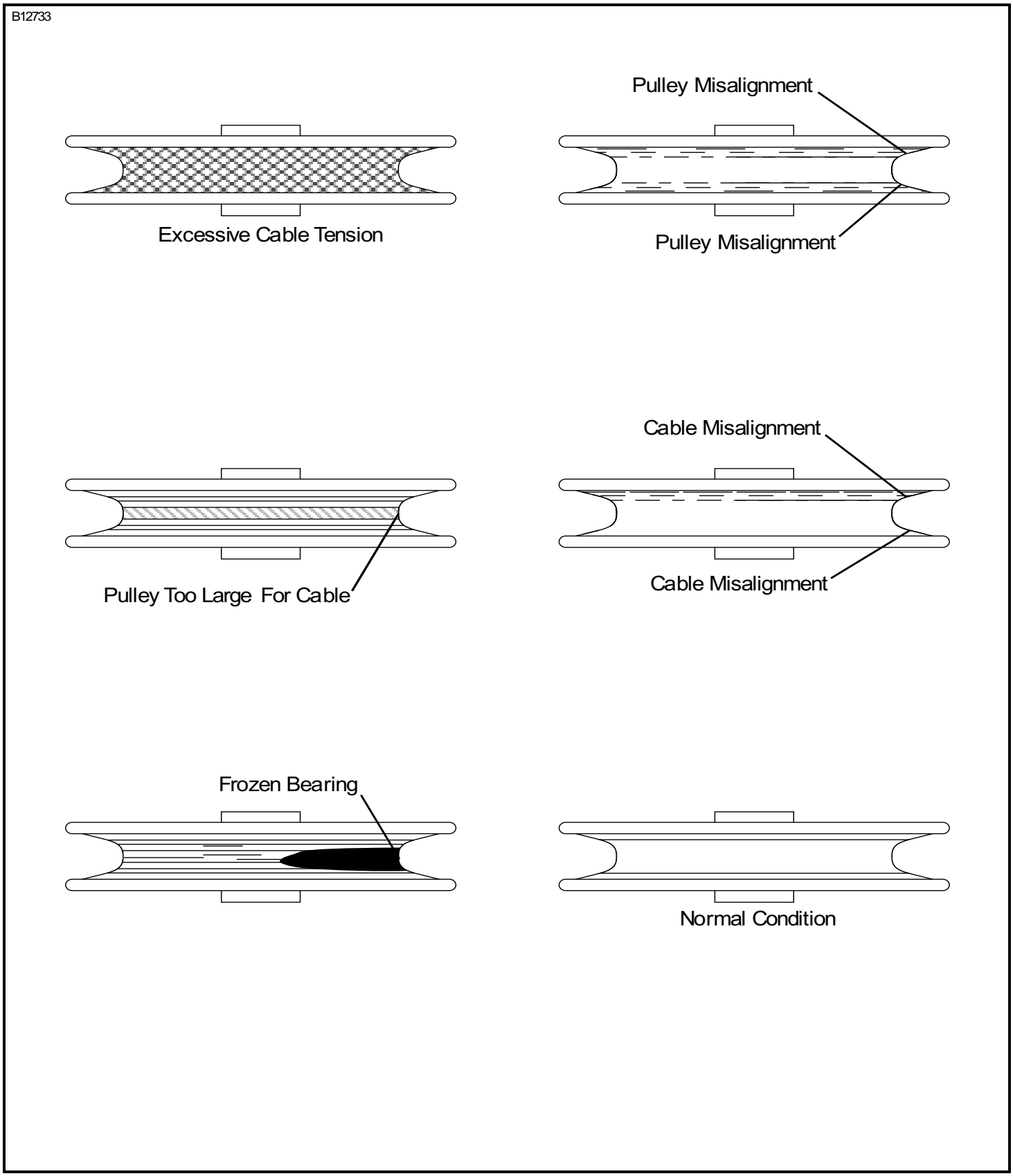
Figure 204 : Sheet 1 : Cable Inspection Technique

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Figure 205 : Sheet 1 : Pulley Wear Patterns



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TURNBUCKLES - MAINTENANCE PRACTICES

1. Turnbuckles

A. A turnbuckle is a device used in cable systems to provide a means of adjusting tension. Turnbuckles have barrel-shaped sleeves with internal left- and right-hand threads at opposite ends. The cables, with terminals attached, are made to such a length that, when the turnbuckle is adjusted to give the specified cable tension, a sufficient number of threads on the terminal ends are screwed into the barrel to hold the load. The clip-locking turnbuckle and its associated parts are identical to standard AN and MS parts except for a slot grooved on the interior of the barrel and the shanks of the forks, eyes, etc. The clip-locking turnbuckle parts have the following drawing numbers:

- MS21251, turnbuckle body
- MS21252, turnbuckle clevis end
- MS21253, turnbuckle clevis end (for bearing)
- NAS649 and NAS651, turnbuckle clip
- MS21254 and NAS648, turnbuckle eye (for pin)
- MS21255 and NAS647, turnbuckle eye end (for wire rope)
- NAS645 and NAS646, turnbuckle fork
- MS21256, turnbuckle barrel locking clip
- AN130-170, turnbuckle assemblies
- MS21259 and MS21260, terminal, wire rope, stud

NOTE: Turnbuckles showing signs of thread distortion or bending should be replaced.

Turnbuckle ends are designed to provide the specified cable tension on a cable system. A bent turnbuckle would place undesirable stress on the cable, impairing the function of the turnbuckle.

2. Turnbuckle Installation

A. When installing cable system turnbuckles, it is necessary to screw both threaded terminals into the turnbuckle barrel an equal amount. It is essential that turnbuckle terminals be screwed into the barrel so that not more than three threads on the terminal are exposed. On initial installation, the turnbuckle terminals should not be screwed inside the turnbuckle barrel more than four threads.

3. Witness Hole

A. Some manufacturers of turnbuckles incorporate a witness hole in the turnbuckle barrel to ensure that the threaded cable terminals are screwed far enough into the barrel. The witness hole can be inspected visually or by using a piece of safety wire as a probe.

4. Safety Methods for Turnbuckles

A. Safety all turnbuckles with safety wire using either the double or single-wrap method or with any appropriately approved special safetying device complying with the requirements of FAA Technical Standard Order TSO-C21. The swaged and unswaged turnbuckle assemblies are covered by AN standard drawings. Do not reuse safety wire. Adjust the turnbuckle to the correct cable tension so that no more than three cable threads are exposed on either side of the turnbuckle barrel.

B. **Double-Wrap Method** - Of the methods using safety wire for safetying turnbuckles, the method described here is preferred, although either of the other methods described is satisfactory. The method of double-wrap safetying is shown in Figure 201.

(1) Use two separate lengths of wire. Run one end of the wire through the hole in the barrel of the turnbuckle, and bend the ends of the wire toward opposite ends of the turnbuckle.

(2) Pass the second length of the wire into the hole in the barrel, and bend the ends along the barrel on the side opposite the first. Spiral the two wires in opposite directions around the barrel to cross each other twice between the center hole and the ends.

(3) Pass the wires at the end of the turnbuckle in opposite directions through the hole in the turnbuckle eyes or between the jaws of the turnbuckle fork, as applicable, laying one wire along the barrel and wrapping the other at least four times around the shank of the turnbuckle and binding the laid wires in place before cutting the wrapped wire off.

(4) Wrap the remaining length of safety wire at least four turns around the shank, and cut it off. Repeat the procedure at the opposite end of the turnbuckle.

(5) When a swaged terminal is being safety wired, pass the ends of both wires through the hole provided in the terminal for this purpose, and wrap both ends around the shank as previously described. If the hole is not large enough to allow passage of both wires, pass the wire through the hole, and loop it over the free end of the other wire, and then

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wrap both ends around the shank as previously described. Another satisfactory double-wrap method is similar to the previous method, except that the spiraling of the wires is omitted as shown in Figure 201.

C. **Single-Wrap Method** - The single-wrap methods described in the following paragraphs and as illustrated in Figure 201 are acceptable but are not the equal of the double-wrap methods.

(1) **Single-Wrap (Spiral)** - Pass a single length of wire through the cable eye or fork or through the hole in the swaged terminal at either end of the turnbuckle assembly. Spiral each of the wire ends in opposite directions around the first half of the turnbuckle barrel, so as to cross each other twice. Thread both wire ends through the hole in the middle of the barrel so that the third crossing of wire ends is in the hole. Spiral the two wire ends in opposite directions around the remaining half of the turnbuckle, crossing them twice. Pass one wire end through the cable eye or fork or through the hole in the swaged terminals, in the manner previously described. Wrap both wire ends around the shank at least four turns each, cutting off excess wire. This method is shown in view (C), Figure 201.

(2) **Single-Wrap** - For the method shown in view (D), Figure 201, pass one length of wire through the center hole of the turnbuckle, and bend the wire ends toward opposite ends of the turnbuckle. Pass each wire end through the cable eye or fork or through the hole in the swaged terminal, and wrap each wire around the shank for at least four turns, cutting off excess wire. After safetying, no more than three threads of the turnbuckle threaded terminal should be exposed.

5. Secured Safety-Wired Turnbuckles

A. Before securing turnbuckles, threaded terminals should be screwed into the turnbuckle barrel until no more than three threads of either terminal are outside the barrel. After the turnbuckle has been adjusted for proper cable tension, two pieces of safety wire are inserted with half the wire length into the hole in the center of the turnbuckle barrel. The safety wires are bent so that each wire extends half the length of the turnbuckle on top and half on bottom. The ends of the wires are passed through the hole in the turnbuckle eyes or between the jaws of the turnbuckle fork, as applicable. The wires are then bent toward the center of the turnbuckle, and each wire is wrapped around the shank four times, binding the wrapping wires in place as shown in Figure 202.

B. When a swaged terminal is being secured, one wire is passed through the hole in the terminal and is looped over the free end of the other wire and both ends wrapped around the shank. All lock wire used in the safetying of turnbuckles should be carbon steel, corrosion-resistant steel, nickel-chromium iron alloy (Inconel), nickel-copper alloy (Monel), or aluminum alloy. For safety cable diameter of safety wire size and material, refer to Table 201.

C. Care should be exercised when safety wiring, particularly where corrosion will present a problem, because smaller wire sizes tend to crack when twisted.

Table 201. Turnbuckle Safetying Guide

Cable Size	Type of Wrap	Diameter of Safety Wire	Material (Annealed Condition)
1/16	Single	0.040	Copper, brass.(See Note)
3/32	Single	0.040	Copper, brass.(See Note)
1/8	Single	0.040	Stainless steel, Monel and "K" Monel.
1/8	Double	0.040	Copper, brass.(See Note)
1/8	Single	0.057 min.	Copper, brass.(See Note)
5/32 and greater	Double	0.040	Stainless steel, Monel and "K" Monel.
5/32 and greater	Single	0.057 min.	Stainless steel, Monel and "K" Monel.
5/32 and greater	Double	0.0512	Copper, brass
NOTE: **Galvanized or tinned steel, or soft iron wires are also acceptable.**			

6. Special Locking Devices

A. Several turnbuckle-locking devices are available for securing turnbuckle barrels such as wire-locking clips. Persons intending to use a special device must ensure that the turnbuckle assembly has been designed to accommodate such devices. A typical unit is shown in Figure 203. When special locking devices are not readily available, the use of safety wire is acceptable.

B. **Assembling and Securing Clip-Lock Turnbuckles** - Wire clip-locking turnbuckles are assembled and secured in the

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following ways (Refer to Figure 204):

(1) Engage the threads of the turnbuckle barrel with the threads of the cable terminal, and turn the barrel until proper cable tension is reached.

(2) Align the slot in the barrel with the slot in the cable terminal.

(3) Hold the lock clip between the thumb and forefinger at the loop end, and insert the straight end of the clip into the opening formed by the aligned slots. Refer to Table 202.

(4) Bring the hook end of the lock clip over the hole in the center of the turnbuckle barrel, and seat the hook loop into the hole.

(5) Apply pressure to the hook shoulder to engage the hook lip in the turnbuckle barrel and to complete safety locking of one end of the turnbuckle.

NOTE: Repeat the above steps to safety lock the opposite end of the turnbuckle. Either lock clips may be inserted in the same turnbuckle barrel hole or they may be inserted in opposite holes. However, do not reverse-wire locking clips.

Table 202. Locking-Clip Application

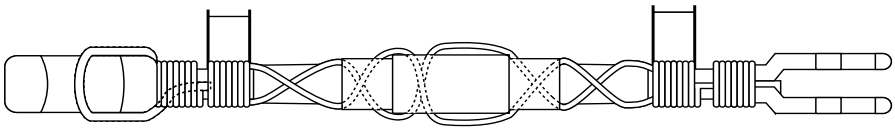
Nominal Cable Diameter	Thread Unf-3tech	Locking Clip MS21256	Turnbuckle Body MS21251
1/16	No. 6-40	-1	-2S
1/8	1/4-28	-1	-4S
1/8	1/4-28	-2	-4L

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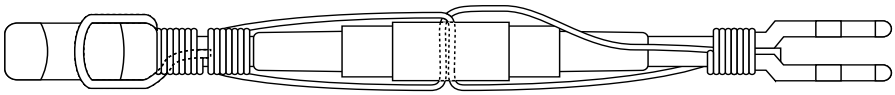
Figure 201 : Sheet 1 : Safetying Turnbuckles

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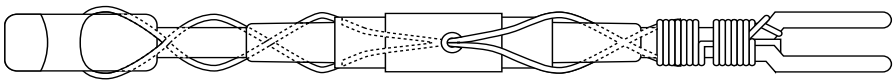
This Applies to All Turnbuckle Wrappings



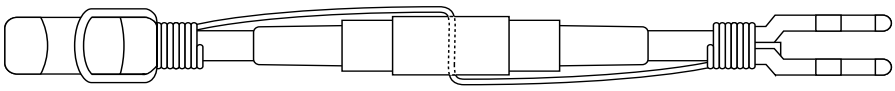
(A) Double Wrap (Spiral)



(B) Double Wrap



(C) Single Wrap (Spiral)

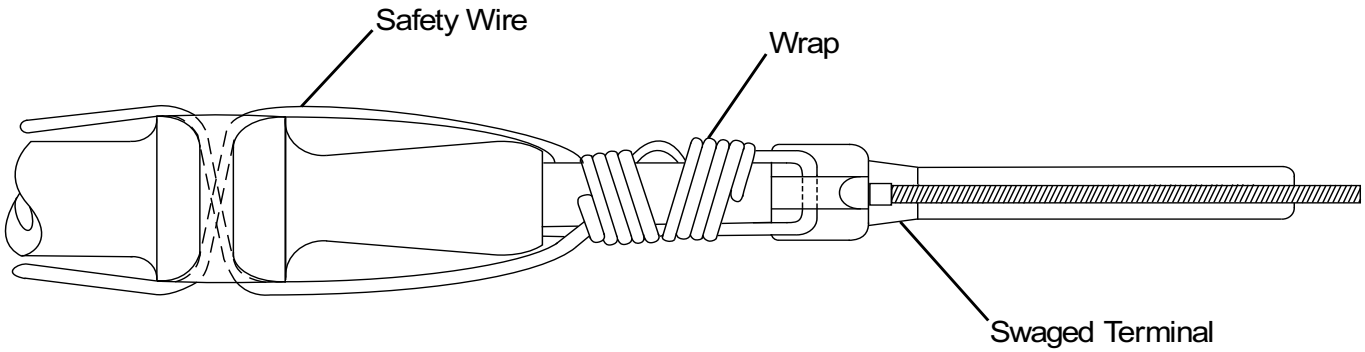


(D) Single Wrap

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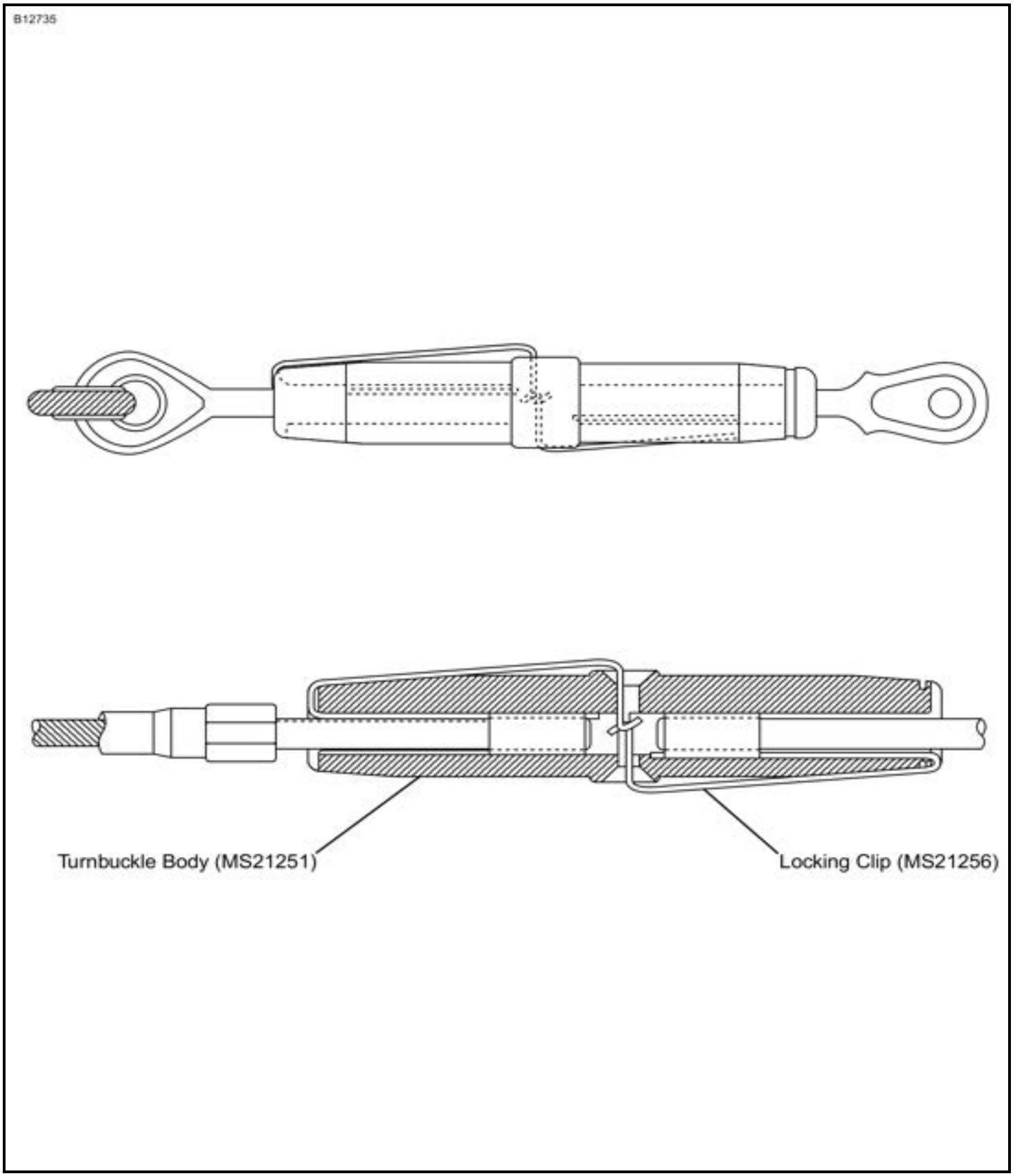
Figure 202 : Sheet 1 : Securing Turnbuckles

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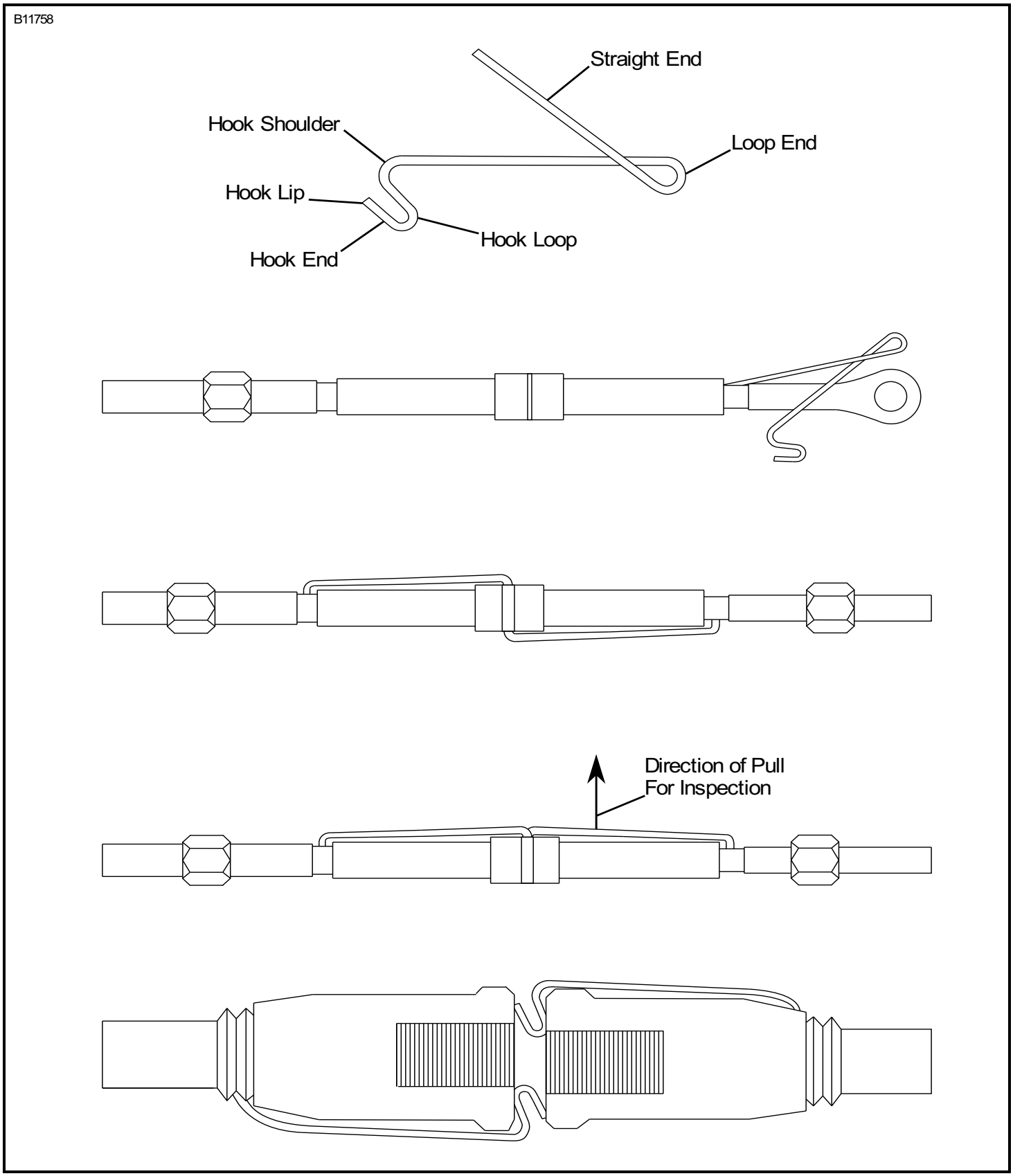
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Figure 203 : Sheet 1 : Clip-Type Locking Device



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Figure 204 : Sheet 1 : Assembling and Securing Clip-Locking Turnbuckles



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SOLVENTS, SEALANTS AND ADHESIVES - MAINTENANCE PRACTICES

1. Solvents, Sealants, and Adhesives

A. Many general purpose solvents, sealants, and adhesives are used in the aircraft.

B. Solvents are used for thinning and cleaning. Use of solvents used for painting are covered in Chapter 51. MPK and alcohol are used for cleaning of composite parts before bonding, and their specific use is described in Chapter 51.

C. Sealants are used for protection, sealing, and thread locking. These items will be specifically called out in specific maintenance manual chapters where their use is required.

D. Adhesives are used in repair and installation. Special purpose adhesives are used for composite repair, composite bonding, upholstery installation, static wick installation, and window installation. Refer to specific maintenance manual chapters for instructions, precautions, and use of these adhesives.

E. Many oils and lubricants are also used.

F. Special purpose cleaners are also listed.

G. Table 201 (Engine and General Purpose Materials) and Table 202 (Special Adhesives, Solvents, and Other Materials) list these items.

Table 201. Engine and General Purpose Materials

Part Number	Product Description of Material	Cessna Specification or Vendor
LI100	Avgas, Gasoline	
Aeroshell 2F	Break-In Oil, Engine, Mil-C-6529 Type II	Aeroshell
Aeroshell 15W50	Oil, Engine	Aeroshell
Z24E024	Oil, Red Hydraulic MIL-H-5606	Esco
	Oil, General Purpose, Lubricating MIL-L-7870	
Aeroshell 22 Mobil 28	Grease, Wheel Bearing MIL-G-81322	Aeroshell Mobil
	Antiseize Compound MIL-T-5544K	Lubtork
DC 7	Silicone Release Compound	Dow Corning
	SAE 10 Oil, Grease Seal Preparation	
Aeroshell 6	Propeller Grease	Aeroshell
	K&N Air Filter Cleaner And Solvent	K&N
	Windex, Glass Cleaner, Instruments And Moving Map Displays	
	P210 Plastic Cleaner, Windows	Sumner Labs
	Permatex No. 2	SX519040 or Loctite
RTV133	Silicone Rubber, Black	E.V. Roberts
RTV162	Silicone Sealant, White, (Type V, Class E)	GE
RTV5818	Silicone Rubber Caulk, Clear 400° F	E.V. Roberts General Electric
440014	Sealant, High Pressure	ND Industries
PR 1428	Fuel Tank Access Panel and Fitting Sealant	PRC Courtalds
AC236	Fuel Tank Sealant	AC Tech
	Paint (Refer to Chapter 51)	SX512701
	Zinc Oxide Primer	SX513840
1522	Fire Resistant Paint (Refer to Chapter 51)	SX512701
	Zinc Chromate Primer	
	Acetone	

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	Alcohol	
	Mek	
325 Solvent UN1268	Cleaning Solvent	

Table 202. Special Adhesives, Solvents and Other Materials

Part Number	Product Description of Material	Cessna Specification or Vendor
MGS418L MGS418H	Adhesive (Structural), Two Part Epoxy	SX513120
LP-89	Conductive Adhesive, Two Part Epoxy	SX518001 Lunar Products
EA9321	Hysol Adhesive, Two Part Epoxy	DexterE.V. Roberts
EA9339	Hysol Adhesive, Two Part Epoxy0	DexterE.V. Roberts
1805	Adhesive, Window Adhesive, Two Part	SX535601Casco Than
1806	Curing Agent, Window Adhesive, Two Part	SX535601Casco Than
1821	Curing Agent, Window Adhesive, Two Part	SX535601Casco Than
ADH105P	Adhesive, Upholstery	Keyston
243	Loctite Threadlocker Medium (Blue) Primer N Or T	Loctite
262	Loctite Threadlocker (Red) Primer N or T	Loctite
542	Loctite Threadlocker (Brown) Primer N	Loctite
609	Loctite Retaining Compound (Green) Primer T	Loctite
620	Loctite Retaining Compound (Green) Primer N	Loctite
660	Loctite Retaining Compound (Silver) Primer N	Loctite
675	Loctite Retaining Compound (Green) Primer T	Loctite
N	Loctite Primer	Loctite
T	Loctite Primer	Loctite

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LIGHTNING PROTECTION AND STATIC DISSIPATION - MAINTENANCE PRACTICES

1. Lightning Protection and Static Dissipation Systems

A. The Cessna 350 (LC42-550FG) airframe is protected during lightning strikes by a combination of electrical grounding and metallic mesh over the composite structure. The lightning strike event is very powerful (hundreds of thousands of amps), of short duration (acting like very high frequency electronics), and of a powerful inductive and magnetic nature that can deform very strong conductive elements. Deviations from the approved design may not provide crew or airframe protection.

B. The metallic mesh conducts the currents of a lightning strike from the attachment point to the arc exit point. The requirements of this mesh are dependent on aircraft location.

C. Chapter 51 defines mesh requirements and mesh repair procedures.

D. **SAFE Strips**

(1) The aircraft interior is specially grounded to avoid large potential differences from one point to another during a lightning strike. This protection is accomplished by grounding bars, grounding straps, and Special Aluminum Foil Edging (SAFE) strips.

(2) SAFE strips are thin self adhesive aluminum foil. They are strategically placed in the cockpit under the upholstery and seats on the composite structure. Refer to Figure 201 and Figure 202. SAFE strips run across the fuselage from one grounding bar to the other.

(3) SAFE Strip Material Requirements:

(a) Adhesive Backed Aluminum Foil P/N 971023 from Alumat Inc.

(b) MS20470AD4-4 Rivets

(c) AN960-4 Washers

(d) Squeegee for rubbing down SAFE strips

(e) Scissors for cutting SAFE strips

(f) Alcohol

(g) Oil-Free Rags

(4) SAFE strips should be replaced or repaired when they are severed across more than one half of the existing width.

(5) SAFE strip repair foil must make contact with the existing foil. This is done by folding over the ends of the repair piece and securing the overlap area that is not secured with the self-adhesive backing of a new overlapping piece. This method is shown in Figure 203. Be sure to adhere to the overlap requirements of 1.0-inch foil to foil contact around the perimeter of the repair.

(6) Clean all surfaces to which SAFE strip material must be applied. Use alcohol and clean, oil-free rags. Wipe surface, turning rag to use a new surface with each wipe. Apply the self-adhesive SAFE strip material by carefully aligning material and rubbing it down with a squeegee. Work slowly and carefully to prevent wrinkles and bubbles. Small wrinkles and bubbles less than 1.0 inch across are acceptable. Large bubbles should be popped with a pin or knife prick and squeegeed down. Cutting out the offending material and replacing it with a suitable patch can repair large wrinkles or creases in the SAFE strip. Be sure to follow the repair guidelines from the previous section.

(7) SAFE strips are grounded to the grounding bars with rivets. Refer to Figure 204. This connection should be inspected. If electrical contact is not confirmed by visual inspection of contact between the washer and SAFE strip, the connection should be removed for replacement. The use of a voltmeter to check the connection will not work. If any connection exists from the SAFE strip and the grounding bar, conductivity will be indicated. The conductivity check does not substantiate the current carrying ability of the connections.

(8) Rivets are MS20470AD4-4 pop type, and secure AN960-4 washers against the SAFE strips.

(9) If complete replacement of a SAFE strip is required, the material should be folded over at the ends where contact with the grounding bar is required. This procedure is shown in Figure 205. Installation of rivets and washers complete the electrical contact connection.

(10) SAFE strips run into the footwell. Inspect this SAFE strip connection by visual confirmation of aluminum foil integrity in the contact area. This detail is shown in Figure 206. Note that the SAFE strip material wraps over the edges of the footwell, from the inside surface to the outside, to make contact with the SAFE strip material on the fuselage floor.

E. **Grounding Straps** - Grounding straps and bars are more conventional conductors. They provide a high current carrying

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capability between major airframe elements. Grounding straps can be found between the engine mount and the fuselage grounding bars, flight control surfaces and wing or stabilizers, gearbox and grounding bar, instrument panel and grounding bar, engine and radio rack, and various avionics components and the radio rack and instrument panel.

(1) Braided Grounding Straps - These are located:

(a) Between the ailerons and wing (one each side)

(b) Between the rudder and vertical stabilizer

(c) Between the fuselage and elevator

(d) Between the gearbox and grounding bar (one each side)

(e) Between the instrument panel and grounding bar (one each side)

(2) Solid Strap Conductors - These are located between the grounding bars and the upper engine mount bolts, one each side.

F. **General Aircraft Conductivity Checks**

(1) The aircraft has nine static wicks. These wicks dissipate the static charge the aircraft develops as it flies through the air. The static charge the aircraft develops is strongly dependent on the atmospheric conditions at the time. Snow is the atmospheric condition most likely to create the largest static charge. Improper static wick function may show up as a crackling sound from the radio speakers.

(2) Check each static wicks (nine each) for low resistance to the nose gear or engine bonding strap. Each should measure less than 10 ohms.

(3) The aircraft fuel filler caps and mounting rings should also be grounded to the rest of the airframe. This is especially important to prevent static discharge during refueling.

(4) Check the fuel cap rings for low resistance to the nose gear or engine bonding strap. This value should be 0 to 500 ohms.

2. Inspection and Repair After Lightning Strike

A. In the event of a lightning strike, or suspected lightning strike, inspect the surface of the aircraft for evidence of damage such as bubbled or bumpy paint.

B. Remove paint and bodywork until mesh is exposed.

NOTE: If damage is noted to the composite material, evidenced by brown or darker fibers, contact Cessna for further instructions.

C. Using a 10x magnifying glass, isolate the entire damaged area of mesh, and a 1-inch square area of damaged mesh within that, evidenced by deformed mesh strands - flat or bulbous in appearance.

D. Count the number of damaged strands in that 1-inch square area and calculate what percentage that is of the entire affected area. If it is less than 2 percent of the total area no mesh repair is necessary. If greater than 2 percent of the total area, a mesh repair is necessary.

E. For mesh repairs, overlap affected area 1 inch beyond damaged area with replacement mesh and apply using Jeffco epoxy.

F. Upon completion of repairs or aircraft exterior inspection, bodywork and paint affected areas in accordance with Section 51-70-00, Paint Repair and Painting.

G. Inspect all hardware associated with damaged areas for magnetism. If no magnetism is noted, re-use hardware.

H. If hardware is found to be magnetized, inspection of respective hardware locations where hardware contacts mesh is required. Wing attachment hardware contacting mesh must have associated bolt holes inspected for damaged mesh. If mesh around the bolt holes is damaged the affected holes and hardware must be increased one size to restore contact of mesh to hardware for proper lightning protection. Enlarge holes using a 5/16 tapered drill ream.

I. Actuate flight controls to ensure proper operation. If defects are noted, inspect for evidence of arcing on control surface attachments and associated hardware and replace as needed.

J. Remove the circuit breaker panel in accordance with Section 25-11-00, Removal of the Circuit Breaker Panel. Remove and replace the avionics bus surge suppressor attached to the front side of the aft circuit breaker panel half.

K. Reinstall the circuit breaker panel and perform one of the following electrical/avionics checkout procedures for all equipment. One procedure is specifically for Basic or Avidyne avionics; the other is for the Garmin G1000 system only. If any equipment is not operating properly, contact Cessna for further instructions for repair or replacement of the equipment.

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L. Verify compass card accuracy.

M. Compliance with Teledyne Continental Motors Service Bulletin No. M88-9, for powerplant lightning strike inspection criteria, is recommended.

3. Checkout Description

A. **Basic or Avidyne Avionics**

(1) 'Buzz Box'

NOTE: It may require the prop to be turned clockwise until an audible click can be heard for the MAGs to work correctly.

(a) Connect 'Buzz Box' to aircraft MAGs connecting to:

1 Right MAG sensor wire to center post on right MAG

2 Left MAG sensor wire to center post on left MAG

3 Ground sensor wire to engine ground

(b) Verify ignition switch positions:

1 Left

2 Right

3 Both

4 Off

WARNING: Clear the area around the prop of personnel and equipment. To avoid the engine from starting, ensure that the spark plugs are not installed or have been disconnected.

(2) Energize the starter with only the left bus energized and verify operation and proper direction of rotation of the engine. Energize the starter with only the right bus energized and verify the operation of the starter motor. Prop should rotate clockwise as viewed from the pilot's seat. The Starter Engaged light on the annunciator panel should be lit when the starter is energized.

(3) Verify operation of the battery switches and essential bus feed. Turn on only the left bus and verify items on the left bus are energized. Also check that the essential bus is energized. Turn on only the right bus and verify items on the right bus are energized. Check again that the essential bus is energized.

NOTE: No ground power is applied for this test.

(4) Connect a 14 VDC (S/N 42001 to 42500) or 28 VDC (S/N 42501 and on) power supply to the ground power plug. (Maintain voltage between 13 VDC and 14 VDC (S/N 42001 to 42500) or between 26 VDC and 28 VDC (S/N 42501 and on) for this test.) Turn on the Crosstie switch and verify that the right bus is energized when the Crosstie switch was turned to "ON" and that both the right and left buses are energized. Verify that the green lights on the left and right voltage regulators are illuminated when the "ALT" switches are turned on. (Voltage regulators are located on the inside of the firewall, right hand side.) Verify the ammeter shows a charge on the corresponding battery when that battery switch is turned on and the ammeter is in the "BATT" mode.

(5) With the Right ALT switch on and no external power applied, check for approximately 10-11 volts (S/N 42001 to 42500) or 24-26 volts (S/N 42501 and On) on the F-1 terminal of the right alternator located at the front of the engine. Terminal F-2 is grounded. With the left 'ALT' switch on check for approximately 10-11 volts (S/N 42003 to 42500) or 24-26 volts (S/N 42501 and On) on the field terminal of the left alternator located at the back of the engine.

(6) With no ground power supplied, energize the avionics master switch and check for voltage on the avionics bus with only the left bus energized and also with only the right bus energized.

CAUTION: Use care when touching hot pitot mast (located under right wing). Ensure no object (e.g. pitot tube cover, adapter, etc.) is attached to the pitot tube. Do not operate pitot heat for more than two minutes.

(7) Turn on the pitot heat and check for heating at the pitot tube. Perform **one** of the following steps:

(a) At the pitot line moisture drain point (center wing), disconnect the drain fitting hose that goes forward to instrument panel.

(b) Attach the pitot tube with a heat resistant adapter connected to flexible hose.

Using Pitot Static test set, raise the pitot system to indicate > 80 kts on the airspeed indicator. Verify that the ammeter load current increases. Turn off the pitot heat and reconnect the hose fitting or remove the pitot tube

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adapter.

(8) Turn on the position lights switch and check all 4 bulbs for illumination and proper color orientation (green right side, red left side).

(9) Turn on strobe lights and verify 3-4 flashes per event. Ensure that the left and right wing flashes alternate between the left and right side.

(10) Turn on the landing light and verify the inboard light illuminates (the one without a diffuser).

(11) Turn on the taxi light and verify the outboard light with the diffuser illuminates.

(12) Door Operational Checkouts:

(a) Close and lock left, right and baggage doors.

(b) Door Open light on annunciator panel should be extinguished.

(c) Open any door and check the flip lights and the step lights stays lit for approximately 10 minutes.

(d) Test the right and baggage doors as described in step (3) above.

(e) With all doors closed and locked, turn on Door Seal Pump Switch, verify pump operation - door seals inflate, and pump turns off when seals are full.

(f) Door seal pump should not cycle (on and off) for at least 30 seconds.

(g) Open left door slowly, dump value should relieve all pressure and door opens.

(h) Again, door seal pump should not cycle (on and off) for at least 30 seconds.

(i) Open right door slowly, dump value should relieve all pressure and door opens.

(j) Turn door seal system off, slowly open either door to verify dump valve operation.

(13) Turn on the Low Boost switch and verify low speed operation of the boost pump.

(14) Momentarily move the back-up pump switch to the arm position. Verify that the blue arm light illuminates and the boost pump operates in high speed. The fuel pump annunciator should also light.

(15) Press the primer button and verify high-speed operation of the boost pump.

(16) Verify operation of the engine gages that can be checked without the engine running:

(a) Fuel level gauge

(b) Man press

(c) Oil temp

(d) Dual amp gauge (turn on a load, should show discharge)

(17) Check ELT remote switch function (leave the ELT Transmitter in the armed mode). Tune the VHF COM to 121.5 MHz and listen through the cabin speaker or the headsets. Turn the ELT switch, located on the right hand knee bolster, to the ON position.

NOTE: Per FCC regulations, the ELT system should only be checked during the first five minutes of the hour.

(18) Test the static and pitot systems for leak rate and for proper operation of the system.

CAUTION: Do not exceed a rate of climb >2000 feet per minute.

(a) Static system leak rate = _____ fpm at 1000 feet above field elevation. (Maximum leak rate is 100 fpm at 1000 feet above field elevation for the static system only.) Raise static system 100 feet above field elevation and hold open the alternate static valve, located on the left side knee bolster (turn to ALT). The system should immediately drop to field elevation.

(b) With switch still in ALT position, raise Pitot System to 200 kts. Maximum leak rate is 50 kts per minute. Pitot System leak rate = _____kpm. After testing Pitot System return switch to the **NORM** position.

(19) Ensure the turn coordinator and attitude gyro flags are not displayed and listen for the rotor of the gyros to spin-up.

(20) **Skip this step if Avidyne Entegra avionics are installed on the aircraft.**Verify OAT, both °F and °C, and voltage readings using top red button. Using the select and control buttons, verify clock operations and functions. (UT, LT, FT, and ET)

(21) Operate the flap switch and verify that the LED in the knob flashes while the flaps are in transition. Check all three positions. Verify the flashing light goes out once the flaps reach the position indicated on the handle.

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(22) Verify operation of trim servos (two axis) for travel, direction of travel, and run them from stop to stop. Ensure trim panel indications agree with external position of the tabs for two axis and that the display properly indicates stop to stop. Verify the operation of the on/off switch. Verify that the pilot's trim switch on the control stick (left side) overrides the co-pilot trim switch on the control stick (right side).

(a) Apply aircraft Battery and Avionics power.

(b) HSI red HDG flag pulls after 2 minutes or less (top left).

(c) Go to Free mode on the KA51B Slaving Controller.

(d) Move CW/CCW switch to CW, compass card moves clockwise and HDG flag appear.

(e) Move Slave/Free switch to Slave mode, compass card rotates back to aircraft heading, the +/- meter shows deflection until the HDG flag pulls.

(f) Go back to Free mode on Slaving Controller and drive the compass card to the CCW position, compass card drives counterclockwise and HDG flag appears during movement.

(g) Move the Slave/Free switch to Slave mode, compass card rotates back to aircraft heading, the +/- meter shows deflection until the HDG flag pulls.

(23) Perform Initial Set-Up Procedures for the Cessna 350 Avionics, as equipped.

(24) Turn on the Avionics Master Switch and verify the operation of:

(a) Pilot's and co-pilot's Audio, VOR, ILS and MKR Navigation systems using the Nav402AP Test set:

1 GPS No. 1 _____

2 GPS No. 2 _____

3 Audio System _____

4 VOR/ILS 1 _____

5 VOR/ILS 2 _____

6 Marker Beacon _____

(b) Perform the Transponder system check-out using the ATC-600A test set or ATC-601 if Mode S equipped:

1 Transponder S/N_____

2 Model No. _____

(c) Check GPS No. 1 and GPS No. 2 for lock and display of position.

(25) Check wire runs and attachments of all wires and coax cables for abrasion potential, support, and wire tying.

(a) Along sides of fuselage _____

(b) Behind the stick grips _____

(c) Behind the instrument panel_____

(d) All flight controls and fluid lines_____

(e) Behind the radio stack_____

(f) Ensure wiring bundles are free and clear of rudder cables on the right side and free and clear of the circuit breaker panel and rudder limiter assembly on the left side._____

(26) Verify security and quality of the switches on the stick grip.

(27) Check the dual ammeter for proper operation. The ammeter should power up in the BATT mode, pressing the button in the lower right should change the mode to ALT. It should show a corresponding discharge when a load is applied to the appropriate bus in the BATT mode.

(28) Check the fuel cap rings for low resistance to the nose gear or engine bonding strap (0 to 500 ohms).

CAUTION: Checking grounds with aircraft power applied will distort resistance reading.

(29) Remove all power from the aircraft. Check each static wick (9 each), for low resistance to the nose gear or engine bonding strap (less than 10 ohms).

CAUTION: Checking grounds with aircraft power applied will distort resistance reading.

(30) Turn on the cabin fan and check for airflow through the selected vents, defroster and floor.

(31) Verify the bus labels on the circuit breaker panel are correct. The top 2 rows are left bus, row 3 is right bus, row 4 is

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essential bus and the bottom row is avionics.

(32) Activate the hot-cold control on the heater panel and verify the firewall shut-off servo works.

(33) Verify the cooling hose is securely fastened and in place behind the instrument panel and the radio rack.

(34) Verify operation of the avionics fan.

(35) To simulate the engine running disconnect oil pressure switch connector (aft left side of engine) and short pins A and B together on connector.

(36) Check the annunciators and aural warning system for operation without ground power connected:

(a) The Left Alt switch turned on, no light or aural warning heard over speakers or headset.

1 Turn Left Alt switch to off, L ALT OFF light is illuminated on annunciator panel. After 2 seconds, "Alternator off" is heard over headsets and speaker.

2 Press ACK button and aural warning is canceled.

(b) The Right Alt switch turned on, no light or aural warning heard over speakers or headset.

1 Turn Right Alt switch to off, R ALT OFF light is illuminated on annunciator panel. After 2 seconds, "Alternator off" is heard over headsets and speaker.

2 Press ACK button and aural warning is canceled.

(c) On center console select left fuel tank:

1 No light on annunciator panel or aural warning is heard over speakers or headset. Fuel Level gauge illuminates green LED for tank selected.

2 Turn Fuel selector to off; FUEL VALVE light is illuminated on annunciator panel. After two seconds fuel valve is heard over headsets and speaker. No LED is lit on Fuel Level gauge.

3 Press ACK button and aural warning is canceled.

4 Repeat previous three steps for the right fuel tank selection.

(d) On aircraft clock press Select button to ET mode.

1 Press Control button to count down from 0:01:00. When timer reaches zero, "Timer at Zero" is heard over headsets and speaker only twice and cancels without pressing the ACK button.

(e) Check that L LOW FUEL, R LOW FUEL, and DOOR OPEN lights are illuminated on annunciator panel.

(f) Check that OIL light on annunciator panel is off.

(g) Remove jumper on Oil Pressure switch and reconnect connector.

(37) With the rudder pedals aligned evenly, activate the rudder limiter by pressing Trim Panel Test button and check that the RDR LMTR annunciator light illuminates and an audible click is heard.

(38) Speed Brake Operational Checks:

WARNING: Clear the left and right wings of personnel and equipment.

(a) Ensure the rudder pedals are aligned evenly and external power is applied>12.4 VDC (S/N 42001 to 42500) or 26 VDC (S/N 42501 and On)

(b) Deploy Speed Brakes by pressing the SPDBRK switch located between the Throttle and Prop levers to the up position.

(c) Speed Brakes deploy, SPDBRK light on annunciator panel illuminates.

(d) Press the test button on the trim panel, Speed Brakes retract; release test button.

(e) Reposition the SPDBRK switch to the down position.

(f) Redeploy the Speed Brakes and move the stall vane lift detector to the stall position full up; the Speed Brakes should not retract.

(g) Reposition the SPDBRK switch to the down position.

(39) Check the test switch on the trim panel to operate the following:

(a) Annunciator panel, all LEDs light.

(b) All trim LEDs light.

(c) Rudder limiter engages.

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(d) All GPS remote annunciators light, if equipped.

(e) Flap Switch LED blinks.

(f) Both Fuel Level gauge LEDs illuminate.

(g) Nav/Com Bypass Switch illuminates.

(40) Check the speaker for operation.

(41) Check the power point for voltage and verify polarity.

(42) Turn on the position lights and verify that the following devices dim: Fuel level gauge LEDs, Trim Panel, Flap Switch LED. Turn off the position lights and verify this action turns off the interior light dimming system.

(43) Verify all four (4) channels of the dimmer, one (1) channel at a time:

(a) **UPPER PNL** (panel) lights; Verify Engine and Flight instruments have (2) bulbs illuminated per instrument. MFD, PFD (or clock) lights and compass light are illuminated and no other lights are on.

(b) **LOWER PNL** (panel) lights; Verify Master switch panel, Heater control panel, Trim panel, and autopilot computer lights and the circuit breaker panel lights are illuminated and no other lights are on.

(c) **FLOOD** lights; Verify glare shield lights are illuminated and no other lights are on.

(d) **SPOT** lights; Verify overhead spotlights are illuminated and on other lights are on. With SPOT lights ON, verify that each lamp turns OFF when using the switch located on the light housing. With the position light switch on, the backlighting on the Flap panel should illuminate.

NOTE: Interior lights only work with position lights rocker switch in the ON position.

(44) Check for painted or plastic coated screws in the instruments panels, and clean instrument dials.

(45) Visually check the integrity of all fuses (4) on the side of the power grid.

(46) Operate the flip lights, front and rear, with the power off and verify correct operation.

(47) Pull all circuit breakers and ensure the labeling agrees with the circuit it supplies by pushing in the circuit breaker. Ensure positive latching of the circuit breaker.

(48) Activate the Nav/Com Bypass Switch with the aircraft power off and ground power disconnected. The switch should illuminate, and the No. 1 GPS, Com, and Nav should turn on. The pilot's headset should receive and transmit on the No. 1 Com. Turn off the Nav/Com Bypass Switch and the GPS, Com and Nav should turn off.

B. **Garmin G1000 System**

(1) 'Buzz Box'

NOTE: It may require the propeller to be turned clockwise until an audible click can be heard for the MAGs to work correctly.

(a) Connect 'Buzz Box' to aircraft MAGs connecting to:

1 Right MAG sensor wire to center post on right MAG

2 Left MAG sensor wire to center post on left MAG

3 Ground sensor wire to engine ground

(b) Verify ignition switch positions:

1 Left

2 Right

3 Both

4 Off

(c) Remove the 'Buzz Box'.

WARNING: Clear the area around the prop of personnel and equipment. To avoid the engine from starting, ensure that the spark plugs are not installed or have been disconnected.

(2) Perform the following tests with the crosstie open: Energize the starter with only the left bus energized and verify operation and proper direction of rotation of the engine. The propeller should rotate clockwise as viewed from the pilot's seat. The Starter Engaged annunciation on the PFD should illuminate when the Starter relays are energized. Energize the starter with only the right bus energized and verify the starter does not engage. Remove the keys from the ignition switch and stow in the proper location.

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(3) Verify operation of the battery switches and essential bus feed. Turn on only the left bus and verify items on the left bus are energized. Also check that the essential bus is energized. Turn on only the right bus and verify items on the right bus are energized. Check again that the essential bus is energized.

NOTE: No ground power is applied for this test.

(4) Connect a 28 VDC power supply to the ground power plug. (Maintain voltage between 26 VDC and 28 VDC for this test.) Turn on the crosstie switch and verify that the left bus is energized when the crosstie switch was turned to 'ON' and that both the right and left buses are energized. Verify that the green lights on the left and right voltage regulators are illuminated when the ALT switches are turned on. (Voltage regulators are located on the inside of the firewall, right hand side.) Verify the ammeter shows a charge on the corresponding battery when that battery switch is turned on.

(5) With the right ALT switch on, check for approximately 24-26 volts on the F-1 terminal of the right alternator. Terminal F-2 is grounded. With the left 'ALT' switch on check for approximately 24-26 volts on the field terminal of the left alternator. Measure the resistance from the left alternator case to the engine block. The total resistance should be less than 1.0 ohm.

(6) With ground power turned off, and using batteries, energize the avionics master switch and check for voltage on the avionics bus with only the left bus energized and also with only the right bus energized, crosstie open.

(7) Apply ground power and turn on the position lights switch. Check all 4 bulbs for illumination and proper color orientation (green right side, red left side).

(8) Turn on the strobe lights and verify 3-4 flashes per event. Ensure that the wing flashes alternate between the left and right sides.

(9) Turn on the landing light and verify the inboard light illuminates (the one without a diffuser).

(10) Turn on the taxi light and verify the outboard light (with the diffuser) illuminates.

(11) Turn on the Vapor Suppress switch and verify low speed operation of the boost pump.

(12) Momentarily move the back-up pump switch to the arm position. Verify that the blue arm light illuminates and the boost pump operates in high speed.

(13) Press the primer button and verify high-speed operation of the boost pump.

(14) Verify operation of the engine gauges that can be checked without the engine running (Cycle the engine instrument circuit breaker to test needle movement):

- (a) Fuel level gauge
- (b) Man press
- (c) Oil temp
- (d) Dual amp gage
- (e) Bus voltage gauge
- (f) Engine temperatures

(15) Door Operational Checkouts:

- (a) Close and lock left, right and baggage doors.
- (b) Door Open annunciation on the PFD should be extinguished.
- (c) Open any door and check the flip lights and the step lights stays lit for approximately 10 minutes.
- (d) Open the right and baggage doors and check the flip lights and the step lights stays lit for approximately 10 minutes.
- (e) With all doors closed and locked, turn on Door Seal Pump Switch, verify pump operation - door seals inflate and pump turns off when seals are full.
- (f) Door seal pump should not cycle (on and off) for at least 30 seconds.
- (g) Open left door slowly, dump value should relieve all pressure and door opens.
- (h) Door seal pump should not cycle (on and off) for at least 30 seconds.
- (i) Open right door slowly, dump value should relieve all pressure and door opens.
- (j) Turn door seal system off, slowly open either door to verify dump valve operation.

(16) Test the static and pitot systems for leak rate and for proper operation.

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CAUTION: Do not exceed a rate of climb>2000 feet per minute.

(a) Static system leak rate = _____fpm at 1000 feet above field elevation. (Maximum leak rate allowed is 100 fpm at 1000 feet above field elevation for the static system only.)

(b) Raise static system 100 feet above field elevation and hold open the alternate static valve, located on the left side knee bolster (turn to ALT); the system should immediately drop to field elevation.

(c) With switch still in ALT position, raise pitot system to 200 kts. Close the valves on the pitot static test pump, then time and observe the pitot system leak rate. Verify that the airspeed tape on the PFD and the mechanical airspeed operate correctly. Maximum leak rate allowed is 25 kts per minute. Pitot System leak rate = _____kpm.

NOTE: Conduct the test for the next task concurrently with this test item.

(17) Check the resistors on the autopilot 485 lines and verify their resistance per the manufacturer specifications.

(18) Replace the TVS diodes on the PFD, GIA, AHRS and standby ADI.

(19) Replace the zener diodes on the oxygen system.

(20) Verify that the PFD is properly configured. While testing the pitot system, run the PFD and mechanical airspeed indicator high enough to verify that the following airspeed limitation markings are correct: (Below 12,000 ft PA)

(a) White Range, 59 - 119 KIAS

(b) Green Range, 71 - 179 KIAS

(c) Yellow Range, 235 KIAS

(d) Red Range, above 235 KIAS

(21) **Pitot Heat Test**

CAUTION: Use care when touching the hot pitot mast. Do not operate pitot heat for more than one minute.

(a) Turn on the pitot heat switch on the instrument panel. Note that the current draw on the amp meter increases. Carefully feel with the back of your hand for actual heating at the pitot tube.

(b) Turn off the pitot heat. **Only leave the pitot heat on long enough to verify the operation of the heating element.**

(22) Ensure the flag of the standby attitude gyro located on the instrument panel, are not displayed. Listen for the rotor of the gyros to spin-up.

(23) Operate the flap switch and verify that the LED in the knob flashes while the flaps are in transition. Check all three positions. Verify the flashing light goes out once the flaps reach the position indicated on the handle.

(24) **Trim Indicators Adjustment**

(a) Verify operation of trim servos (two axis) for travel, direction of travel, and run them from stop to stop. Verify the operation of the on/off switch.

(b) Ensure the trim indications agree with the external position of the tabs for the aileron and elevator trim and that the display properly indicates stop to stop.

(c) If trim panel adjustment is required contact Cessna for instructions. If adjustments are made, re-verify operation of trim servos (two axis) for travel, direction of travel, and run them from stop to stop. Re-verify the operation of the on/off switch as well.

(25) Engage the Autopilot with the audio panel on at the same time. Disconnect the autopilot, while listening for the audio level of the AP DISC through the aircraft speaker.

(26) Perform Initial Set-Up Procedures for the Cessna 350 Avionics, as equipped.

(27) **Avionics Tests**

(a) Comm 1 and 2 - Refer to Table 201.

(b) GPS 1 and 2 - Refer to Table 202.

(c) PFD - Refer to Table 203.

(d) MFD - Refer to Table 204.

(e) Audio System - Refer to Table 205.

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(f) Marker Beacon (NAV-401AP) - Refer to Table 206.

(g) MFD (NAV-401AP) - Refer to Table 207.

(h) NAV (NAV-401AP) - Refer to Table 208.

(i) ILS (NAV-401AP) - Refer to Table 209.

(j) Autopilot (NAV-401AP) - Refer to Table 210.

(k) Flight Director - Refer to Table 211.

(l) Auto Trim - Refer to Table 212.

Table 201. Avionics Test - Comm 1/2

Comm 1/2:

[ ]	[ ]	TX/RX 123.225
[ ]	[ ]	T/R QUALITY
[ ]	[ ]	SIDE TONE
[ ]	[ ]	MAN SQUELCH
[ ]	[ ]	PILOT'S PTT
[ ]	[ ]	COPILOT'S PTT

Table 202. Avionics Test - GPS 1/2

GPS 1/2:

[ ]	[ ]	Verify that GPS 1 and GPS 2 are receiving a signal
[ ]	[ ]	Sat Lock
[ ]	[ ]	Setup per procedure
[ ]	[ ]	TIS traffic data - TIS will not be available unless a terminal radar site is nearby providing data. Most of the time, this will not be the case. If TCAD is installed TCAD must be failed before TIS will be displayed.
[ ]	[ ]	DX to PFD, MFD
[ ]	[ ]	DX from PFD
[ ]	[ ]	S/W and data cards revisions match
[ ]	[ ]	Verify GPS positions are valid on MFD

Table 203. Avionics Test - PFD

PFD:

[ ]	[ ]	Proper Setup for Cessna 350
[ ]	[ ]	RX valid GPS position data from GPS 1,2
[ ]	[ ]	Knob operation
[ ]	[ ]	Air Data operation
[ ]	[ ]	Autopilot Annunciations - Verify no red AFCS or PFT on powerup.
[ ]	[ ]	Valid AHRS
[ ]	[ ]	Cross talk with MFD - Verify that the message and alert box has no messages that state a backup path is being utilized.

Table 204. Avionics Test - MFD

MFD:

[ ]	[ ]	Proper setup for aircraft type
[ ]	[ ]	RX valid GPS position data from GPS 1,2
[ ]	[ ]	Cross talk with PFD - Verify that the message and alert box has no messages that state a backup path is being utilized.

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[ ] [ ] Receives Stormscope or WX data as installed

[ ] [ ] Engine instrumentation data present and valid

[ ] [ ] Traffic - For TCAD verify PFD message box does not display TCAD system failure message (approx. one minute for TCAD to go online and failure messages to clear). For TIS verify Mode S transponder is functioning by checking the transponder annunciation on the PFD. If there is no red "X" and indicates normally, TIS will be functional.

Table 205. Avionics Test - Audio System

Audio System:

[ ] [ ] Pilot's ICS

[ ] [ ] Co-Pilot's ICS

[ ] [ ] Pass 1,2 ICS

[ ] [ ] ISO/CREW functions

[ ] [ ] Pilot's/Co-pilot's side tone quality and level

[ ] [ ] Emergency TX on Comm 1 with audio panel off

[ ] [ ] Nav 1, 2 Audio

[ ] [ ] Bose Headsets **REPEAT ALL AUDIO SYSTEM CHECKS**

Conduct the following tests using the NAV-401AP test set.

Table 206. Avionics Test - Marker Beacon (NAV-401AP)

Marker Beacon:

[ ] [ ] Outer Marker - 400 Hz tone, Blue Light

[ ] [ ] Middle Marker - 1300 Hz tone, Amber Light

[ ] [ ] Inner Marker - 3Khz tone, White Light

[ ] [ ] Remote Display

[ ] [ ] Hi/Lo sensitivity tests, Hi/Lo Split

Table 207. Avionics Test - MFD (NAV-401AP)

MFD:

[ ] [ ] Proper setup for aircraft type

[ ] [ ] RX valid GPS position data from GPS 1,2

[ ] [ ] Cross talk with PFD - Verify that the message and alert box has no messages that state a backup path is being utilized

[ ] [ ] Receives Stormscope or WX data as installed.

[ ] [ ] Engine instrumentation data present and valid.

Table 208. Avionics Test - Navigation Systems 1,2 - NAV (NAV-401AP)

Navigation Systems 1,2 :

NAV:

[ ] [ ] Receive 112.80

[ ] [ ] Flags Out of view

[ ] [ ] CDI (Left / Center / Right)

[ ] [ ] To/From operation

[ ] [ ] 30 degree accuracy checks

Table 209. Avionics Test - Navigation Systems 1,2 - ILS (NAV-401AP)

ILS:

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[ ] [ ] Receive 108.1
[ ] [ ] Localizer (Left / Center / Right)
[ ] [ ] Localizer flag pulled
[ ] [ ] Glide slope bars (Up / Center / Down)
[ ] [ ] Glide slope valid
[ ] [ ] Autopilot coupling and tracking checks in Nav Mode, testing above functions
[ ] [ ] Nav Audio
[ ] [ ] ILS sensitivity

Table 210. Avionics Test - Navigation Systems 1,2 - Autopilot (NAV-401AP)

Autopilot:

[ ] [ ] Master switch operation
[ ] [ ] HDG bug operation
[ ] [ ] NAV (Tracks D-BAR)
[ ] [ ] CRS (Tracks Course Datum Pointer, No signal applied)
[ ] [ ] APR mode engaged on ILS freq.
[ ] [ ] Localizer Left Right (Stick Tracks)
[ ] [ ] Glide Slope soft capture
[ ] [ ] G/S bars up/down, stick tracks to glide slope bars
[ ] [ ] ALT hold mode engages and holds
[ ] [ ] ALT PRESELECT engages and tracks from PFD
[ ] [ ] VS engages and tracks from PFD controls
[ ] [ ] AP Disconnect (Both control sticks)
[ ] [ ] Disconnect audio
[ ] [ ] Remote Annunciator on PFD

Table 211. Avionics Test - Navigation Systems 1,2 - Flight Director (NAV-401AP)

Flight Director:

[ ] [ ] Command Bars engage, and track in Autopilot pitch mode
[ ] [ ] AP FD mode, Command bar tracks with autopilot
[ ] [ ] FD mode; Command Bars track, Servo clutches disengage

Table 212. Avionics Test - Navigation Systems 1,2 - Auto Trim (NAV-401AP)

Auto Trim:

[ ] [ ] Hold slight aft and forward pressure and verify autotrim in the opposite direction on the MFD.

(28) Physically verify the security of the BNC connections on the NAV signal splitter, and the back of the radios.

(29) Check the speaker for operation.

(30) Check ELT remote switch function (leave the ELT Transmitter in the armed mode). Tune the VHF COM to 121.5 MHz, and listen through the cabin speaker or the headsets. Turn the ELT switch, located on the right hand knee bolster, to the ON position.

NOTE: Per FCC regulations the ELT system should only be checked during the first five (5) minutes of the hour.

(31) **Transponder Test** Perform the transponder test and checkout using the ATC 600A or ATC 601-2 test set. The ATC-600A Transponder/DME test set is to be used for testing Garmin AT model SL 70 or Garmin model GTX 327 Mode A/C transponders. The ATC 600A is not to be used for testing of the Garmin model GTX 33 mode S transponder. The ATC-601-2 Mode S Transponder test set is required for testing Garmin model GTX 33 Mode S transponder, and

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may also be used for testing Garmin model GTX 327 Mode C transponder.

NOTE: A DI or Q inspector must witness the transponder test.

Record the model and serial number of the transponder below.Make/Model_____/_____ Serial Number_____

(32) Turn aircraft power OFF, check the fuel cap rings for low resistance to the nose gear or engine bonding strap, (0 to 500 ohms).

CAUTION: Checking grounds with aircraft power, or ground power applied will distort resistance reading.

(33) Remove all power from the aircraft. Check each static wick or mounting pad (9 ea.), for low resistance to the nose gear or engine bonding strap, (less than 10 ohms).

(34) Turn on the cabin fan and check for airflow through the selected vents, defroster and floor.

(35) Verify the bus labels on the circuit breaker panel are correct. The top 2 rows are essential bus, row 3 is left bus, row 4 is right bus and the bottom row is avionics.

(36) Activate the hot-cold control on the heater panel and verify the firewall shut-off servo works.

(37) Verify operation of the avionics fan.

(38) Check the annunciator and aural warning system for operation without ground power connected. Start the engine.

(a) The left Alt switch turned on, no light or aural warning heard over speakers or headset.

1 Turn left alt switch to off, L ALT OUT annunciation on the PFD. After 2 seconds left alternator out is heard over headsets and speaker.

2 Press Alert button and aural warning is canceled. The left alt switch turned on, no light or aural warning heard over speakers or headset.

3 Turn right alt switch to off, R ALT OUT annunciation on the PFD. After 2 seconds right alternator out is heard over headsets and speaker.

4 Press Alert button and aural warning is canceled.

5 Turn the right alt switch to 'on'.

6 Shut down the engine.

(b) On the center console, select left fuel tank:

1 No annunciations or aural warning is heard over speakers or headset. Fuel level gauge illuminates blue indication for tank selected.

2 Turn fuel selector to off, FUEL VALVE annunciation on the PFD. After two seconds fuel valve is heard over headsets and speaker. There will be no indication next to the fuel level gauge.

3 Press Alert button and aural warning is canceled.

(c) On the center console, select right fuel tank:

1 No annunciations or aural warning is heard over speakers or headset. Fuel level gauge illuminates blue indication for tank selected.

2 Turn fuel selector to off, FUEL VALVE annunciation on the PFD. After two seconds fuel valve is heard over headsets and speaker. There will be no indication next to the fuel level gauge.

3 Press Alert button and aural warning is canceled.

(d) Check that DOOR OPEN annunciates on the PFD when one of the three doors are open.

(39) Speed Brake Operational Checks:

WARNING: Clear the left and right wings of personnel and equipment.

(a) Apply external power > 26 VDC.

(b) Deploy the Speed Brakes by pressing the SPDBRK switch located between the Throttle and Prop levers to the up position.

(c) When the Speed Brakes deploy, SPEED BRAKES annunciates on the PFD.

(d) Reposition the SPDBRK switch to the down position.

(e) Redeploy the Speed Brakes and move the stall vane lift detector to the stall position full up. The Speed Brakes will retract.

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(f) Reposition the SPDBRK switch to the down position.

(40) Check the test switch on the overhead. The test switch will illuminate the push-button indication on the left side of the cockpit, illuminate the flap handle, and activate the rudder limiter. The rudder limiter activation will cause a chime to play and an amber caution to be displayed on the PFD.

(41) Turn on the position lights and verify that the following devices dim: flap panel switches, 5 pack switches, and flap switch LED.

(42) Operate the flip lights, front and rear, with the power off and verify correct operation.

(43) Verify all four (4) channels of the dimmer, one (1) channel at a time:

(a) **UPPER PNL** (panel) lights; MFD, PFD, Audio Panel, Remote Keypad, standby instrument lights and compass light are illuminated and no other lights are on.

(b) **LOWER PNL** (panel) lights; Verify Master switch panel, Heater control panel, Trim panel, and the circuit breaker panel lights are illuminated and no other lights are on.

(c) **FLOOD** lights; Verify glare shield lights are illuminated and no other lights are on.

(d) **SPOT** lights; Verify overhead spotlights are illuminated and no other lights are on.

(e) With the position light switch on, the backlighting on the Flap panel should illuminate.

(44) Visually check the integrity of all fuses (4) on the power grid located in the aft avionics compartment.

(45) Pull all circuit breakers and ensure the labeling agrees with the circuit it supplies by pushing in the circuit breaker. Ensure positive latching of the circuit breaker.

(46) Check the power point for voltage and verify polarity.

(47) Check for painted or plastic coated screws in the instruments panels, and clean instrument dials.

(48) Verify security and quality of the switches on the stick grip.

(49) Verify the cooling hose is securely fastened and in place behind the instrument panel and the radio rack.

(50) Verify AHRS calibration and perform AHRS calibration on the PFD if necessary.

(51) Check wire runs and attachments of all wires and coax cables for abrasion potential, support, and wire tying:

(a) Along sides of fuselage

(b) Behind the stick grips

(c) Behind the instrument panel

(d) All flight controls and fluid lines

(e) Behind the radio stack

(f) Ensure wiring bundles are free and clear of rudder cables on the right side and free and clear of the circuit breaker panel and rudder limiter assembly on the left side.

4. Repair of Static Wicks

A. The nine static wicks on the aircraft are located at the tips of the wings, horizontal, outboard trailing edge of the ailerons, outboard trailing edge of the elevator, and the top of the rudder. The rudder static wick is mounted on a threaded stud and bonded electrically and mechanically to the aluminum counterweight at the top of the rudder. The other static wicks are mounted on static wick pads. These pads are bonded electrically and mechanically to the structure, and the static wick is attached with screws to this pad.

B. If the required conductivity is not achieved, these components must be replaced or re-bonded with conductive adhesive.

(1) Conductive Adhesive Material Requirements

(a) Equipment:

1 Respirator mask - Commercial

2 Rubber gloves - Commercial

3 Sandpaper, 120 and 400 grit - Commercial

4 Applicator spatula - Commercial

5 18 oz. mixing container - Commercial

6 Shop towel - Commercial

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(b) Materials:

1 Conductive Adhesive - SX518001

2 Acetone, MPK, or Ethyl Acetates - Commercial

C. **Bonding Process** - The following process describes mounting pad and thread rod styles of the static wick mount installations. Follow the steps in the order presented.

(1) Mounting Pad Static Wick Procedure

(a) Mark the location of retainers on the aircraft skin per the corresponding drawing. Do not exceed 1/8 inch between the aft end of the mounting pad and the trailing edge of the appropriate control surface.

(b) Remove paint and primer, if any, from area to be bonded using 120 grit sandpaper. Expose the lightning protection mesh without sanding through the mesh material.

(c) Degrease the aircraft surface and bottom of the retainer with a grease free solvent, such as acetone, MPK, or ethyl acetate. Do not contaminate the cleaned area.

(d) Use either approved conductive adhesive per SX518001:

1 Mix one part LP-89 Part A Resin with one equal part LP-89 Part B Paste Catalyst. Mix thoroughly until a smooth, creamy consistency is reached. Pot life is 30 minutes at 70° F after mixing.

2 Take a premixed cartridge of Eccobond 64C A/B from the freezer. Allow it to thaw at room temperature. Work life is approximately 30 minutes from when the adhesive is removed from the freezer.

(e) Abrade the clean, dry aircraft surface and the bonding surface of the static wick retainer with 200 to 600 grit paper. Do not abrade through the lightning protection mesh of the aircraft. Remove all oxide from the retainer. Dry thoroughly and wipe the surface with a clean towel to remove loose particles.

(f) After removing the oxide, immediately apply a coat of adhesive to the retainer. The time between completion of the sanding process and the application must not exceed 5 minutes.

(g) Upon completion of the previous step, immediately apply the retainer to the aircraft surface. Slightly twist the retainer while applying light pressure, to ensure thorough wetting and to squeeze out excessive adhesive. Once the retainer is in place, do not disturb the bond further. Movement will increase the bond resistance and may weaken the mechanical strength.

(h) Remove excessive adhesive, leaving a fillet of adhesive around the entire edge of the retainer.

(i) Use masking tape to temporarily secure the mounting pad during curing.

(j) Cure the adhesive for a minimum of 24 hours at 70° F. Cure may be accelerated by applying 150° F for one hour.

(k) Ensure that all retainers are secure, with no evidence of cracks in the adhesive fillet between the retainer and the aircraft surface.

(l) After the mounting pad has cured, check the electrical resistance to the lightning mesh with a digital voltmeter as close to the mount pad as possible. The resistance should be 0 to 2.0 ohms.

(2) Rudder Threaded Rod Static Wick Procedure

(a) Locate the static wick installation position in the vertical stabilizer counterweight.

(b) Degrease the threaded rod, receptor threads in the rudder counterweight, and the receptor treads in the static wick with a grease-free solvent such as acetone, MPK, or ethyl acetate. Do not contaminate the cleaned area.

(c) Use either approved conductive adhesive per SX518001:

1 Mix one part LP-89 Part A Resin with one equal part LP-89 Part B Paste Catalyst. Mix thoroughly until a smooth, creamy consistency is reached. Pot life is 30 minutes at 70° F after mixing.

2 Take a premixed cartridge of Eccobond 64C A/B from the freezer. Allow it to thaw at room temperature. Work life is approximately 30 minutes from when the adhesive is removed from the freezer.

(d) Place a small amount of adhesive on 0.5 in. of one end of the threaded rod. Install the threaded rod into the stabilizer counterweight, adhesive end first, to a depth of approximately 0.5 in. to the mechanical stop.

(e) After the threaded rod has cured, check the electrical resistance to the counterweight material as close to the threaded rod as possible. Resistance should be 0 to 2.0 ohms.

(f) After the aircraft has been painted, install the static wick as described below. Ensure that the threaded rod is

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masked for the aircraft paint process.

(g) With a wooden applicator, place adhesive on the first 0.1 inch of the threaded rod above its base, to allow a removable bond to the tip of the static wick.

(h) Insert the static wick onto the threaded rod, and ensure that it threads on the full length to its mechanical stop, allowing contact with the 0.1 inch of adhesive.

(i) Cure the adhesive for a minimum of 24 hours at 70° F. Cure may be accelerated by applying 150° F for 1 hour.

(j) Ensure that the static wick is secure.

D. **Inspection Procedures** - Visually inspect all retainer installations thoroughly. Ensure that the retainer has no evidence of cracks in the adhesive fillet between the retainer and the aircraft surface. Ensure that all static wicks are secure.

E. **Rework** - If a mounting pad has evidence of cracks in the adhesive fillet that exceed the maximum length criteria, the static wick mounting pad must be removed and re-installed. If the threaded rod-style static wick is not secure, remove, clean, and re-install the static wick threaded rod as per procedure.

F. **Acceptance Criteria**

(1) ADHESIVE - The adhesive application shall be acceptable if there is a smooth surface fillet between the retainer and the aircraft surface, with no evidence of cracks or unbonded areas in the adhesive of over 0.1 in. in length.

(2) IMPROPER CURE - The adhesive application shall be acceptable if there is no evidence of tackiness in the adhesive after the proper cure time.

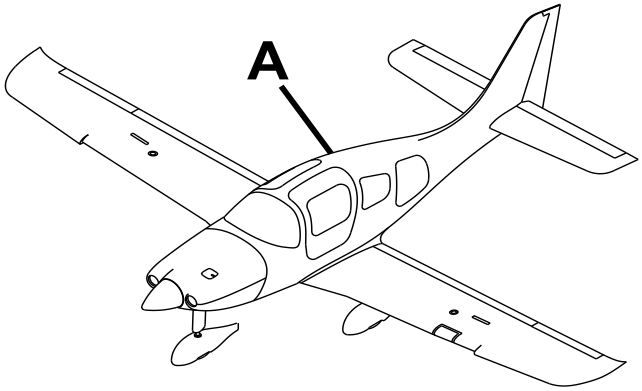
(3) SECURE INSTALLATION - The static wick mount installation shall be acceptable if there is no evidence of improper bonding of the static wick pad or threaded rod allowing any movement of the static wick or evidence of cracking and tackiness of the adhesive.

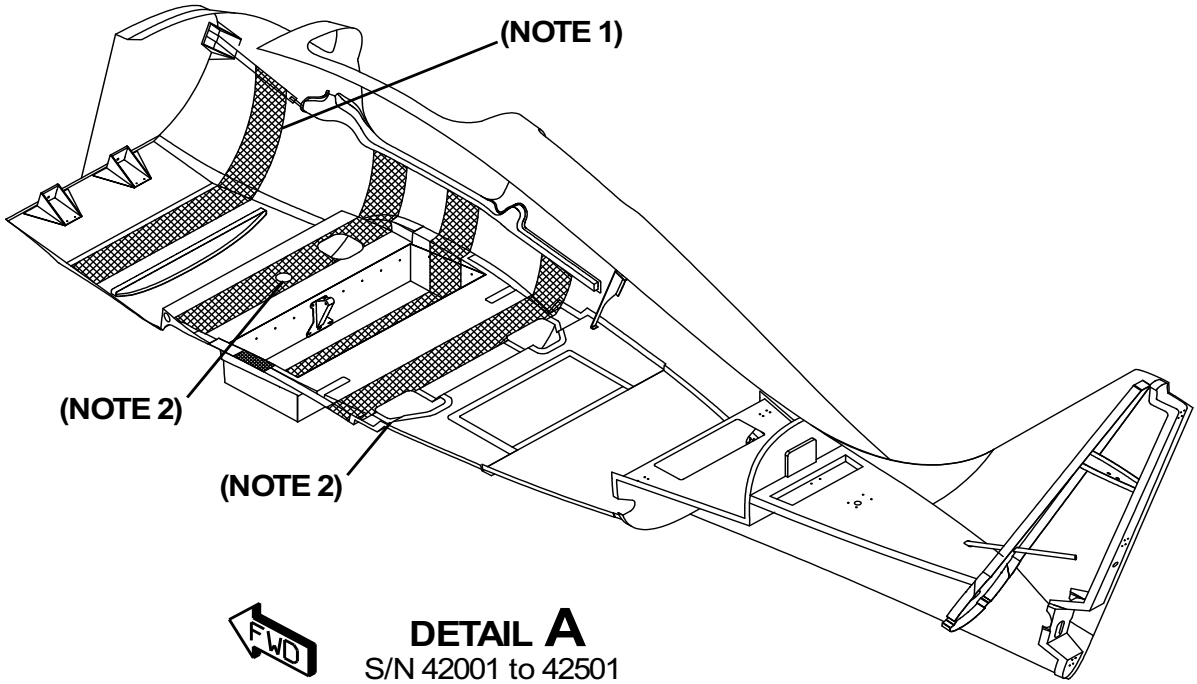
(4) ELECTRICAL RESISTANCE - The static wick mount installation shall be acceptable if the electrical resistance of the mounting connection is no more than 2.0 ohms, as measured from the mount to the lightning mesh or counterweight.

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Figure 201 : Sheet 1 : Placement of SAFE Strips (S/N 42001 to 42501)

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DETAIL A
S/N 42001 to 42501

NOTE 1: Aluminum foil should cover from left fuselage skin at lightning bar, across floor panels to right fuselage skin a lightning ba r.

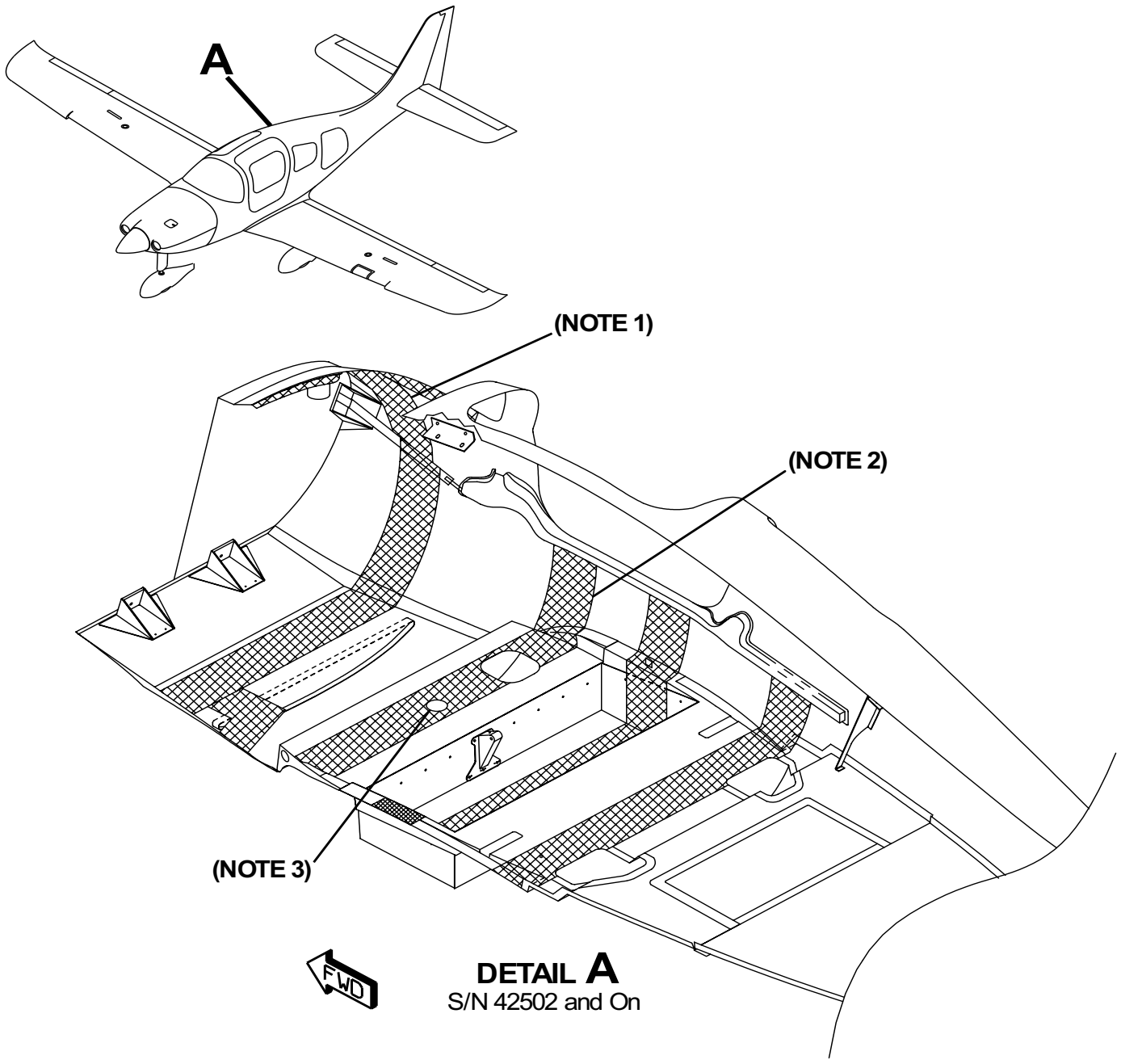
NOTE 2: Trim aluminum foil at all cutouts.

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Figure 202 : Sheet 1 : Placement of SAFE Strips (S/N 42502 and On)

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NOTE 1: Upper strips of aluminum foil should run upward and forward to overlap with foil from covered defrost duct, both sides.

NOTE 2: Aluminum foil should cover from left fuselage skin at lightning bar, across floor panels to right fuselage skin at lightning bar.

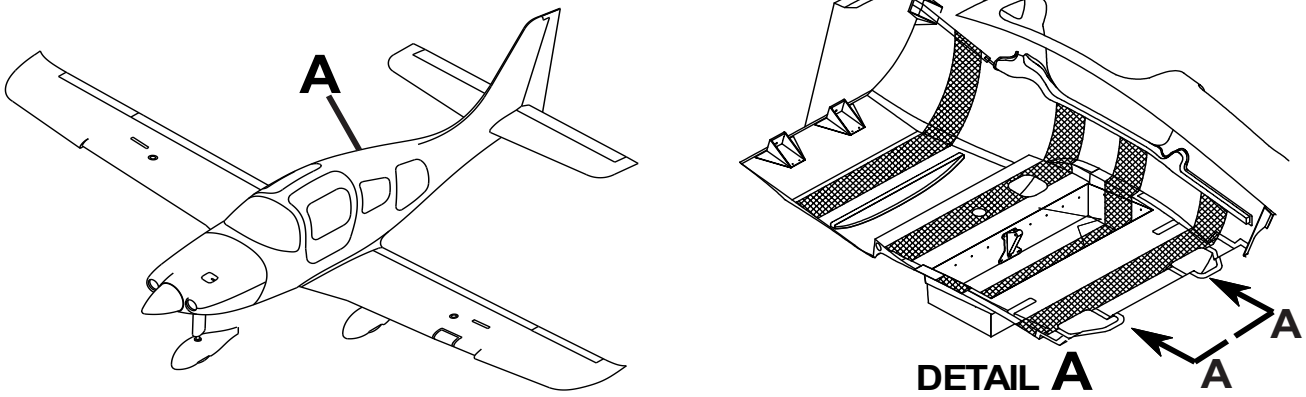
NOTE 3: Trim aluminum foil at all cutouts.

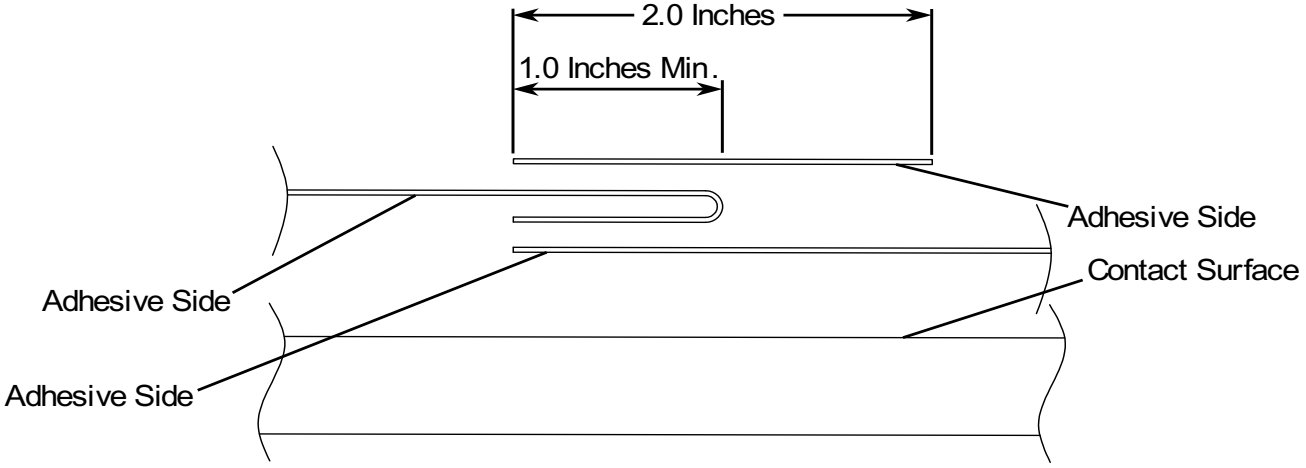
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Figure 203 : Sheet 1 : Folding SAFE Strip

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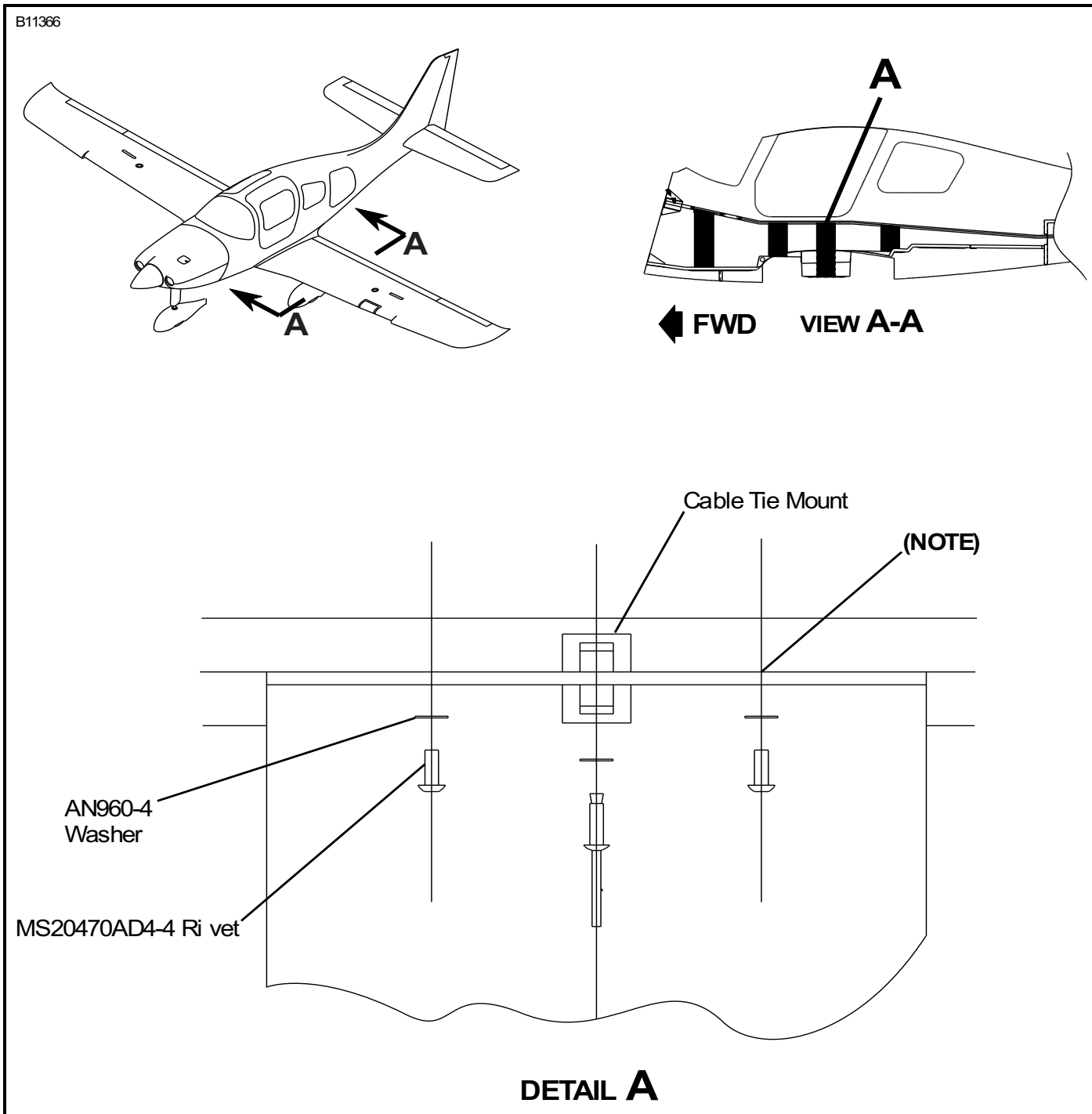


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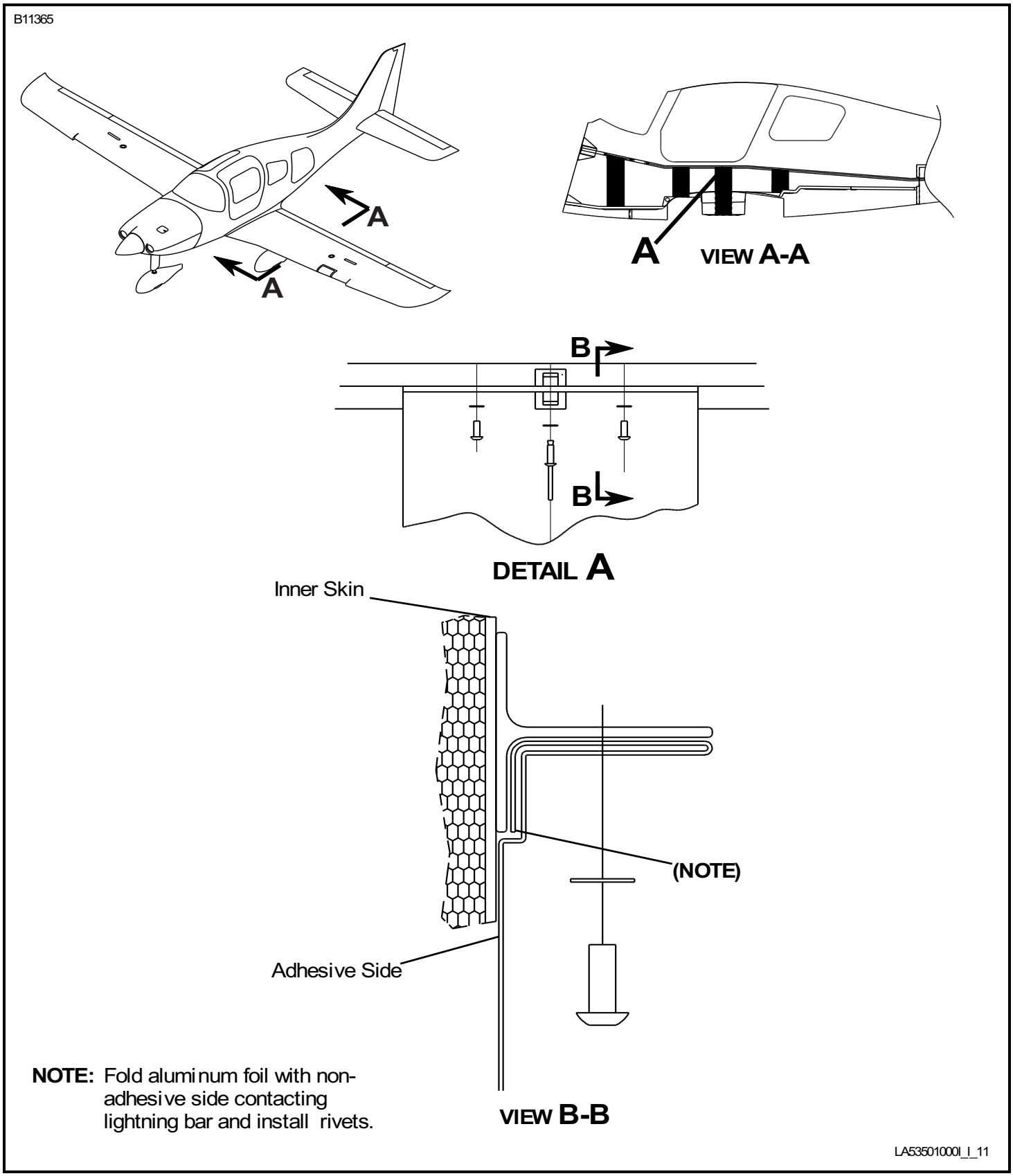
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Figure 204 : Sheet 1 : Grounding SAFE Strip to Grounding Bar



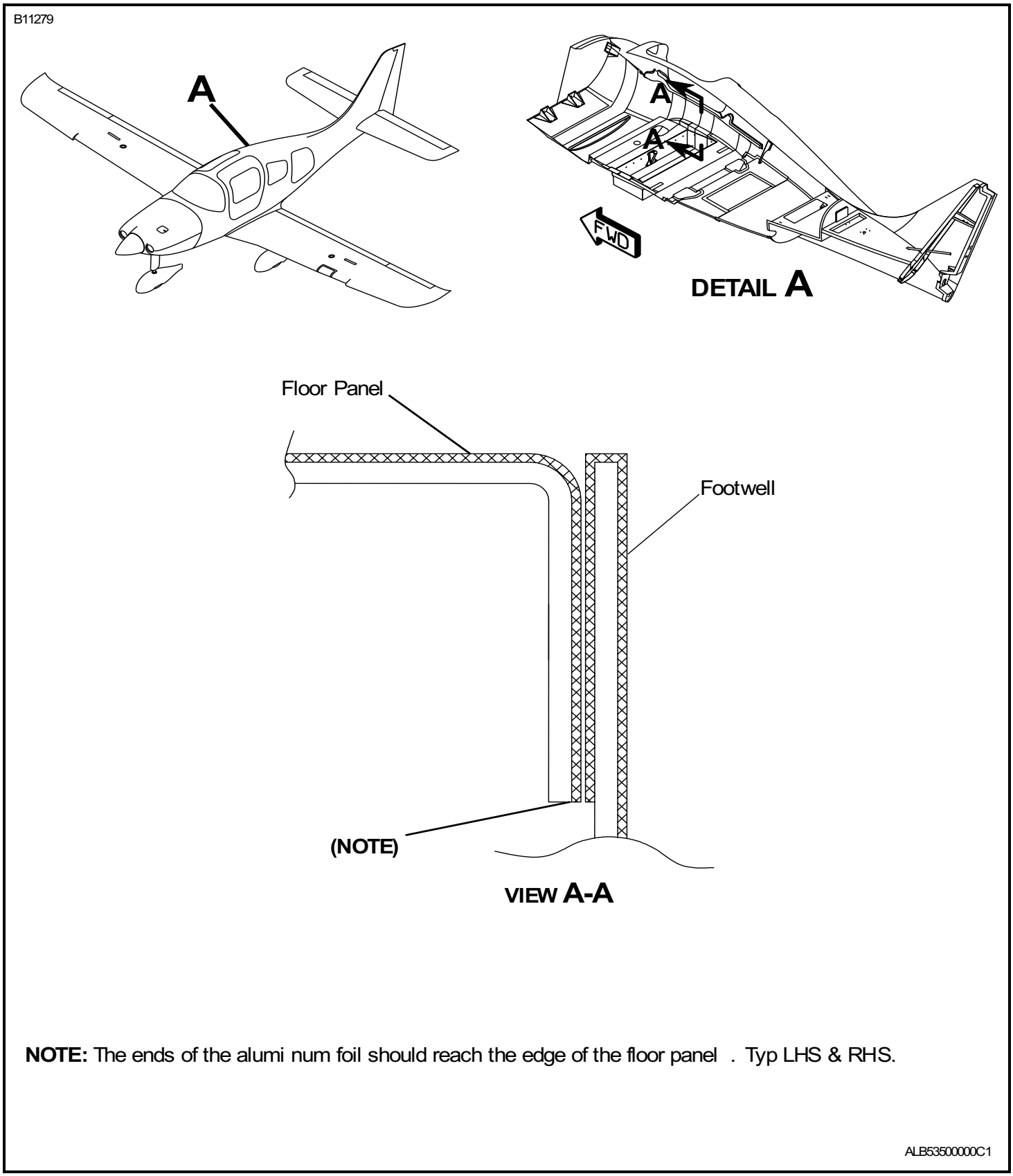
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Figure 205 : Sheet 1 : Folding SAFE Strip



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Figure 206 : Sheet 1 : Installing SAFE Strip in Footwell



NOTE: The ends of the alumi num foil should reach the edge of the floor panel . Typ LHS & RHS.

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REPAIR OF CLICKBOND NUTPLATES - MAINTENANCE PRACTICES

1. Repair of Clickbond Nutplates

A. Clickbond nutplates are a special type of nutplate that is bonded to the surface rather than being riveted to the structure. If a Clickbond nutplate should fail, it can be replaced with another Clickbond nutplate, or a riveted nutplate under certain conditions, as specified in Table 201. This repair procedure can be used in lieu of specific factory instructions.

Table 201. Permitted Areas of Clickbond or Riveted Nutplate Repair

Clickbond PartNumber	Application	Assembly DrawingNumber
CB6009CR4-1P	Baggage Net Installation	LB53255000
CB6014CR3-1P	Baggage Net Installation	LB53255000
	Cowling	LA53600000
	Instrument Panel Installation	LB53310000
CB6014CR08-1P	Access Panel Installation, Fuselage	LB53650000

B. **Defect Scenarios**

(1) The nut in the Clickbond nutplate is stripped or damaged.

(2) The bonding of the Clickbond to the surface has failed.

C. **Repair Procedure No. 1** - Bond a new Clickbond of the same type as a replacement for the one that failed.

(1) Remove the old Clickbond nutplate by prying it off.

(2) Sand the substrate to prepare a smooth surface using 150 grit sandpaper.

(3) Clean the surface with a dry-wipe or clean, pressurized air.

(4) Bond on a new nutplate. Use Clickbond adhesive kit CB92 or adhesive Hysol 9321. Follow manufacturer's instructions.

D. **Repair Procedure No. 2** - Replace the Clickbond with riveted nutplate.

(1) Sand the adhesive residue down to create an even surface for the nutplate. Take care not to damage the structural material below.

(2) Use a nutplate from Table 202 to determine the proper riveted nutplate replacement.

Table 202. MS Nutplate Replacements for Clickbonds

ClickbondNutplate	MS Nutplate	Drill Rivet Holes	Rivet Hole Spacing From Center Hole
CB6009CR4-1P	MS21075-4	0.098 in.	0.281 in.
CB6014CR3-1P	MS21075-3	0.098 in.	0.250 in.
CB6014CR08-1P	MS21075-08	0.098 in.	0.234 in.

NOTE: The old Clickbond nutplate can be reused if the nut still has sufficient self-locking friction. Clean and sand the surface on the Clickbond nutplate for good adhesion, and follow the procedures discussed in Repair Procedure No. 1.

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SWAGELOK FITTINGS - MAINTENANCE PRACTICES

1. Swagelok Fittings

A. The information in this section is taken from the *Swagelok Tube Fitter's Manual*. The Swagelok manual should be referenced if there is any question about the information presented in this manual. Formal training from Swagelok or Cessna is recommended before using any Swagelok fittings.

B. The Swagelok fittings used in Cessna aircraft provide a consistent tube connection for both fuel and hydraulic systems without the need for flaring tubing. These flareless, mechanical grip fittings consist of a fitting body, front ferrule, back ferrule, and nut, as seen in Figure 201. The fittings are received from Swagelok pre-assembled and normally ready for use, with the exception being the bulkhead fittings used in some systems and in other specific applications. With the installation of the bulkhead fittings, one end of the fitting must be disassembled to install the fitting and the removed end reassembled prior to installation of the tubing. Figure 201 shows the correct orientation of the body, ferrules and nut for proper installation.

C. When installing Swagelok fittings, always maintain the cleanliness of the hose, fittings, and tube interiors. Protect the threads from damage and dust.

D. **Swagelok Fitting Installation**

(1) Visually inspect the fitting to verify that the ferrules are in place and installed correctly as shown in Figure 202 and Figure 203. When looking inside the end of the fitting, three lines should be seen.

(2) Insert the tubing into the tube fitting such that the tubing firmly rests on the shoulder of the fitting as shown in Figure 204.

(3) Using a permanent marker, mark the 6 o'clock position on the nut and extend onto the tube for reference as shown in Figure 205. In addition, with the tube fully inserted in the fitting and the nut finger-tight, mark on the tube the intersection between the tube and the nut.

(4) While holding the fitting body with a backup wrench, tighten the nut 1-1/4 turns as shown in Figure 206. *For 1/16 in., 1/8 in., and 3/16 in., 2 mm, 3 mm, and 4 mm fittings, only a 3/4 turn from finger-tight is necessary.*

(5) Verify installation per the Swagelok Fitting Gauging section.

E. **Swagelok Fitting Gauging** - After the Swagelok fittings are installed, the design of the fitting allows the installation to be verified using a go no-go gauge. This gauge may only be used for standard fitting and unions. For bulkhead fittings and other special fittings the gauge cannot be used because the distance from the body to the nut is variable.

CAUTION: Only use Swagelok gap inspection gauges on Swagelok tube fittings. The gauge will not work properly on other tube fittings.

NOTE: Swagelok gap inspection gauges should not be used after pre-swaging or hydraulic swaging.

NOTE: Swagelok gap inspection gauges should be used only for the initial installation of the fitting and not for re-installation of fittings or installation of pre-made ends.

(1) Figure 207 shows the use of the Swagelok inspection gauge and the indication of a correctly installed fitting. Note that if the gap inspection gauge does not fit between the nut and body hex, the fitting is sufficiently tightened.

(2) Figure 208 shows an improperly installed fitting. Note the gap inspection tool goes into the slot between the nut and the body hex. Additional tightening is required or further investigation may be required.

F. **Swagelok Fittings Retightening (Installing Pre-Swaged Fittings)**The Swagelok fittings may be disconnected and retightened many times and still provide a reliable, leak-proof seal if the following instructions are followed.

(1) Clean both parts of the fitting, including the tubing in the swaged area, and inspect for scratches, burrs, etc. If the tube and sealing area are damaged, replace part. Refer to Figure 209.

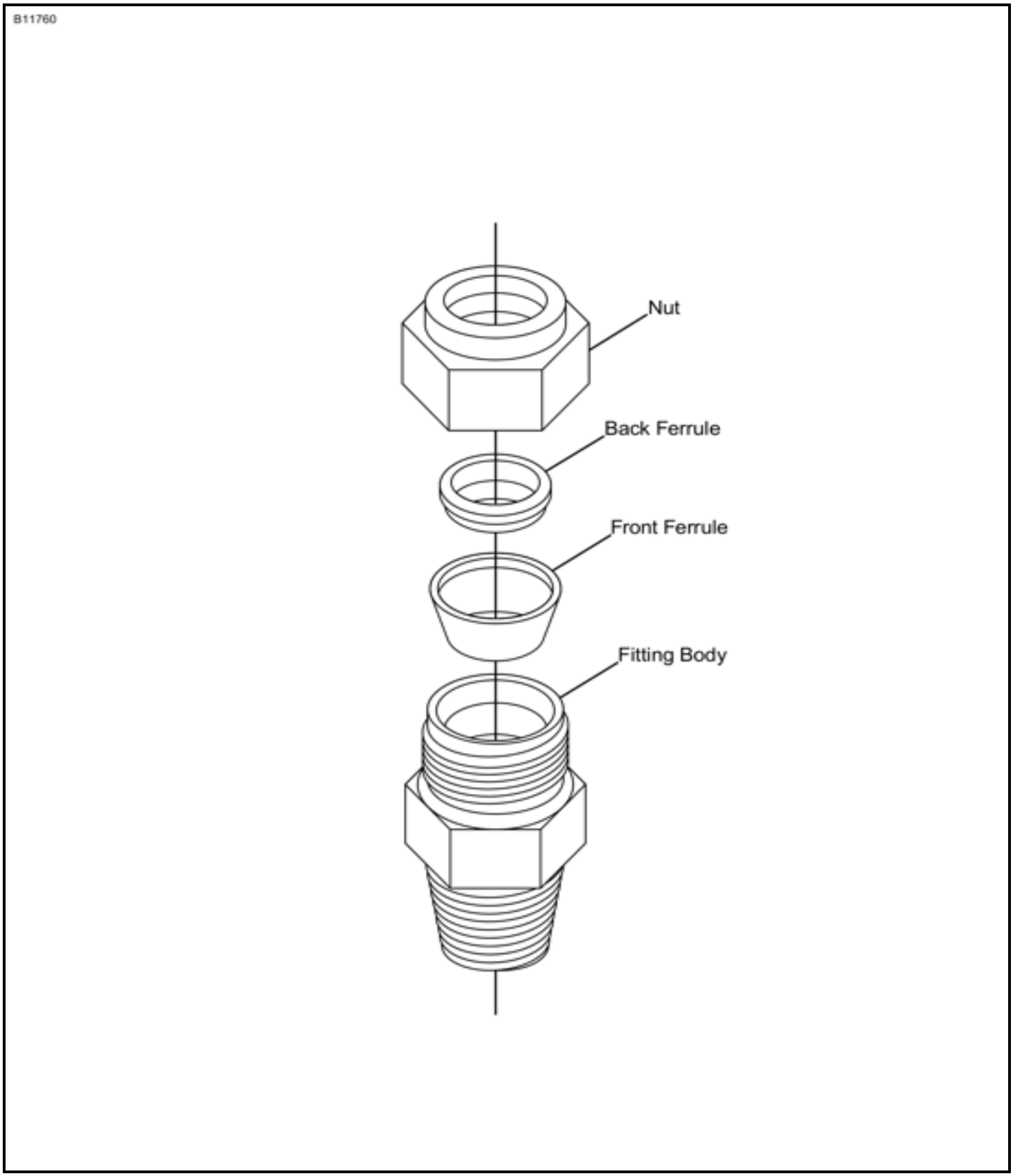
(2) Fully insert the tubing with the pre-swaged ferrules into the fitting body as shown in Figure 210, and finger-tighten the nut onto the fitting body.

(3) Rotate the nut to the original position with a wrench, stopping when an increase in resistance is felt. Tighten slightly past the original position using a wrench. Figure 211 shows a fully-tightened fitting.

NOTE: Smaller tubing sizes require less tightening to reach the previously pulled-up position. Larger tubing sizes require more tightening. Wall thickness can also affect the amount of torque required to retighten the fittings.

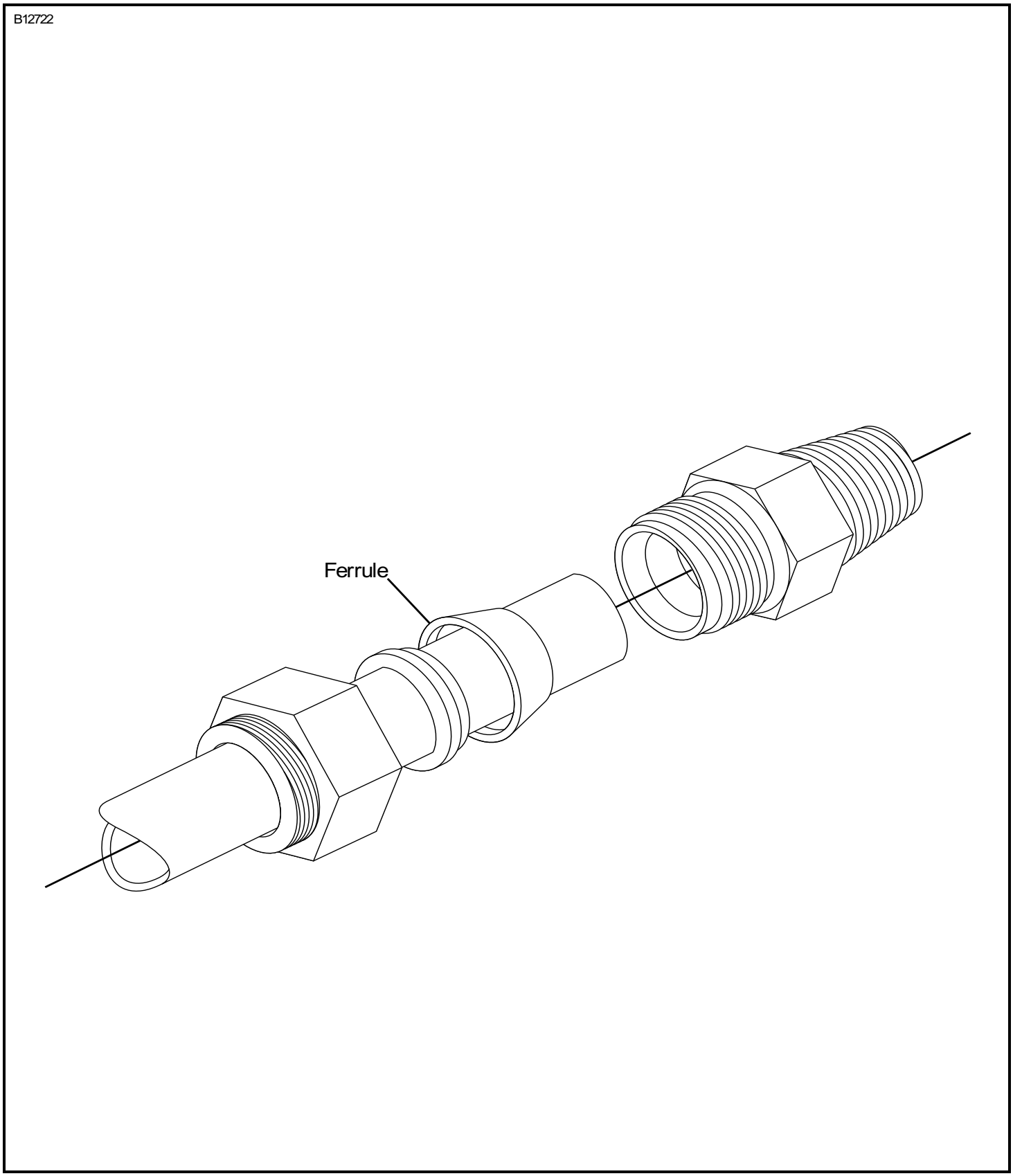
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Figure 201 : Sheet 1 : Swagelok Fitting (Exploded View)



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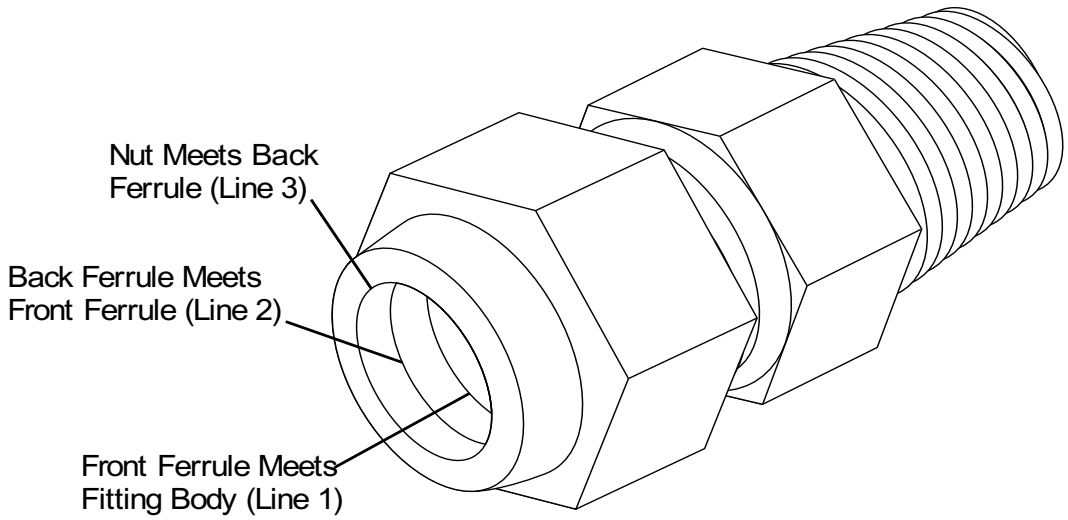
Figure 202 : Sheet 1 : Proper Ferrule Position



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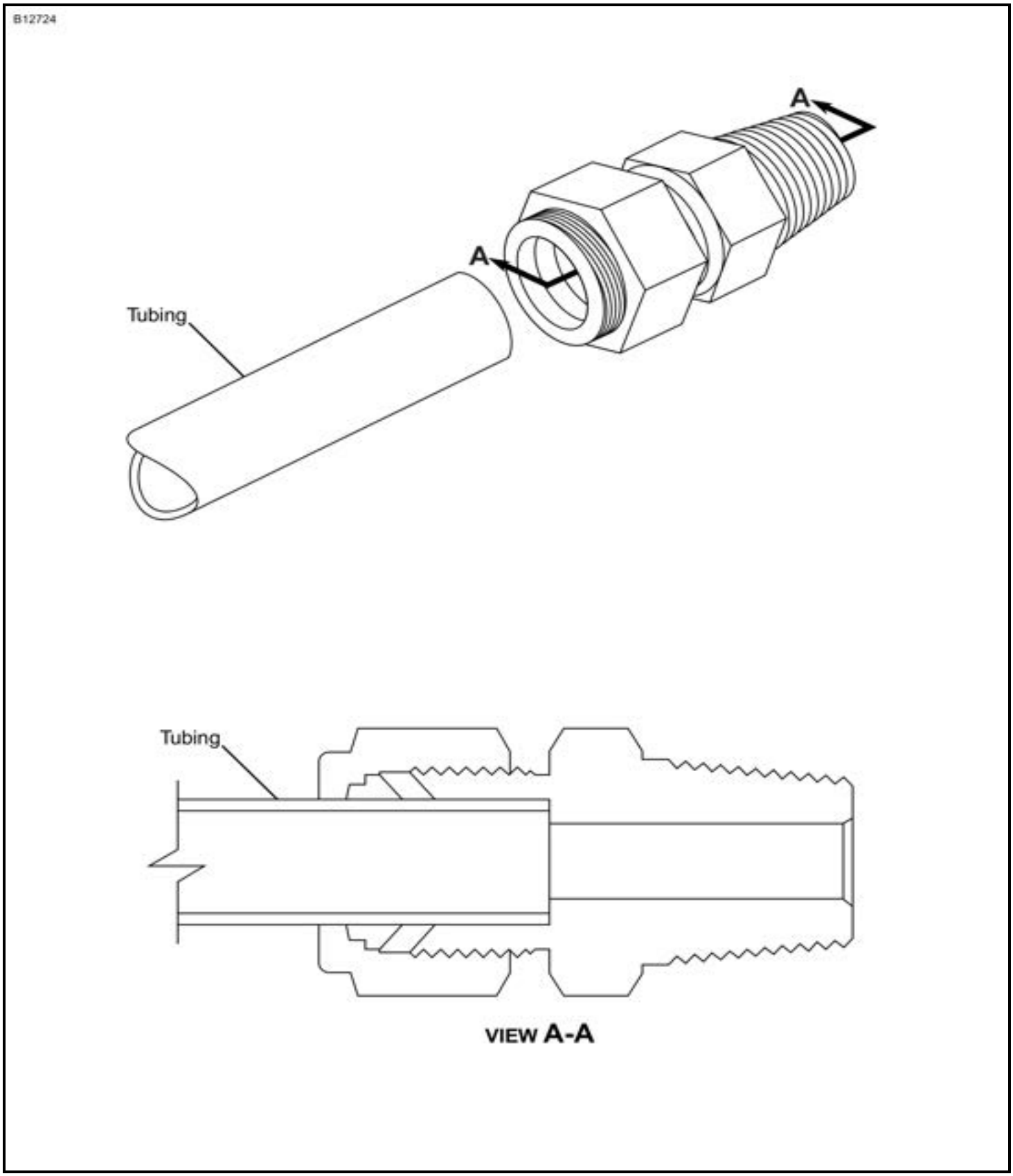
Figure 203 : Sheet 1 : Swagelok Fitting Visual Inspection

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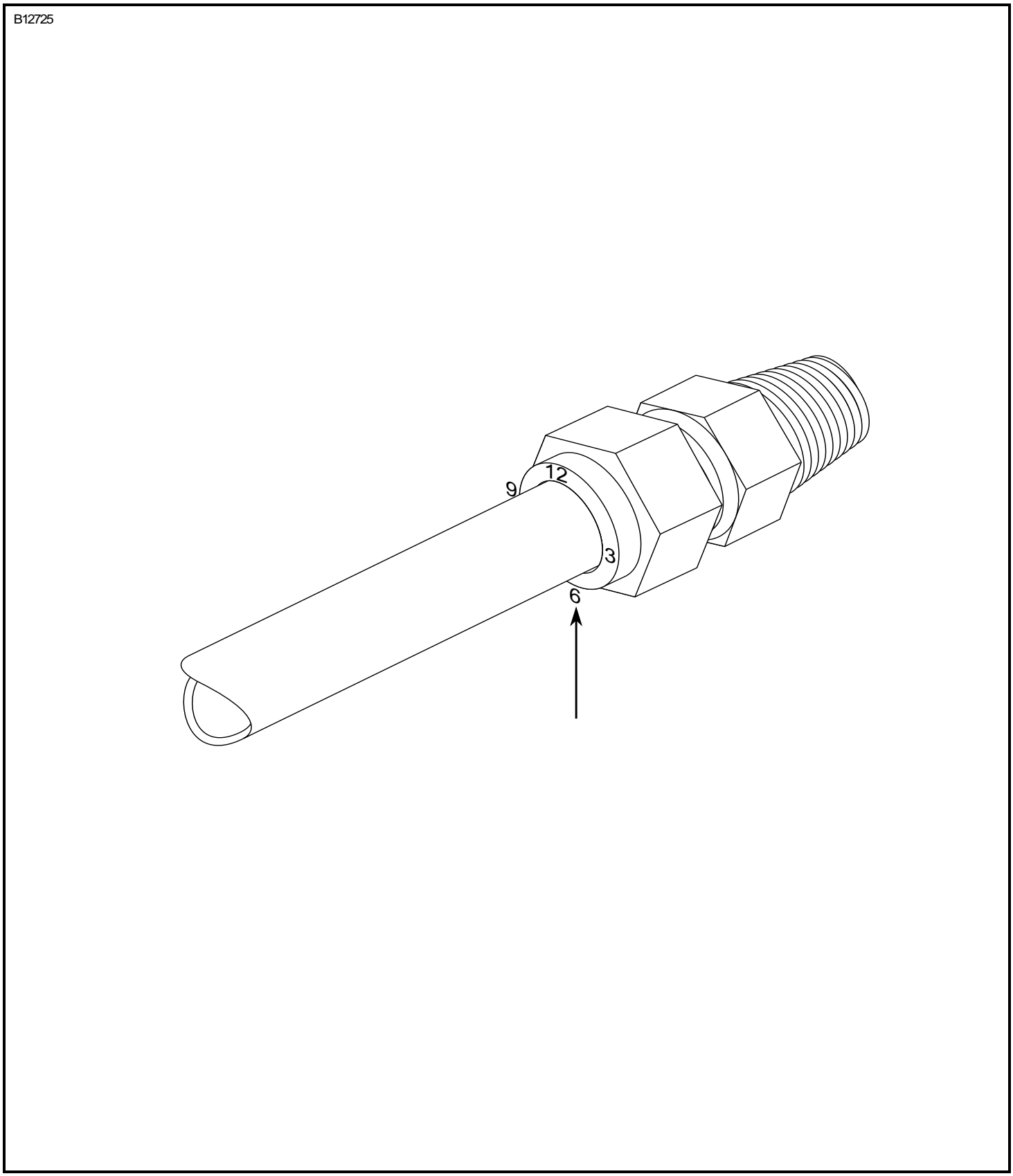
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Figure 204 : Sheet 1 : Fully Insert Tubing into Fitting



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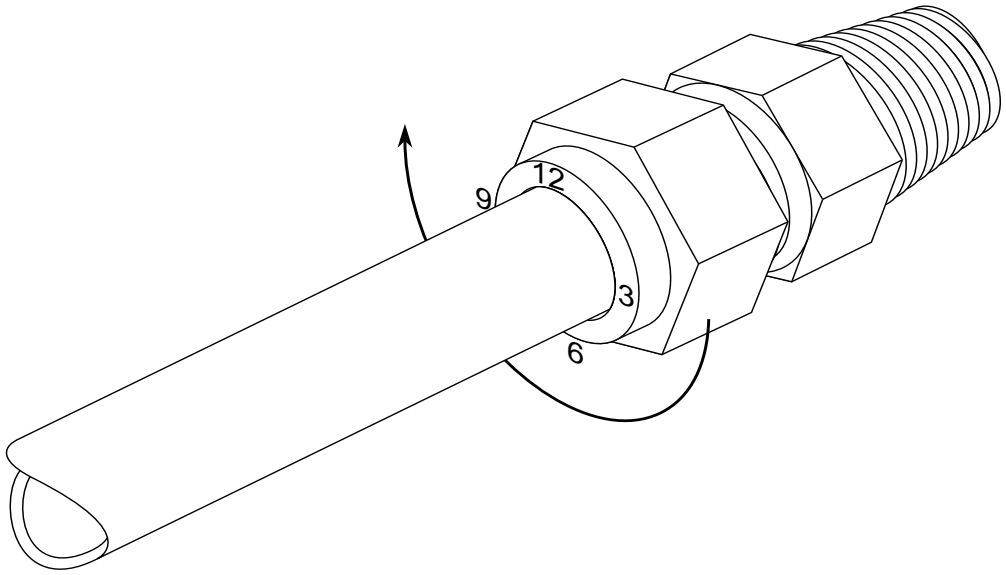
Figure 205 : Sheet 1 : Mark Nut with a Permanent Reference Mark



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Figure 206 : Sheet 1 : Tighten Nut 1-1/4 Turns for New Fittings

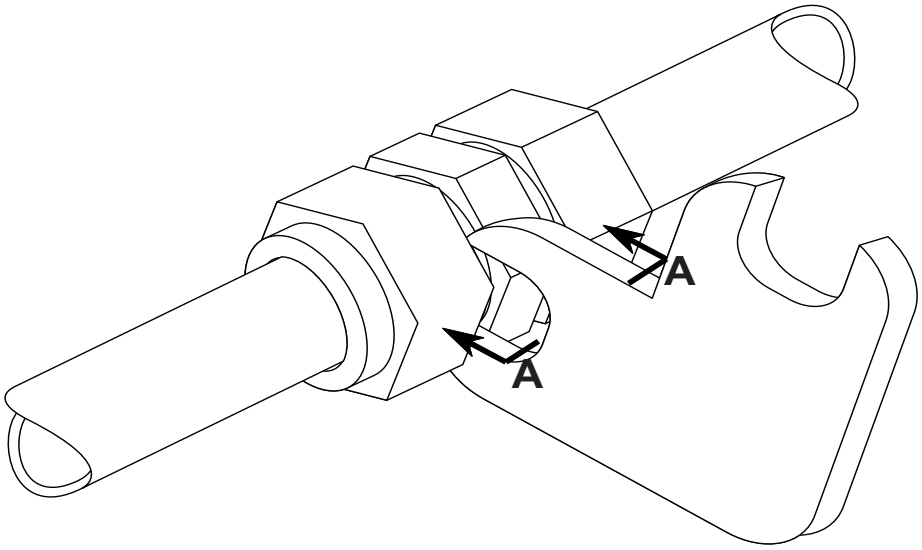
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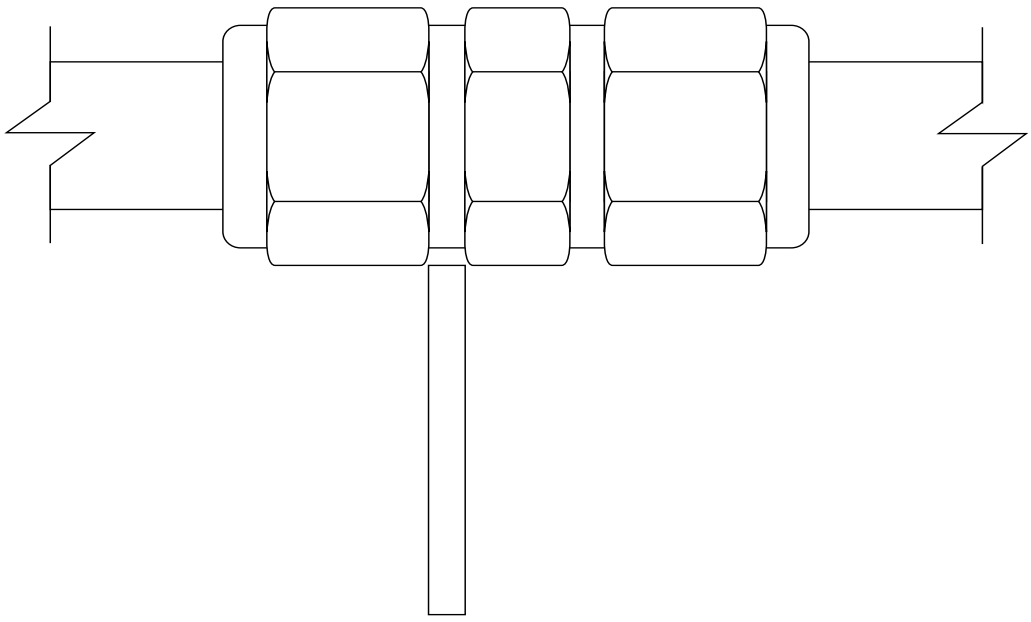


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Figure 207 : Sheet 1 : Insert Swagelok Gauge to Verify Correct Torque of Newly Installed Fitting

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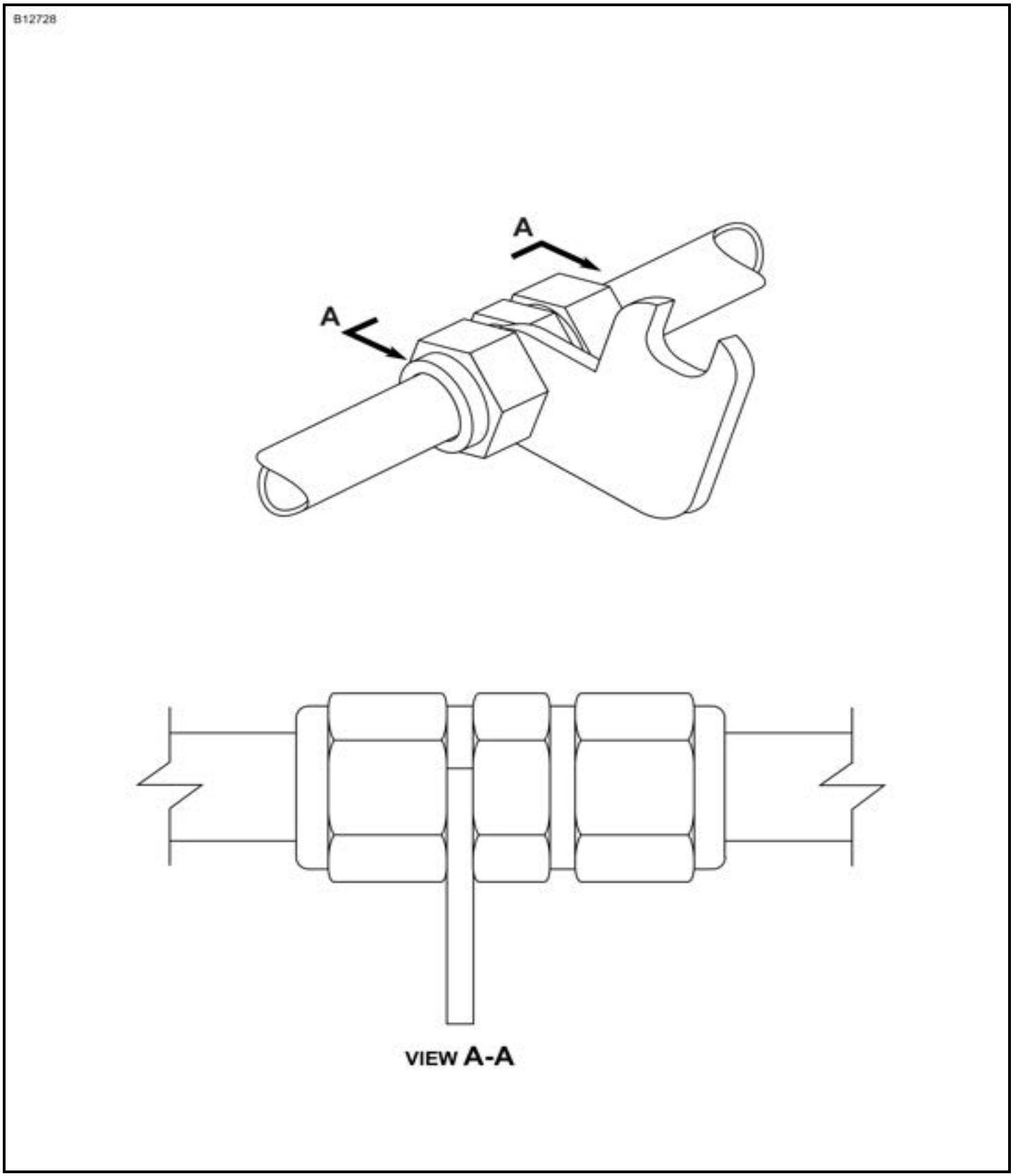




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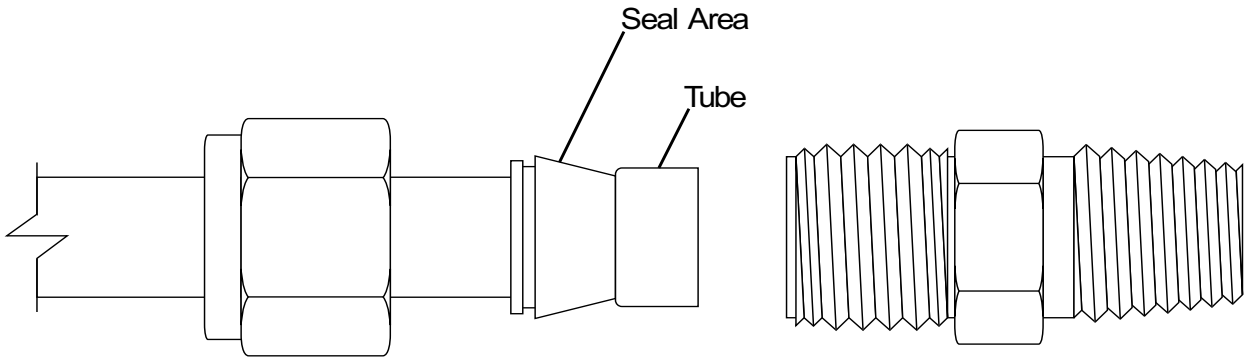
Figure 208 : Sheet 1 : Gauge Shows Fitting is Not Correctly Installed and Torqued



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Figure 209 : Sheet 1 : Disconnected Pre-Swaged Swagelok Fitting

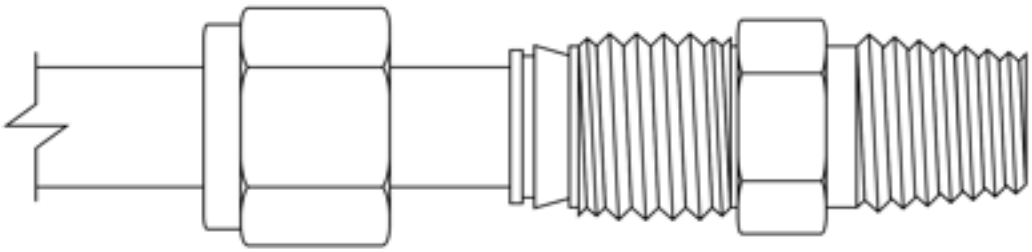
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Figure 210 : Sheet 1 : Inserted Pre-Swaged Swagelok Fitting

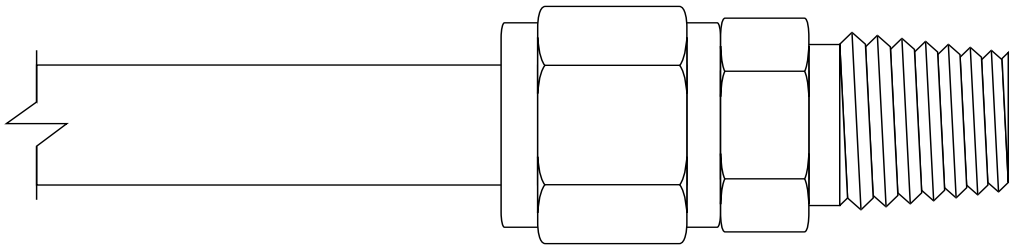
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Figure 211 : Sheet 1 : Tightened Pre-Swaged Swagelok Fitting

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TAP TESTING - DESCRIPTION AND OPERATION

1. General

A. This section contains information needed to complete tap testing procedures on the airplane. Information required to complete inspections of specific locations on the airplane can be found in the applicable section for that area of structure.

2. Inspection Requirements

A. Before the tap test inspection is started, you must comply with the following requirements:

(1) You must have a blunt-ended tap hammer (or equivalent) that weighs 1.0 ounce, +0.3 or -0.3 ounce.

(2) The inspector must be capable of distinguishing and differentiating audible pitch changes produced during the tap test.

(3) The airplane must be moved to an area free from loud noise produced by traffic, power equipment, aircraft or any other source that would require raised voices for normal conversation.

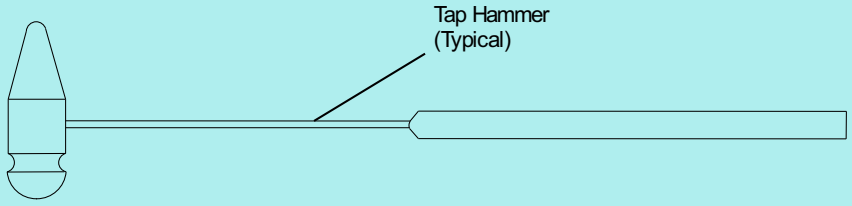
3. Tap Testing

A. Complete a Tap Test of the applicable structure.

(1) Thoroughly clean and dry the inspection surface to remove any loose dirt, contaminates and grease.

(2) Complete the tap test with the use of a blunt-ended tap hammer or equivalent with a head that weighs 1.0 ounce, +0.3 or -0.3 ounce.

B18310



CAUTION: Do not use excessive or aggressive force or a hammer with a sharp edge to strike the surface of the airplane. Structural damage will occur if too much force or a sharp-edged hammer is used to complete the tap test.

(3) Use gentle and consistent force with the tap hammer to lightly tap the surface in each 1-inch^2 area throughout the inspection zone.

(4) Compare similar structures within the inspection zone for changes in the acoustic response pitch.

NOTE: Areas of the fuselage that have structure behind them will have a higher response pitch than areas that do not have structure behind them.

(5) A "flat" or "dead" response pitch is considered unacceptable.

(6) Larger areas that are disbonded or delaminated can cause the tap hammer to rebound to a lower height than it would on intact structure.

(7) Use moderate pressure from your hand or several fingers to push on an area of structure.

(a) A faint "crackling" sound can often be heard if the area has core damage.

(8) Put a mark on damaged areas to indicate the location and size of the area where the damage is found.

(9) Put a mark on any areas where you are uncertain whether damage exists to indicate that the area is suspect. Additional Nondestructive Inspection (NDI) procedures must be completed to evaluate the area more closely.

(10) Cessna recommends that maintenance personnel that complete tap tests be composite repair technicians trained in accordance with SRM standards or certified NDI technicians.

4. Required Chapter 4 Inspection Areas

A. The inspections listed in Chapter 4 require that tap testing be completed in the following areas:

(1) **Fuselage Bonding Seams** - The fuselage halves are split into left and right symmetrical sides and include the areas from the firewall back to and including the vertical stabilizer. The seam extends from centerline to 2 inches on the right side of centerline. The seam tapers to 1.75 inches at approximately FS 240 and tapers again to 1.5 inches at

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approximately FS 261.5. The upper fuselage bond seam from the top of the windshield to a point 48 inches aft of the top of the windshield is an area of extremely thick ply buildup and can not be effectively tap tested. This area is only to be visually inspected.

(2) **Wing Skin Leading Edge and Trailing Edge Seams** - The wing skin seams are at the leading and trailing edges of the wing. The leading edge seam is 2 inches wide and centered about the leading edge. The trailing edge should be tap tested on the top and bottom of the edge from the edge to 2 inches inboard.

(3) **Wing Skin to Ribs and Spars** - The location of each rib and spar is shown in Section 57-10-00, Location of Spars and Ribs. The top and bottom wing skin at the areas of the ribs and spars should be tap tested.

(4) **Wing Flaps Leading and Trailing Edge Seams** - The flap seams are at the leading and trailing edges of the flap. The leading edge is 1 inch wide and extends 1 inch up from the leading edge chord plane of the flap. Because the bond extends onto core as shown in Figure 1, tap testing is only valid for the first 0.5 inches of the bond. The trailing edge should be tap tested on the top and bottom of the edge from the edge to 1.5 inches forward.

(5) **Aileron Leading And Trailing Edge Seams** - The aileron seams are at the leading and trailing edges of the ailerons. The leading edge seam is 1.0 inch wide, starting 1 inch along the surface from the leading edge of the aileron. In the area around the drive hinge attachment pocket, tap test approximately 1.0 inch from the edge of the part to verify the bonding surface. The trailing edge should be tap tested on the top and bottom of the trailing edge from approximately 0.90 inches from the edge to approximately 2.0 inches from the trailing edge as shown in Figure 2.

(6) **Rudder Leading and Trailing Edge Seams** - The rudder seams are at the leading and trailing edges of the rudder. The leading edge seam is the width of the flat leading edge of the rudder. The trailing edge seam should be tested as follows:

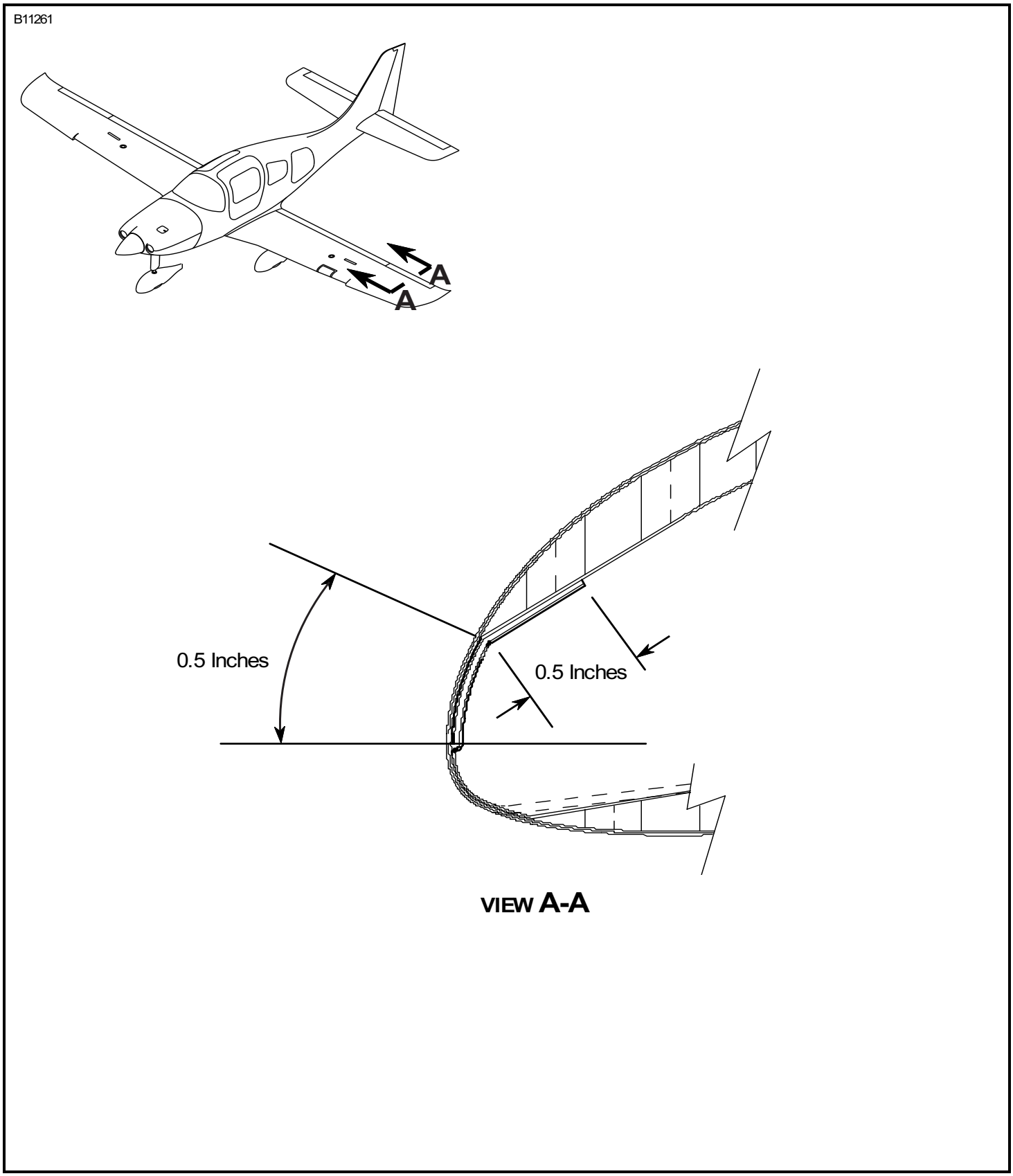
(a) The top 14 inches should be tested by tapping on the right hand side a distance of 1.5 inches inboard from the trailing edge.

(b) The remainder of the trailing edge should be tested 1.0 inches inboard from the trailing edge.

(c) Refer to Figure 3 for areas of tap testing.

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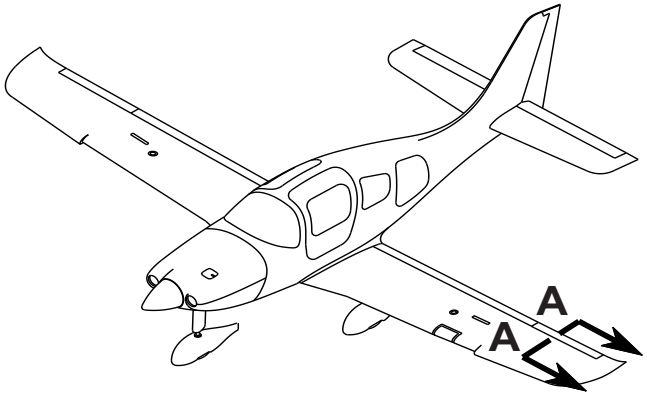
Figure 1 : Sheet 1 : Wing Flap Leading Edge Tap Testing Area

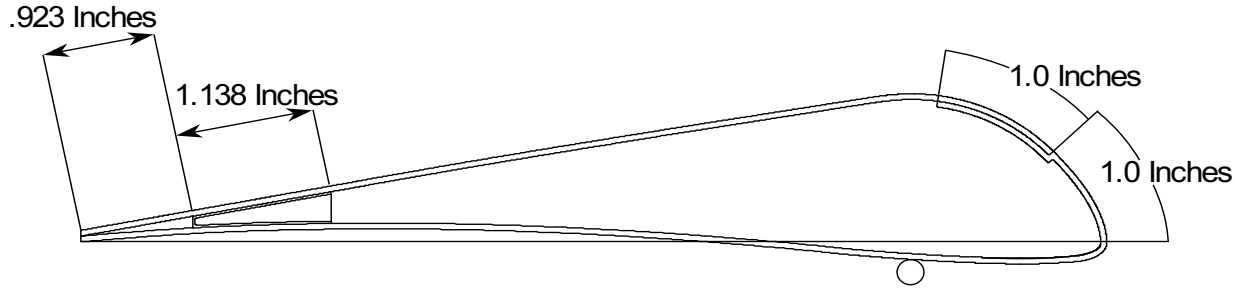


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Figure 2 : Sheet 1 : Aileron Tap Test Areas

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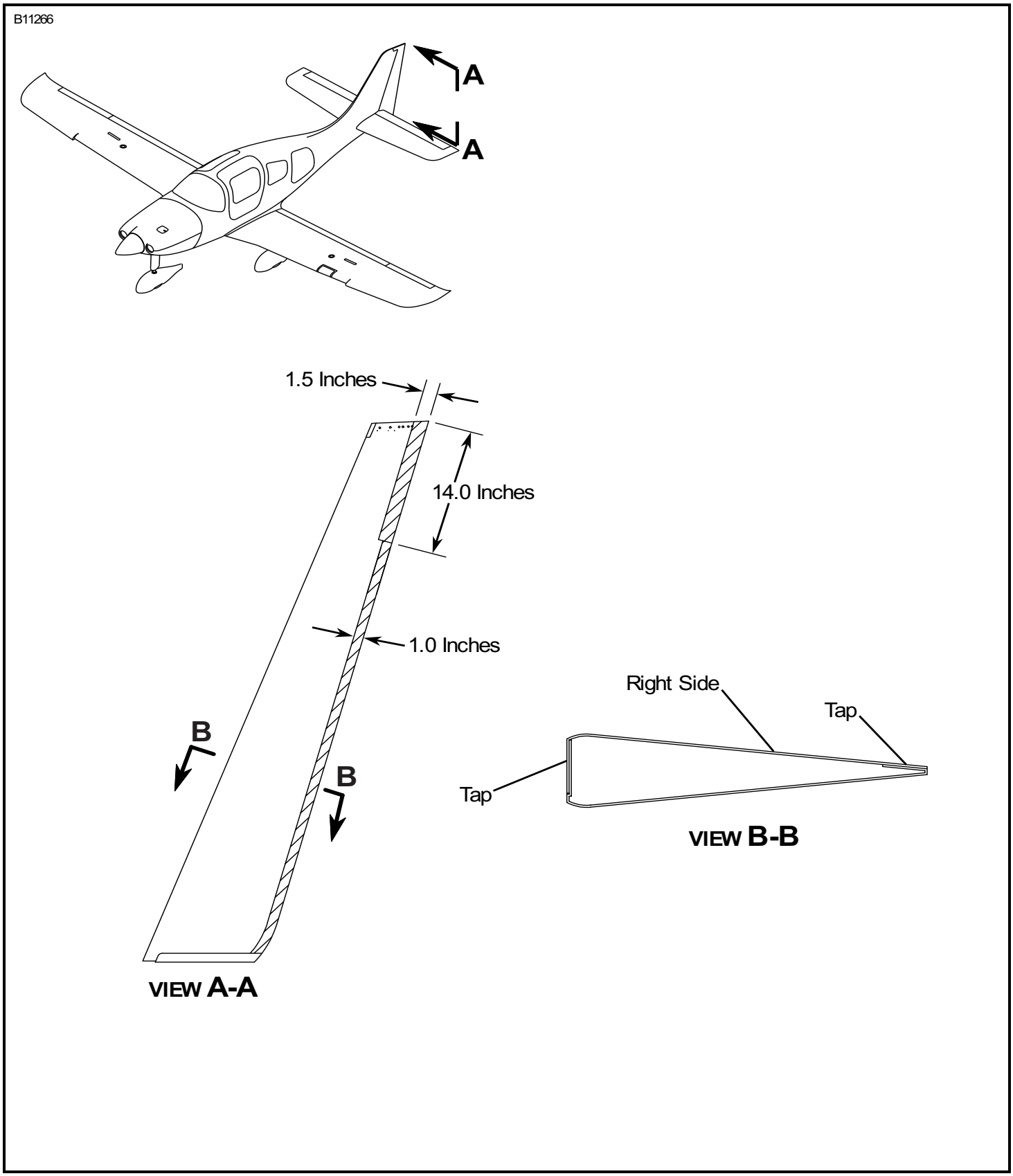




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Figure 3 : Sheet 1 : Rudder Tap Test Areas



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STRUCTURAL INSPECTIONS - DESCRIPTION AND OPERATION

1. General

A. This section contains information needed to complete inspections on the structure of the airplane. The procedures given here are the general requirements that must be followed to correctly complete inspections. More specific information relating to the area that is to be inspected can be found in the inspection procedure for each component or area of the airplane.

2. Visual Inspections

A. Equipment.

(1) The following equipment can be used to complete required visual inspections. Not all equipment is required for each inspection and additional tools or equipment can be required for inspections of specific areas of the airplane:

- Flashlight or other light source capable of producing 100 foot-candles at the inspection surface
- Inspection mirror
- Borescope
- Camera or camcorder

B. Personnel.

(1) The following are requirements that the inspector must meet to complete visual inspections of the airplane structure and components:

- Near Vision of 20/25 (Snellen) at 16 inches +1 or -1 inch or equivalent, in at least one eye, natural or corrected
- Capable of distinguishing and differentiating colors
- Familiar with the appearance of visible damage and have training and experience on visual inspection of aircraft structure

C. Access Requirements.

(1) Direct visual examination must be used whenever possible. For areas that are not accessible, the use of a borescope or digital camera or camcorder attached to a pole is recommended.

(2) The examination must be completed at a distance of not more than 24 inches from the surface of the component that is to be inspected.

(3) The examination must be completed at an angle that is within 30 degrees of perpendicular to the surface that is to be inspected.

D. Visual Inspection Procedure.

(1) Clean the inspection surfaces thoroughly to remove all loose materials, dirt, contamination and grease.

(2) Use a towel to dry the inspection surface.

(3) Use a light source held approximately perpendicular to the inspection surface to examine the structure or component.

(4) Use a light source held and an angle of approximately 15 degrees of parallel with the inspection surface to examine the structure or component.

(5) Examine the structure or components for the following:

- Paint that is blistered, swollen, cracked, deformed or missing
- Security of parts or attached components
- Discoloration
- Bucking
- Bulging
- Dents
- Oil-canning
- Wrinkles
- Chafing
- Foreign objects
- Fluid leaks.

(6) Put a mark on any damaged areas that shows the location and type of damage found.

(7) Put a mark on any areas where you are uncertain whether damage exists to indicate that the area is suspect.

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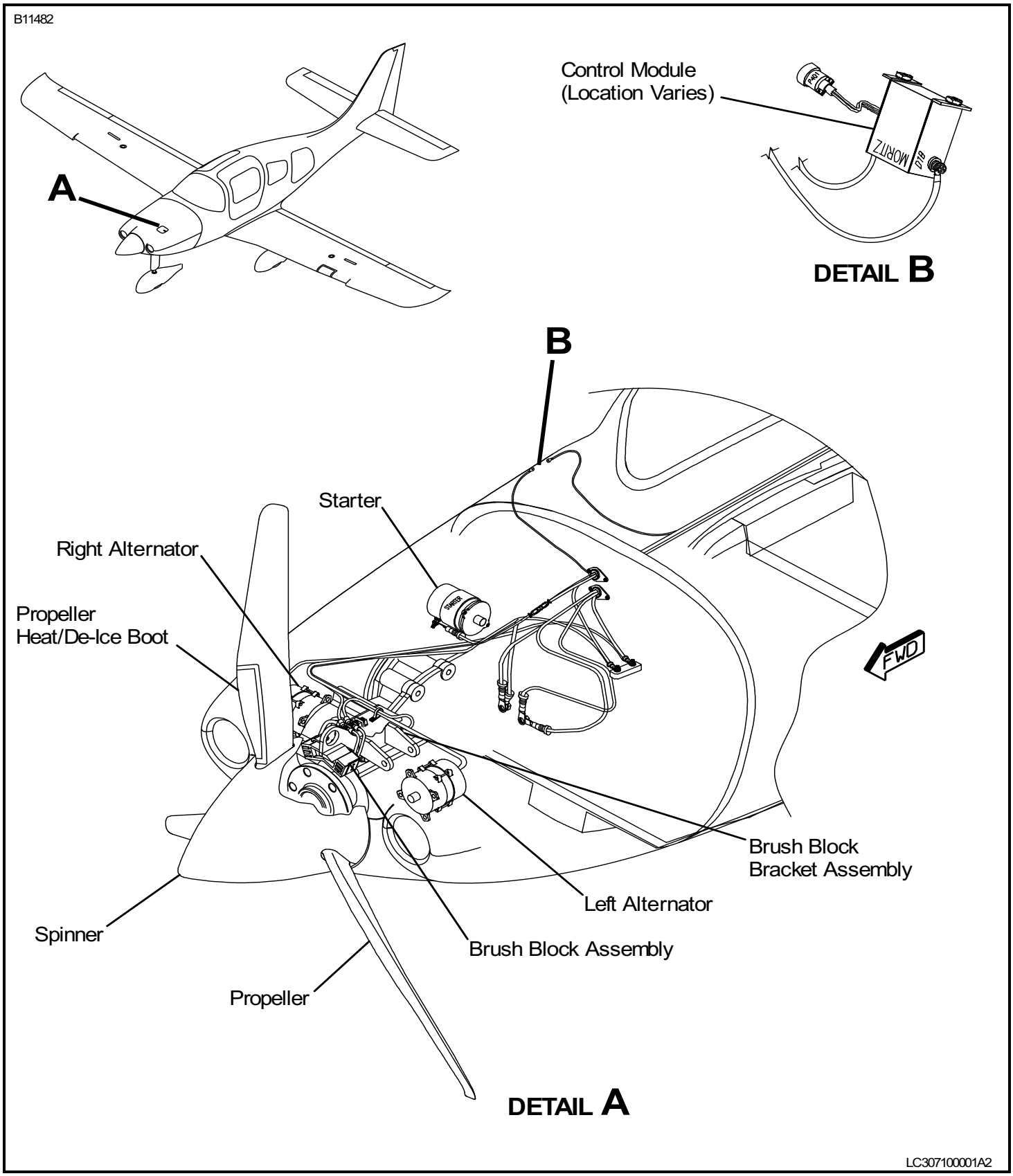
ICE AND RAIN PROTECTION - GENERAL

1. Propeller Heat/De-Ice - General

A. The Propeller De-Ice system is intended to remove ice from the propeller blades of the aircraft. The system is intended as an aid during inadvertent flight in icing conditions. The system is not intended to allow flight into known icing conditions.The Propeller De-Ice system is comprised of the propeller blade electric heater boots, slip ring assembly, propeller heat control module, brush block and bracket assembly, control switch with integral annunciator, 30 amp fuse on the power grid, and miscellaneous covers, hardware, and wire. Refer to Figure 1. The pilot controls the system by the Prop De-Ice Control Switch. After setting the Control Switch to the ON position the controller will conduct a self test and then illuminate the annunciators ON and PROP in green. The system operates on a 90 second on, 90 second off, electrical cycle.

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Figure 1 : Sheet 1 : Propeller Heat/De-Ice System



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PROPELLER HEAT - MAINTENANCE PRACTICES

1. Brush Block

A. **Brush Block Removal** - (Refer to Figure 201.)

(1) Remove the top engine cowling per Chapter 71-10-00.

(2) Remove two insulated terminal screws and star washers, and disconnect the brush leads from the insulated terminal of the brush block bracket.

(3) While holding the brush block, remove the two screws, washers, and spacers attaching the brush block to the brush block bracket.

(4) Slowly pull the brush block away from the slip ring to relieve pressure on the brushes.

(5) Remove the brush block taking care not to damage the brushes.

B. **Brush Block Installation** - (Refer to Figure 201.)

(1) Press the brushes into the brush block.

CAUTION: Use care when handling the brushes. Skin oils may contaminate the brush material and reduce brush service life.

(2) With the brushes compressed into the brush block, wrap small tie straps around the brush block to hold the brushes in place. Refer to Figure 202.

(3) Attach the brush block to the brush block bracket using two AN-503-10-20 screws, AN960-10 or NAS1149F0363P washers, and Northcoast DI0029 spacers. Leave the attachment screws just loose enough to allow movement of the brush block assembly fore and aft with minimal effort.

(4) Attach the wire terminal ends for the top brushes to the forward insulated terminal screw and for the middle brushes to the middle insulated terminal screw. Refer to Figure 203. Place star washers under the head of the screws and tighten snug. The bottom brush leads are not used and should be coiled and secured using tie straps.

(5) Clean the slip rings using methyl n-Propyl ketone, acetone, or spray de-greaser.

(6) View the alignment of the brushes and the slip ring from the side. The brush should ride in the center of the slip ring. Refer to Figure 204. If alignment is necessary, install NAS1149F0316, NAS1149F0332, or NAS1149F0363 washers between the spacer and brush block to achieve correct alignment. Different thickness washers may be stacked together.

(7) Cut and discard the tie straps holding the brushes into the brush block to allow the brushes to contact the slip ring. Press the brush block up to the slip ring and tighten the mounting screws to allow the brush block to be adjusted with moderate force. Adjust the brush block so the left side has approximately .06 inch clearance and the right side has approximately .09 inch clearance when viewed from the front of the engine looking down. Refer to Figure 205. Tighten the mounting screws snug and secure with .032 safety wire per MS33540.

C. **Brush Block Maintenance** - Inspect the brush block during the 100 hour and/or annual inspections. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance.

(1) Inspect for grease, fuel, oil, or carbon deposits on the brushes and brush block. Clean with methyl n-Propyl ketone or replace components if unable to be cleaned.

(2) Measure brush wear. Replace the brushes if wear exceeds 50 percent or if less than 5/16 inch of brush material remains.

(3) Inspect for uneven wear, chipping, or broken brushes. Replace components as necessary.

(4) Verify brush to slip ring alignment per Brush Block Removal procedures.

D. **Brush Block Repair** - The brushes are non-repairable and should be replaced if found to be defective.

2. Slip Ring

A. **Slip Ring Removal**

(1) Remove the upper and lower cowling per Chapter 71-10-00.

(2) Remove the brush block per the Brush Block Removal procedures.

(3) Remove the spinner, propeller, and left alternator drive pulley per Chapter 61-10-00.

(4) Remove the slip ring.

B. **Slip Ring Installation**

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(1) Place the slip ring on the engine crankshaft flange.

(2) Install the left alternator drive pulley, propeller and spinner per Chapter 61-10-00.

C. **Slip Ring Maintenance**

(1) Clean the slip ring with MPK or other suitable solvent.

(2) Visually inspect the slip ring for excessive wear, grooves, rough surfaces, cracks, burned or discolored areas, and oil or dirt deposits. Inspect for any de-bonding of the copper rings from the slip ring plate

(a) If uneven wear or wobble is noted, verify that slip ring run out is less than 0.005 inches

(b) Grooves should be less than 0.007 inches deep.

(c) Small defects can be removed with crocus cloth. The rings should have a minimum 32 micro finish.

D. **Slip Ring Repair** - The slip ring can be machined to achieve run out and surface finish requirements. The minimum slip ring total thickness is 0.437 inch. The Maximum allowable slip ring run out is 0.005 inch. Surface finish should be 32 micro-inch or less.

CAUTION: Excessive heat build-up during machining can cause the copper ring to delaminate from the epoxy. Use appropriate cutting fluid and feed rates to minimize heat build-up during any machining operation on the slip rings

3. Brush Block Bracket

A. **Brush Block Bracket Removal** - (Refer to Figure 201.)

(1) Remove the top engine cowling per Chapter 71-10-00.

(2) Remove the No.2 (left) alternator per Chapter 24-30-00.

(3) Remove the brush block per Brush Block Removal Section.

(4) Remove the two AN5-24 bolts, two MS21042L-5 nuts, four AN960-516 washers, and four Northcoast DI-00030 spacers attaching the No. 2 (left) alternator bracket and brush block bracket to the crankcase. Refer to Figure 201.

(5) Remove the bracket.

B. **Brush Block Bracket Installation** - (Refer to Figure 201.)

(1) Install the brush block bracket and No.2 (left) alternator bracket using two AN5-24 bolts, two MS21042L-5 nuts, four AN960-516 washers, and four Northcoast DI-00030 spacers

(2) Torques nuts 180 to 220 in.-lb.

(3) Install the brush block per instructions in Brush Block Installation section.

(4) Install the No.2 (left) alternator per instructions in Chapter 24-30-00.

(5) Install the top engine cowling per instructions in Chapter 71-10-00.

C. **Brush Block Bracket Maintenance** - Inspect the brush block bracket for cracks during the 100 hour and/or annual inspections. If cracks are present, replace the bracket. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance.

4. Propeller Heater Boot

A. Propeller heater boots installed on the aircraft are manufactured by SMR Technologies Inc. or Hartzell Propeller Inc. If you are uncertain of the manufacturer of the propeller heater boots installed on the aircraft contact Cessna for guidance.
Remove and install propeller heater boots per instructions in the latest revision of SMR Technologies Report SMR-97-33-013, *Propeller De-Icer Installation and Maintenance Manual* or *Hartzell Propeller Manual 182* (61-12-82), Propeller Electrical De-ice Boot Removal and Installation Manual, as applicable.
Repair or service propeller heater boots per instructions in the latest revision of SMR Technologies Report SMR-97-33-013, *Propeller De-Icer Installation and Maintenance Manual* or *Hartzell Propeller Manual 181* (30-60-81), *Propeller Ice Protection System Component Maintenance Manual*, as applicable.
Copies of these documents can be obtained from the sources that follow:

SMR Technologies Inc.
93 Nettie Fenwick Rd.
Fenwick, WV 26202
Telephone: 304.846.6636

Hartzell Propeller Inc.
Attn: Product Support Administrative Assistant
One Propeller Place
Piqua, Ohio 45356-2634

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Toll Free: 800.767.6899	Telephone: 937.778.4200
Fax: 304.846.6268	Fax: 937.778.4391
Website: www.lceshield.com	Website: www.hartzellprop.com

(1) **Heater Boot Removal** - (Refer to Figure 206 or Figure 207.)

(a) Remove the propeller spinner per instructions in Chapter 61-10-00.

(b) Remove the cowling per instructions in Chapter 71-10-00.

(c) Remove the propeller heater boots per the latest revision of SMR Technologies Report SMR-97-33-013 or *Hartzell Propeller Manual 182* (61-12-82), as applicable. Make a record of lead wire routing and attachment hardware for reinstallation.

(2) **Heater Boot Installation** - (Refer to Figure 206 or Figure 207.)

(a) Inspect the heater boot leads for proper resistance. Resistance must measure 1.56 to 1.72 ohms (14 V) or 4.7 to 4.9 ohms (28 V) at 77±5° Fahrenheit.

(b) Inspect the individual components for damage prior to installation.

(c) Install the propeller heater boots per the latest revision of SMR Technologies Report SMR-97-33-013 or *Hartzell Propeller Manual 182* (61-12-82), as applicable.

(d) Ensure the SMR boots are positioned so the heat zone is located 1.7±0.060 inches from the edge of the propeller hub; this will place the edge of the boot approximately 1.25 inches from the edge of the propeller hub. Ensure the Hartzell boots are positioned so the edge of the boot is 1.25 inches from the edge of the propeller hub.

(e) Route and attach the heater boot leads.

1 SMR Boots

a Route the heater boot leads between the propeller hub and the inside diameter edge of the spinner bulkhead.

b Loosely install the lead clamps using screws, washers, and nuts over the heater boot leads and onto the forward side of the spinner bulkhead.

c To relieve tension caused by centrifugal force, gently pull the heater boot leads up thought the lead clamps far enough that the lead rests against the inside surface of the spinner dome. Tighten the clamp hardware 20 to 25 in-lbs to secure the leads.

2 Hartzell Boots

a Route the heater boot leads between the propeller hub and the spinner bulkhead. Do not twist the boot lead wires or allow them to be twisted.

b Loosely install the loop clamps using screws, washers, and self-locking nuts.

c To relieve tension caused by centrifugal force, gently pull the heater boot leads up thought the loop clamps far enough that the lead rests against the inside surface of the spinner dome. Tighten the clamp hardware 22 to 25 in-lbs to secure the leads.

d Attach the slip ring wire harness to the spinner bulkhead using bushings, washers, screws, and self-locking nuts. If a self-locking nut configuration is not already present for attaching the boot leads to the spinner bulkhead, replace the top terminal nut with an MS21083C06 self-locking nut; replace the screw with a Hartzell screw (P/N B7035-14) as necessary to provide two thread protrusion through the self-locking nut. Orient wire terminals at the spinner bulkhead at a normal (perpendicular) angle to the spinner dome. Refer to Figure 207. Torque nut 10 to 12 in-lbs.

(f) Connect the terminal ends of the propeller boot leads (SMR boots) or the slip ring wire harness (Hartzell boots) to the slip ring terminals with 6-32 washers and nuts. Refer to Figure 208. For each heater lead use the two outer terminals only. The inner terminal is not used. Torque the retaining nuts 10 to 12 in.-lb.

CAUTION: Do not bend or damage the heater leads.

(g) Verify that propeller pitch change over the allowable range will not cause binding or stress on the heater leads.

(h) Install large tie straps over the three heater boot leads and around the propeller hub. Route the leads over the ridge in the hub. Ensure the leads are not pulled tight against the ridge to prevent chaffing. The leads should

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have a slight gap over the ridge edges of approximately 0.125 inch. Offset the ends of the tie straps 180 degrees to minimize any imbalance. Tighten the tie straps snug. Refer to Figure 209 for clarification and for spacing of the tie straps.

(i) Install the cowling per Chapter 71-10-00.

(j) Install the propeller spinner per Chapter 61-10-00.

(3) **Heater Boot Maintenance** - Inspect the heater boots during the 100 hr and/or annual inspections. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance.

(a) Visually inspect the heater boots for wrinkles, loose or torn areas, and/or abrasions or cuts along the leading edge and on the thrust face (flat) of the propeller blades.

(b) Inspect the boot leads for signs of abrasion or damage. Inspect the lead clamps for tightness.

(c) Test the resistance of the heater boot elements. Disconnect the boot leads from the slip ring terminals. Place an ohmmeter across the boot leads. The ohmmeter should indicate 1.56 to 1.72 (14 V) or 4.7 to 4.9 ohms (28 V) of resistance at 77±5° F. If the reading is not in this range replace the heater boot.

(4) **Heater Boot Repair** - The heater boots should be serviced and repaired per SMR Technologies Report No. 97-33-013 or Hartzell Propeller Manual 181 (30-60-8812), as applicable.

5. Control Module

A. The control module is located in one of three places. If Avidyne avionics are installed the control module is attached to the standby battery bracket (refer to Figure 210, Sheet 1) or to the right side instrument panel brace (refer to Figure 210, Sheet 2). If the Garmin G1000 system is installed the control module is located behind the instrument panel on the right hand side.

(1) **Control Module (Basic or Avidyne Option)**

(a) **Control Module Removal**

1 Remove the co-pilot's side kidney panel per Chapter 25-11-00.

2 Disconnect the cabling to the control module.

3 Remove two screws, washers, and nuts attaching the control module to the mounting bracket.

(b) **Control Module Installation** - Installation is the direct reverse of removal.

(2) **Control Module (Garmin G1000 Option)** - Refer to Figure 211. For the Garmin G1000 option, the control module is located behind the instrument panel on the right hand side.

(a) **Control Module Removal**

1 Remove the glare shield per Chapter 25-11-00.

2 Disconnect the cabling to the control module.

3 Remove two 8-32 pan head screws attaching the control module to the mounting bracket.

(b) **Control Module Installation** - Installation is the direct reverse of removal.

(3) **Control Module Maintenance** - Inspect the control module during the 100 hr and/or annual inspections. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance. The propeller de-ice system has a high current draw; use ground power or start the engine.

(a) With aircraft power on, actuate the propeller de-ice system by pressing the de-ice control switch. Verify the power on self test performs correctly.

1 When the propeller de-ice control switch is powered on, the ON and FAIL annunciators will illuminate for one second and extinguish for one second. The on-off cycle will repeat during the six second self test. After the self test is completed the PROP and ON annunciators will illuminate steady for 30 seconds, indicating the DEICE is in the ON cycle. The 'Prop' annunciator will remain illuminate whenever the Propeller de-ice control switch is powered on.

(b) Inspect attachment of the control module to the support bracket for loose hardware.

(c) Inspect the wiring for damage or abrasion.

(4) **Control Module Repair** - The control module is non-repairable and should be replaced if found to be defective.

6. Testing

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A. Before testing, assure that all electrical connections are secure and the propeller has been reinstalled. When the Prop De-Ice Control Switch is powered on, the Prop De-Ice Control Module will perform a self-test. The ON and FAIL annunciators will illuminate for one second and extinguish for one second during a six second cycle. The PROP DE-ICE annunciator will remain illuminated whenever the prop de-ice control switch is powered on. Cycle testing should be performed with the cowling and spinner installed and the engine running.

CAUTION: Do not run the engine with the spinner removed. Damage to the propeller boot leads will occur.

(1) **Ground Test** - The ground test is best accomplished with two persons. One person actuates the system from the cockpit and the second person verifies heater output at the propeller.

(a) **Basic Operation Test** - The Basic Operation Test verifies the general electrical connections.

1 Connect a source of external power to the aircraft.

2 Power the right bus

3 Actuate the prop de-ice switch, and wait for the controller set test to complete.

4 Assure the ON annunciator is illuminated.

5 Verify the three propeller de-icers are generating heat by carefully touching the boots.

6 After the system is verified to be heating, deactivate the prop de-ice switch.

CAUTION: Do not operate the propeller de-ice system for prolonged periods. Overheating and damage will occur.

(b) **Cycle Time Test** Perform the Ground Cycle Test with the engine running to allow for proper cooling of the propeller boots. When the pilot presses the propeller de-ice switch, the prop de-ice system will perform a self-test for six seconds. After the self-test is completed the PROP and ON annunciators will illuminate steady for 30 seconds, indicating the propeller deice system is in the ON cycle. After 30 seconds the propeller de-ice ON annunciator will flash, indicating the propeller de-ice is in the OFF cycle. Propeller de-ice 'ON' will continue to cycle 30 seconds steady state, and 30 seconds flashing until the pilot de-selects the propeller de-ice switch

1 Actuate the propeller de-ice switch and wait for the controller self test to complete.

2 The 'ON' annunciator should remain steady for 30 seconds. Time the ON portion of the cycle with a stopwatch.

CAUTION: The propeller boot heaters will become extremely hot during operation. Exercise care when touching the propeller boots or adjacent areas.

3 When the ON annunciator begins to flash, the system is in the OFF portion of the cycle. Time this portion of the cycle with a stopwatch. The time should be 30 seconds. Verify the three propeller boots are cooling off by carefully touching the boots.

4 Repeat the cycle time test to verify consistent in the cycle times.

(c) **Fault Indications** - The Troubleshooting paragraph of this chapter charts the fault indications along with probable causes and remedies.

1 The FAIL light and the ON light illuminate at the same time during the test - This situation indicates the Propeller De-Ice system is drawing less than 18 amps. Check for loose or open circuits, or heater or brush failure.

2 The FAIL light illuminates and the ON light extinguishes during the test - This situation indicates the current draw is too high, above 30 amps. If this condition occurs, check for a short to ground in or around the brushes or power leads. Another cause could be a faulty propeller de-ice system control module, or no or low power at the bus terminal of the control module.

NOTE: The propeller heaters will be hotter at the root than at the outboard section; this is normal due to varying watt density in the heating element and is not cause for concern.

(2) **In-Flight Testing** It is recommended that the propeller de-ice system be operated during flight periodically to test the system. It is suggested that the system be turned off when not needed or when clearing the runway after landing. The propeller de-ice system may be activated prior to take off. Assure that the current draw is between 19 and 25 amps.

(a) Actuate the propeller de-ice switch and wait for the controller self test to complete.

(b) The 'ON' annunciator should remain steady for 90 seconds. Time the ON portion of the cycle with a stopwatch.

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(c) When the 'ON' annunciator begins to flash, the system is in the OFF portion of the cycle. Time this portion of the cycle with a stopwatch. The time should be 90 seconds. Verify the three propeller boots are cooling off by carefully touching the boots.

(d) Repeat the cycle time test several times to verify consistent cycle times.

7. Propeller Heat/De-Ice - Troubleshooting

Table 201. Propeller Heat/De-Ice - Troubleshooting

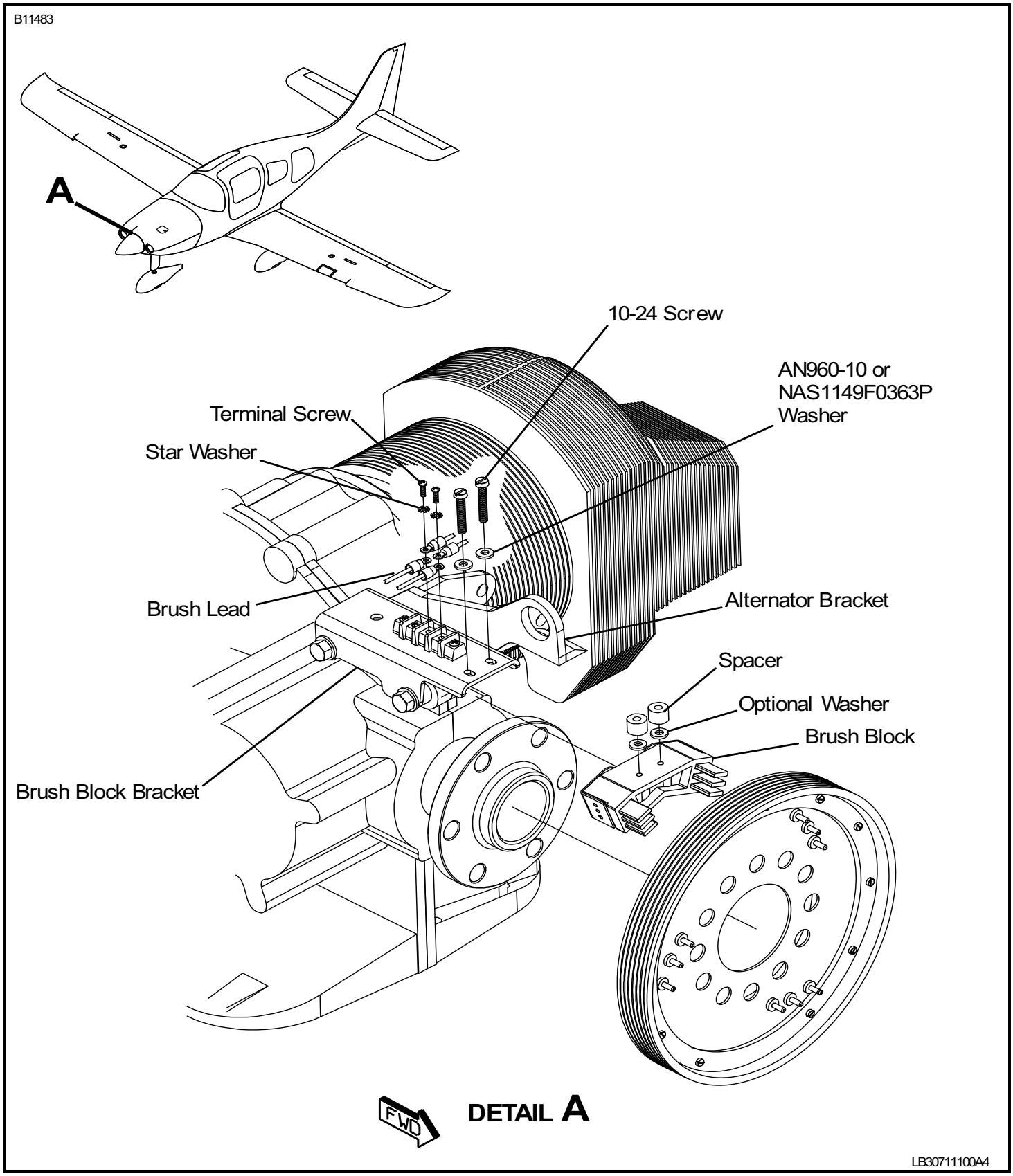
Annunciator Lights	Trouble Indication	Probable Cause	Inspection	Remedy
ON - Steady On FAIL - Steady On	Low Current Draw (Under 18 A)	Heater boot open.	Electrically isolate each boot and check for continuity.	Replace boot if internal open.
				Tighten boot electrical connection at slip ring terminal.
		Brush failure.	Inspect brushes for condition and proper contact with slip ring.	Replace brushes.
				Align brushes.
		Open in power wiring.	Inspect wiring to verify condition. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
ON - Off FAIL - Steady ON	High Current Draw (Over 30 A)	Short to ground in power wiring.	Inspect wiring to verify condition. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
		Short to ground at brush block.	Inspect brushes for condition and short to ground.	Replace brushes.
				Align brushes.
		Short between slip ring commutators.	Inspect slip ring assembly to verify electrical isolation.	Replace slip ring assembly.
	Low Voltage at Control Module	Loose connection at fuse or Control Module.	Inspect wiring to verify condition. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
		Faulty control module.	Test control module.	Replace control module.
	Radio Noise or interference when the De-Ice system is ON	Brushes arcing.	Inspect brushes for evidence of burning.	Replace brushes.
				Align brushes.
		Loose connection.	Inspect wiring to verify condition. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
		Wire too close to radio equipment.	Inspect wiring to verify location.	Reroute wiring to at least 8 inches from radio equipment and antenna leads.
		Defective fuse.	Place jumper across fuse.	If radio noise disappears, replace the fuse.

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ON - Off FAIL - Steady ON		Defective control switch.	Place jumper across pins 1 and 8 and pins 4 and 10 on the control switch.	If radio noise disappears, replace the control switch.
	Rapid or uneven brush wear or brush chipping	Brush assembly out of alignment.	Inspect brushes for condition and proper contact with slip ring.	Align brushes.
		Slip Ring runout.	Verify slip ring runout is less than 0.005 inches	Replace slip ring.
		Slip Ring surface defects.	Visually inspect surface condition.	Resurface slip ring if within limits.
				Replace slip ring.
		Oil leak at propeller hub or engine.	Visually inspect slip ring for oil.	Correct oil leak condition.
	System does not actuate when control switch is selected.	Faulty control switch.	Bypass control switch and verify system operation.	Repair wiring as necessary. Tighten all connections.
		Faulty or blown fuse.	Replace fuse. If fuse blows again inspect wiring for shorts.	Replace fuse.

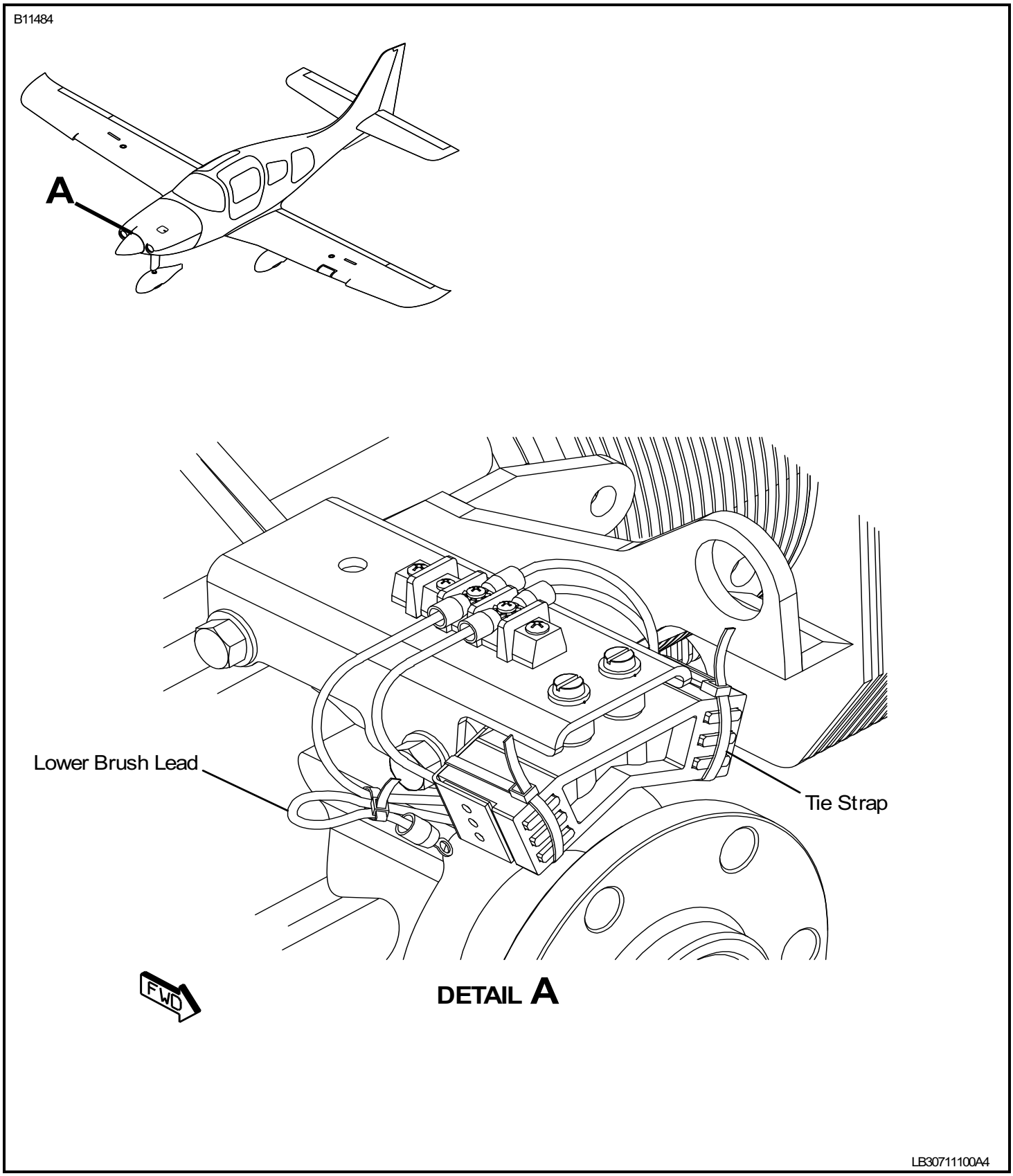
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Figure 201 : Sheet 1 : Brush Block



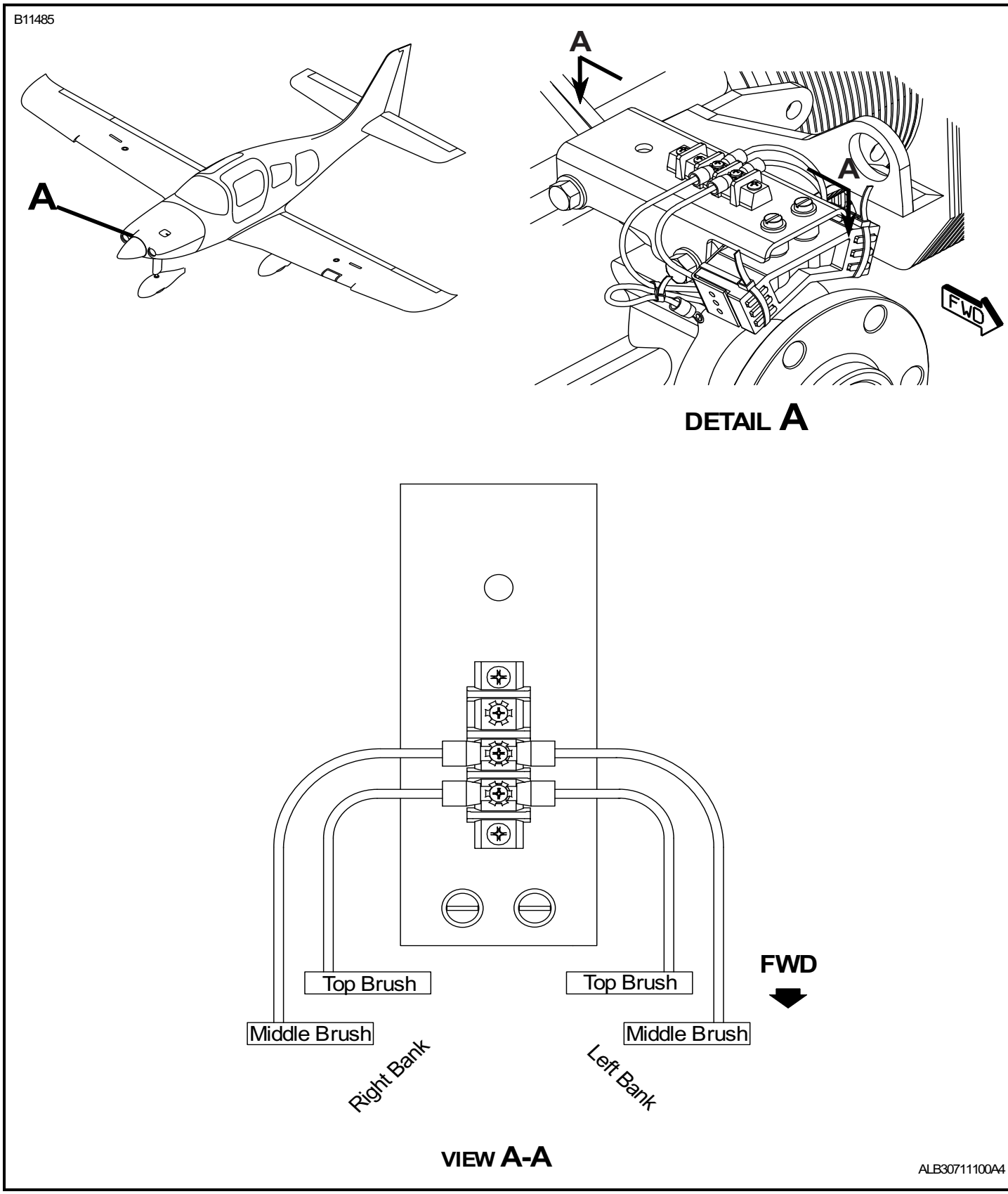
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Figure 202 : Sheet 1 : Brush Block Tie



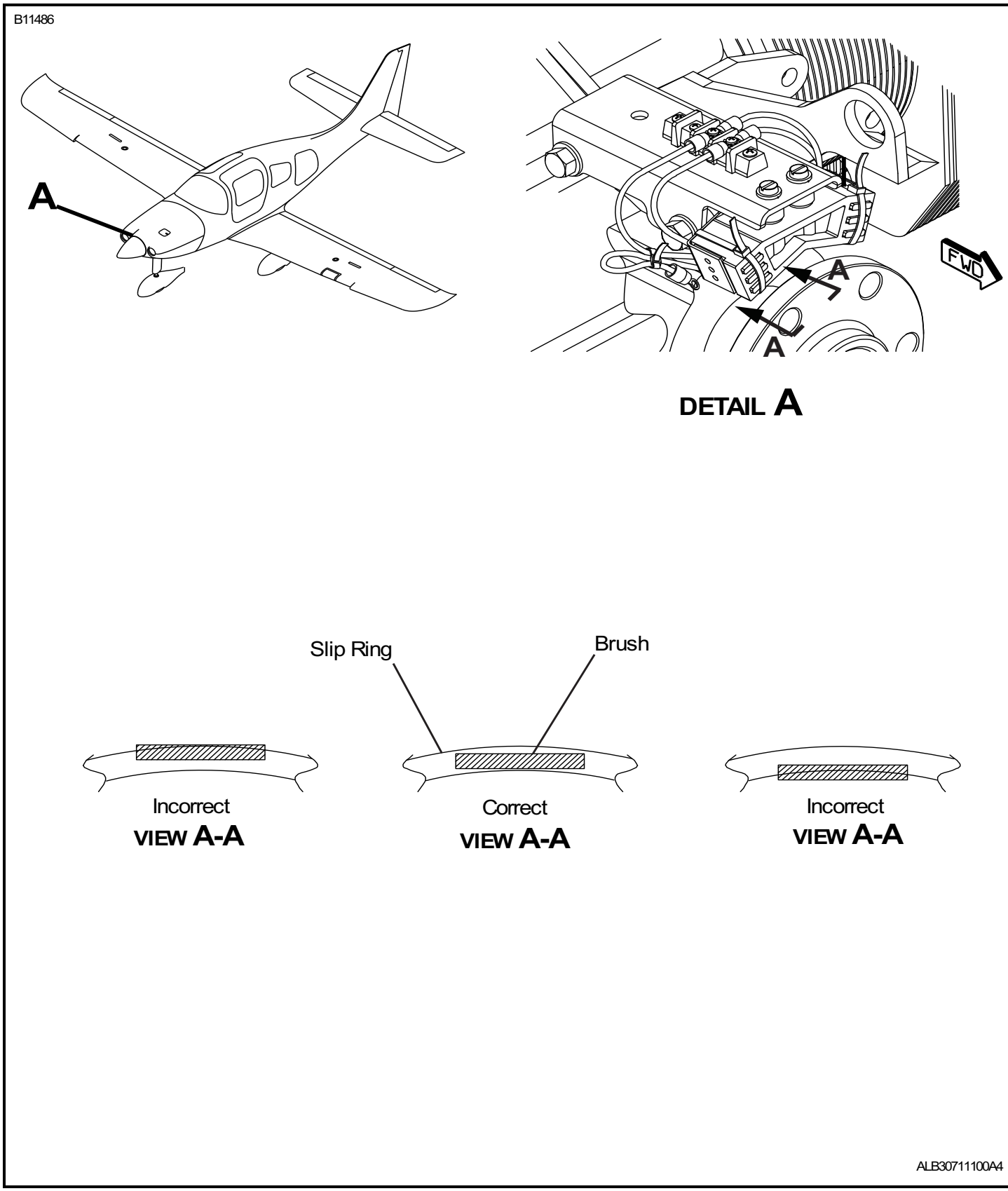
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Figure 203 : Sheet 1 : Brush Wire Routing



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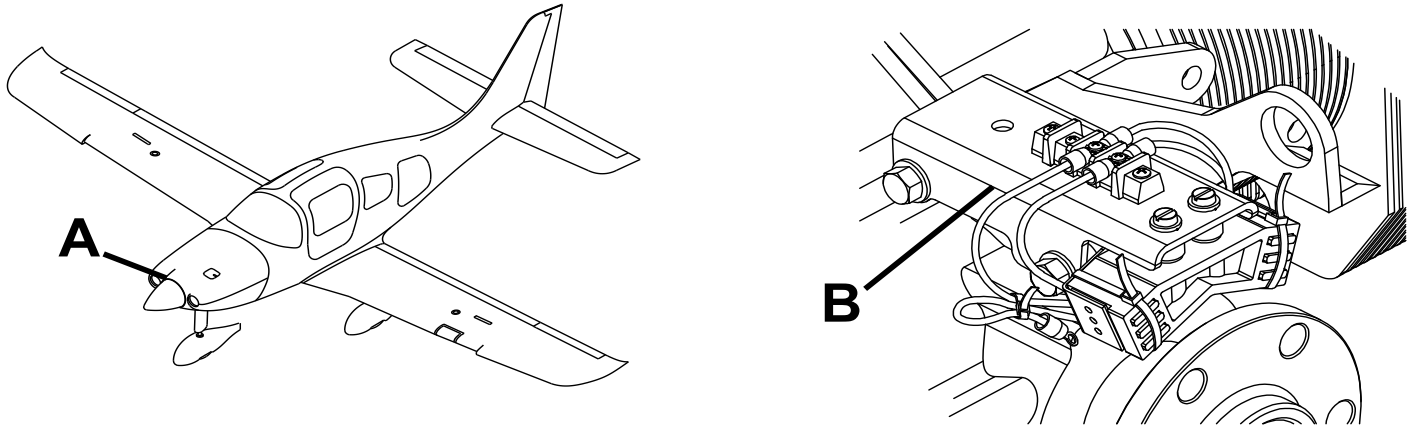
Figure 204 : Sheet 1 : Brush Alignment on Slip Ring



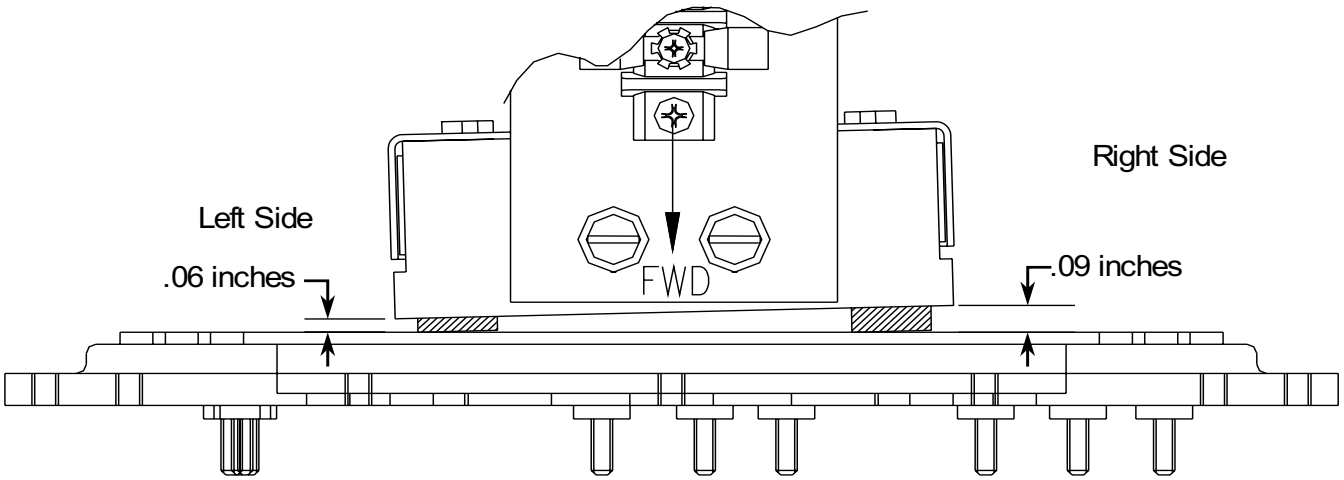
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Figure 205 : Sheet 1 : Brush Block Alignment

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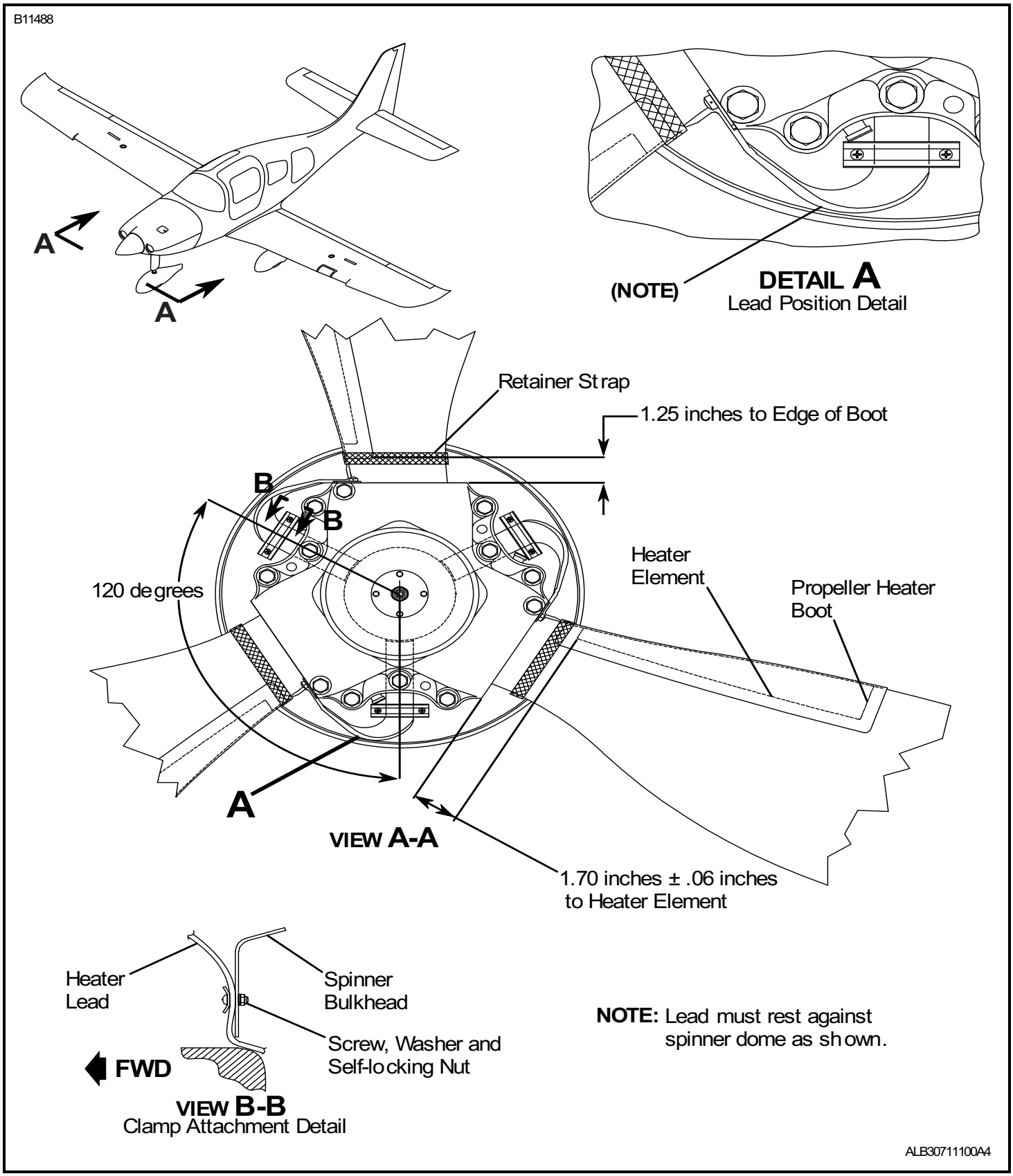
DETAIL A



DETAIL B

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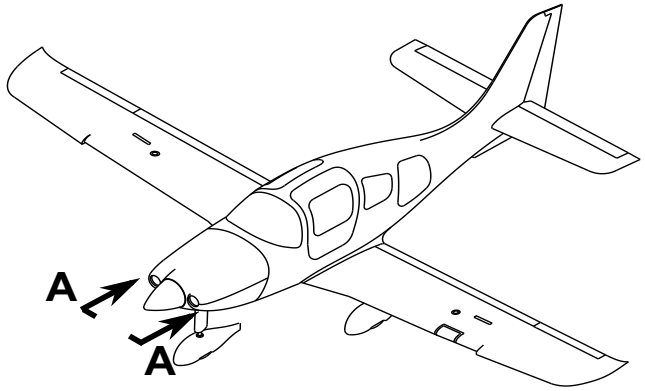
Figure 206 : Sheet 1 : SMR Technologies Inc. Propeller Heater Boot

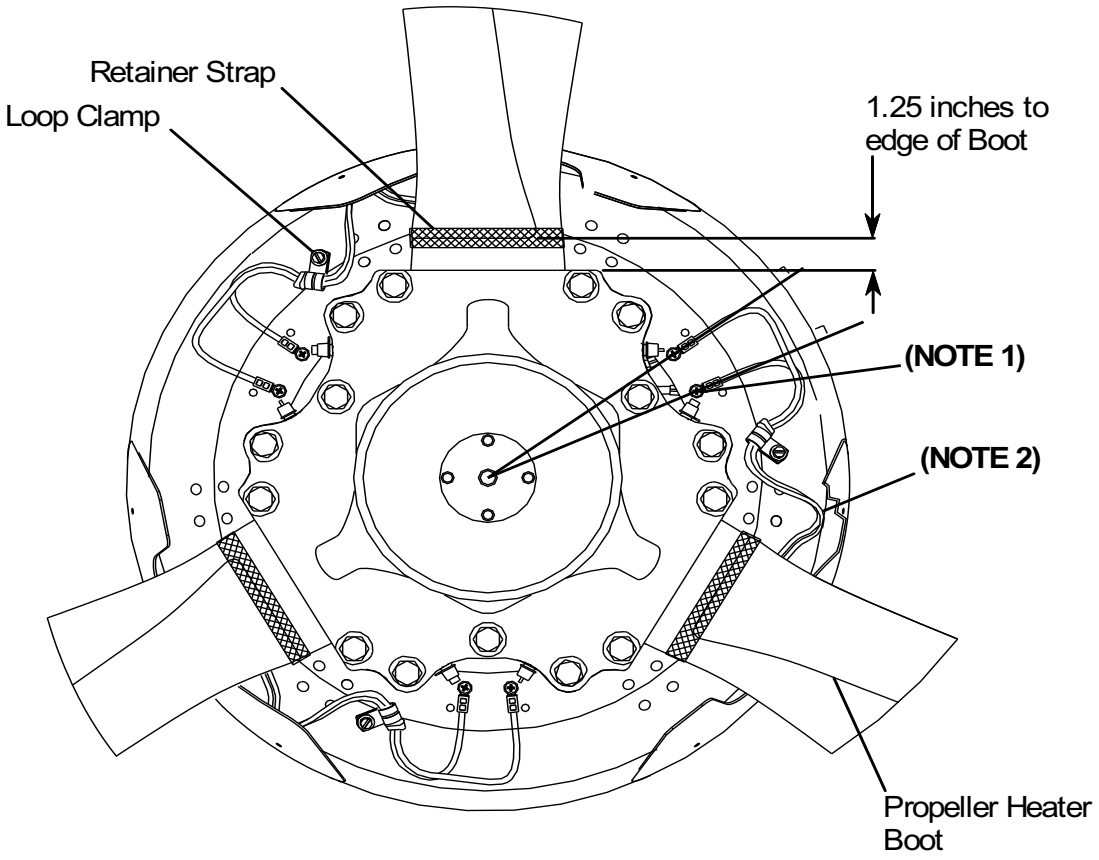


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Figure 207 : Sheet 1 : Hartzell Propeller Inc. Propeller Heater Boot

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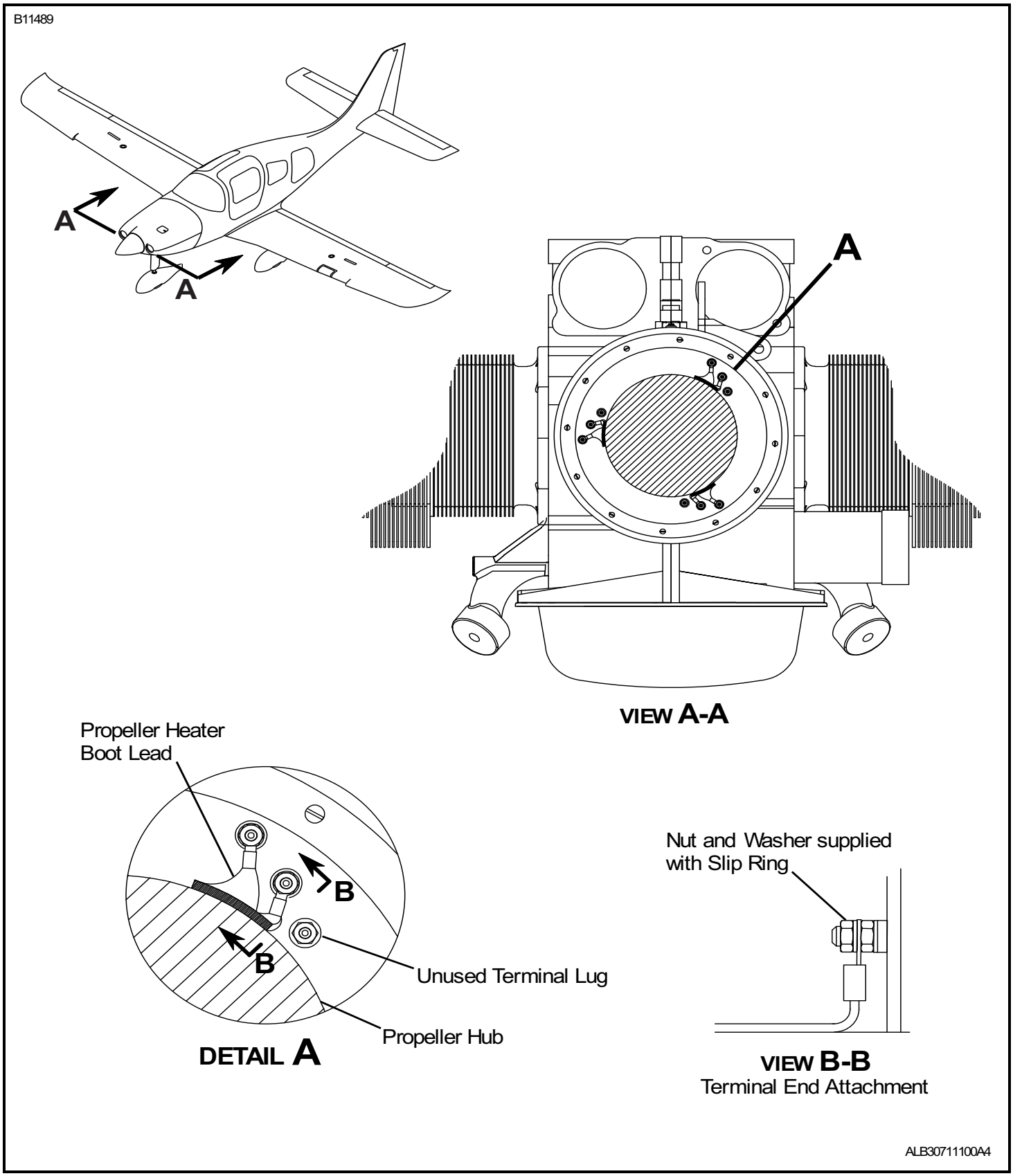
VIEW A-A

NOTE 1: Position wire terminal at a normal (perpendicular) angle to spinner dome as shown.
NOTE 2: Lead must rest against spinner dome as shown.

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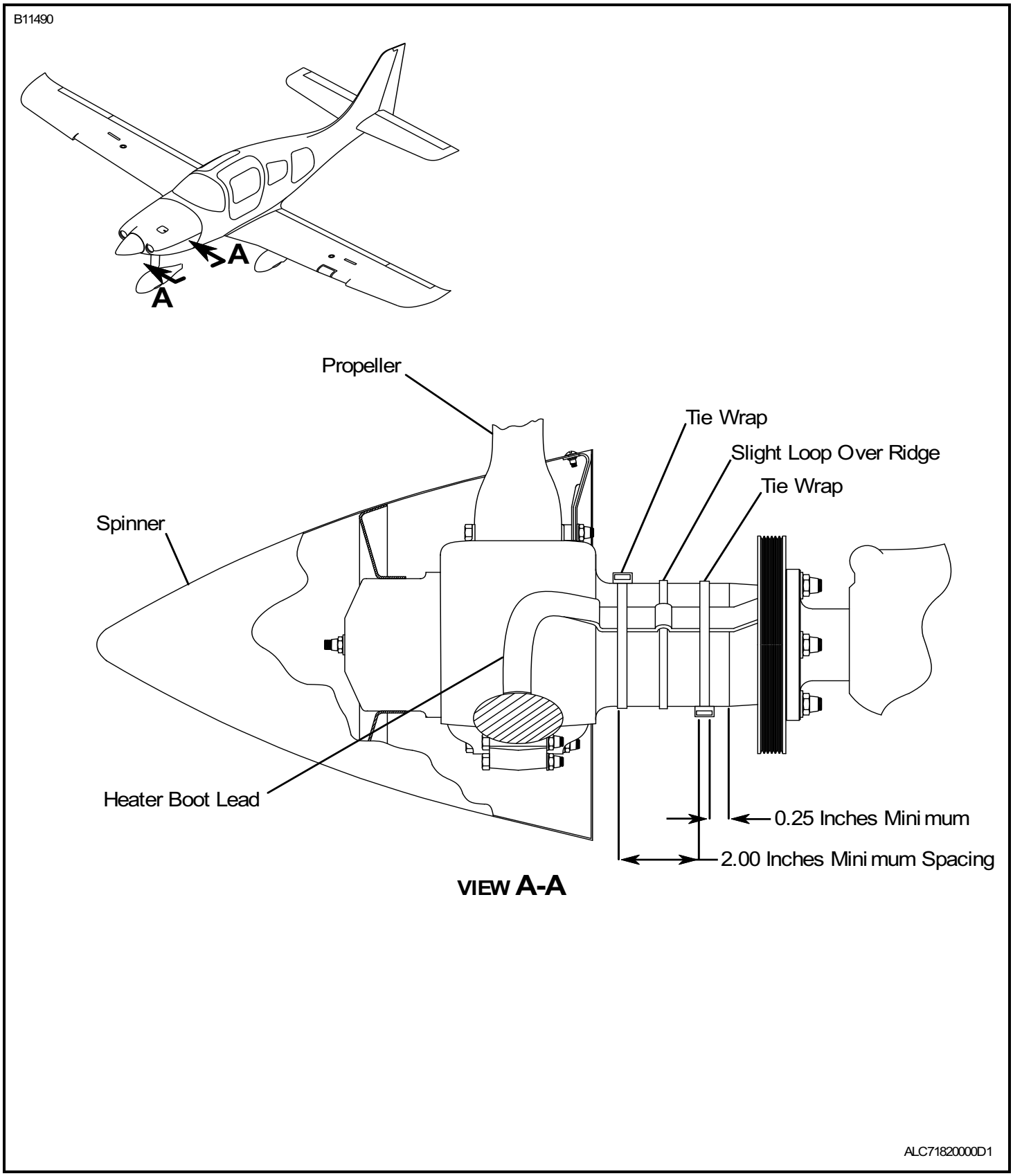
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Figure 208 : Sheet 1 : Heater Boot Lead Terminal Attachment



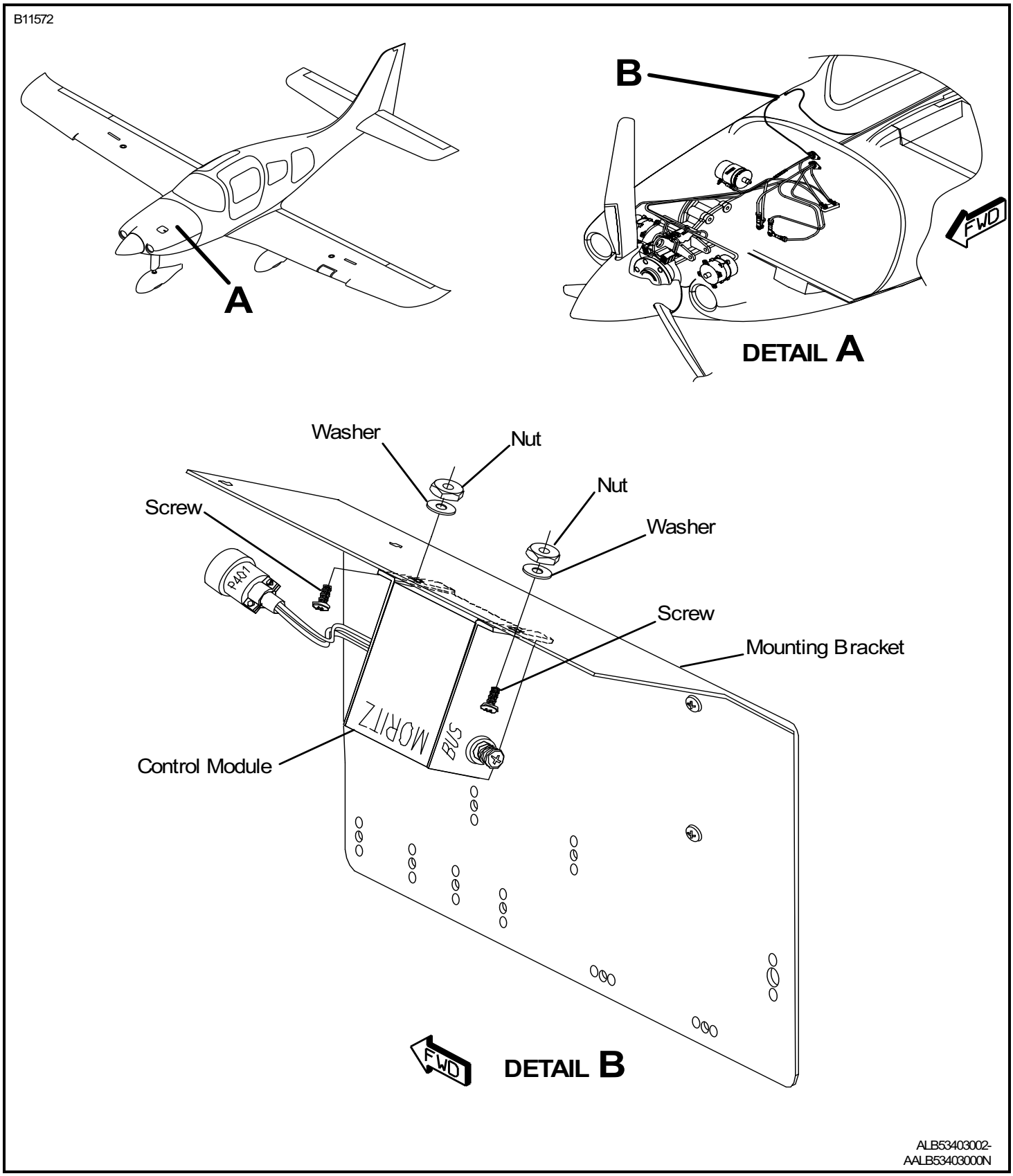
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Figure 209 : Sheet 1 : Heater Boot Lead Attachment



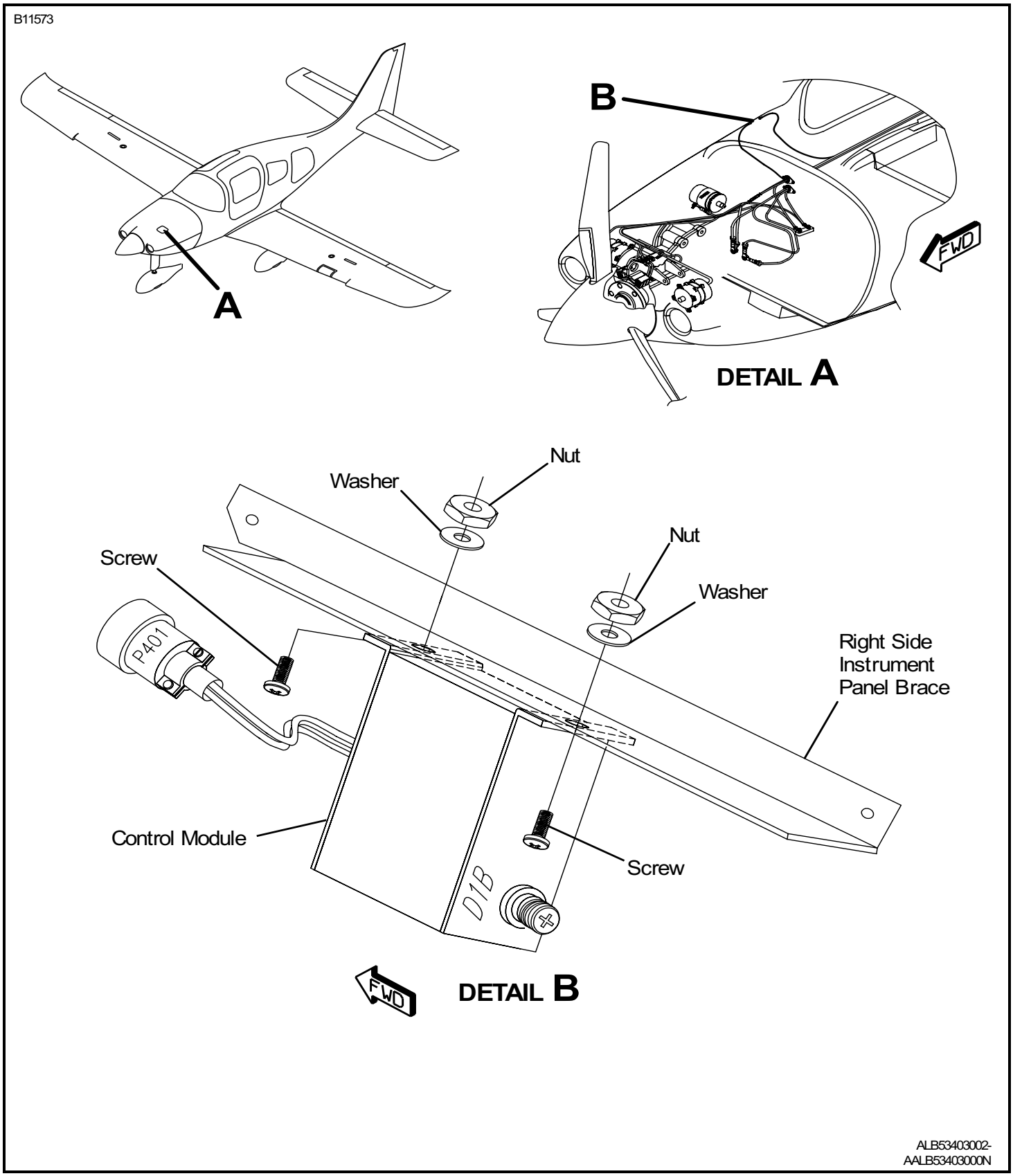
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Figure 210 : Sheet 1 : Control Module (Basic or Avidyne Option)



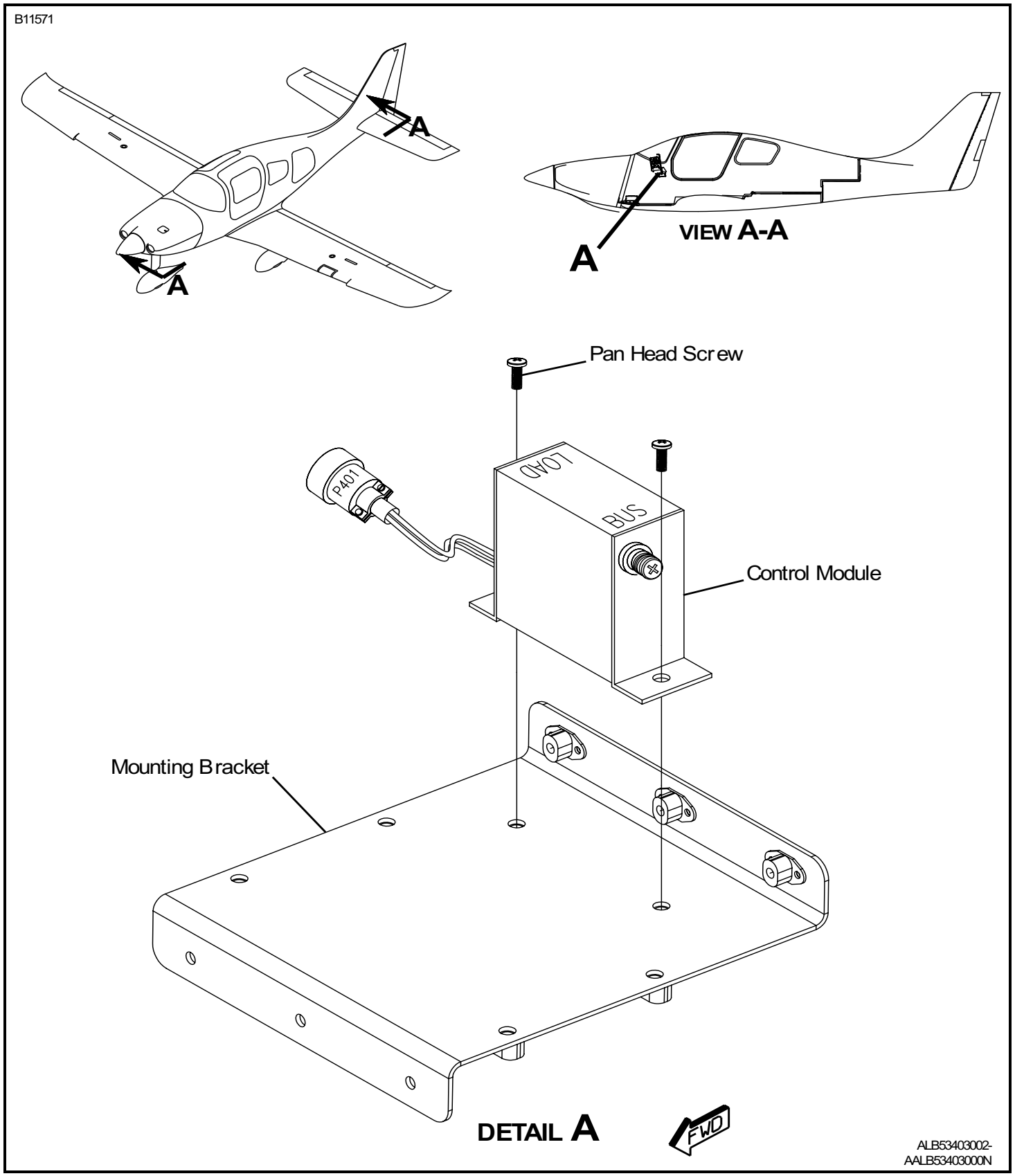
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Figure 210 : Sheet 2 : Control Module (Basic or Avidyne Option)



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Figure 211 : Sheet 1 : Control Module (Garmin G1000 Option)



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ELECTRICAL POWER - GENERAL

1. Electrical System - General

A. **Electrical Power System** - The electrical system in this aircraft consists of two independent buses, which are referred to as the left bus and right bus. Two alternators provide charging power for the batteries, as well as system power. The batteries will also provide additional power in the event of an over demand situation where the requirements on the system are greater than what can be provided by the alternator. The left and right buses in turn feed the avionics and essential buses. A summary of the busses and equipment powered by each bus is shown in Table 1 (Basic or Avidyne option) or Table 2 (Garmin G1000 option). A functional diagram of the electrical system is shown in Figure 1 (Basic or Avidyne option) or Figure 2 (Garmin G1000 option).

B. Five current limiters protect the alternators and bus outputs. In addition, the left and right buses are physically isolated inside a covered area mounted to the firewall. Left and right bus controls, grounds, and outputs are routed through separate holes, connectors, and cable runs so any failure on one bus will not affect the operation of the other bus.

C. Control of the buses is via the master switch panel located on the lower left portion of the instrument panel (Basic or Avidyne) or located in the cockpit overhead (Garmin G1000). There is also a crosstie switch on this panel, which will restore power in the event of failure of the right or left systems. For example, if the alternator or some other component on the left side should fail, the crosstie switch will restore power to the electrical items on the left bus by connecting the left bus to the right bus.

D. As its name may suggest, power to the essential bus is never affected, provided power from at least one bus (left or right) is available. The essential bus is diode fed, i.e., current will only flow in one direction, from both the right bus and the left bus allowing the essential equipment to have two sources of power.

E. **Avionics Bus**

(1) **Basic or Avidyne Option** - The avionics bus provides power to the Audio/Voice, GPS 1, GPS 2, Nav/Com 1, Com 2, Transponder/Encoder/Equipment Fan, HSI or Traffic, Autopilot, Map or MFD, and Weather.

(2) **Garmin G1000 Option** - The avionics bus provides power to the Audio/MKR, Integrated, Avionics No. 2, Com No. 2, Transponder, Avionics Fan, Traffic, Autopilot, MFD, and Weather.

F. **Left Bus**

(1) **Basic or Avidyne Option** - The left bus provides power for the Aileron Trim, Pitot Heat, SpeedBrakes, Engine Instruments, Rudder Limiter, Carbon Monoxide Detector, Oxygen, Position Lights, Landing Light, Left Voltage Regulator, Clock, Cabin Fan, and PFD.

(2) **Garmin G1000 Option** - The left bus provides power for the Aileron Trim, Pitot Heat, SpeedBrakes, Rudder Limiter, Position Lights, Landing Light, Left Voltage Regulator, and Fan.

G. **Right Bus**

(1) **Basic or Avidyne Option** - The right bus provides power for the Strobe Lights, Taxi Light, Right Voltage Regulator, Door Seal, Power Point, PFD, and Elevator Trim.

(2) **Garmin G1000 Option** - The right bus provides power for the Strobe Lights, Taxi Light, Right Voltage Regulator, Door Seal/Power Point, Carbon Monoxide Detector, Oxygen, Display Keypad, and Air Conditioning.

H. **Essential Bus**

(1) **Basic or Avidyne Option** - The essential bus is diode-fed from either the right or the left bus and provides power for the Attitude Horizon, Turn Coordinator, Panel Lights, Annunciators, Left Bus Relays, Fuel Pump, Stall Warning, Flaps, and the Right Bus Relays.

(2) **Garmin G1000 Option** - The essential bus is diode-fed from either the right or the left bus and provides power for the PFD, AHRS, Elevator Trim, Panel Lights, Standby Instrument Lighting, Air Data Computer, Engine Airframe, Integrated Avionics No. 1, Com No. 1 Left Bus Relays, Fuel Pump, Stall Warning, Flaps, Standby ADI, and the Right Bus Relays.

I. **Battery Bus**

(1) **Basic or Avidyne Option** - Four items are connected to the battery bus. These items will operate even if the left and right buses are turned off since they are directly connected to either the left or right battery. The Nav/Com bypass switch, Hobbs Meter, and ELT, are connected to the right battery and the courtesy lights/flip lights are connected to the left battery. The Hobbs Meter, ELT, and courtesy lights/flip lights are each protected by a 3-amp fuse which is not accessible from the cockpit.

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(2) **Garmin G1000 Option** - Three items are connected to the battery bus. These items are directly wired to the battery, but the term battery bus is used for consistency. These items will operate even if the left and right buses are turned off since they are directly connected to the right or left battery. The items directly wired to the right battery include the Hobbs Meter, and ELT. The courtesy lights/flip lights are directly wired to the left battery. A 3-amp fuse protects each component and is not accessible from the cockpit.

CAUTION: For extended service life, disconnect the batteries to prevent them from being continuously charged by external power when the aircraft is in maintenance.

CAUTION: Do not replace the sealed lead acid batteries with wet lead acid batteries.

CAUTION: Never "jump-start" an aircraft that has a "dead" or discharged battery. It takes approximately three hours to recharge a fully discharged battery with the aircraft generating system or external power.

(3) **Nav/Com Bypass Circuit** - The Nav/Com bypass switch is an alternate action device that activates when depressed and needs to be depressed again to cycle off. Activation of the Nav/Com bypass switch turns on the No. 1 GPS, No. 1 Com, and No. 1 Nav by connecting them directly to the right battery, bypassing the power grid, circuit breaker panel, and ass ociated wiring. The Nav/Com bypass switch is protected by two 5-amp fuses.

Table 1. Summary of Buses and Equipment (Basic or Avidyne Option)

Bus	Bus Component	Circuit Breaker
Avionics Bus	Audio/Voice	5 amp
	GPS 1	5 amp
	GPS 2	5 amp
	Nav/Com1	10 amp
	Com 2	10 amp
	Transponder/Encoder and Equipment Fan	5 amp
	HSI (5 amp) or Traffic (4 amp)	5 or 4 amp
	Autopilot	5 amp
	Map or MFD	10 amp
	Weather	5 amp
Left Bus	Aileron Trim	1 amp
	Pitot Heat (fuse amperage depends on Pitot installed)	7.5 amp or 10 amp
	SpeedBrakes	3 amp
	Engine Instruments	3 amp
	Rudder Limiter	5 amp
	Carbon Monoxide Detector	2 amp
	Oxygen	3 amp
	Position Lights	10 amp
	Landing Light	4 amp
	Left Voltage Regulator	5 amp
	Clock, and Cabin Fan	7.5 amp
	PFD Power	10 amp
	Air Conditioning	20 amp
	Strobe Lights	10 amp
	Taxi Light	4 amp

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Right Bus	Right Voltage Regulator	5 amp
	Door Seal/Power Point	5 amp
	PFD Power	10 amp
	Elevator Trim	1 amp
	Air Conditioning	15 amp
Essential Bus	Attitude Horizon	3 amp
	Turn Coordinator	3 amp
	Panel Lights	7.5 amp
	Annunciators	3 amp
	Left Bus Relays	5 amp
	Fuel Pump	10 amp
	Stall Warning	5 amp
	Flaps	10 amp
	Right Bus Relays	5 amp
Battery Bus	Nav/Com Bypass Switch	Two 5 amp
	ELT	3 amp
	Courtesy Lights/Flip Lights	3 amp
	Hobbs Meter	3 amp

Table 2. Summary of Buses and Equipment (Garmin G1000 Option)

Bus	**Bus Component**	**Circuit Breaker**
Avionics Bus	Audio/MKR	5 amp
	Integrated Avionics No. 2	5 amp
	Com No. 2	5 amp
	Transponder	5 amp
	Avionics Fan	3 amp
	Traffic	3 amp
	Autopilot	5 amp
	MFD	5 amp
	Weather	3 amp
Left Bus	Aileron Trim	2 amp
	Pitot Heat	7.5 amp
	SpeedBrakes	3 amp
	Rudder Limiter	5 amp
	Position Lights	5 amp
	Landing Light	5 amp
	Left Voltage Regulator	5 amp
	Fan	5 amp
Right Bus	Strobe Lights	5 amp
	Taxi Light	2 amp
	Right Voltage Regulator	5 amp
	Door Seal/Power Point	5 amp

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Right Bus	Carbon Monoxide Detector	2 amp
	Oxygen	3 amp
	Display Keypad	2 amp
	Air Conditioning	15 amp
Essential Bus	PFD	5 amp
	AHRS	5 amp
	Elevator Trim	2 amp
	Panel Lights	7.5 amp
	Air Data Computer	5 amp
	Engine Airframe	5 amp
	Integrated Avionics No. 1	5 amp
	Com No. 1	5 amp
	Left Bus Relays	1 amp
	Fuel Pump	5 amp
	Stall Warning	2 amp
	Flaps	10 amp
	Standby Attitude Horizon	3 amp
	Right Bus Relays	1 amp
Battery Bus	Hobbs Meter	3 amp
	ELT	3 amp
	Courtesy Light	3 amp

2. Troubleshooting Procedures for the Electrical System

A. **Symptom** - System does not charge or discharge showing on ammeter.

B. **Probable Cause** - Bad alternator, voltage regulator, or system short.

C. **Action** - Charging system problems associated with an alternator should be analyzed in the manner of: Pre-Voltage Regulator Checks, Voltage Regulator Checks, and Post Voltage Regulator Checks.

D. **Pre-Voltage Regulator Checks** - Check the affected alternator switch, voltage regulator circuit breaker, wires, and terminals for contact resistance build-up. Resistance should be less than 0.1 ohms.

E. **Voltage Regulator Check** - Check that there is power coming into and going out of the voltage regulator. The fault/function indicator on the unit is designed to alert the user to the condition of the alternator/voltage regulator.

(1) A red light on the unit, with the affected master switch on means there is a ground short in the alternator field circuit or the voltage regulator's field control transistor is shorted.

(2) A green steady light on the unit with the engine running means there is power coming out of the voltage regulator, but the alternator field or field wire to the voltage regulator is open.

(3) A green fast flickering light on the unit with the engine running means power input devices (like circuit breaker, switch, etc.), and the field systems are functioning correctly.

(4) A green slow flickering light on the unit with the engine running means the power input devices (like switch, etc.) are defective or have higher than normal resistance.

(5) No light on the unit with the affected master switch on means that one power input device (like the switch, circuit breaker, or wiring) or the voltage regulator is defective.

F. **Post-Voltage Regulator Checks**

(1) Check the affected alternator by measuring the resistance of the field. Resistance from field to ground on the alternator should be from 3.5 to 5.0 ohms. Check the resistance of the meter leads before measuring field.

(2) Check for "flying" short and other intermittent problems by slowly rotating the affected alternator while measuring the

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field resistance. A drop below 3.0 ohms could indicate a bad alternator that might damage the voltage regulator.

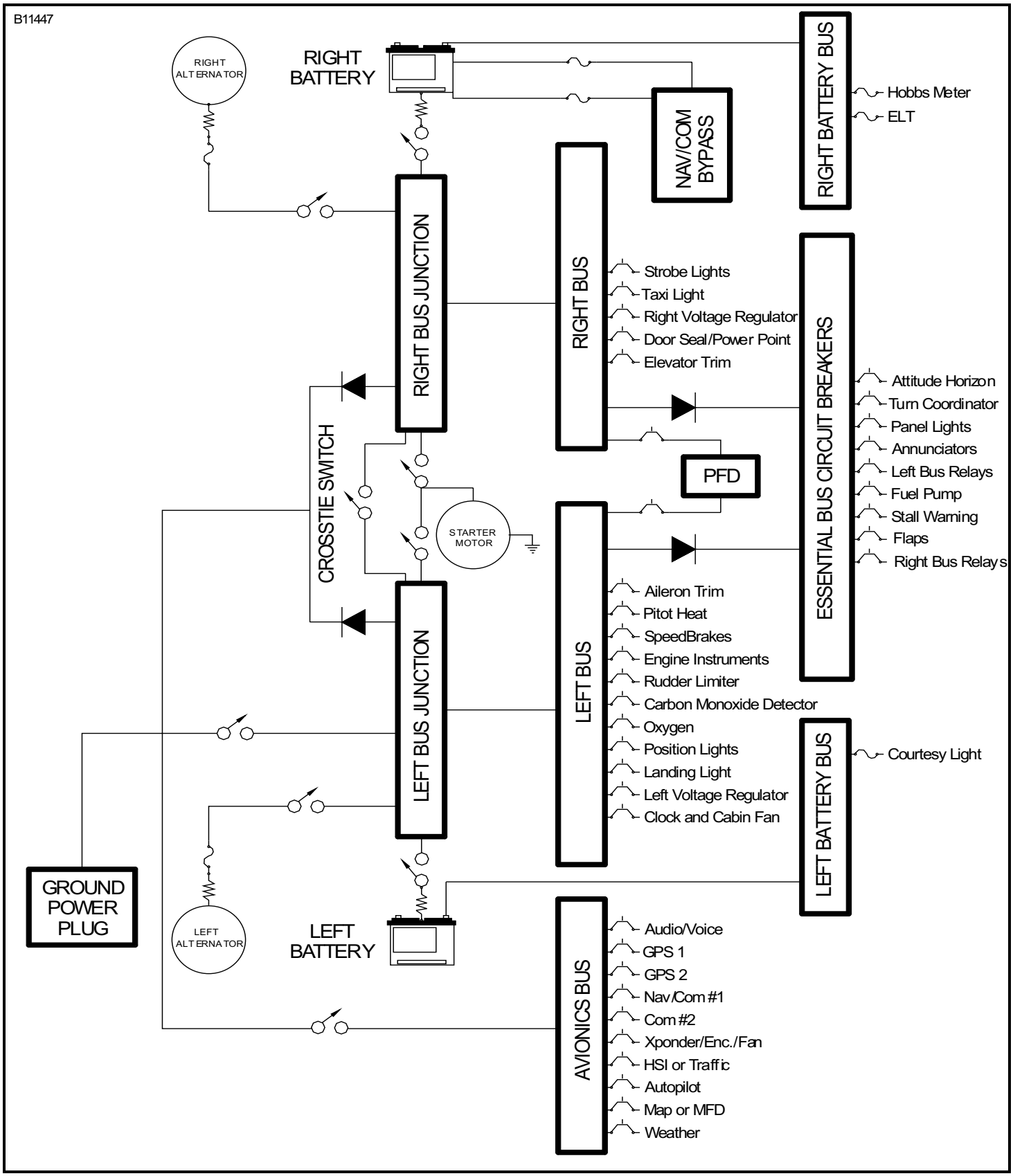
(3) Check the condition of the alternator's field, terminals, and wires connecting it to the voltage regulator and the aircraft charging system.

(4) Check the batteries' relays for proper operation and connections.

(5) Check the affected alternator relay for proper operation and connections.

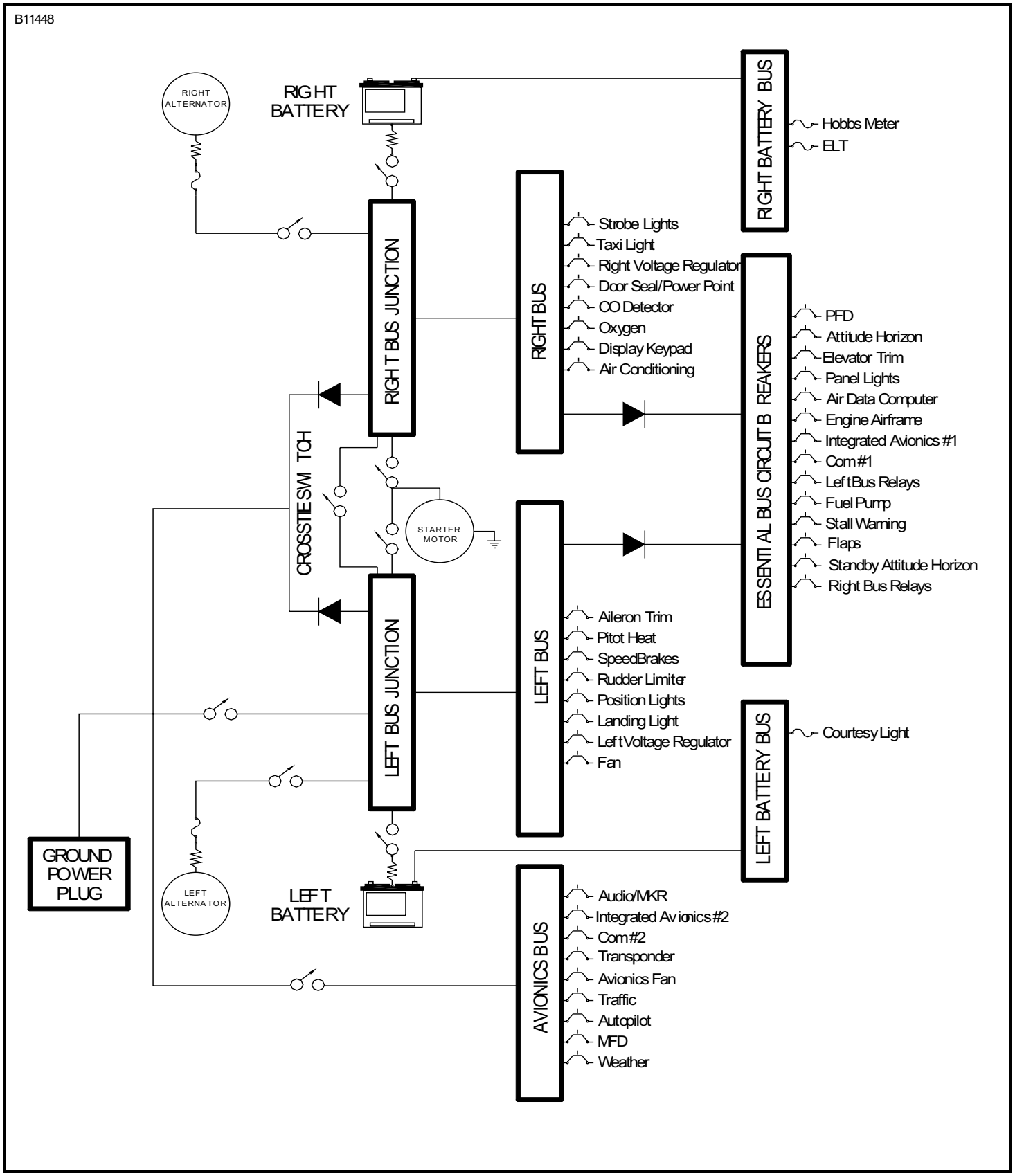
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Figure 1 : Sheet 1 : Electrical System (Basic or Avidyne Option)



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Figure 2 : Sheet 1 : Electrical System (Garmin G1000 Option)



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DC GENERATION - MAINTENANCE PRACTICES

1. DC Generation

A. The Cessna 350 electrical system requires two alternators for the operation of electrical equipment. The No. 1 alternator comes with the engine, is located at the front of the engine on the right side, and runs the right bus. The No. 2 alternator is the added alternator, is located on the front, left of the engine, and runs the left bus.A third alternator may be installed on the aircraft, as required, for optional equipment. This alternator is located at the rear of the engine on the right side.

(1) **No. 1 (Right) Alternator** (Refer to Figure 201.)

(a) **Removal of No. 1 (Right) Alternator**

1 Disconnect both aircraft batteries.

2 Disconnect the two connectors attached to the alternator.

3 Remove the plain washer, lock washer, and nut from the four mounting studs.

4 Discard the lock washers.

5 Remove the alternator from the mounting studs.

6 Remove the gasket from the mounting studs.

(b) **Installation of No. 1 (Right) Alternator**

1 Install a new gasket on the alternator.

2 Install the alternator assembly on the mounting studs with four plain washers, four new lock washers, new gasket, and four plain nuts.

3 Torque nuts 180 to 220 in.- lbs.

4 Connect two connectors to the alternator.

5 Reconnect both aircraft batteries, start the aircraft, and check the alternator for proper operation.

(c) **No. 1 (Right) Alternator Maintenance Procedures**

1 The torque required to slip a new elastomer coupling must be a minimum of 180 in.-lbs. (15 ft.-lbs.) torque measured after 45 degrees of revolution at a rate of 1 to 2 degrees per second. If you can hand-turn the gear relative to the black housing, it is below the requirements, and the coupling assembly must be replaced. (Refer to Figure 202.)

2 Slippage must occur at the outside diameter of the elastomer with no damage to the elastomer.

3 On couplings that have been in service for more than 25 hours, slippage torque must not be less than 140 in.-lbs. (11.7 ft.-lbs.) torque. If you can hand-turn the gear relative to the black housing, it is below the requirements, and the drive hub assembly must be replaced. (Refer to Figure 202.)

(d) **Alternator Brushes** - Every 1000 hours inspect the alternator brushes. The alternator must be replaced if the brushes are worn more than 75 percent.

1 Disconnect both aircraft batteries.

2 Disconnect the battery wires connected to the alternator.

3 Remove the alternator.

4 Remove the brush assembly per the latest revision of Teledyne Continental Motors, Inc. Form No. X30531-3.

5 Inspect the brushes.

6 Re-install the brush assembly and reassemble the alternator per Teledyne Continental Motors, Inc. Form No. X30531-3.

7 Install the alternator.

8 Connect the battery wires.

9 Reconnect both aircraft batteries, start the aircraft, and check the alternator for proper operation.

(2) **No. 2 (Left) Alternator** (Refer to Figure 203.)

(a) **Removal of No. 2 (Left) Alternator**

1 Disconnect both aircraft batteries.

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2 Disconnect the wires attached to the alternator.

3 Remove the two bolts (one bolt with Plane Power alternator), washers, and alternator tension plate attaching the alternator to the banana bracket.

4 Remove the belt from the alternator pulley.

5 Remove the self locking nut, bolt, washers, and pivot spacer connecting the alternator to the mounting bracket. Note the position of the washers for reinstallation.

6 Remove the alternator.

7 Retain all hardware for use in reinstallation.

(b) **Installation of No. 2 (Left) Alternator**

WARNING: Ensure that there is a minimum clearance of 1/2 inch between the upper cowling skin and the alternator case, fan, belt, and pulley.

1 Attach the alternator to the mounting bracket. Ensure the washers are installed so that the edge of the belt is parallel to the face of the drive pulley by ±1.0 degree.

2 Install the alternator belt.

3 Insert the alternator tension plate into the banana bracket and hand tighten the hardware attaching the assembly to the alternator.

4 Connect a torque wrench to the alternator pulley retention nut. With an assistant holding the propeller to prevent the engine crank shaft from rotating, tighten the alternator tension plate until the belt no longer slips on the alternator pulley and force applied to the torque wrench is 13 to 17 ft-lbs. Do not over-tighten the belt. Torque the alternator tension plate bolt(s) 12 to 15 ft-lbs (Kelly alternator) or 165 to 180 in-lbs (Plane Power alternator); ensure that a minimum of two threads extend past the Kelly alternator body, or the bolt does not bottom out in the Plane Power alternator (McMaster-Carr 90108A415 washers as required). Safety wire the Plane Power alternator tension plate bolt to the bracket.

5 After the belt tension has been fully adjusted, torque the nut attaching the banana bracket to the propeller governor bracket and the nut attaching the alternator to the mounting bracket 15 to 20 ft.-lbs.

6 Wiring Connections

a Kelly Alternator

1. Connect the wires to the alternator and torque the attachment nuts 10 to 12 in.-lbs.

b Plane Power Alternator

NOTE: The F2 terminal is to remain grounded with the ground strap UNLESS the aircraft voltage regulator is a type “A” regulator using a 2-wire field circuit, in this case remove and discard the ground strap from the F2 terminal and connect wiring from the voltage regulator to the F1 and F2 terminals, torque to 20 in-lbs.

1. Install the battery wire with the MS25171-2S terminal nipple onto the 6 mm output terminal and torque to 50 in-lbs.
2. Install the ground wire to any of the three 5 mm studs on the rear of the alternator and torque to 35 in-lbs.
3. Install the field wire with the MS25171-1S terminal nipple to the F1 terminal on the rear of the alternator and torque to 20 in-lbs.
4. If the aircraft is equipped with an “alternator out light” circuit, connect that wire to the AUX terminal and torque to 20 in-lbs. Otherwise leave the AUX terminal open.

7 Reconnect both aircraft batteries, start the aircraft, and check the alternator output for proper operation.

(c) **Alternator Pulley**

1 **Removal of Alternator Pulley**

a Disconnect both aircraft batteries.

b Disconnect the battery wires attached to the alternator.

c Remove the bolts attaching the alternator tension plate to the banana bracket and remove the belt from the alternator pulley.

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d Remove the pulley retention nut and washer from the alternator drive shaft.

e Remove the pulley.

2 **Installation of Alternator Pulley**

a Kelly Alternator

1. Installation of the pulley is the reverse of removal.
2. Torque the pulley retention nut 35 to 45 ft-lbs.

b Plane Power Alternator

1. Apply Loctite 272 to the threads and shaft.
2. Install the pulley retention nut and washer, and torque to 55±4 ft-lbs.

c Reinstall and tension the alternator belt per the Installation of Alternator section.

d Connect the battery wires to the alternator.

e Reconnect both aircraft batteries, start the aircraft, and check the alternator output for proper operation.

(3) **Accessories Alternator** - (Refer to Figure 204.)

(a) **Removal of Accessories Alternator**

1 Disconnect both aircraft batteries.

2 Disconnect the battery wires attached to the alternator.

3 Loosen the adjustment rod and remove the belt from the alternator pulley. The adjustment rod ends are threaded such that rotating the rod itself will loosen or tighten the ends equally.

4 Remove the bolt and washer connecting the adjustment rod to the alternator.

5 Remove the nut, bolt, and washer connecting the alternator to the mounting bracket. Remove the belt cutter.

6 Retain all hardware for use in reinstallation.

(b) **Installation of Accessories Alternator**

1 Ensure the alternator cooling fan kit, spacer, and alternator drive pulley are installed on the alternator. Verify the nut is torqued 350 to 400 in-lbs.

2 Place the alternator into the alternator bracket on the rear of the engine. Insert the bolt through the alternator pivot. Place the belt cutter, washer, and nut onto the pivot bolt. Do not tighten at this time.

3 Align the belt onto drive pulley and alternator pulley.

4 Ensuring that the adjustment rod ends are assembled into the rod equal distance, attach the adjustment rod to the alternator. Tighten the bolts. For the bolt attaching the alternator to the adjustment rod, protrusion of two bolt threads is not required and no visible female threads on the alternator are allowed; lockwire the bolt.

5 Tighten, or loosen, the adjustment rod until the breakaway (slip) torque at the alternator drive pulley is 440 to 480 in.-lbs. on a new belt. After the belt has been run for a few hours the minimum belt tension should not be below 300 in.-lbs. After final belt adjustment has been made the torque jam nuts on the adjustment rod to hold the proper adjustment.

CAUTION: Do not use the alternator drive nut to check belt tension. Use Kelly Aerospace tool to check alternator belt breakaway torque.

6 Connect the battery wires to the alternator.

7 Reconnect both aircraft batteries, start the aircraft, and check the alternator output for proper operation.

(c) **Alternator Pulley**

1 **Removal of Alternator Pulley**

a Disconnect both aircraft batteries.

b Disconnect the battery wires attached to the alternator.

c Loosen the adjustment bracket and remove the belt from the alternator pulley.

d Remove the pulley retention nut and washer from the alternator drive shaft.

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e Remove the pulley.

2 **Installation of Alternator Pulley**

a Installation of the pulley is the reverse of removal.

b Torque the pulley retention nut 350 to 400 in.-lbs.

c Connect the battery wires to the alternator.

d Reinstall and tension the alternator belt per the Installation of Accessories Alternator section.

e Reconnect both aircraft batteries, start the aircraft, and check the alternator output for proper operation.

(d) **Drive Pulley Removal and Installation**

1 Remove the nut and lock washer from the starter adaptor drive shaft.

2 Remove the alternator drive pulley.

3 Installation is the reverse of removal. Torque the nut 700 to 720 in.-lbs. Ensure a minimum clearance of .050 inches between the drive pulley and the end of the starter adapter stud.

(4) **Pulley Maintenance Procedures**

(a) The pulleys should be free from large nicks, scratches and excessive warpage. If warpage exceeds .050 inches, the pulley must be replaced. No shimming is permissible. Small scratches and nicks are acceptable as long as they do not present a hazard to the belt. Blending and burnishing of small scratches and nicks is permissible.

(b) The drive pulley for the No. 2 alternator must be replaced if elongated screw holes are present. The maximum elongation permissible is 1/4 diameter where runout is within .050 inches maximum (.025 inch either way).

(c) If there is excessive black, sooty residue on the pulleys or surrounding area, immediately remove the belt from service and assure that the pulleys are properly aligned.

(d) The belt must be replaced if there is any evidence of fraying or dry rot.

(e) Do not apply "belt treatments" spray or otherwise to the belt. Any spillage of ester based oils, turbine oils or MIL-H-5606 hydraulic fluid on the belt must be immediately removed with hot soapy water (household dishwashing soap or equivalent).

(5) **Alternator Brushes** -Every 1000 hours inspect the alternator brushes. The Kelly alternator must be replaced if the brushes are worn more than 75 percent. Replace the Plane Power alternator brush assembly if the brushes extend less than .250 inches from the edge of the brush holder case.

(a) Disconnect the aircraft battery.

(b) Disconnect the battery wires connected to the alternator.

(c) Remove the bolts attaching the alternator tension plate to the banana bracket or loosen the adjustment rod and remove the belt from the alternator pulley.

(d) Loosen the self-locking nut connecting the alternator to the mounting bracket and rotate the alternator upward allowing access to the alternator brush block.

(e) Kelly Alternator

1 Remove the two brush block retainer screws and washers.

2 Remove the brush block. The brushes are spring loaded and may fly out of the brush block.

3 Inspect the brushes.

4 Reinsert the brushes into the brush block. Depress the brushes, thus loading the springs, and insert a rod or wire through the hole above the brush seats to hold the brushes in place.

5 Install the brush block into the alternator.

6 Withdraw the rod or wire and listen for a "click" to indicate the brush has properly seated. There should be one click per brush.

7 Repeat steps 5 through 7 until the brushes seat properly.

8 Replace the retaining screws and washers and torque per alternator manufacturer’s specifications.

(f) Plane Power Alternator

1 Examine the brushes per the manufacturer’s instructions.

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(g) Reinstall and tension the alternator belt per the Installation of the No. 2 (Left) Alternator or the Installation of Accessories Alternator sections.

(h) Connect the battery wires to the alternator.

(i) Reconnect the aircraft battery, start the aircraft, and check the alternator output for proper operation.

2. DC Indicating Instruments

A. The following information applies to aircraft equipped with the Basic or Avidyne avionics option only. DC monitoring for aircraft equipped with the Garmin G1000 option is displayed on the MFD.

(1) **Dual Ammeter/Loadmeter Description of Operation** - The ammeter is in the engine instrument panel in the center-left position. The ammeter is a dual presentation gauge that measures the condition of both batteries and alternators in terms of charging or discharging. The ammeter is selectable to show either the condition of the batteries or the alternators. When power is first applied, the indicator defaults to the "BATT" indicator. When the button on the lower left side of the instrument is pressed, the indicator switches to the "ALT" indicator. Pressing the button will toggle between each indication. When the battery mode is selected, "BATT" is illuminated by a white light. When the alternator mode is selected, "ALT" is illuminated by a white light. The range of the indications run from a +60 amps to -100 amps in 30 amp increments. While there is no placarded operating range, under most conditions the instrument should indicate a positive charging state. The master switches for the left and right bus systems must be on for the ammeter to operate.

(2) **Removal of Dual Ammeter/Loadmeter**

(a) Remove the engine instrument panel per instructions in Section 25-11-00, Engine Instrument Panel.

(b) Unplug the cable going into the back of the ammeter.

(c) Unscrew the four screws surrounding the ammeter.

(3) **Installation of Dual Ammeter/Loadmeter**

(a) Put the ammeter in its slot, and screw in the four screws attaching the instrument to the engine instrument panel.

(b) Plug the cable into the back of the ammeter.

(c) Install the engine panel per instructions in Chapter 25.

(4) **Voltmeter Description of Operation (S/N 42001 to 42500)** - The voltmeter is located in the top left corner of the flight instrument panel. This instrument contains three separate indications: a voltage reading, an outside air temperature provided by the top window, and a multi-function timepiece in the lower window. The voltmeter displays the left system's bus voltage only, and under normal conditions should indicate about 14.2 volts. At 16 volts, the voltage regulator will take the left alternator off-line, and at approximately 8 volts, electrical equipment connected to the left bus will cease to operate or will operate erratically. The voltmeter is electrically powered and will not operate if the left master switch is turned off. Refer toSection 31-00-00, Figure 1, for a drawing of the voltmeter. This voltmeter is not present on aircraft S/N 42501 and on.

(5) **Removal of Voltmeter**

(a) Remove the glare shield, and remove the connector installed into the back of the unit.

(b) Unscrew the four screws surrounding the voltmeter, and remove it.

(c) Slide the unit out of the flight instrument panel.

(6) **Installation of Voltmeter**

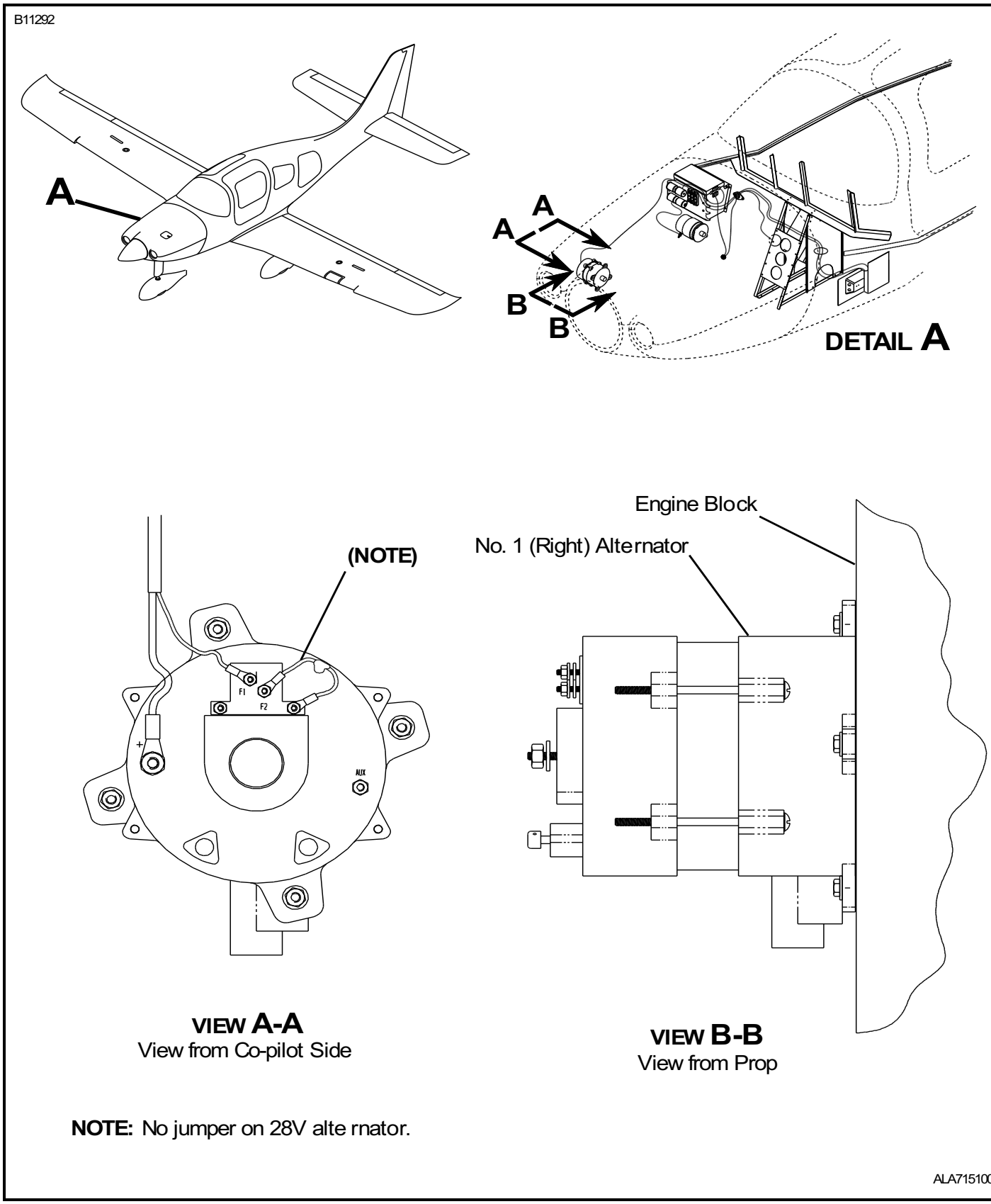
(a) Install the connector into the back of the voltmeter.

(b) Slide the unit into its slot in the flight instrument panel, and screw in the four screws.

(c) Install the glare shield.

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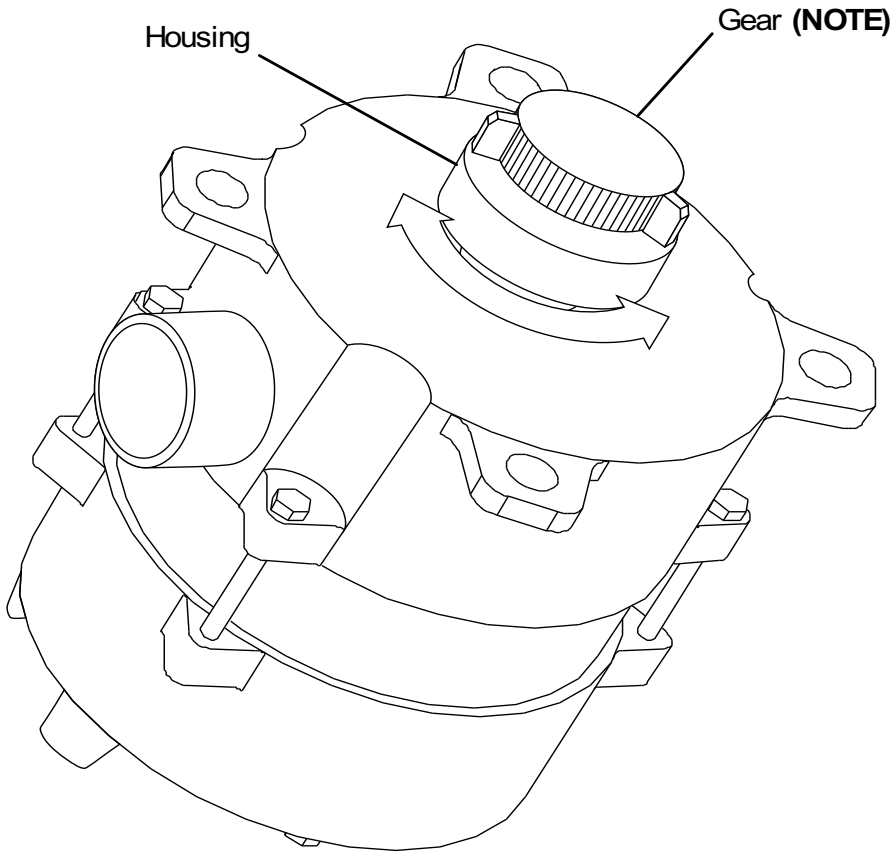
Figure 201 : Sheet 1 : No. 1 (Right) Alternator



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Figure 202 : Sheet 1 : Alternator Drive Hub

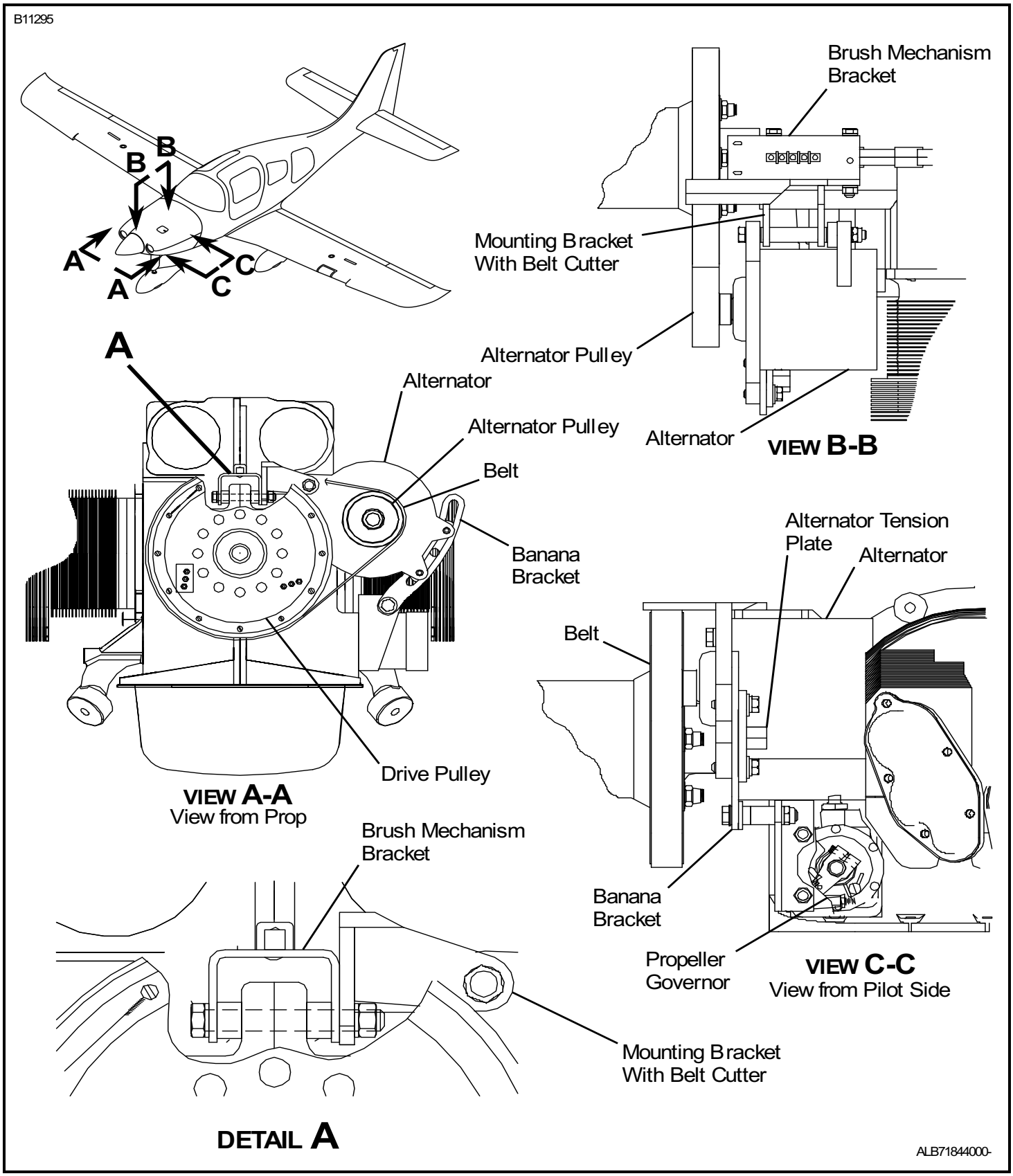
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NOTE: Turn gear while holding housing .

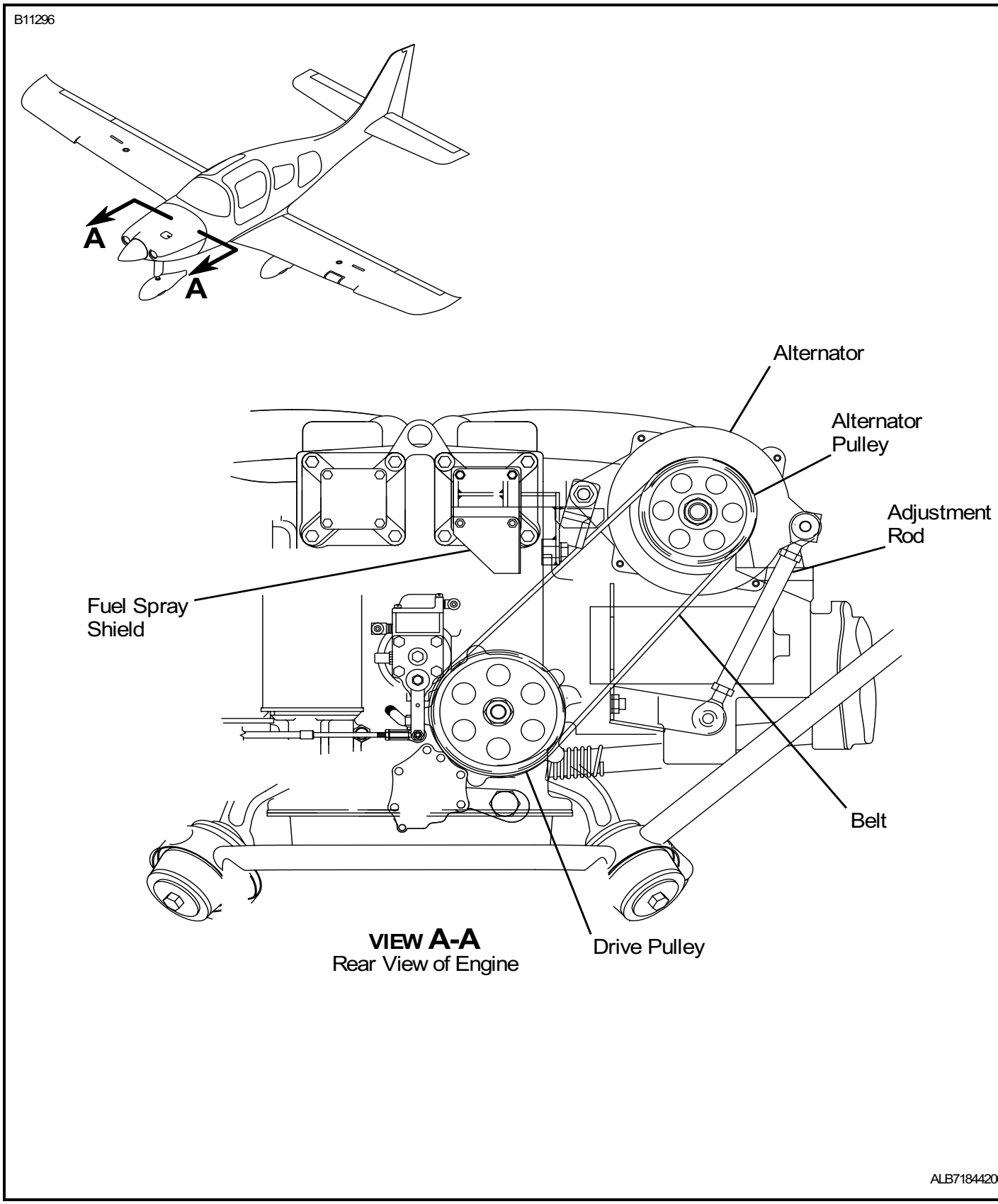
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Figure 203 : Sheet 1 : No. 2 (Left) Alternator



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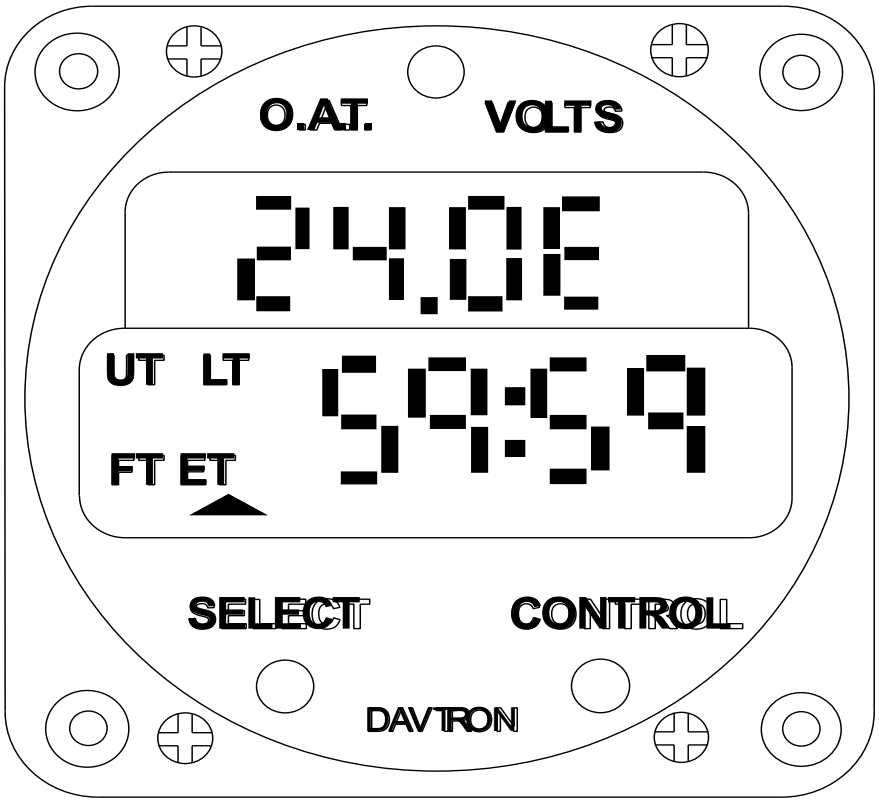
Figure 204 : Sheet 1 : Accessories Alternator



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Figure 1 : Sheet 1 : Voltmeter/OAT/Clock

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BATTERY - MAINTENANCE PRACTICES

1. Normal Battery System (S/N 42001 to 42500)

A. **Applicability** - This procedure is based on the Concorde™ RG-1215 batteries. Refer to battery manufacturer's instructions for batteries other than the one specified.

B. **Removing the Battery** (Refer to Figure 201.)

CAUTION: To reduce the chance of personal injury and possible equipment damage, remove the negative wire before removing the positive wire.

(1) For S/N 42001 to 42022, loosen the three bolts securing the battery box lid to the battery box, and open the lid. For S/N 42023 and on, remove the four bolts securing the battery box cover to the battery box and remove the cover.

(2) Disconnect the negative connector, then the positive connector for each battery. Refer to the detail of battery cable attachment in Figure 202.

(3) Remove the battery.

C. **Installing the Battery**

(1) Place the battery in the battery box.

WARNING: Failure to reconnect the battery connectors properly could result in reversing the polarity of the battery.

(2) Reconnect positive connector, then connect the negative connector for each battery. Torque bolts 36 to 40 in.-lbs. Refer to the detail of battery cable attachment in Figure 202.

(3) For S/N 42001 to 42022, close the battery box cover over the batteries, and install the three bolts. Torque 30 to 36 in.-lbs. For S/N 42023 and on, replace the battery box cover and install the four bolts. Hand tighten the bolts until snug.

CAUTION: Do not replace the sealed lead acid batteries with wet lead acid batteries.

NOTE: Do not use any type of material or sealant to seal holes or gaps in the battery box, or the open area formed where battery wiring passes through a grommet. These spaces must remain clear to ensure ventilation of the batteries.

D. **Test Procedure for Reserve or Emergency Capacity**

(1) Make sure the battery is charged.

(2) With the battery temperature above 59° F, discharge the battery at the 80-85 percent test rate. (Refer to FAR 23.1309, 23.1351, 23.1353(h)).

(3) The minimum end point after one hour of the time established for the airframe essential power requirement of discharge must be 9 volts. If the battery fails to deliver the above rated ampere-hour capacity, it has reached its end of life. The battery is no longer considered airworthy and must be replaced.

(4) Allow the battery to cool to room temperature before recharging.

E. **Maintenance Procedures for Battery**

(1) The batteries must be boost charged every 90 days when in storage.

(2) Batteries that have not been recharged every 90 days are to be conditioned. First discharge the battery at the test rate. Recharge with a constant current charge at the C/10 rate (1/10 of the C rate amperes) for eighteen hours or until the battery voltage reaches 2.5 volts per cell. Allow the battery to rest for at least one hour and re-test per the Test Procedure above.

(3) The battery should be recharged when the open circuit voltage is below 2.08 volts per cell.

(4) The battery should be recharged at 1/2 of the C rate with a constant potential (CP) or constant voltage charger regulated at 2.35 volts per cell. The battery is charged when the charge current stabilizes for one hour.

(5) It is recommended the emergency capacity test procedure be performed after twelve months or six hundred hours of operation, whichever comes first. To ensure continued airworthiness, another emergency capacity test should be performed every twelve months or every two hundred hours.

2. Normal Battery System (S/N 42501 and On)

A. **Applicability** - This procedure is based on batteries approved by Cessna. Refer to battery manufacturer's instructions for batteries other than those approved by Cessna.A battery charging circuit has been added to the power grid on aircraft built on or after mid 2007, retrofit available S/N 42501 and On, that will energize the battery relays and allow ground power

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charging of flat or discharged batteries without removing the battery(ies) from the aircraft. Batteries will start charging when the ground power unit is connected and the master and crosstie switches are ON. The master and crosstie switches must be turned off before removing the ground power plug. Flat or discharged batteries on aircraft without the battery charging circuit must be removed from the aircraft before charging. The circuit does not affect the operation of the master and crosstie switches.

B. **Removing the Battery** (Refer to Figure 203.)

CAUTION: To reduce the chance of personal injury and possible equipment damage, remove the negative wires before removing the positive wires.

(1) Remove the seven bolts and washers securing the battery box cover to the battery box and remove the cover.

(2) Disconnect the negative connector, then the positive connector for each battery. Refer to the detail of battery cable attachment in Figure 204.

(3) For each battery, remove the bolt and washer attaching the battery hold down and remove the hold down.

(4) Remove the battery.

C. **Installing the Battery** (Refer to Figure 203.)

WARNING: Failure to reconnect the battery connectors properly could result in reversing the polarity of the battery.

CAUTION: Do not replace the sealed lead acid batteries with wet lead acid batteries.

CAUTION: The battery cables must be tied to tie blocks to keep them from chafing against the cowling.

NOTE: Do not use any type of material or sealant to seal holes or gaps in the battery box or the open area formed where battery wiring passes through a grommet. These spaces must remain clear to ensure ventilation of the batteries.

(1) Place the battery in the battery box.

(2) Reinstall the battery hold down and tighten the bolt snug.

(3) Reconnect positive connector, then connect the negative connector for each battery. Torque nuts or bolts, as applicable, 36 to 40 in-lbs. Refer to the detail of battery cable attachment in Figure 204.

(4) Replace the battery box cover and secure with bolts and washers. Tighten bolts snug.

D. **Battery Charging** - Use this procedure if the aircraft is not equipped with the battery charging circuit, is equipped with the circuit but ground power is not available, or the battery requires bench charging.

(1) Remove the battery from the aircraft.

(2) Using an AA Portable Power Corp. Smart Fast Charger (1.5 A) for 24V Lead Acid Battery (model No. CH-LA2415, or equivalent), connect the charger to the battery ensuring correct polarity.

(3) Connect the charger to AC power. The following will result:

- The green LED will turn orange.
- When the LED is orange the battery is charging.

CAUTION: Ensure the battery does not overheat. The battery temperature, measured at the case surface, should not exceed 100° Fahrenheit.

(4) When the LED turns green the battery is fully charged.

(5) The Smart Fast Charger may be left on the battery after the battery is full charged.

(6) After the battery is fully charged, turn off the Smart Fast Charger by removing the AC power plug.

(7) Disconnect the battery.

(8) Allow the battery to cool down to room temperature.

E. **Battery Discharge Test**

(1) Fully bench charge the battery.

(2) Prior to connecting to the battery, ensure that the load tester is off.

(3) The battery should be discharged at the 1 hour rate (check the battery amphour rating) to an end voltage of 1.67 volts per cell or 20 volts. Measure the time. The battery must achieve at least 80 percent of the rated time (or 48 minutes at the 1 hour rate). If the first discharge time is less than 48 minutes, repeat the cycle one more time.

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(4) If the second discharge fails to deliver at least 48 minutes, the battery should be rejected. Return the battery to the vendor for servicing or replacement if the warranty is still valid. If not, scrap the battery.

(5) Once the battery has passed all required inspections and after it is fully recharged, the battery is ready for installation.

F. **Battery Maintenance**

WARNING: The battery must be removed from the installation and serviced in a well ventilated designated area. During servicing, the battery will generate oxygen and hydrogen gases, which can be explosive under the right conditions.

WARNING: These batteries contain sulfuric acid which is highly corrosive and which can cause serious physical injury if it comes in contact with skin or if inhaled. It can also cause serious eye injury or blindness if it comes into contact with the eyes.

WARNING: Caution must be exercised to avoid damage to the exterior case which could allow the contents to escape or come in physical contact with external materials or personnel. If a battery case is found to be damaged, handle the battery with care and avoid contact with the skin. Inspect all areas adjacent to the battery for evidence of corrosion.

(1) After initial installation, a capacity check of the battery is required at 800 hours or 11±1 months, whichever comes first with subsequent capacity checks performed every 400 hours or 6±1 months.

(2) Visually inspect the battery for any signs of cracks, corrosion, unusual terminal pin wear or discoloration on the pins.

(3) Before reconnecting to the aircraft, ensure the terminals have not worn or become loose.

G. **Battery Reconditioning** - Delaying the recharge of even a partially discharged battery can make it resist the charger for some time. When put on a charger the displays may indicate that the battery is fully charged, but when you attempt to use it again it acts as if completely discharged.
Recharging immediately after use will significantly increase the service life of the battery. It is recommended that the battery be recharged within 24 hours of being discharged. The amount of capacity regained with each cycle will eventually level out and this stabilization is an accurate indication of the battery's useful capacity.

(1) Charge the battery for 24 hours to ensure that it has gone through a complete charging cycle.

(2) Perform a discharge test.

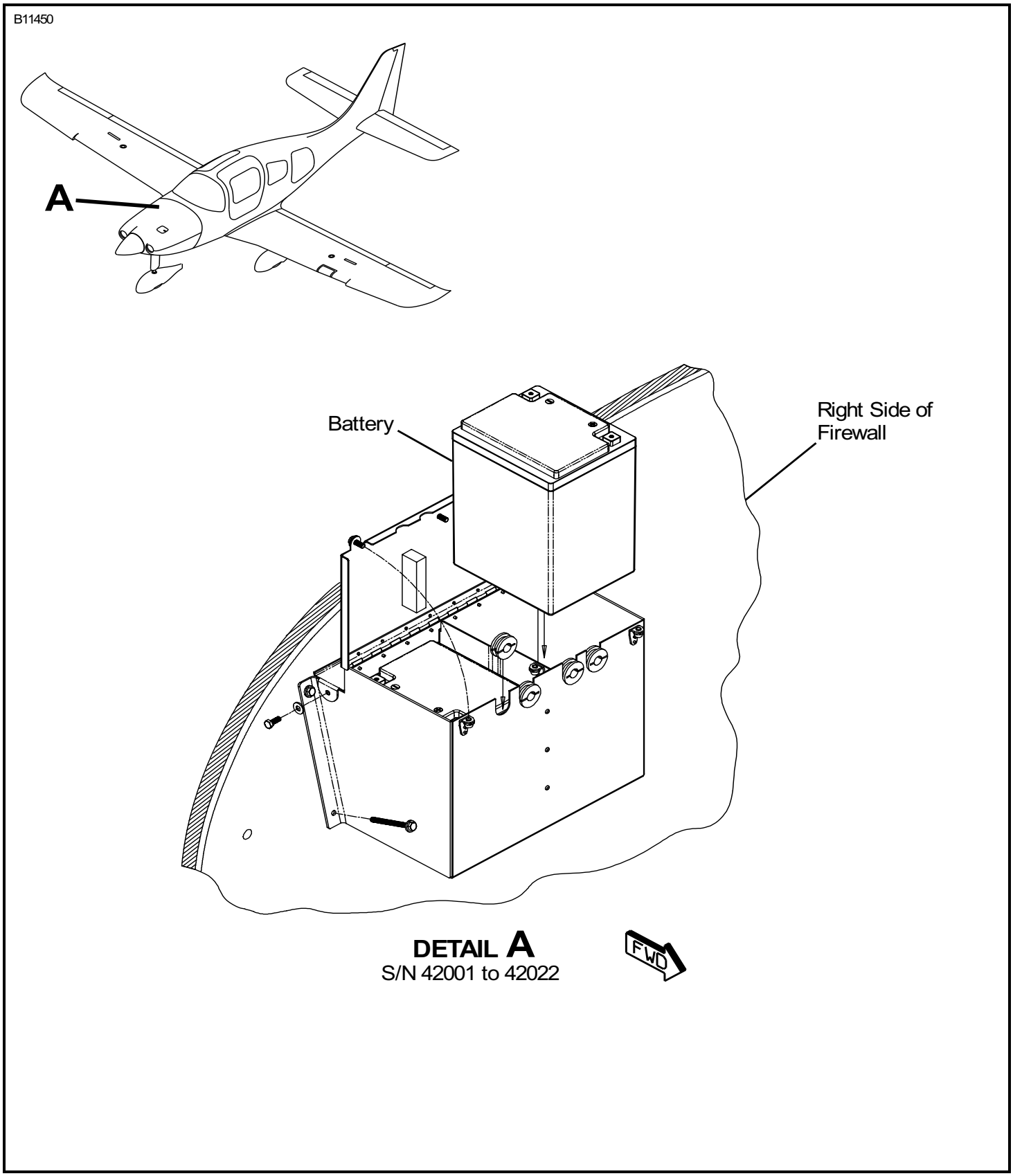
(3) If the battery fails the discharge test, charge the battery again for 24 hours.

(4) Perform a discharge test.

(5) Repeat this charge/discharge process 4 or 5 times. The capacity of the battery should increase appreciably with four to five charge/discharge cycles. If the battery still fails to measure up to capacity, return the battery to the vendor for servicing or replacement if the warranty is till valid. If not, scrap the battery.

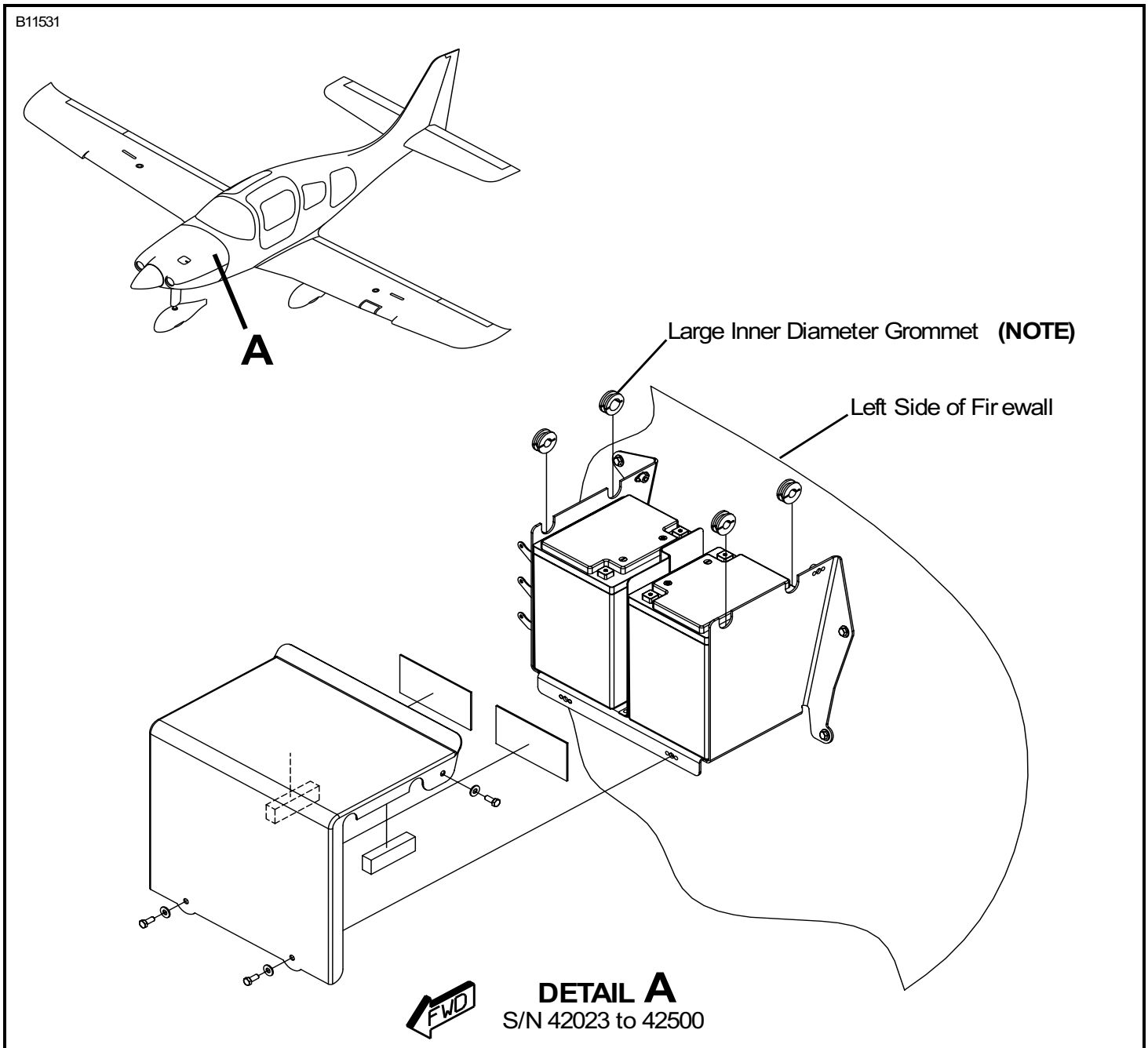
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Figure 201 : Sheet 1 : Battery Installation (S/N 42001 to 42500)



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Figure 201 : Sheet 2 : Battery Installation (S/N 42001 to 42500)

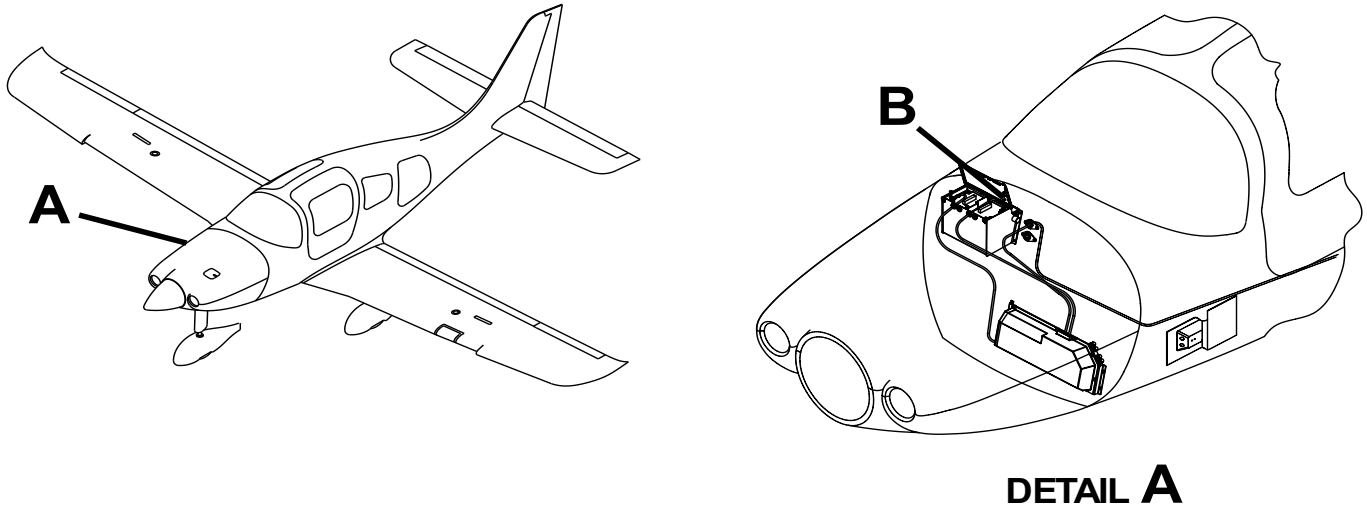


NOTE: Used on Fire Jacket Protected Lead.

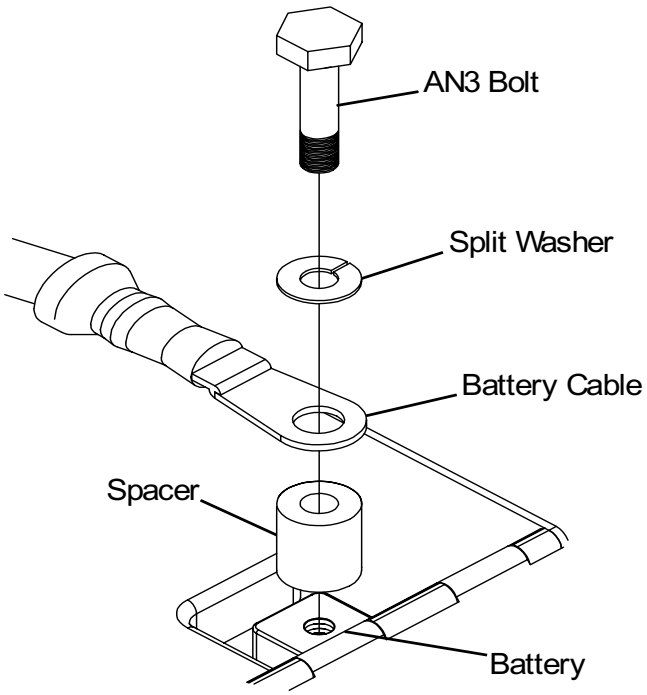
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Figure 202 : Sheet 1 : Battery Cable Attachment (S/N 42001 to 42500)

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DETAIL A

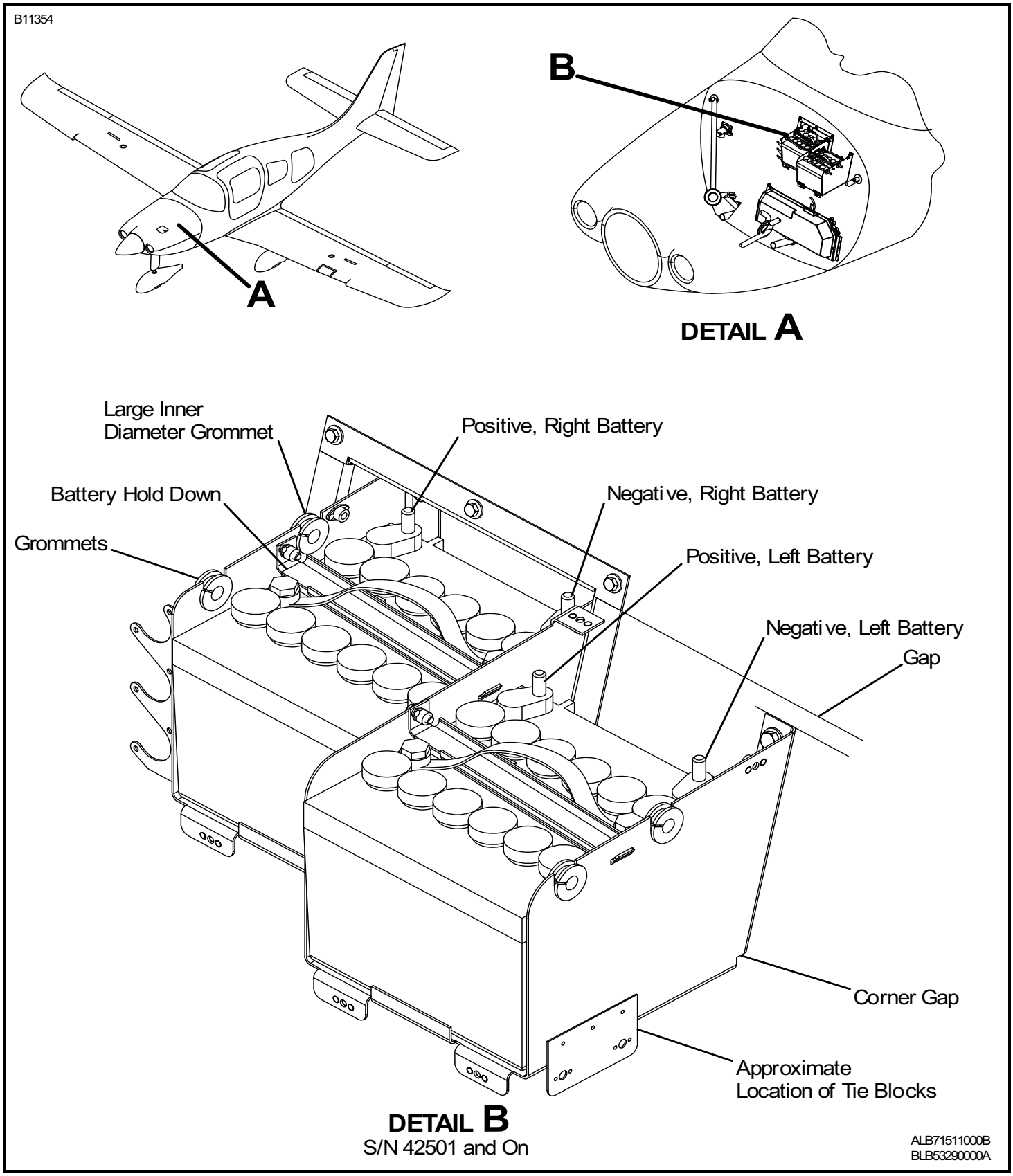


DETAIL B
S/N 42001 to 42500

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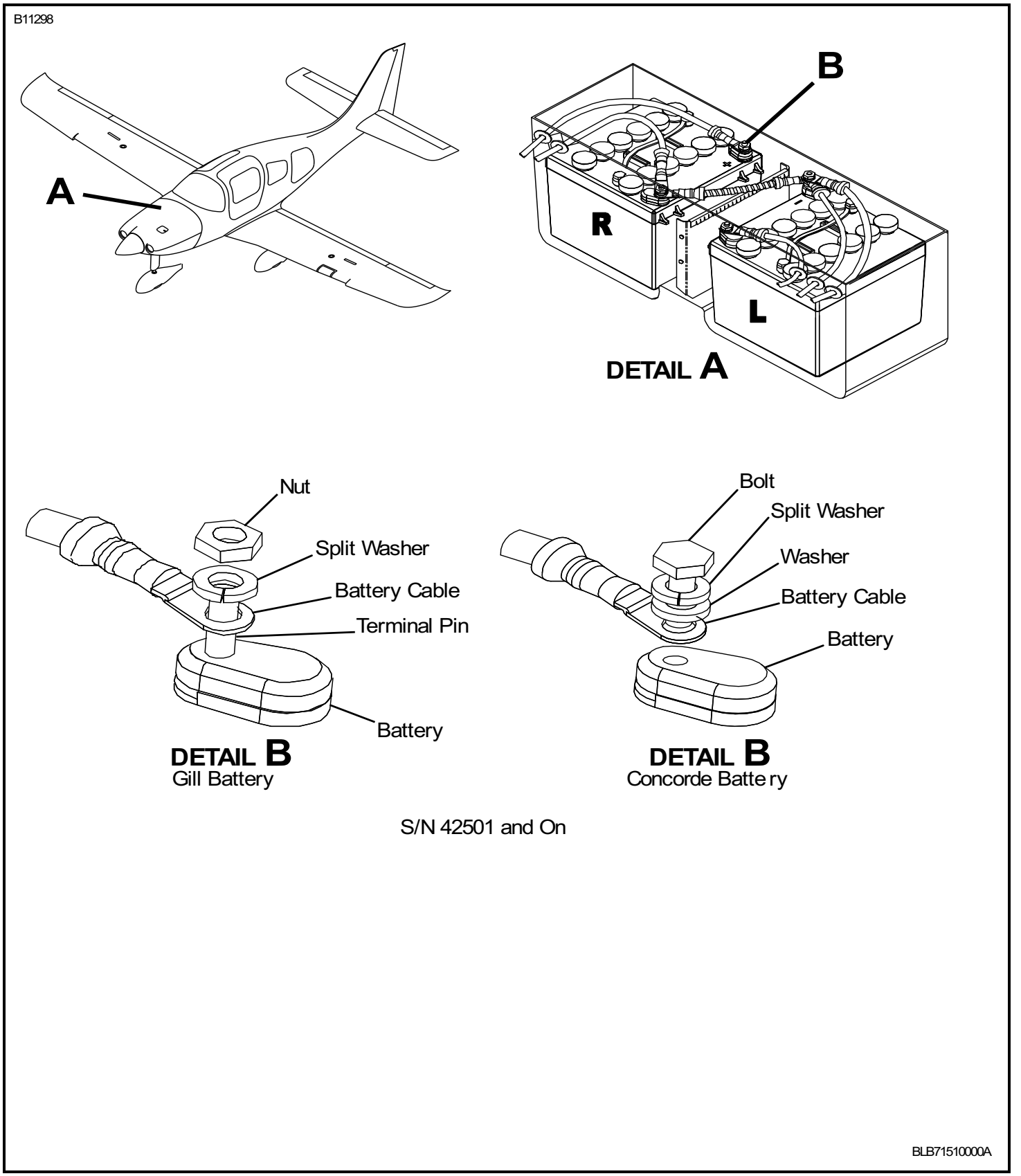
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Figure 203 : Sheet 1 : Battery Installation (S/N 42501 and on) (Cover Not Shown)



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Figure 204 : Sheet 1 : Battery Cable Attachment (S/N 42501 and On)



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GROUND POWER - MAINTENANCE PRACTICES

1. Ground Power Plug

A. Removal and Replacement of the Ground Plug Relay

(1) Remove the cover from the power grid by removing the two 8-32 pan head screws and washers.
(2) Verify that the aircraft power is off.
(3) Remove the large power cables and mark the location.
(4) Remove the relay.
(5) Install a new transorb (surge suppressor) on the new relay.
(6) Move the reverse current diode to the new relay.
(7) Reassemble in the reverse order.

B. Removal and Replacement of the Power Plug Receptacle (S/N 42001 to 42500)

(1) Turn the aircraft master switch on and fully extend the flaps.
(2) Turn the aircraft master switch off and placard "Do Not Turn On Power".
(3) Remove the four 8-32 screws attaching the step cover, and remove the cover.
(4) Remove two 1/4-28 screws, washers, and self-locking nuts attaching the receptacle. Refer to Figure 201, Sheet 1.
(5) Pull the power plug receptacle down to gain access to the solder joint.
(6) Unsolder with a propane torch adjusted to low.
(7) Re-solder a new plug, and slide the boot over the solder joint.
(8) Reassemble in reverse order. Torque attachment screws 140 to 180 in.-lbs.

C. Removal and Replacement of the Power Plug Receptacle (S/N 42501 and On)

(1) Turn the aircraft master switch off and placard "Do Not Turn On Power".
(2) Remove the hat rack carpet per instructions in Section 25-30-00, Hat Rack Carpet.
(3) Remove the screws securing the baggage bulkhead access panel and remove the panel.
(4) Remove the nuts, lock washers, and washers attaching the cables to the receptacle. Refer to Figure 201, Sheet 2.
(5) Remove the four 6-32 pan head screws and washers, attaching the receptacle. Refer to Figure 201, Sheet 2. Note the location of the washers for reinstallation. The purpose of the washers is to align the receptacle outer door flush with the fuselage.
(6) Reassemble in reverse order. Torque 6-32 screws 10 in.-lbs.

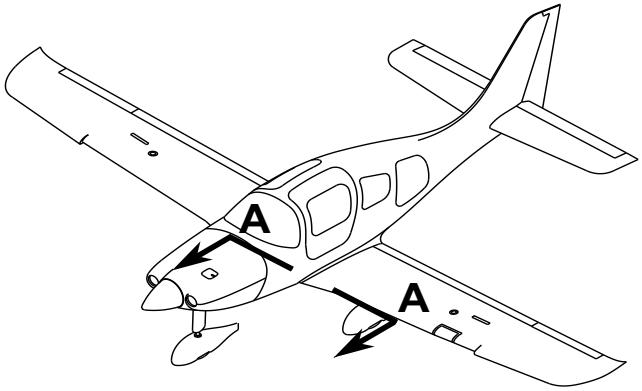
D. Ground Power Plug Troubleshooting

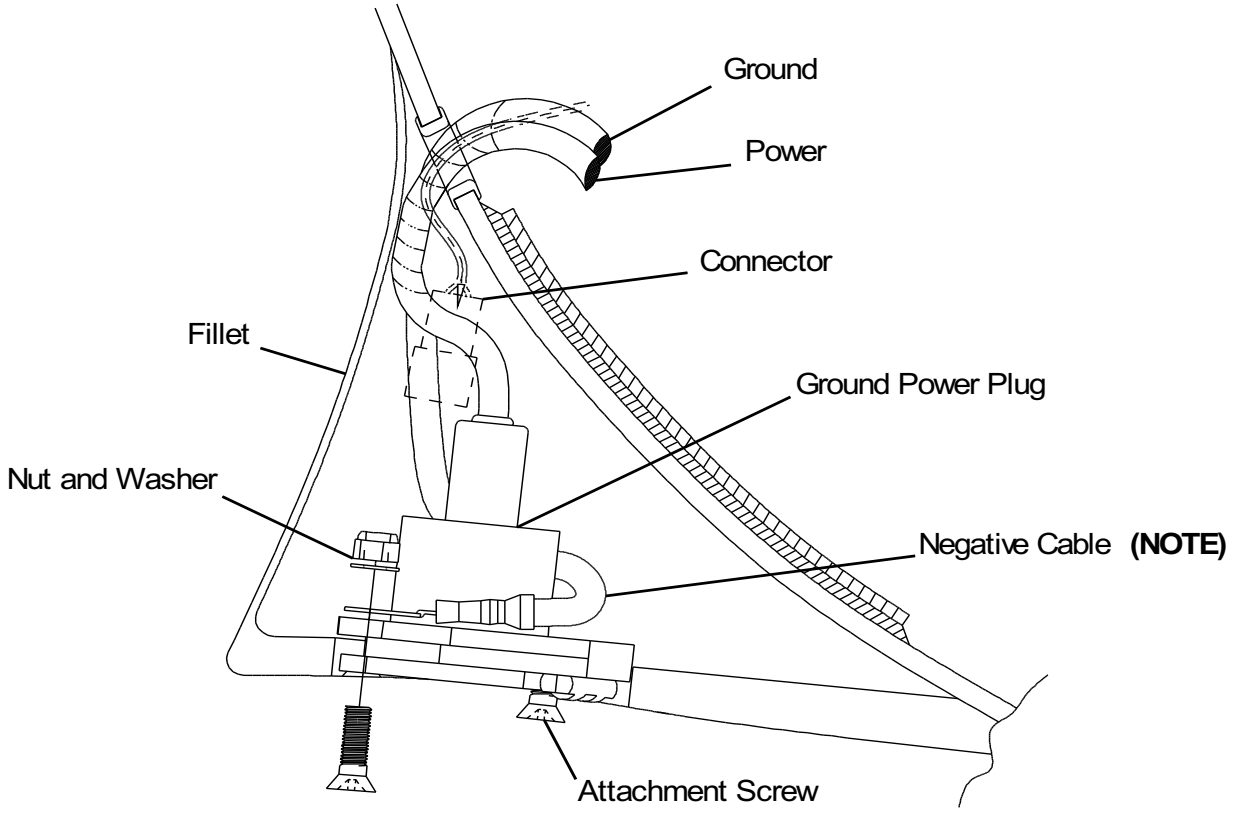
(1) If the ground power plug relay will not engage:
    (a) The reverse current diode mounted on the relay may be open or have loose connections. Replace the relay or tighten the connections.
    (b) The transorb (surge suppressor) is shorted. Replace transorb.
    (c) Ground at the power plug or at the engine block is disconnected or loose. Reconnect or tighten the ground.
    (d) Contamination in the ground power plug does not allow the connector to go in all the way. Replace plug.
    (e) Ground Power relay windings are open. Replace relay.
(2) Ground power relay will not disengage.
    (a) Relay contacts are welded together internally. Replace relay.
    (b) BATT switches are turned on. Turn off BATT switches.

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Figure 201 : Sheet 1 : Ground Power Plug Installation

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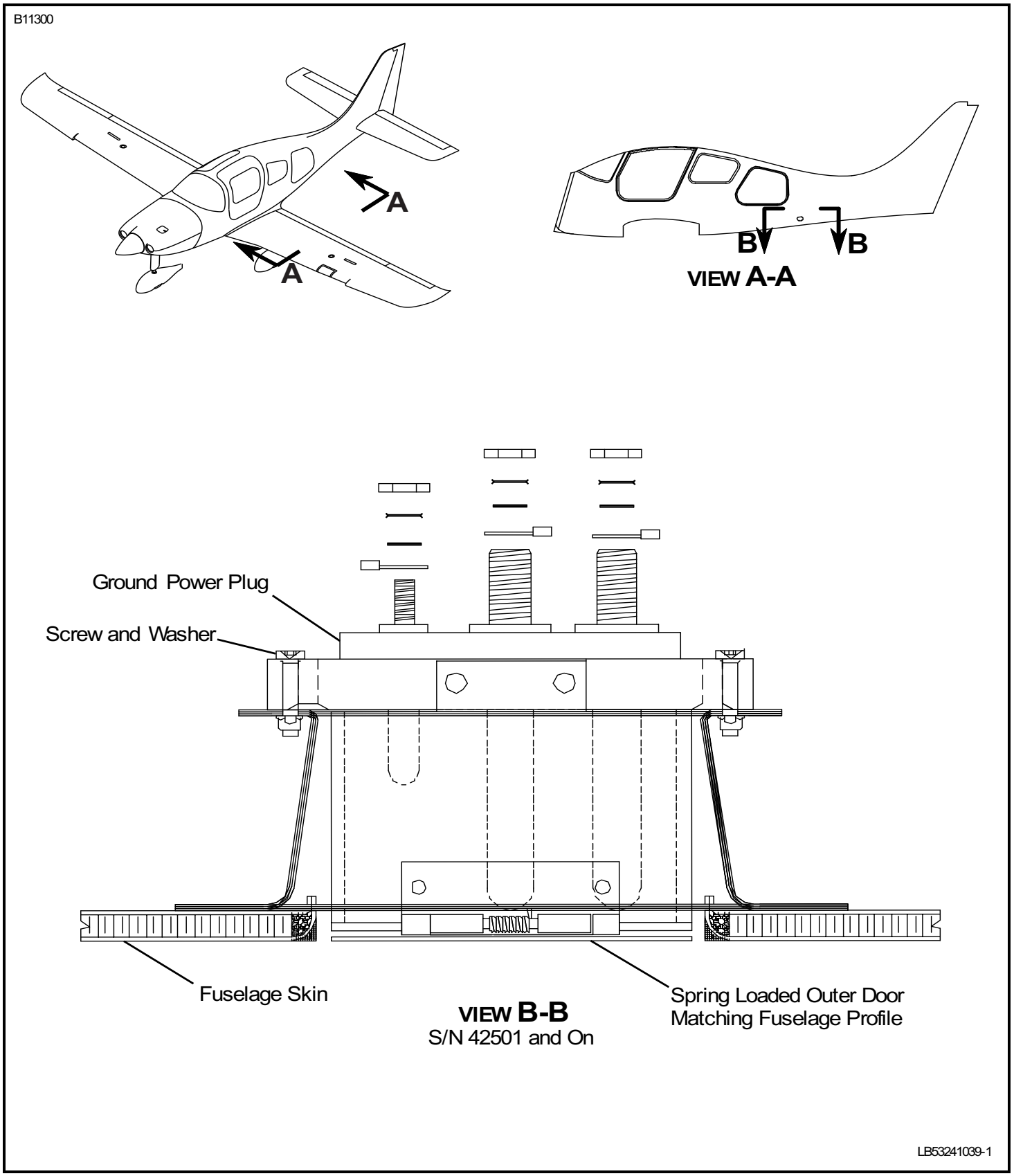
VIEW A-A
S/N 42001 to 42500

NOTE: May be attached to either bolt of the ground power plug.

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Figure 201 : Sheet 2 : Ground Power Plug Installation



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ELECTRICAL LOAD DISTRIBUTION - MAINTENANCE PRACTICES

1. Electrical Load Distribution

A. **Removal of Voltage Regulator** (Refer to Figure 201.)

(1) The voltage regulators are attached to each other and a metal plate connected to the inside firewall on the copilot's side or attached under the instrument panel on the copilot's side. Remove the glare shield as necessary for access to the regulators.

(2) Unhook the wires connected to the voltage regulators.

(3) Remove the three pan-head screws attaching the top voltage regulator to the bottom voltage regulator, and remove the top voltage regulator.

(4) Remove the three standoffs securing the bottom voltage regulator to the metal plate, and remove the voltage regulator from its mounting.

B. **Installation of Voltage Regulator**

(1) Place the bottom voltage regulator in its mounting.

(2) Install the three standoffs to secure the bottom voltage regulator to the plate. The standoffs must protrude through the nutplate flush or better.

(3) Place the top voltage regulator over the standoffs and install the three pan-head screws and six washers.

(4) Reconnect the wires to the voltage regulator.

C. **Removal of Rocker and Master Switch Panels**

(1) Steps for the removal of the rocker and master switch panels can be found in Section 25-11-00.

D. **Installation of Rocker and Master Switch Panels**

(1) Steps for the installation of the rocker and master switch panels can be found in Section 25-11-00.

2. Circuit Breaker Panel

A. Steps for the removal and installation of the circuit breaker panel can be found in Section 25-11-00, Circuit Breaker Panel.

3. Power Grid

A. The power grid is locate on the firewall in the engine compartment on the pilot's side. The power grid does not require any service or maintenance. However, it should be visually inspected every 1000 hours for frayed or chafed wiring and for evidence of corrosion and/or overheating.

4. Electric A/C Interlock Assembly

A. The Electric A/C Interlock Assembly, located under the left side of the baggage shelf, is a component of the air conditioning system (ACCS) when the system has the optional electrically driven compressor. Refer to Section 21-65-00 for a description of the ACCS.

(1) **Removal of the Electric A/C Interlock Assembly** (Refer to Figure 202.)

(a) Remove the baggage shelf carpet per instructions in Section 25-30-00, Removal of the Baggage Floor Carpet.

(b) Remove the Elevator Interconnect Access Panel and/or the Electronics Bay Access Panel per instructions in Section 25-31-00.

(c) Disconnect all electrical connections to the interlock assembly.

(d) Disconnect the 3/4-inch interlock cooling SCAT tube attached to the interlock assembly.

(e) Remove 4 screws and washers attaching the interlock assembly bracket to the shelf.

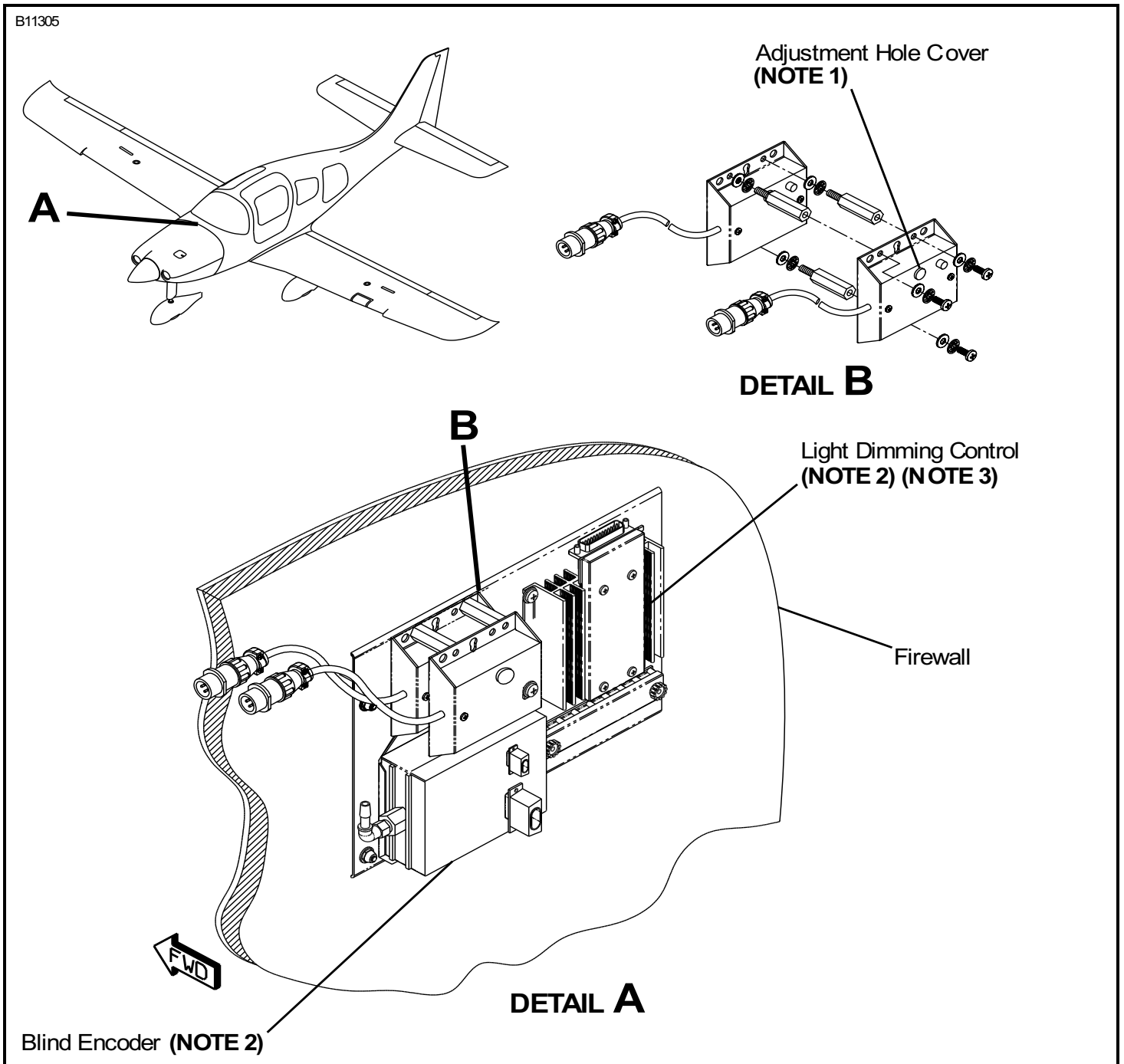
(f) Remove the interlock assembly.

(2) **Installation of the Electric A/C Interlock Assembly** - Installation is the reverse of removal.

(3) **Maintenance of the Electric A/C Interlock Assembly** - The Electric A/C Interlock Assembly does not require maintenance.

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Figure 201 : Sheet 1 : Voltage Regulator Installation

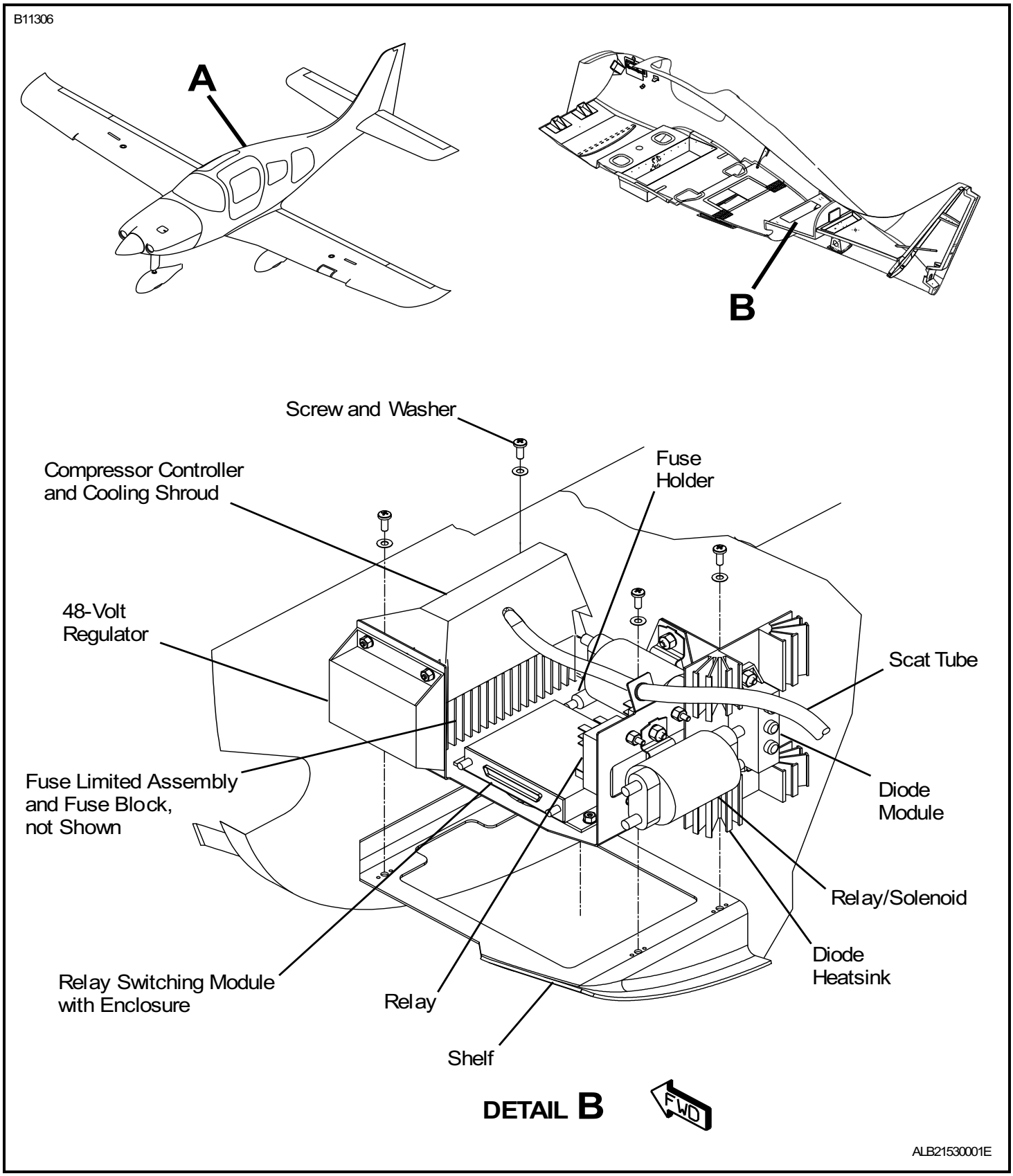


NOTE 1: Adjust voltage to 14.2 VDC (S/N 42001 to 42500) or 28.4 ± .4 VDC (S/N 42501 and on).
NOTE 2: Not present with Garmin G1000.
NOTE 3: Installed position may be 90 degrees from shown.

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Figure 202 : Sheet 1 : Electric A/C Interlock Assembly



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INDICATING/RECORDING SYSTEMS - GENERAL

1. Voltmeter/OAT/Clock

A. If the Avidyne Entegra Primary Flight Display (PFD) or the Garmin G1000 system is installed in the aircraft, the Voltmeter/OAT/Clock described by the following paragraphs will not be installed in the aircraft.

(1) **Voltmeter/OAT/Clock** - This instrument contains three separate indications. The upper window is dedicated to voltmeter and outside air temperature readings, and the lower window is a multifunction timepiece. Refer toFigure 1 for a drawing of the instrument. There are three control buttons on the face of the indicator. Pressing the top button, labeled O.A.T. and VOLTS, will cycle the reading in the upper Liquid Crystal Display (LCD) from a voltage reading, to an outside air temperature in degrees F, to an outside air temperature reading in degrees C. The lower two buttons, labeled SELECT and CONTROL, are for the multifunction clock and are discussed in Section 7 of the POH/AFM.

(2) **The digital clock** displays four time modes: universal time (UT), sometimes referred to as Greenwich Mean Time (GMT), local time (LT), flight time (FT), and elapsed time (ET). The time mode selected is indicated in the lower digital display window on the left side with a flashing underscore below the selected time mode. In Figure 1 the ET time mode is selected. Pressing the SELECT button several times cycles the display through the four time modes in the order they are listed.

(3) **To test the clock**, hold the SELECT button in for three seconds. This will cause the display to read '88:88' with all four modes underscored. When electrical power is first turned on, the top display will indicate the system's voltage, and the clock will display the mode it was in when the electrical power was turned off. When the electrical system of the airplane is not on, the time and other settings such as flight time are held in the memory of the clock by a backup AA battery. The battery has a three-year life, but it must be replaced every two years with a good 1.5 volt battery. Incorrect time and mode readings during power up are indications of an expended or defective backup battery. If temperatures are below -20°F, the numbers in the clock will not appear.

(4) **Initial Set-Up** To Set UT (Universal Time) (see Note), use the following instructions:

(a) Press the SEL button one or more times until UT is displayed on the left hand side of the lower window.

(b) Press both the SEL and CTL buttons at the same time to enter the set mode. The tens of hours digit will start flashing.

(c) Press the CTL button one or more times to change the tens of hours digit to the correct digit of Universal Time (refer to Note 1).

(d) Press the SEL button once to move to the single hour digit, which will start flashing.

(e) Press the CTL button one or more times to change the single hour digit to the correct digit of Universal Time.

(f) Press the CTL button one or more times to change the tens of minutes digit to the correct digit of Universal Time.

(g) Press the SEL button once to move to the single minute digit, which will start flashing.

(h) Press the CTL button one or more times to change the single minute digit to the correct digit of Universal Time.

(i) Press the SEL button to exit the set mode. The lighted annunciator will resume its normal flashing indicating the UT clock is running.

(5) To Set LT (Local Time) use the following instructions:

(a) Press the SEL button one or more times until LT is displayed on the left hand side of the lower window.

(b) Press both the SEL and CTL buttons at the same time to enter the set mode. The tens of hours digit will start flashing.

(c) Press the CTL button one or more times to change the tens of hours digit to the correct digit of Local Time (refer to Note 2).

(d) Press the SEL button once to move to the single hour digit, which will start flashing.

(e) Press the CTL button one or more times to change the single hour digit to the correct digit of Local Time.

(f) Press the SEL button to exit the set mode as the minutes are already synchronized with the UT clock and can not be set in Local Time.

NOTE: (1) The times of various events, particularly astronomical and weather phenomena, are often given in "Universal Time" (abbreviated UT) which is sometimes referred to, now colloquially, as "Greenwich Mean Time" (abbreviated GMT). The two terms are often used loosely to refer to time kept on the

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Greenwich meridian (longitude zero), five hours ahead of Eastern Standard Time. Times given in UT are almost always given in terms of a 24-hour clock.

NOTE: (2) Local Time is eight hours later (PST) or seven hours later (PDT) than the Universal Time just set.

2. Hour Meter - General

A. **Indicating** - Flight time is recorded by an hour meter - most frequently referred to as the Hobbs. If basic instruments only or Avidyne avionics are installed, the Hobbs is installed in the right knee bolster, next to the ELT remote switch. If the Garmin G1000 system is installed, the Hobbs is installed on the left side of the center console. The instrument is DC powered. DC power for the hour meter bypasses the master switch and will operate with the master switch in the off position, provided sufficient battery power is available. The system is protected by a fuse located in a fuse block next to the battery. The meter will record up to 9999.9 hours. Flight time is measured by either an oil pressure switch or an air switch, depending on which option is installed.

B. **Oil Pressure Switch** - There is an oil pressure switch next to the oil pressure transducer as shown in Section 31-30-00, Figure 201. The switch is closed when oil pressure is introduced to the system. If the battery is adequately charged, a ground is established for the Hobbs (the other side of the Hobbs is powered from the 3 amp fuse on the power grid), and the time the engine is in operation is recorded and displayed on the meter.

C. **Air Switch** - There is an air switch located either under the standby battery bracket or next to the terminal junction rail behind the co-pilot's side of the instrument panel if Avidyne avionics are installed, or behind the instrument panel on the co-pilot's side if the Garmin G1000 system is installed, that activates at 2 in. of Hg or approximately 60 kts. This switch is plumbed into the pitot line for its activation.

NOTE: To disable the Hobbs during a pitot static check, remove the Hobbs fuse from the power grid. This will stop the accumulation of "false" time on the Hobbs. Replace the fuse after the test.

3. Documentation Of Hobbs Meter Replacement

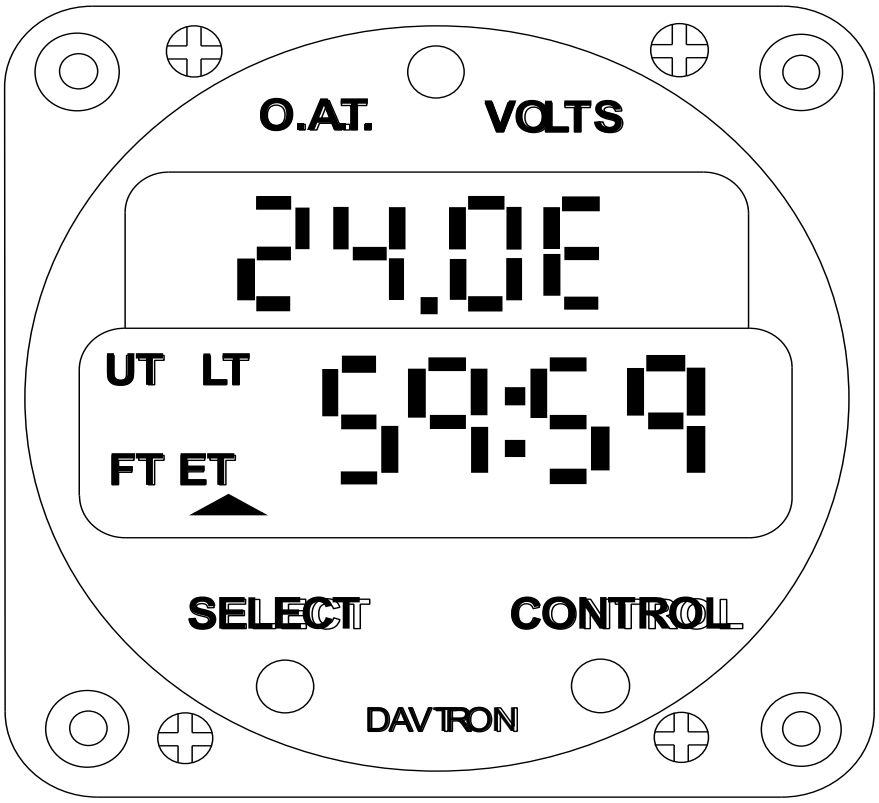
A. **Placard** - If a Hobbs meter becomes inoperative, it must be replaced as soon as possible. The time shown on the defective meter plus the best estimate of the additional time in service must be placarded in an area near the Hobbs. For example, if the meter had 627.9 hours when it became inoperative and an additional 2.1 hours of time in service was flown before replacement, then the placarded time would be 630 hours. A statement like *630 hours is the current Hobbs Meter reading* is an appropriate placard.

B. **Logbooks** - Similar statements like those in the Placard paragraph above, along with the date and time of the discrepancy, must be adequately documented in the airframe and engine logbooks.

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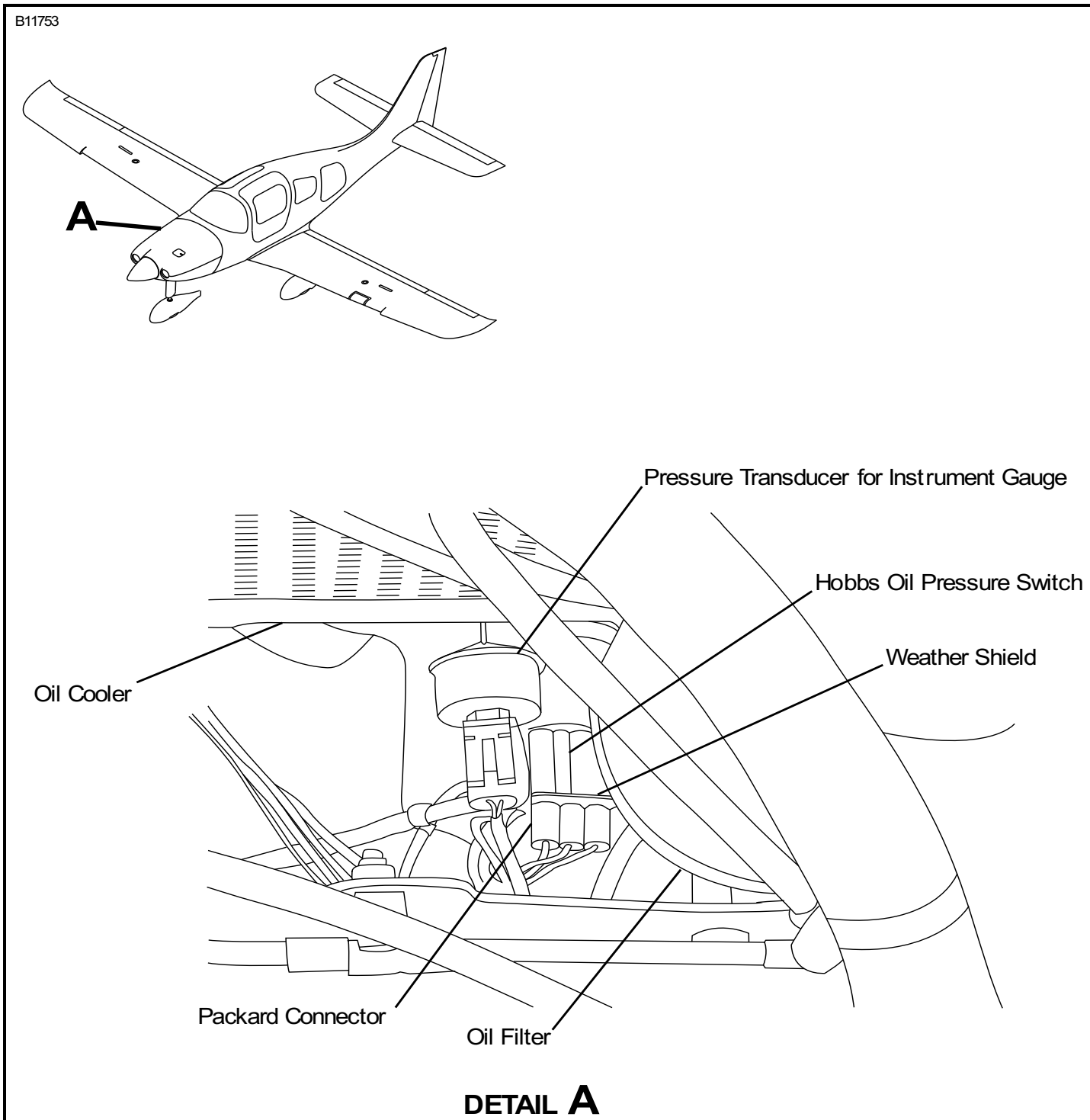
Figure 1 : Sheet 1 : Voltmeter/OAT/Clock

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Figure 201 : Sheet 1 : Hobbs Oil Pressure Switch



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CLOCK - MAINTENANCE PRACTICES

1. Removal, Installation, and Maintenance of the Clock

A. Removal of the Clock

(1) Remove the glare shield per instructions in Chapter 25-11-00, Glare Shield, and remove the connector installed in the back of the clock.

(2) Unscrew the four panel screws surrounding the clock.

(3) Remove the clock from the backside of the flight instrument panel.

B. Installation of the Clock

(1) Install the connector into the back of the clock.

(2) Align the clock with the screw holes in the flight instrument panel, and install the four screws. The screws should be snug tight.

(3) Reinstall the glare shield.

C. **Maintenance of the Clock** - No special maintenance of the clock is necessary other than the biennial replacement of the backup battery.

NOTE: It is not necessary to remove the clock to replace the backup battery. The clock is in the top portion of the flight instrument panel, and when the glare shield is removed, the AA battery can be accessed.

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INDICATING/RECORDING SYSTEMS - MAINTENANCE PRACTICES

1. Removal, Installation, and Maintenance of the Hour Meter

A. Removal of the Hour Meter

(1) If basic instruments only or Avidyne avionics are installed in the aircraft, remove the right knee bolster per instructions in Chapter 25-11-00. If the Garmin G1000 system is installed, remove the access panel on the right side of the center console.

(2) Remove the connector installed in the back of the hour meter. Be sure to label the location of each wire.

(3) Unscrew the two screws on the back of the knee bolster, and slide the unit out.

B. Installation of the Hour Meter

(1) With the hour meter partially inserted through the hole in the knee bolster or center console, as applicable, install the connectors (per labeling) into the back of the hour meter.

(2) Slide the unit into its slot flush with the bolster or center console, as applicable, and install the two screws. The screws should be snug tight.

(3) Reinstall the knee bolster per instructions in Chapter 25-11-00 or replace the center console access panel, as applicable.

C. Hour Meter Maintenance - No special maintenance of the hour meter is required.

2. Removal and Installation of the Oil Pressure Switch

A. Switch Removal - Use the following steps for removal of the oil pressure switch. Refer to Figure 201.

(1) Remove the top engine cowling per instructions in Chapter 71-10-00.

(2) Remove the Packard electrical plug from the back of the switch.

(3) Use an open-end wrench to screw the 1/8 inch pipe-threaded switch out of the engine case.

B. Switch Installation - Use the following steps for installation of the oil pressure switch.

(1) Install the 1/8 inch pipe-thread pressure switch into the engine case. Torque switch 60 to 80 in.-lbs.

(2) Connect the Packard connector to the oil pressure switch. Ensure there is a red weather shield around the Packard switch.

(3) At an oil temperature reading of 75°F or higher, run the engine at about 1700 RPM to pressure test the system.

(4) Shut down the engine and check for oil leaks.

(5) Reinstall the engine cowl.

3. Removal and Installation of the Air Switch

A. Removal of the Air Switch

(1) If basic instruments only or Avidyne avionics are installed in the aircraft, remove the glare shield and right side kidney panel per Chapter 25-11-00. If the Garmin G1000 system is installed, remove the glare shield per Chapter 25-11-00.

(2) Remove the mounting hardware, wires, and pitot line.

(3) Cap the pitot line, and connect the wires together if the aircraft is going to be operated with the switch removed.

B. Installation of the Air Switch

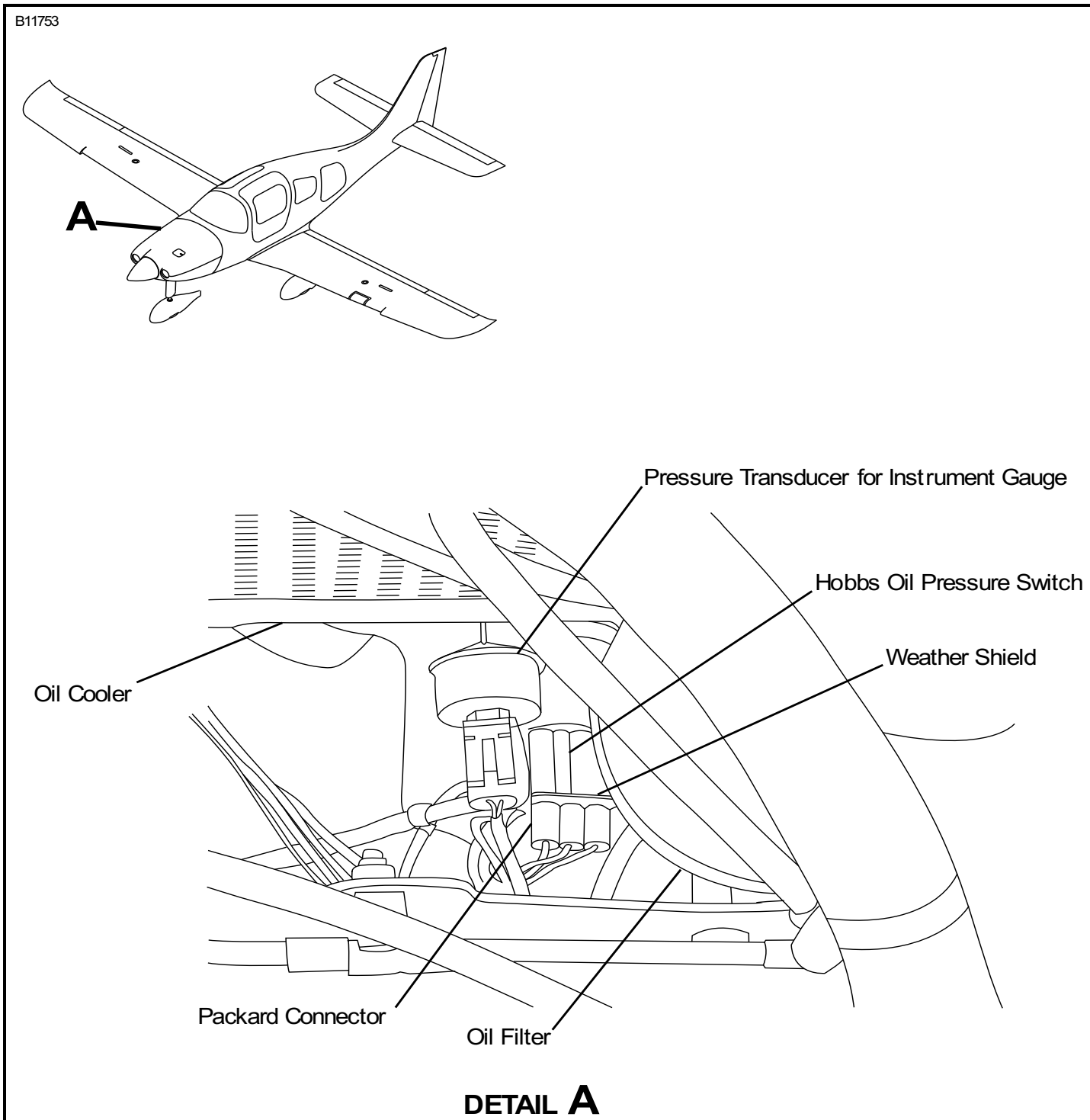
(1) Connect the pitot line, wires (to normally open and common), and mounting hardware.

(2) Test the pitot system for leaks. The system cannot exceed 500 fpm at 1000 ft. above field elevation.

(3) Replace the glare shield and kidney panel, as applicable, per instructions in Chapter 25-11-00.

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Figure 201 : Sheet 1 : Hobbs Oil Pressure Switch



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S-TEC 55X AUTOPILOT WITH AUTOTRIM - GENERAL

1. General

A. **System Overview** - The S-Tec 55X autopilot is a two-axis system that controls movement around the pitch and roll axes. Information for movement around the roll axis is obtained from the turn coordinator of the airplane. The roll axis of the System 55X has heading select, VOR/Localizer front and back course intercept and tracking. It is also interfaced with the GPS and Nav system that provides standard autopilot outputs.

B. The System 55X derives pitch axis information from a solid-state absolute pressure transducer and an internal, sensitive accelerometer that provide precise pitch axis altitude hold, automatic/manual glideslope intercept and capture, and vertical speed commands totally independent of the aircraft's artificial horizon gyro. The System 55X autopilot delivers accurate altitude, vertical speed, and vertical acceleration data to the system's pitch computer, regardless of aircraft flight attitude.

2. The System 55X Flight Guidance Programmer/Computer

A. The System 55X Flight Guidance Programmer/Computer serves the function of converting pilot commands to logic signals for the roll and pitch computer functions. As the pilot enters the desired mode by pressing the appropriate mode selector switch, the computer acknowledges the mode, causing the appropriate annunciator to illuminate.

B. The roll computer receives signal inputs from the Directional Gyro or HSI, VOR/LOC and GPS deviation indicators, and the turn coordinator. The roll computer computes roll servo commands for stabilization, turns, radial intercepts, and tracking.

C. The pitch computer receives signal inputs from the altitude pressure transducer; accelerometer; glideslope deviation indicator; and vertical speed modifier control, optional altitude selector/alerter, or altitude/vertical speed selector if installed. The pitch computer computes pitch servo commands for vertical speed, altitude hold, and glide slope intercept and tracking. Sensing for trim annunciation or automatic elevator trim is provided by the pitch servo. Drive for the elevator trim servo is provided by the pitch computer. Refer to Figure 2 for a diagram of the autopilot system.

3. Preflight Self-Test Procedure (S-Tec 55X Autopilot)

A. The System 55X incorporates a self-test that must pass before the autopilot can be engaged. To perform the test, aircraft DC electrical power must be on and the avionics master on. Place the Autopilot Master switch to the on position, and observe that all annunciators of the Programmer/Computer and remote display illuminate for approximately five seconds, then extinguish. When the system is first turned on, the Flight Guidance Programmer/Computer and Annunciator will display all functions as shown in Figure 1. When the self-test is complete, a RDY (ready) annunciation is displayed. Should a fault be detected, the FAIL annunciator will remain on and the auto pilot will not operate.

4. Checking The Autotrim System (S-Tec 55X)

A. Place the Trim Master Switch in the on position.

B. Press and release the HDG and VS switches and ensure that HDG and VS illuminate on the annunciator.

C. Rotate the VS knob clockwise. The control stick should move slowly out (pilot may have to assist a heavy yoke).

D. Rotate the VS knob counterclockwise. The control stick should move slowly in.

E. Press the autopilot disconnect switch (APD) on the control stick (refer toFigure 3) and verify that the autopilot disconnects.

F. Engage HDG mode and move the DG or HSI HDG bug left and right. The control stick should follow the HDG bug.

(1) If HSI equipped, center the course arrow under the lubber line and push the NAV button. Move the course arrow on the HSI left then right. The control stick should follow the course arrow. Channel a valid VOR signal and move course arrow just enough to deflect the left/right needle 1 or 2 dots. The control stick should follow the Course Deviation Indicator (CDI) left/right needle during the test. (This test is only valid if the left/right needle is centered with the course arrow under the lubber line)

(2) If DG equipped, center the HDG bug under the lubber line. Channel a valid VOR signal. Move the OBS to cause left/right CDI needle deflection. The control stick should follow the left/right needle movement.

G. Press the REV mode switch. The control stick should respond opposite to the course arrow and CDI left/right needle inputs.

H. Press the ALT (Altitude Hold) switch. Slowly pull aft (nose up) on the control stick. Autotrim should run nose down with TRIM flashing on the remote annunciator and the autopilot computer/programmer after approximately 3 seconds. Slowly move the control stick forward (nose down). After 3 seconds, Autotrim should move nose up with TRIM flashing on the remote annunciator and the autopilot computer/programmer after approximately 3 seconds.

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I. Place the Trim Master Switch in the off position.

5. Manual Electric Trim Test

A. Place the Trim Master switch in the 'ON' position.

B. Move the elevator trim switch (hat switch on the control stick) up and down. The autopilot should disengage. The RDY (ready) annunciation will flash, then display a steady indication, and the trim should operate in the commanded direction. (The elevator trim switch will disengage the autopilot only when a pitch mode is engaged.) Re-engage the heading and VS modes, and press the Autopilot disconnect switch (APD). The autopilot should disengage. As before, the RDY indication will flash, then annunciate steady. An audible tone should be heard when the autopilot is disconnected.

6. If Flight Director Equipped

A. Place the Autopilot Master Switch in SELECT FD position. Verify the roll, pitch, and trim servos are disengaged. The steering bar should be in view on the attitude indicator.

B. Engage HDG mode. Move the HDG bug 45 degrees left. The roll steering bar should slowly indicate a left steering command. Repeat the same test for the right side.

C. Engage VS mode. Select 1500 FPM rate of climb. Verify the pitch steering bar moves slowly up. Repeat the same test for the down direction.

D. Place the Autopilot Master Switch in SELECT FD/AP. The servos should re-engage.

E. Place the Trim Master switch in the 'ON' position.

F. Place the Manual Electric Trim Command switch to MOVE FWD or AFT. The autopilot should disconnect.

G. The Manual Electric Trim Command switch will disconnect the autopilot only if there is a Pitch Mode engaged.

7. Autopilot Disconnect Switch

A. The autopilot disconnect switch is a spring loaded rocker switch on the top left side of the pilot's control stick and is normally operated with the thumb of the left hand. Pressing either the top or the bottom portion of the rocker switch will disengage the autopilot. (Note: Operating the elevator trim switch will also disconnect the autopilot.)

8. S-Tec 360 Autopilot Altitude Preselect (Optional Equipment)

A. **General Overview** - The S-Tec 360 Autopilot Altitude Preselect (AAP) enhances use of the S-Tec System 55X Autopilot, but is not a required component of the autopilot system. While the S-Tec 360 has a number of special functions like altitude and decision height alerting, the primary features of the AAP is the ability to preselect an assigned or desired altitude. When the altitude command is sent to the autopilot, the airplane will climb or descend at a preset rate to the preset altitude and thereafter maintain that altitude.

B. It is important to understand that the S-Tec 360 unit does not control the autopilot. Vertical speed and altitude commands for the autopilot are preprogrammed into the AAP unit but are not commanded until the corresponding functions on the autopilot display unit are engaged. The two primary advantages of the AAP unit are first the ability to preprogram altitude and climb settings for later execution, and second the barometric calibration feature discussed in the next paragraph.

C. Altitude information is obtained from the encoder altimeter of the airplane. The AAP unit has a baro calibration feature, which allows the pilot to correct the pressure altitude from the encoder for local pressure variations. Essentially, there are two Kollsman displays in the airplane, the static system altimeter and the S-Tec 360 display unit. If the encoding altimeter is properly calibrated and the pilot routinely updates the altimeter settings of the AAP unit, the autopilot will maintain the correct altitude above sea level.

D. The method of altitude hold is different when using the altitude hold feature of the S-Tec System 55X autopilot. When the altitude hold function is engaged on the autopilot without input from the AAP, the autopilot senses the current atmospheric pressure at the instant altitude hold is engaged, and thereafter, maneuvers the airplane to maintain that pressure. If the airplane flies into a low pressure area, the autopilot will command the airplane to descend. In this situation, the pilot must use the vertical speed control on the autopilot to compensate for the normal en route and diurnal changes in atmospheric pressure.

E. The S-Tec System 55X Autopilot and 360 AAP both have altitude (ALT) and vertical speed (VS) functions. During the study of the pilot's guide on the following pages, it is important to distinguish which unit the ALT or VS discussion is applicable. If the term **autopilot** is used, this refers to the buttons on the S-Tec System 55X autopilot. If the terms **Selector**, **Selector Unit** , or **AAP** are used, it refers to the S-Tec 360 unit shown in Figure 5. In general, the AAP is used to **select**, **input**, or make altitude and/or vertical speed commands **active**. The autopilot, when **engaged**, executes the commands. Understanding the meaning of a few terms that are used in the S-Tec Pilot's Guide is helpful.

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(1) The term **selected altitude** refers to the altitude selected or preset in the AAP Unit for the autopilot to intercept and/or maintain.

(2) The **operate mode** is the normal state of the S-Tec 360 system. In this mode, preset ALT and VS commands are executed when the pilot engages the corresponding commands on the autopilot. The **Entry Mode** is used to input altitude, or other related settings.

F. **S-Tec 360 System Description** - The S-Tec Liquid Crystal Display (LCD) Altitude and Vertical Speed Selector are designed for use with S-Tec 55X Autopilot. The system is used to pre-select an altitude, a rate of climb, or a rate of descent, which is then performed by the autopilot. In addition, the selector provides an altitude alert mode, a decision height (DH) alert mode, an altitude readout from the encoder, barometric calibration, and a manual mode. This supplemental section provides information on the features and functions of the system and operating instructions for its proper use. A labeled drawing of the display unit is shown in Figure 4.

G. The LCD Altitude and Vertical Speed Selectors are integrated into a single panel-mounted unit, which contains the display, the operating switches, and the computer electronics. The system is designed to interface with the autopilot and the encoder, which provides a standard altitude output in increments of 100 feet.

9. S-TEC 360 Theory of Operation

A. **Encoder Interface** - The Altitude Selector Computer reads and decodes raw altitude information from the altitude encoder. The uncorrected altitude information is adjusted by changing the setting in the barometric calibration. When the information from the selected altitude matches the corrected decoded altitude information from the encoder, the altitude selector computer signals the autopilot to electrically engage the altitude hold mode of the autopilot.

B. The vertical speed selector provides an electrical output to the autopilot pitch flight guidance computer that is proportional to the amplitude and polarity (direction) of the vertical speed. For example, +500 FPM climb VS would produce a plus (+) voltage in an amount representing 500 FPM. The autopilot compares the existing vertical speed with the selected vertical speed and maneuvers the airplane to match these signals.

C. **System Diagram** - Figure 5 contains a block display of how the various avionics components function with the S-Tec 360 system.

10. S-TEC 360 Ground Testing

A. **Self-Test** - When power is applied to the system, an internal self-test of the computer electronic elements, the display, annunciators, and the altitude alerter audio tone is conducted. Successful test conclusion is indicated by a display of 29.9 in the selector unit. The self-test cycle does not check the encoder for proper operation. However, the preflight check procedures outlined in Chapter 34 provide a method to determine proper operation of the encoder.

B. **Power Switches** - There are no separate on/off switches for the S-Tec 360 Autopilot Altitude Preselect. The S-Tec 360 is powered when the system and avionics master switches are on. In addition, the Trans call SSD 120 Blind Encoder/Digitizer does not have a power switch. The encoder is turned on by the avionics master switch. There are certain operational limitations of the encoder which are discussed in Chapter 34.

C. **Preflight Inspections** - The following preflight procedure provides an operational test of the entire system, including the encoder, the altitude selector, and the autopilot. A successful test is indicated by the autopilot switching from VS Mode to ALT Hold Mode as the selected altitude is matched to field elevation.

(1) System, Autopilot, and Avionics Master Switches - ON

(2) Autopilot ALT Select Button - ON

(3) Altimeter - Set to local altimeter setting or field elevation, as appropriate.

(4) Altitude Selector -

(a) Observe that the self-test cycle is complete. When first powered, the system will display all annunciations for approximately five seconds, ending with an audible tone. Thereafter, the system will display a barometric setting of 29.9 with the barometric annunciator flashing.

(b) Rotate the selector input knob to set barometric setting to the nearest 0.1 in. of Hg (for millibars push on barometric switch).

(c) Push the ALT Switch to display ALT SEL. With the SEL flashing, rotate the selector knob to input an altitude 300 to 400 ft. lower or higher than the indicated altitude.

(d) Push the VS Switch to activate VS Selector, rotate the selector switch knob to input desired climb (+) or descent (-) speed.

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(e) Push the ALT Switch to access the altitude set mode - ALT SEL will be annunciated.

(5) Autopilot

(a) Engage the HDG Mode

(b) Simultaneously depress VS and ALT switches on the autopilot, and observe that the VS and ALT annunciators both illuminate.

(c) Rotate the altitude selector knob on the AAP unit and change the selected altitude so that it matches the field elevation. The VS annunciation on the autopilot should extinguish when the setting on the altitude selector is within 100 ft. of the altitude indicated on the altimeter. Extinguishing of the VS annunciation with the ALT remaining on indicates the altitude hold mode is engaged. If altitude engagement does not occur within 100 ft. of indicated altitude, readjust the barometric calibration.

(6) Disengage Autopilot - Adjust the altitude selector to the desired cruise altitude and set the vertical speed to an appropriate climb setting for use after takeoff.

(7) Autopilot Preflight - Conduct autopilot preflight per the instructions in S-Tec System 55X Pilot's Guide.

11. S-TEC Autopilot Integration With Avidyne Flightmax Entegra Primary Flight Display (PFD)

A. Whenever the S-Tec System 55X Autopilot or the Avidyne PFD are replaced, perform setup procedures indicated in Chapter 34.

12. Instructions For Continued Airworthiness (S-Tec Corp.)

A. **Description** - The S-Tec flight control system includes the following items:

(1) Roll servo

(2) Pitch servo

(3) Trim servo (elevator)

B. **Installed Equipment** - Panel or remotely mounted pitch/roll/yaw computers include the following:

(1) Panel mounted controllers, indicators, switches, and breakers

(2) Barometric pressure transducer/static source

C. **Servos** - Servo installations utilize aluminum brackets to secure the servos to the airframe. Attachment to the aircraft primary flight controls and trim systems is accomplished through cable and push rod assemblies.

D. **Controls Operation Information** - Operation of the autopilot and other systems is described in the FAA-Approved *S-Tec Flight Manual Supplement* (P/N 891830S). Specialized controls, annunciation, operation and interpretation are covered in this required document and in the *S-Tec Pilot's Operating Handbook* (P/N 87109) that supplements the approved AFMS. This information is also contained in the POH/AFM.

E. **Servicing Information** - Approved S-Tec dealers holding the appropriate FAA certification must accomplish all servicing of items. Locations and access to the components installed are described and depicted in the installation drawings and Installation Instructions ST- 852. Removal and replacement of components should be determined by functional checks indicated in S-Tec *AFM Supplement* 891830S, *Pilot's Operating Handbook* (P/N87109) and the 'Post Installation Ground Checks' document, number 05110.

F. **Maintenance Instructions** - Condition and/or airworthiness inspections required under FAR Part 43, or other FAA approved programs, should include several items regarding the S-TEC autopilot system installed in the aircraft:

(1) Visual Inspections

(a) Flight Control Components - Check control cables, pulleys, servos, and associated equipment for condition, attachment, clearance, and proper operation. Replace cables that have broken strands or evidence of corrosion. Check cables for proper tension.

(b) Electrical Wiring and Components - Check for security, chafing, damage and attachment. For replacement of any of the autopilot wiring, cables or associated components, see the *S-Tec Master Drawing List* 921139 for the appropriate drawing or document associated with the action to be performed.

(2) System Operation Check after Maintenance - Refer to the Preflight Self-Test Procedure (S-Tec 55X Autopilot), Checking The Autotrim System, Manual Electric Trim Test, and If Flight Director Equipped sections.

13. Troubleshooting Information

A. Troubleshooting this equipment should only be accomplished by authorized S-TEC Dealers, holding the appropriate FAA

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certification, with required test equipment and service data. System function should be determined through functional checks indicated in the *S-Tec AFM Supplement* 891830S, *Pilot's Operating Handbook* (P/N87109) and the 'Post Installation Ground Checks' document, number 05110.

14. Special Issues

A. **Removal and Replacement Information** - All components can be removed with common tools and practices.

B. **Diagrams/Special Inspections** - Refer to *S-Tec Master Drawing List* 921139 for all diagrams. There are no special inspection requirements.

C. **Application of Protective Treatments** - Servos and transducers should be removed prior to application of corrosion (or other) treatments. Panel-mounted components should not be exposed to these treatments.

D. **Structural Fasteners** - Refer to S-Tec Document No. 761017, Installation, S-Tec System 55/55X Autopilot

E. **Special Tools**

(1) **Crimp Tools**

A crimp tool and positioner/locator meeting MIL Specification M22520/1-01 or equivalent are required to ensure consistent, reliable crimp contact connections for the rear d-sub connectors. These tools are available from ITT Cannon or other vendors:

Insertion Tool:	ITT Part No. 274-7048-000 (Desc. CIET-22D-01)
Crimp Tool (HD):	ITT Part No. 995-0001-584 (Desc. M22520/2-01)
Locator Tool:	ITT Part No. 995-0001-244 (Desc. M22520/2-07)
Locator Tool (HD):	ITT Part No. 995-0001-739 (Desc, M22520/2-06)
Locator Tool (HD):	Desc. M22520/2-09

(2) **S-TEC Special Tool**

Clutch Adjustment Spanner	Part Number 6622	Drawing No. 6622 and 1
Clutch Adjustment Spanner	Part Number 66228	Drawing No. 66228 and 1N/A

F. **Normal Category Aircraft** - Electrical load must be maintained within 80 percent of generator capacity for systems installed (AC 43.13-1B). Electrical loads are described in the Component Weights and Current Drain section of the S-Tec Installation Instructions ST-852.

G. **Overhaul Time Limitations - N/A**

H. **Airworthiness Limitations** - Limitations are listed in the Limitations Section of AFM Supplement(s).

I. **Revisions** - The S-Tec Service Letter program will be utilized to inform aircraft operators of significant changes to this ICA. Contact S-Tec Corporation at 1-800-USA-STEC.

15. Autopilot Roll Servo

A. General - The autopilot roll servo is located under the flap drive bracket. It is accessed through the wing controls access panel located under the fuselage at the aft end of the wing saddle, near the wing trailing edge. The servo is attached to the bracket at an angle. This angle aligns the servo drive capstan grooves with the bridle cable. The bridle cable is attached to the aileron interconnect pushrod with two saddle assemblies. Refer to Section 27-10-00 for a discussion of the aileron system. Refer to Figure 6 for a picture of the aileron roll servo.

B. **Removal** - The roll servo bridle cable must be detached to remove the roll servo. Remove the MS-21256-1 locking clips from the barrel of the turnbuckle. Detach this cable from both ends by loosening the turnbuckle and removing the clevis pins. Retain the MS20392-2C-11 clevis pin and NAS1149F0332P washer for reuse. A new MS24665-134 cotter pin will be required for reinstallation.

(1) **Bridle Cable Removal and Inspection** - The bridle cable does not have to be removed from the servo drive capstan for most maintenance procedures. Check cable for signs of wear or corrosion. Replace cable assembly if required. Autopilot installation bulletins must be followed for this process.

(2) **Unplug Roll Servo** - Cut the nylon tie used to secure the electrical wiring. Unplug the autopilot roll servo electrical connection at the connector.

(3) **Servo Removal** - The servo is attached to the bracket with four AN3-4A bolts and MS20365-1032 nuts with NAS1149F0332P washers. These are located on the corners of the servo attach flange. Three of these bolts can be

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seen in Figure 6. When the four AN3-4A bolts and MS20365-1032 nuts with NAS1149F0332P washers are removed, the servo can be removed.

C. **Servo Installation** - Installation is the reverse of removal.

(1) Adjust bridle cable tension to proper 30±2 lbs. value.

(2) The four AN3-4A bolts and MS20365-1032 nuts are torqued 30 to 36 in.-lbs.

(3) Be sure to check cable tension and installation per autopilot installation requirements.

16. Autopilot Pitch Servo

A. **General** - The autopilot pitch servo is located under the baggage bulkhead. It is accessed through the baggage bulkhead access panel located under the upholstery on the hat rack. The servo is attached to a composite bracket mounted at an angle to the fuselage centerline. This angle aligns the servo drive capstan grooves with the bridle cable. The bridle cable is attached to the elevator final drive pushrod with two saddle assemblies. Refer to Figure 7.

B. **Removal** - The pitch servo bridle cable must be detached to remove the roll servo. Remove the MS-21256-1 locking clips from the barrel of the turnbuckle. Detach this cable from both ends by loosening the turnbuckle and removing the clevis pins. Retain the MS20392-2C-11 clevis pin and NAS1149F0332P washer for reuse. A new MS24665- 134 cotter pin will be required for reinstallation.

(1) **Bridle Cable Removal and Inspection** - The bridle cable does not have to be removed from the servo drive capstan for most work. Check cable for signs of wear or corrosion. Replace cable assembly if required. Autopilot installation bulletins must be followed for this process.

(2) **Unplug Pitch Servo** - Cut the nylon tie used to secure the electrical wiring. Unplug the autopilot pitch servo electrical connection at the connector.

(3) **Servo Removal** - The servo is attached to the bracket with two MS24694-S50 flat head screw on the upper two mounting locations and two AN3-4A bolts with NAS1149F0363P washers on the lower two mounting locations. Each screw and bolt is installed into a clip nut (ESNA RM52LHA4972-8-02) attached to the servo mounting flange. These mounting bolts are located on the corners of the servo attach flange, and the bolts can be seen in Figure 8. Remove the servo.

C. **Servo Installation** - Installation is the reverse of the removal.

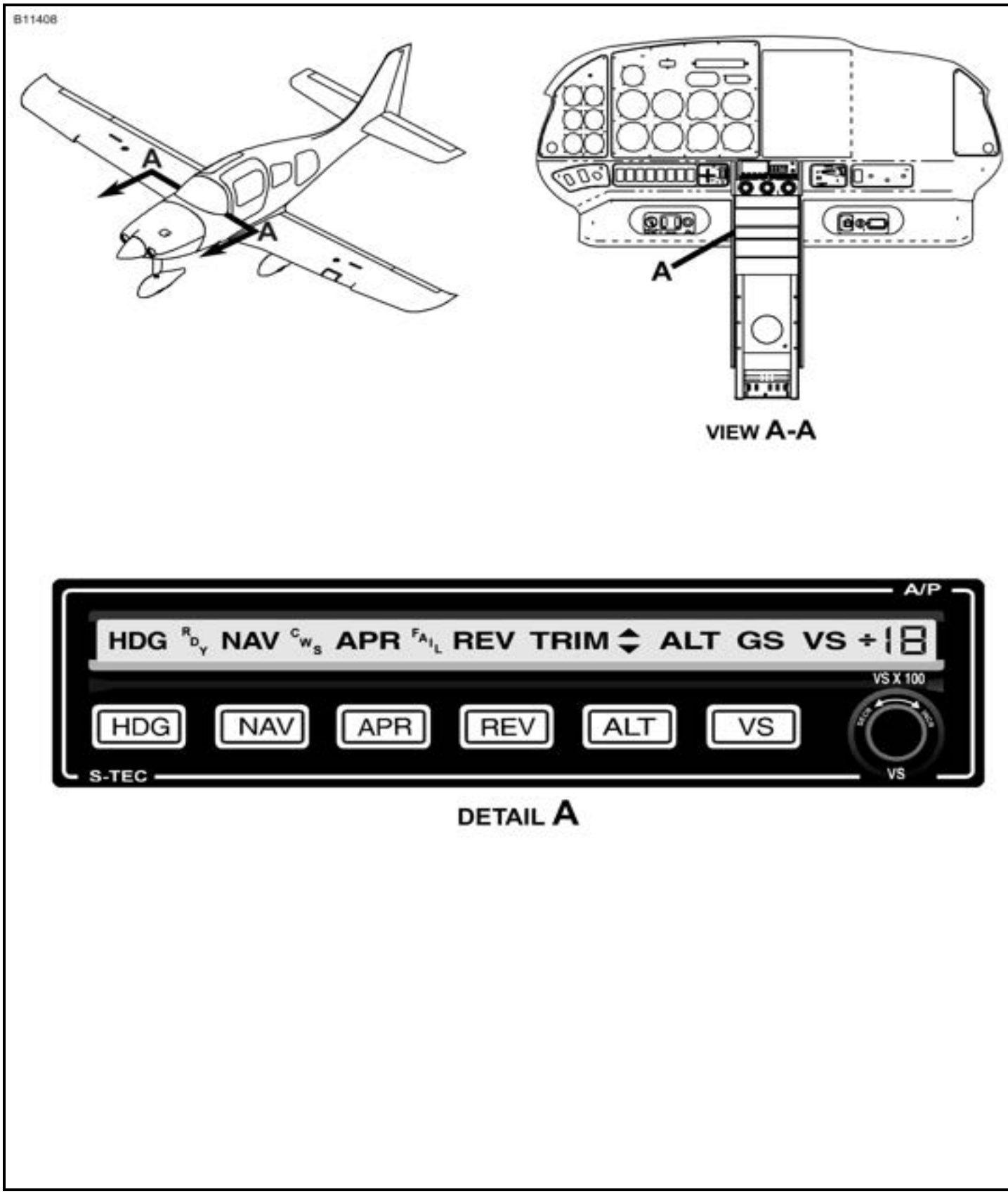
(1) Adjust bridle cable tension to proper 30±2 lbs. value.

(2) Torque all bolts 30 to 36 in.-lbs.

(3) Be sure to check cable tension and installation per autopilot installation requirements.

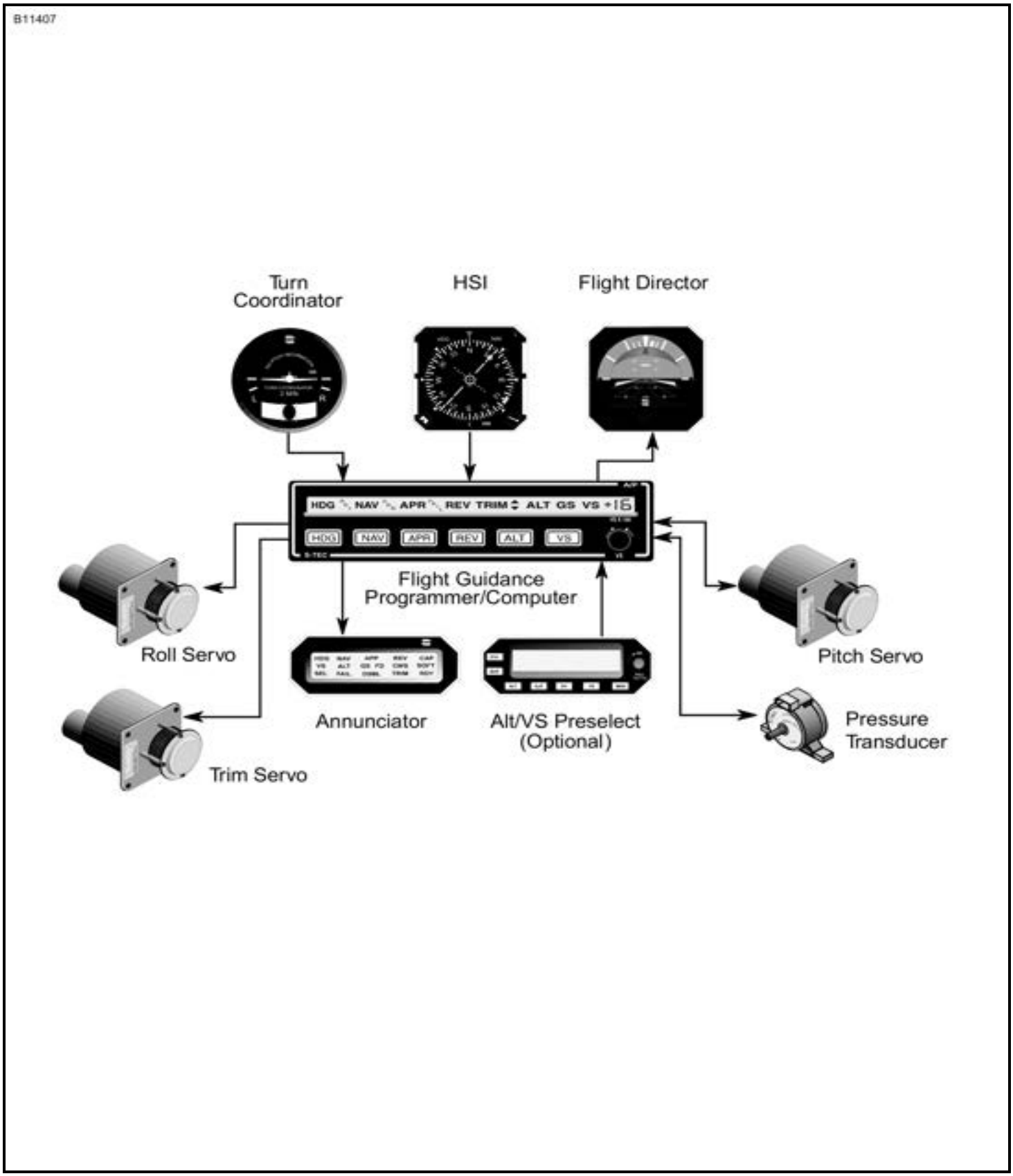
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Figure 1 : Sheet 1 : S-Tec 55X Control Panel



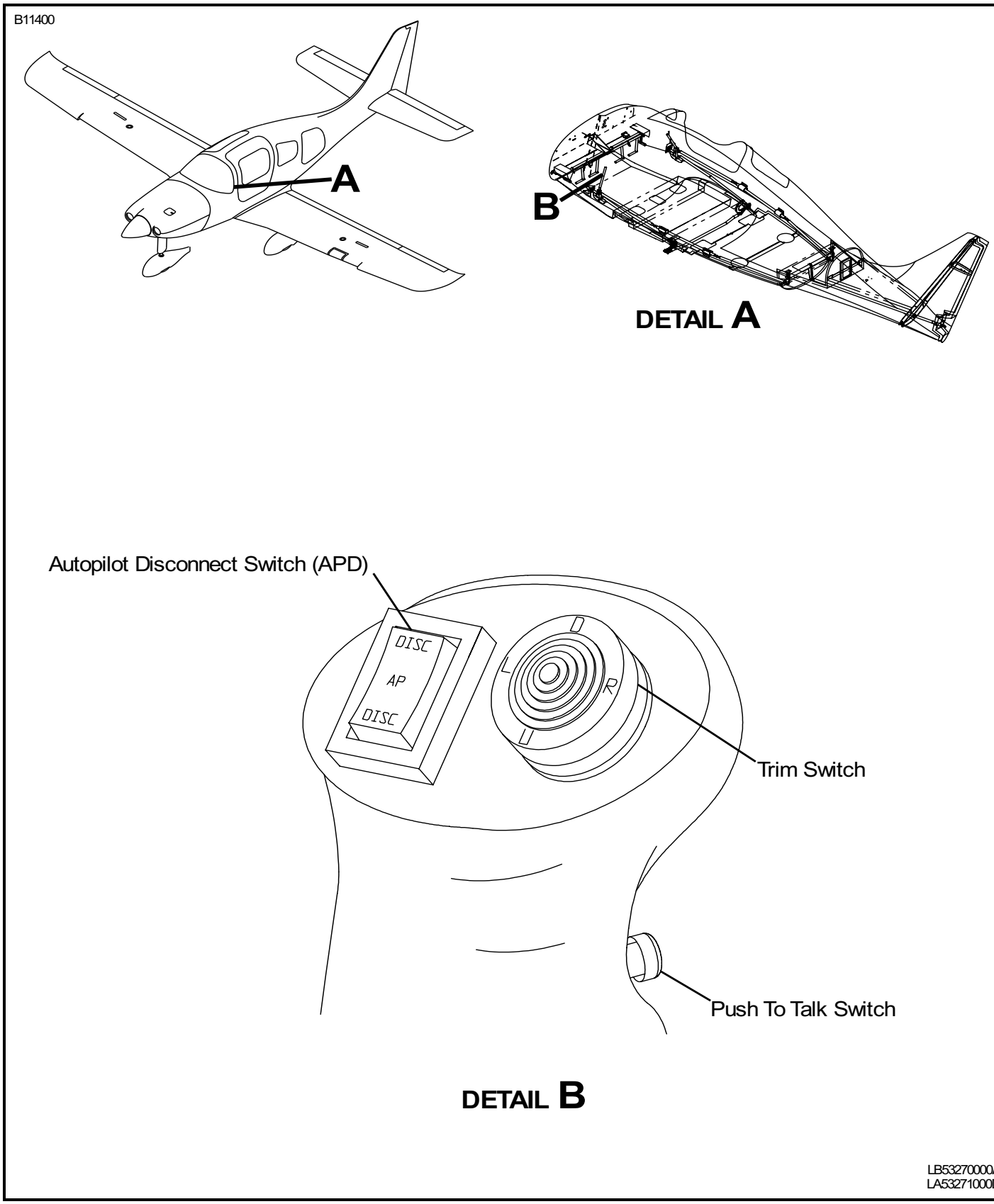
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Figure 2 : Sheet 1 : S-Tec 55X Autopilot System



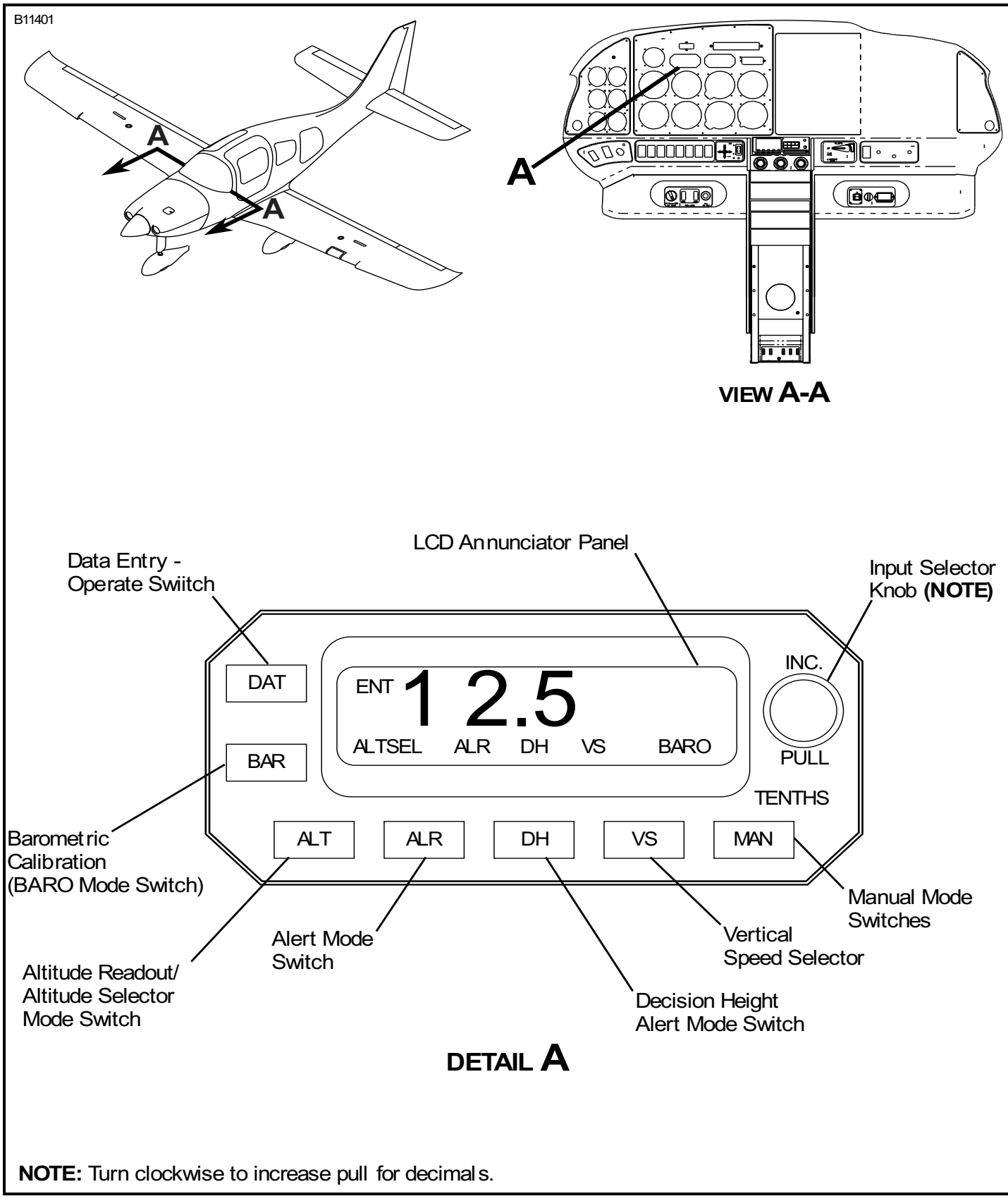
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Figure 3 : Sheet 1 : Autopilot Switches on the Control Stick



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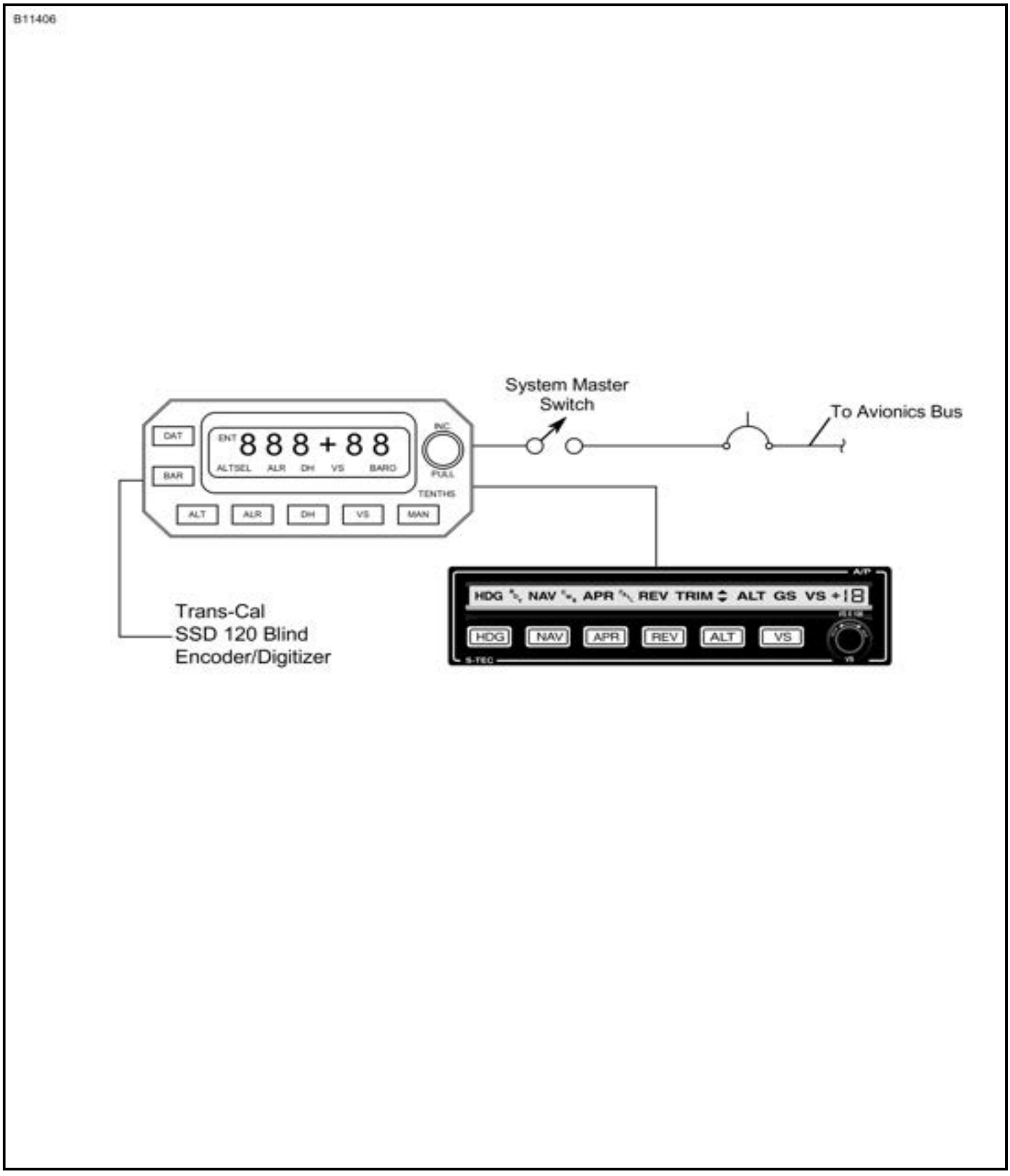
Figure 4 : Sheet 1 : S-Tec 360 AAP



NOTE: Turn clockwise to increase pull for decimals.

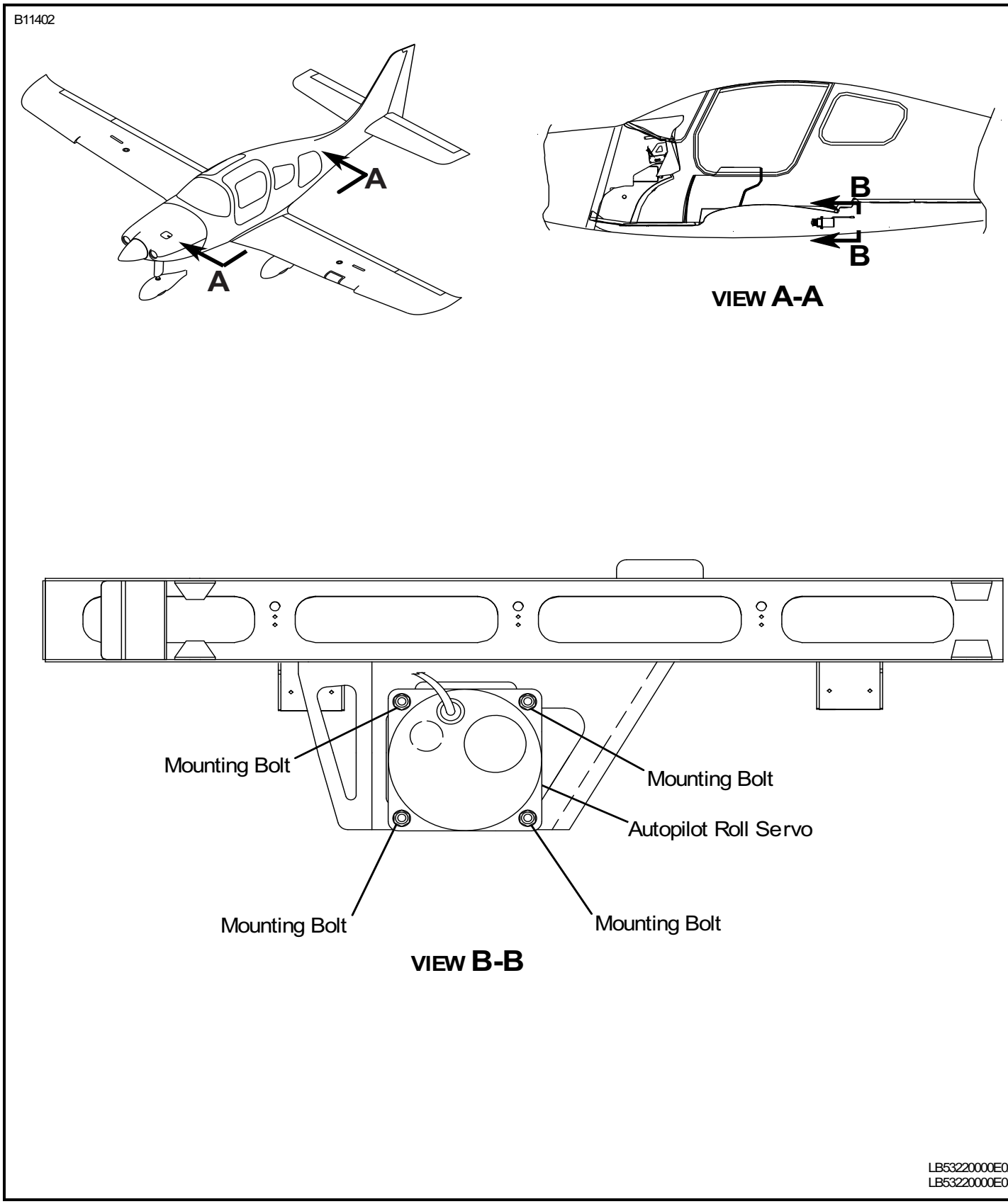
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Figure 5 : Sheet 1 : Block Diagram of S-Tec 360 System



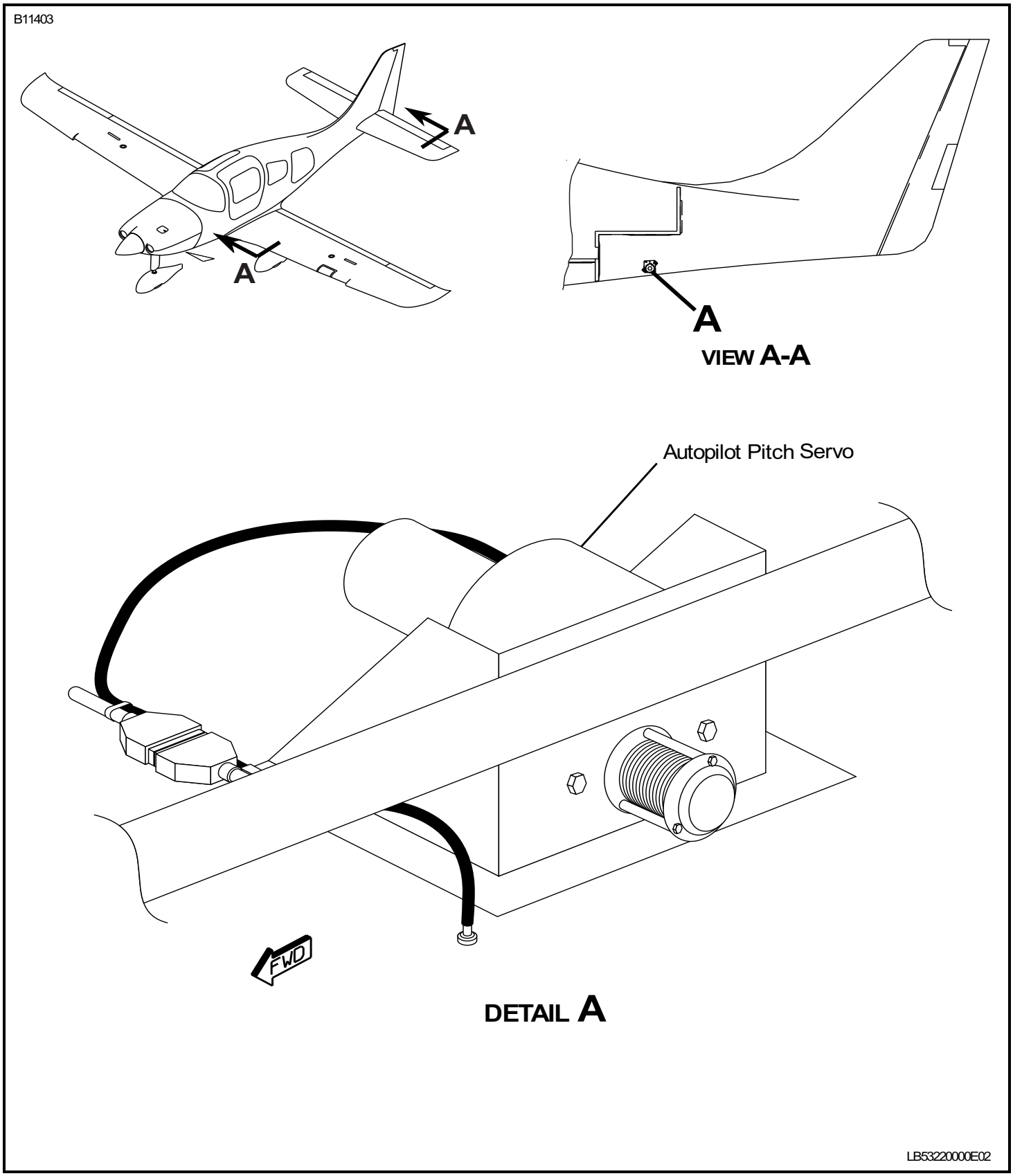
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Figure 6 : Sheet 1 : Autopilot Roll Servo



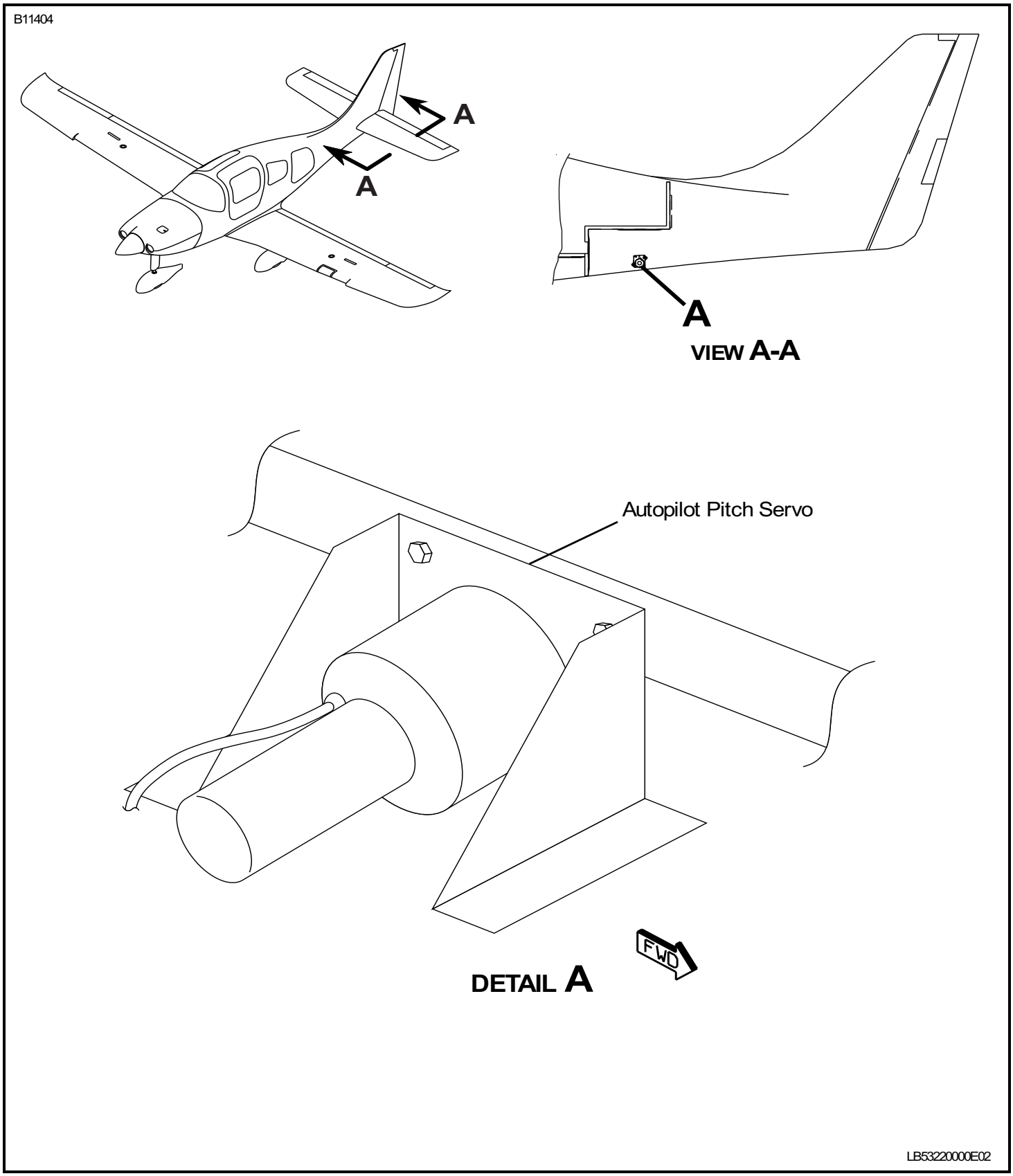
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Figure 7 : Sheet 1 : Autopilot Pitch Servo



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Figure 8 : Sheet 1 : Autopilot Pitch Servo



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GARMIN GFC 700 AUTOPILOT - MAINTENANCE PRACTICES

1. General

A. The GFC 700 is a two-axis fail-safe digital flight control system. There is no single LRU with the name GFC 700; rather, GFC 700 refers to an integrated autopilot and flight director system, with functions provided by multiple G1000 LRUs and servos. Refer to Figure 201. The following functions are provided by the GFC 700 in this installation:

- Flight Director
- Autopilot
- Manual Electric Trim

B. **Flight Director** - The Flight Director operates within the no. 1 GIA 63/GIA 63 W and uses data from the G1000 system, including air data, attitude, and navigation data, to calculate commands for display to the pilot and for the Autopilot. Flight Director command bars and mode annunciations are sent to the PFD through a high-speed Ethernet connection for display to the pilot. The Flight Director operates independently of the Autopilot, and allows the pilot to hand-fly the command bars, if desired.

C. **Autopilot** - The Autopilot operates within the two GSA 81 servos. Flight Director data is processed within the two servos and turned into aircraft flight control surface commands. The autopilot cannot operate unless the Flight Director is engaged.
The following is a summary of the autopilot functions provided by each LRU:

- GDU 1040 PFD - Displays the Flight Director command bars and the autopilot mode annunciations.
- GDU 1042/GDU 1044 MFD - Provides controls for the autopilot functions.
- GIA 63/GIA 63W - Performs the calculations to display the Flight Director command bars on the PFD and sends control movement commands to the GSA 81 autopilot servos.
- GSA 81 - Actuates the control surfaces based on commands received from the GIA 63/GIA 63W.
- GSA 82 Trim Adapter - Operates the elevator trim tab.

D. **Configuration and Testing** - After removal and replacement, LRUs must be configured and tested per the *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03.

CAUTION: When removing and/or replacing any G1000 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.
Before removing any G1000 LRU, it is required that the technician verify the LRU software part number and version against the Required Equipment List RB011002.

2. Autopilot Disconnect Switch

A. The autopilot disconnect switch is a push button switch on the top left side of the pilot's control stick and the top right side of the co-pilot's control stick, and is normally operated with the thumb. Refer to Figure 202. Pressing the switch will disengage the autopilot and trim system. (Note: Operating the elevator trim switch in the overhead will also disconnect the autopilot.)

3. Troubleshooting

A. Troubleshoot the GFC 700 system per the *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03.

4. GSA 81 Autopilot Roll Servo

A. Refer to Figure 203.
The GSA 81 is mated to the GSM 85 Servo Mount to form a single servo unit
The design of the servo assembly allows the servo portion (GSA 81) to be removed from the capstan (GSM 85) without the need to de-rig the aircraft control cables. The roll servo is located beneath the rear passenger seats.

(1) **Removal of the GSA 81 Autopilot Roll Servo**

(a) Ensure aircraft power is off. Unplug any auxiliary power supplies.

(b) On the underside of the wings at the fuselage remove the Flap and Aileron Push-Rod Access Panel.

(c) Disconnect servo connector P680.

(d) Remove the two bolts holding the servo to the servo mount. Refer to Figure 204.

(e) Remove the unit.

(f) Remove excess grease build-up from the single servo output gear using a lint-free cloth.

NOTE: It is not necessary to remove all of the grease from the output gear, only the excess

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grease. DO NOT USE SOLVENTS TO CLEAN THE OUTPUT GEAR!

(g) Using a brush or other applicator, apply a thin coat of grease to the servo output gear. Use Aeroshell 33 MS (preferred) or Aeroshell 17 grease.

(2) **Installation of the GSA 81 Autopilot Roll Servo**

(a) Set unit in place.

(b) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

(c) Install the two bolts holding the servo to the servo mount.

(d) Connect servo connector P680.

(e) Configure and test the GSA 81 per the *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03.

(f) Operate all controls through full travel and ensure that no binding or restriction occurs.

(g) Reinstall the Flap and Aileron Push-Rod Access Panel.

5. GSM 85 Autopilot Roll Servo Mount

A. **Removal of the GSM 85 Autopilot Roll Servo Mount**

(1) Remove the GSA 81 servo per the GSA 81 Autopilot Roll Servo section.

(2) Loosen the bridle cable turnbuckle and remove the clevis pins.

(3) Remove the four screws holding the mount to the servo bracket.

(4) Remove the unit with the bridle cable.

B. **Installation of the GSM 85 Autopilot Roll Servo Mount**

(1) Prior to installation, adjust and verify the slip clutch torque is set to 24±4 in-lbs. for the roll servo mount or 50±7 in-lbs. for the pitch servo mount.

(a) Place the servo mount onto the Garmin Servo Adjustment Fixture (011-01085-XX) and secure with toggle clamps. Refer to Figure 205.

(b) Install the cable, supplied with the fixture, between the capstan under test and the fixture capstan.

(c) Remove slack in the cable with the tension adjust knob.

(d) Tighten the thumbscrews on the fixture capstan.

(e) Attach a socket to a calibrated torque wrench of the appropriate range and place the assembly on top of the fixture capstan. Adjust the wrench support as required to support the torque wrench in a level, horizontal position. Ensure the torque wrench is set to ZERO while the motor and solenoid are OFF.

(f) Connect a 24 VDC power supply to the fixture.

(g) Place the solenoid switch to ON and move the direction switch to the CW or CCW position. Allow the capstan to rotate about one revolution in each direction. Adjust the castle nut on the servo mount until the torque value indicates 24±4 in-lbs. for the roll servo mount or 50±7 in-lbs. for the pitch servo mount.

(h) Switch the motor direction back and forth as necessary to allow adjustment until the required torque range is achieved in both directions. Ensure the castle nut aligns with one of the holes in the shaft.

(i) Observe the slip clutch torque reading for at least two full revolutions in each direction.

(j) Ensure the minimum and maximum torque readings are within tolerance of 24±4 in-lbs. for the roll servo mount or 50±7 in-lbs. for the pitch servo mount.

(k) When finished, remove the servo mount from the servo adjustment fixture.

(2) Install a new cotter pin through the capstan nut and servo mount shaft.

(3) Set the unit in place.

(4) Install the four screws holding the mount to the servo bracket.

(5) Install the bridle cable. Wrap the turnbuckle end approximately 270 degrees over the capstan. Wrap the fork end approximately 450 degrees over the capstan. Set the cable tension to 25 to 60 lbs. The bridle cable should make two complete revolutions around the capstan and the lock ball should be mated in its seat. System rigging to be accomplished with controls neutral.

(6) Safety wire the turnbuckle using a single wrap method with 0.040 safety wire or a double wrap method using 0.030 safety wire.

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(7) Install the GSA 81 servo per the GSA 81 Autopilot Roll Servo section.

6. GSA 81 Autopilot Pitch Servo

A. Refer to Figure 206.
The GSA 81 is mated to the GSM 85 Servo Mount to form a single servo unit.
The design of the servo assembly allows the servo portion (GSA 81) to be removed from the capstan (GSM 85) without the need to de-rig the aircraft control cables. The pitch servo is located beneath the aft baggage area.

(1) **Removal of the GSA 81 Autopilot Pitch Servo**

(a) Ensure aircraft power is off. Unplug any auxiliary power supplies.

(b) Remove the baggage bulkhead access panel located under the hat rack per Chapter 25.

(c) Disconnect servo connector P681.

(d) Remove the two bolts holding the servo to the servo mount. Refer to Figure 206.

(e) Remove the unit.

(f) Remove excess grease build-up from the single servo output gear using a lint-free cloth.

NOTE: It is not necessary to remove all of the grease from the output gear, only the excess grease. D0 NOT USE SOLVENTS TO CLEAN THE OUTPUT GEAR!

(g) Using a brush or other applicator, apply a thin coat of grease to the servo output gear. Use Aeroshell 33 MS (preferred) or Aeroshell 17 grease.

(2) **Installation of the GSA 81 Autopilot Pitch Servo**

(a) Set unit in place.

(b) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

(c) Install the two bolts holding the servo to the servo mount.

(d) Connect servo connector P681.

(e) Configure and test the GSA 81 per the *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03.

(f) Operate all controls through full travel and ensure that no binding or restriction occurs.

(g) Reinstall the baggage bulkhead access panel and the hat rack per Chapter 25.

7. GSM 85 Autopilot Pitch Servo Mount

A. **Removal of the GSM 85 Autopilot Pitch Servo Mount**

(1) Remove the GSA 81 servo per the GSA 81 Autopilot Pitch Servo section.

(2) Remove the two clamps holding the bridle cable to the main control cable.

(3) Remove the four screws, two washers, and two self-locking nuts holding the mount to the servo bracket.

(4) Remove the unit with the bridle cable.

(5) A tapered shim is attached to the face of the servo mount by two 10-32 x 17/32 screws and two 10-32 x 25/32 screws. Remove shim and retain for installation.

B. **Installation of the GSM 85 Autopilot Pitch Servo Mount**

(1) Prior to installation, adjust and verify the slip clutch torque is set to 50±7 in-lbs. per the GSM 85 Autopilot Roll Servo Mount Removal paragraph.

(2) Install tapered shim.

(3) Set the unit in place.

(4) Install the four screws, two washers, and two self-locking nuts holding the mount to the servo bracket. Apply Loctite 242 to the two screws without washers and nuts.

(5) Install the bridle cable. Wrap the turnbuckle end approximately 270 degrees over the capstan. Wrap the fork end approximately 450 degrees over the capstan. Set the cable tension to 25 to 60 lbs. The bridle cable should make two complete revolutions around the capstan and the lock ball should be mated in its seat. System rigging is to be accomplished with controls neutral.

(6) Install the GSA 81 servo per the GSA 81 Autopilot Pitch Servo section.

8. GTA 82 Trim Adapter

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A. The Garmin GTA 82 Trim Adapter is a remote mounted device that is used to allow the GFC 700 to drive a pitch trim actuator provided by the airframe manufacturer. The GTA 82 adapter is located under the instrument panel on the pilot side beneath the standby instruments. The trim adapter is powered by the Essential Bus and interfaces with two GIA 63/GIA 63W Integrated Avionics units through serial communication on separate RS-485 ports. Refer to Figure 207.

B. **Removal of the GTA 82 Trim Adapter:**

(1) Remove four screws securing the GTA 82 to the panel. The GTA 82 is located under the instrument panel on the pilot side beneath the standby instruments.

(2) Disconnect connector.

(3) Remove unit.

C. **Replacement of the GTA 82 Trim Adapter:**

(1) Set unit in place.

(2) Install four screws to secure unit to panel.

(3) Connect connector.

(4) After installation of unit, reload the software and the autopilot gain files.

9. Inspection

A. **Annual**

(1) Move the flight controls through their full range of motion. Check for any signs of binding or resistance.

(2) Engage the autopilot and check the pitch, roll, and trim servos for normal operation.

(3) Press the autopilot disconnect switch on the pilot control stick and verify that the autopilot disengages and a disconnect tone is played through the aircraft speaker. Repeat for the copilot stick.

(4) Visually inspect each Servo Mount (GSM 85), moving the control surface through its full range of motion. Ensure there is no binding in the control or bridle cables. Check the cables for fraying or corrosion, and check the mount brackets for cracks or wear.

(5) Check that bridle cable tension is 25 to 60 lbs., and slip clutch settings for each capstan are in accordance with GSM 85 Autopilot Roll Servo Mount Removal.

(6) Perform servo checks per the *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03.

B. **1000 Hour** - Perform a visual inspection every 1000 airframe hours or 12 months, whichever occurs first, in conjunction with the next scheduled inspection and check for corrosion, damage, or other defects for each of the items listed below. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

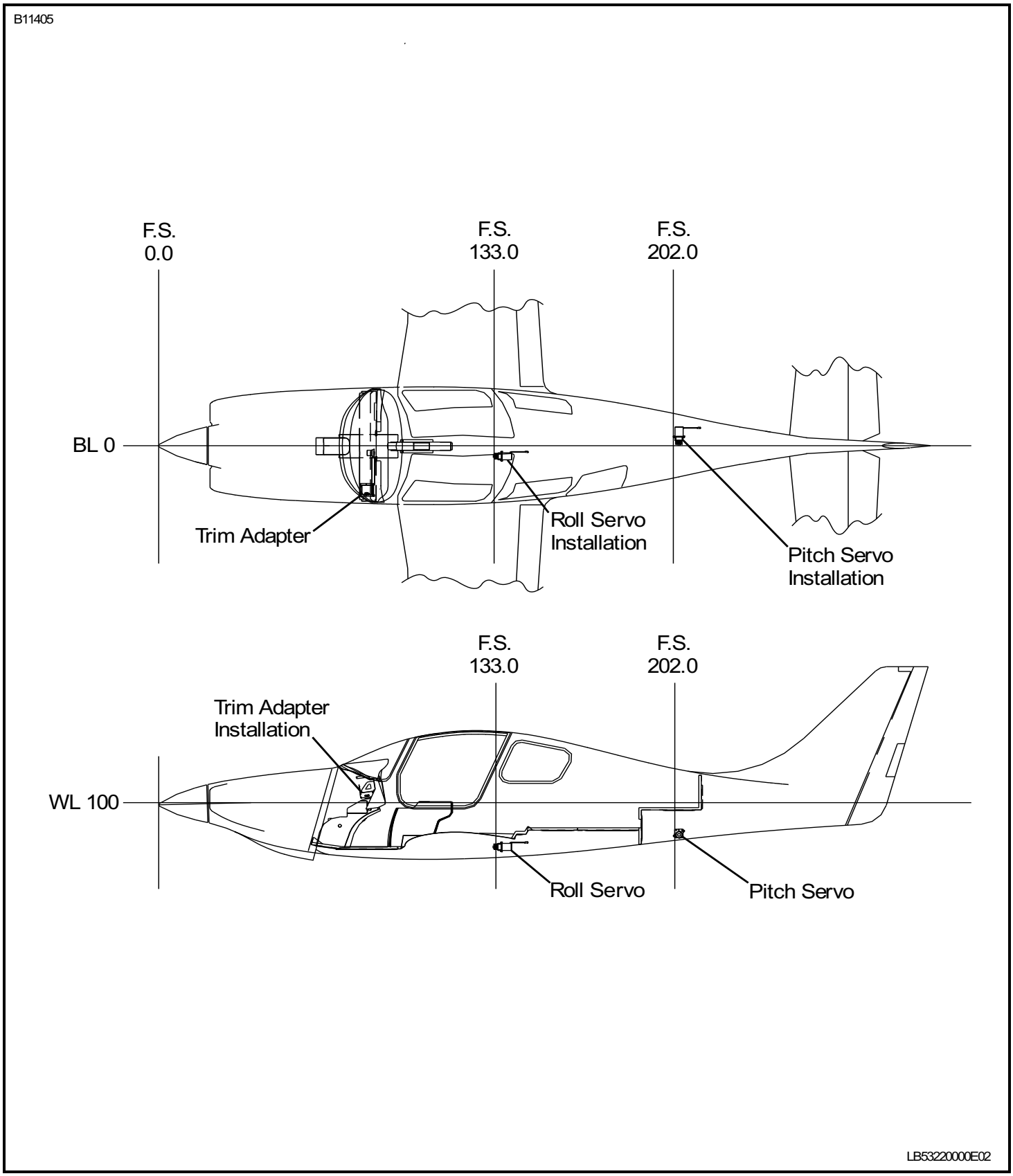
(1) **GSA 81 Servo Actuator** - Separate each GSA 81 from the GSM 85 Servo Mount (pitch, roll). Clean and re-grease the output gear every 1000 hours or 3 years, whichever occurs first, in accordance with the *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03. Check connector and wiring.

(2) **GSM 85 Servo Mount** - Inspect the GSM 85 Mounts. Look for excessive wear in the capstan, and bridle cables.

(3) **GTA 82 Trim Adapter** - Inspect the GTA 82 unit and connector for corrosion or other defects.

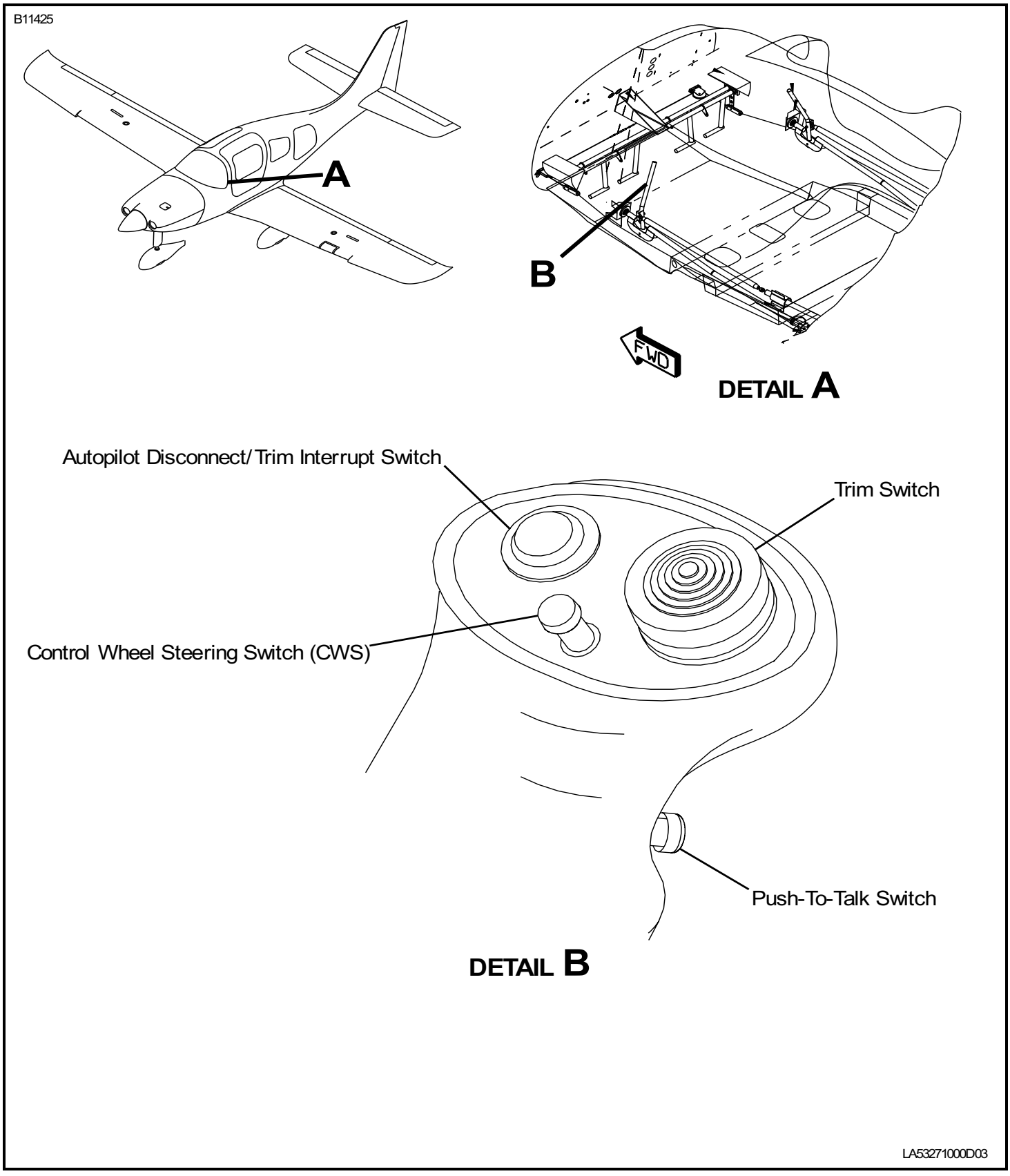
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Figure 201 : Sheet 1 : GFC 700 Autopilot



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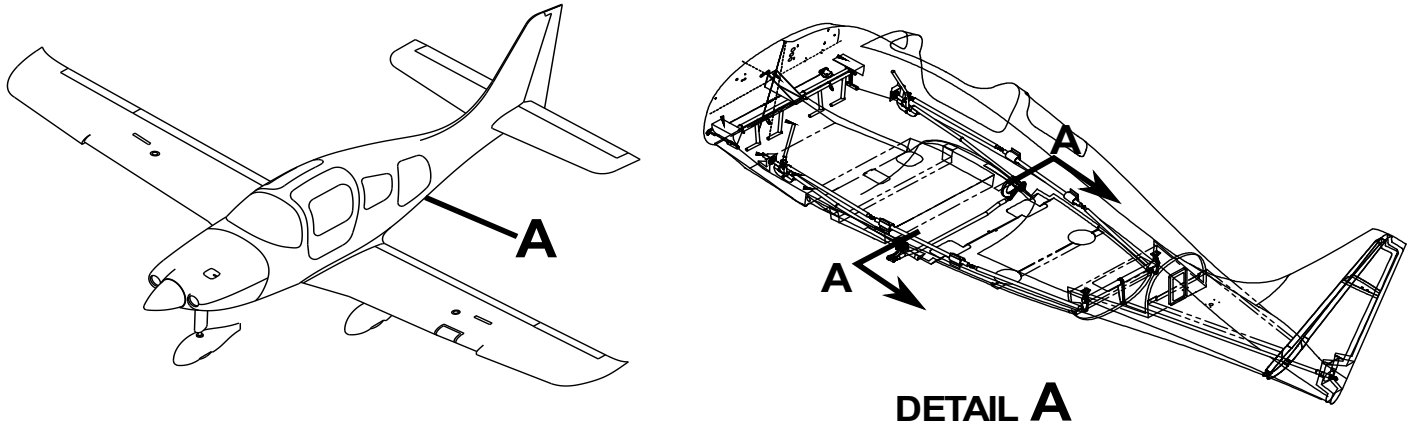
Figure 202 : Sheet 1 : Autopilot Switch on the Control Stick



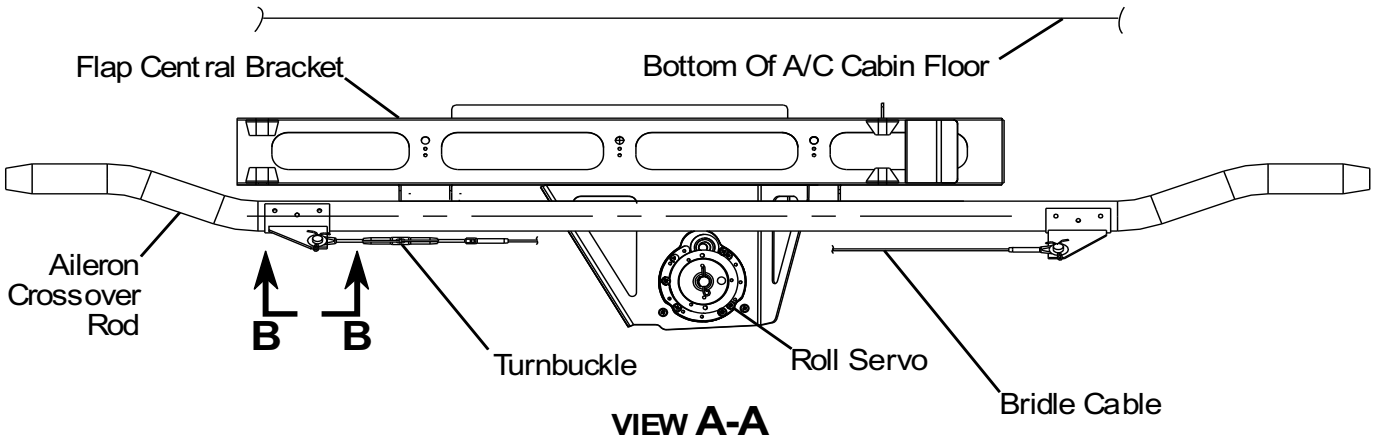
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Figure 203 : Sheet 1 : Roll Servo

B11426

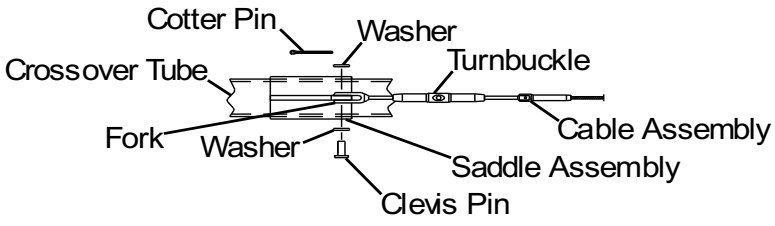


DETAIL A



VIEW A-A

Flap Actuator Hardware Removed For Clarity
Capstan Cover Removed For Clarity



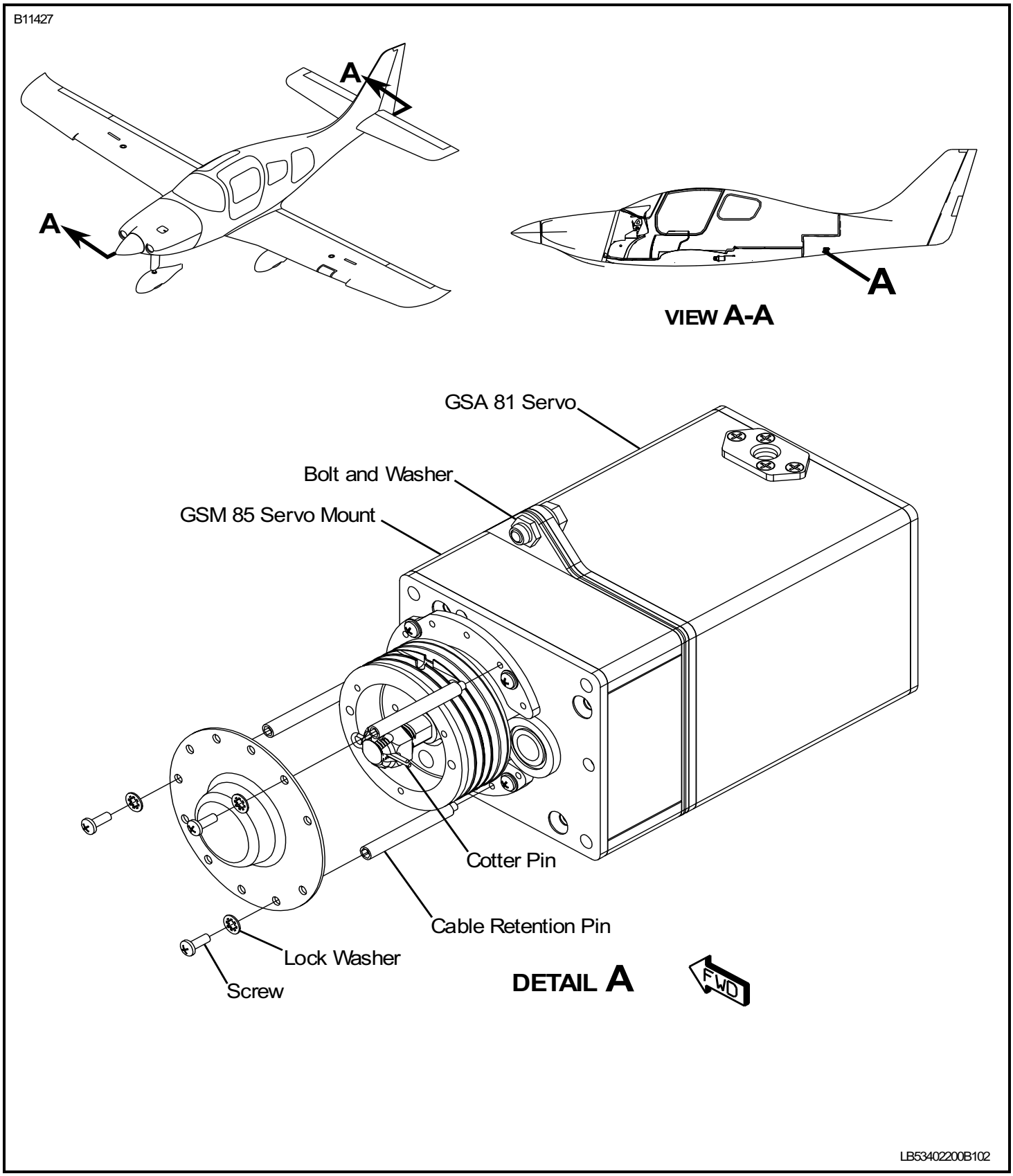
VIEW B-B

Rivets Removed For Clarity
(Fastener Installation Normal)

ADC53270000E01
AALB53220000E02
BBLB53220000E04

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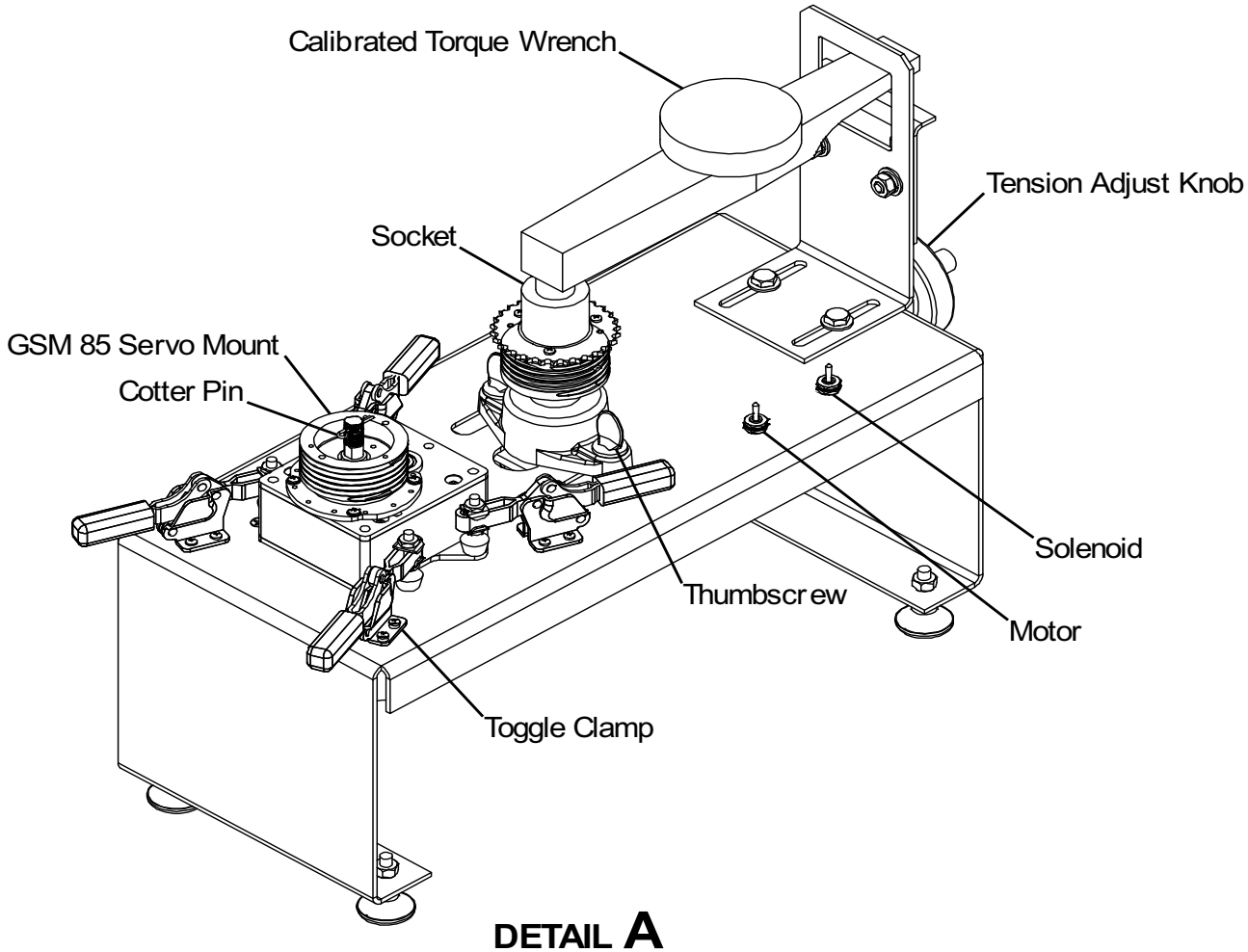
Figure 204 : Sheet 1 : Servo Assembly



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Figure 205 : Sheet 1 : Garmin Servo Adjustment Fixture

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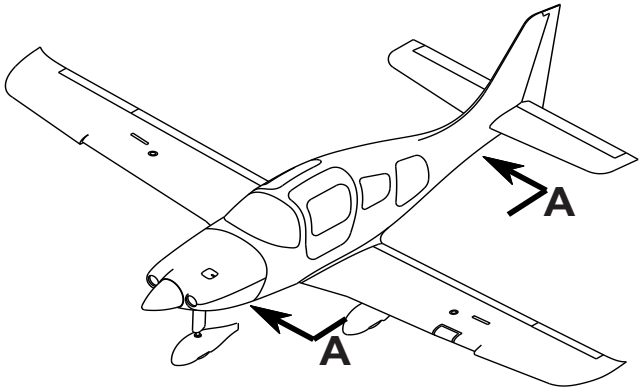
DETAIL A

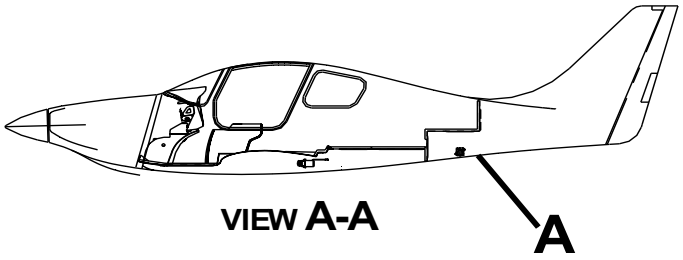
LB53402200B102

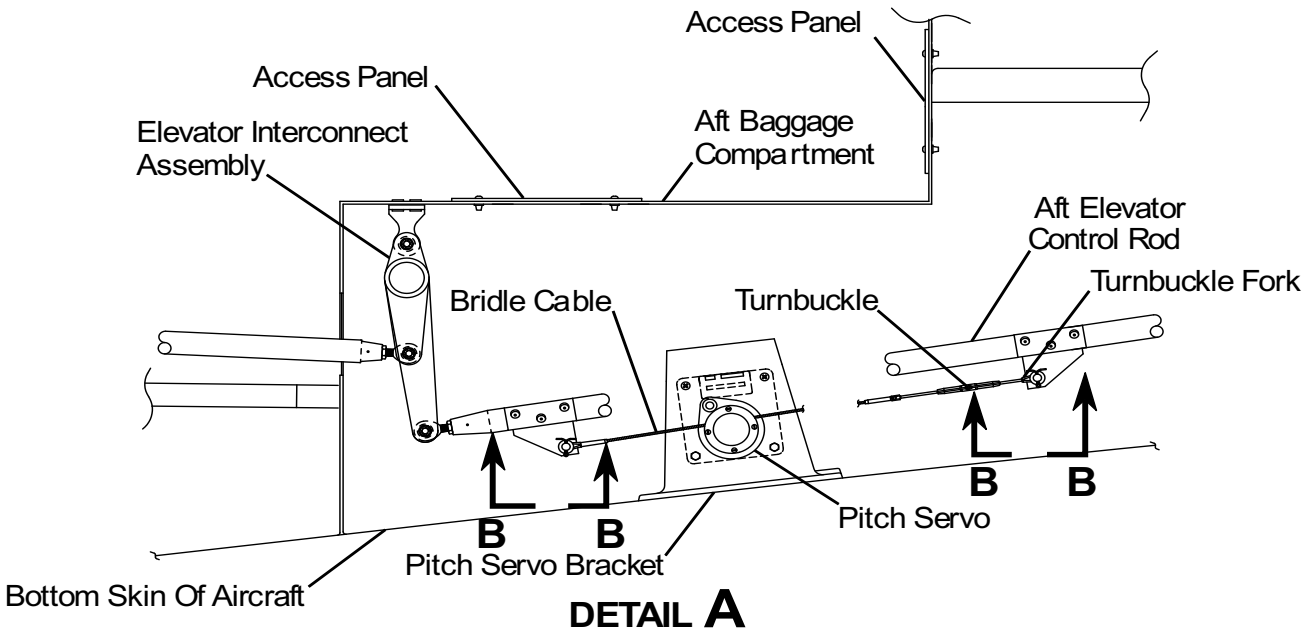
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Figure 206 : Sheet 1 : Pitch Servo

B11428

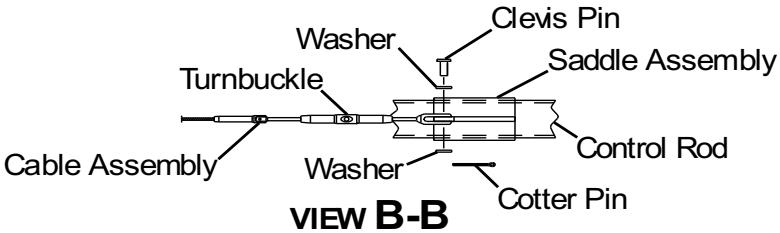






DETAIL A

View Looking Right From Pilot Side Of A/C
Shown with Elevator In Full Down Position



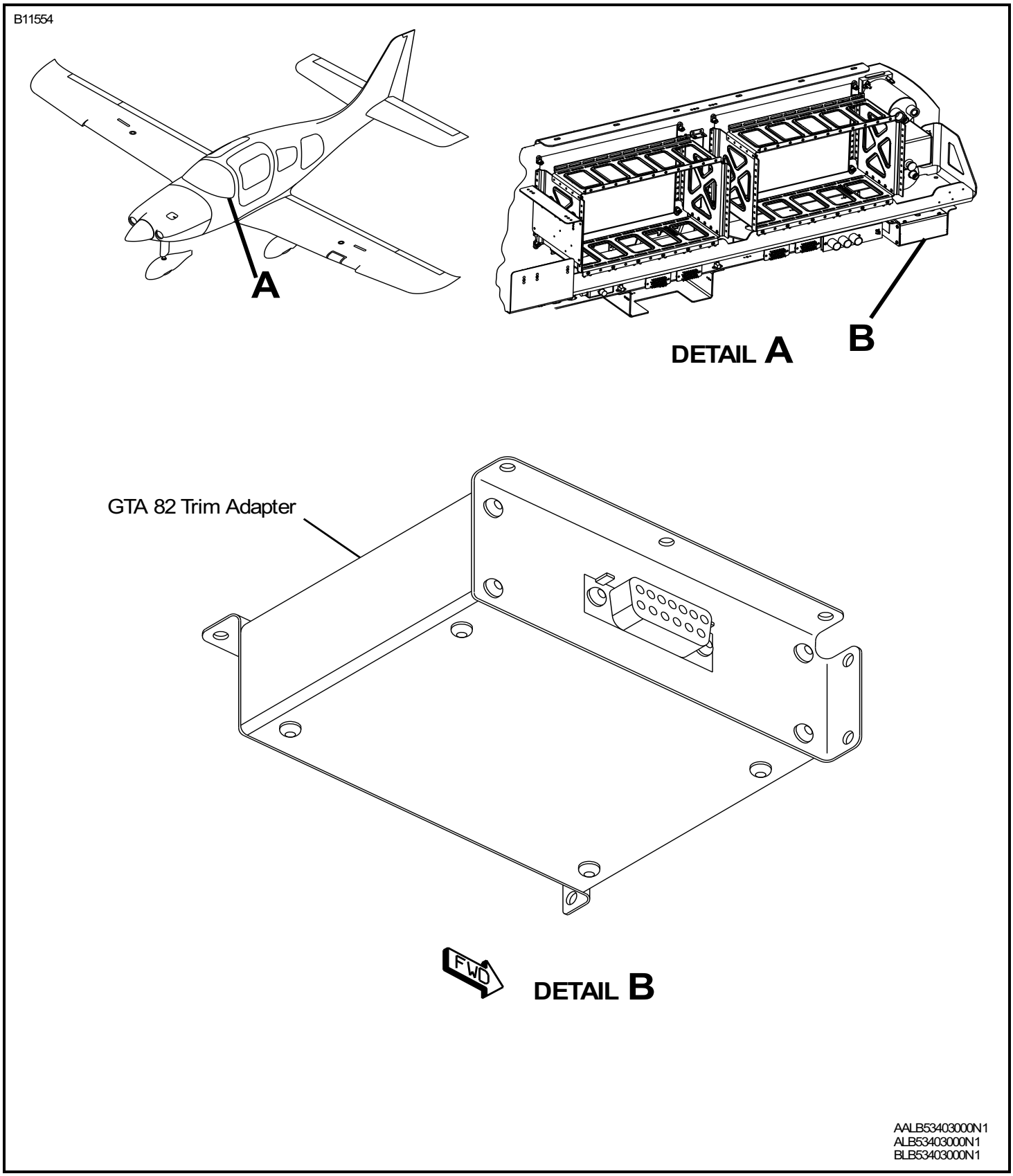
VIEW B-B

View Looking Up, Rivets Removed For Clarity
(Fastener Installation Normal)

LB53220000E02
LB53220000E05

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Figure 207 : Sheet 1 : GTA 82 Pitch Trim Adapter



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FIRE PROTECTION - MAINTENANCE PRACTICES

1. General - Fire Extinguisher Description and Limitations

A. The airplane's fire extinguisher is located below the copilot's seat in a metal bracket and is mounted horizontally. The extinguisher is stored with the top of the unit near the middle of the airplane so that it is accessible from the pilot's seat. Refer to Figure 201

B. The extinguisher is filled with a 1211/1301 Halon mixture (commonly called Halonaire) that chemically interrupts the combustion chain reaction rather than physically smothering the fire. The hand extinguisher is intended for use on Class B (flammable liquids, oil, grease, etc.) and Class C (energized electrical equipment) type fires.

C. The installed fire extinguisher is an H_3R, Inc. Halon blend model 1200. The unit has a 5BC U/L rating and contains 2.5 lbs. of recycled Halon with a 14-16 second discharge time. It has been demonstrated that 2.5 lbs. of extinguishing agent, when totally discharged at one time within the cabin of the airplane during flight, can be safely ventilated from the cabin within an FAA prescribed time period. The ventilation time period is important since Halon has some levels of toxicity. For this reason, using other types of fire extinguishers, with lower U/L ratings, different extinguishing agents, or greater weights of Halon is not approved.

D. **Temperature Limitations** - The fire extinguisher has temperature storage limitations that may need to be considered depending on the operating environment of the airplane. If it is anticipated that the cabin temperature will exceed the extremes shown in Table 201 below, the extinguisher must be removed and stored in a more temperate location.

Table 201. Temperature Limits for Fire Extinguisher Storage

Temperature Extremes	Maximum/Minimum Temperatures
Lowest Cabin Temperature	-40° F (-40° C)
Highest Cabin Temperature	120° F (49° C)

2. Inspecting the Charge of the Fire Extinguisher

A. Since the fire extinguisher does not have a pressure gauge to indicate its charge, it must be weighed at the annual or 100 hour inspection intervals to ensure there is an adequate amount of extinguishing agent. The extinguisher unit has a gross weight of 3.33 lbs. (±1 oz.) when it is installed in the airplane and must be replaced if the gross weight is less than 3.1 lbs. Use the following steps to determine if the fire extinguisher is properly charged.

(1) Remove the extinguisher from the airplane.

(2) Place the unit on a sensitive scale that measures weight within about half an ounce (14 grams).

(3) Verify that the gross weight of the unit is 3.1 lbs. (1420 grams) or more.

B. The variation between the weight of 3.33 pounds and the minimum allowable weight of 3.1 lbs. equates to a difference of 0.23 lb. or almost 105 grams. The fire extinguisher has a calibrated loss and can lose up to 4 grams of agent per year, or 24± grams during the extinguisher's useful six year life. Clearly, a loss of over 105 grams is an indication of a serious problem, and the extinguisher must be replaced.

C. In any event, the extinguisher must be replaced after six years in service. It is not cost effective to reclaim the extinguisher's Halon, disassemble and inspect the unit, replace worn or defective parts, and then recharge it.

3. Inspecting the Condition of the Fire Extinguisher

A. Check the following items on the fire extinguisher and then reinstall in the airplane:

(1) Check condition of tamper seal for security.

(2) Check for damage to the discharge lever.

(3) Check condition of the valve nozzle.

(4) Check legibility and security of instructions decal.

(5) Ensure there are no dents or deformation to the cylinder.

(6) Note the extinguisher's gross weight on the 100 hour or annual inspection form.

4. Inspection of the Fire Extinguisher Bracket

A. Inspect the bracket mounting screw holes for evidence of fatigue and pull-through.

B. Inspect the clamp mechanism for evidence of fatigue, and look for bends or crimps in the circular band. Install the extinguisher.

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C. Ensure the latching mechanism functions properly. Tug on the extinguisher a few times. The extinguisher should remain in place even when a moderate force is applied. If the extinguisher has a tendency to break out of the bracket when a moderate force is applied to the unit, replacement of the bracket is required.

5. Removal and Installation of the Fire Extinguisher Bracket

A. The fire extinguisher bracket is mounted to the aft side of the footwell on the copilot's side of the airplane. It is attached with two 1/4 - 28 screws, two flat head washers, and two 1/4 - 28 self-locking nuts.

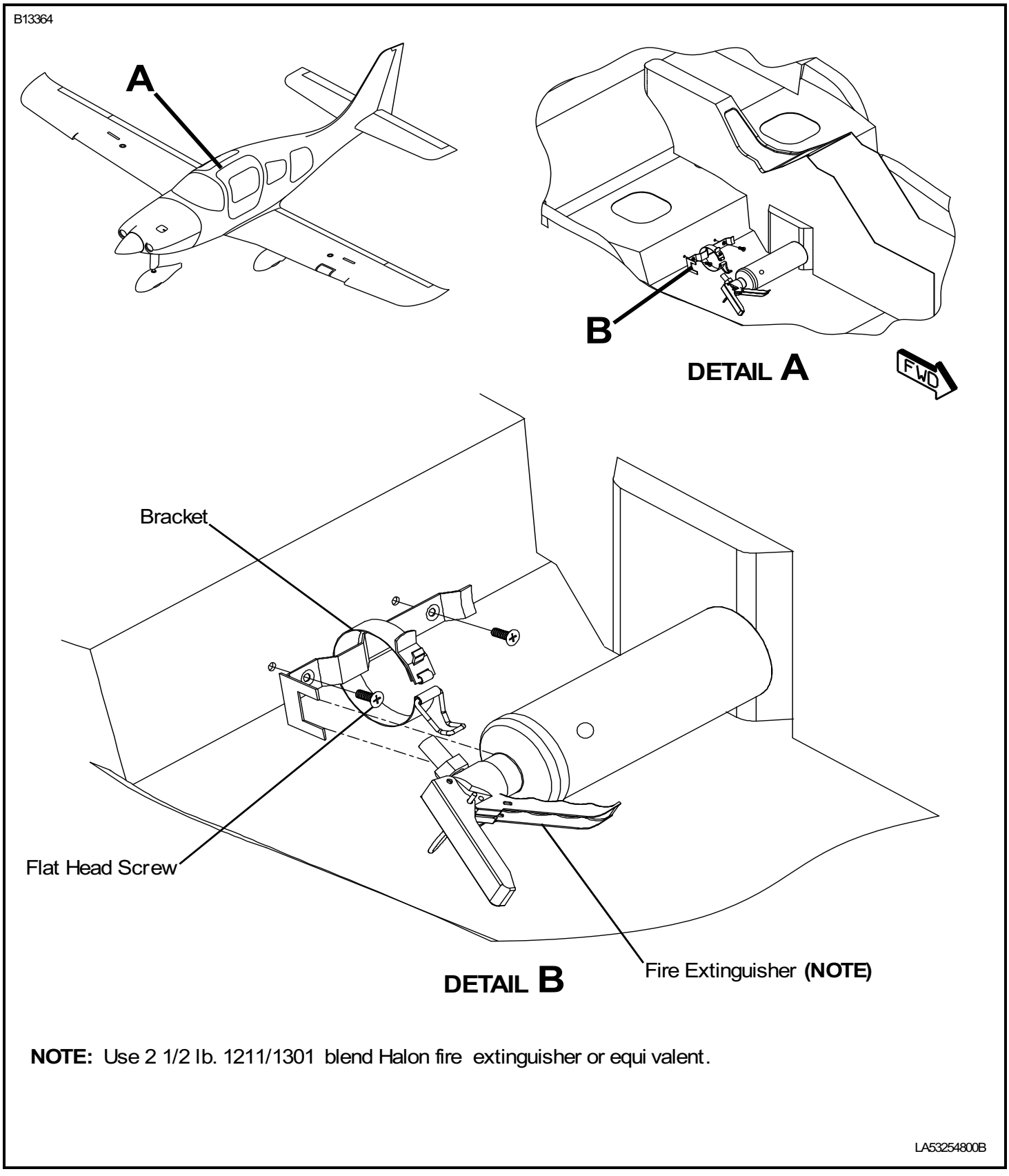
(1) The bracket is mounted by inserting the screws into the countersunk portion of the bracket and then inserting it through the pre-drilled holes in the panel of the footwell. The washer and self-locking nut are applied to the back side of the panel and torqued to 72 in.-lbs.

(2) The bracket is uninstalled by removing the two 1/4 - 28 self-locking nuts and flat head washers on the backside of the mounting panel.

B. Access to the back side of the mounting panel is available from two locations. There is an access panel under the copilot's seat, which requires removal of the seat (Refer to Chapter 25) and an access panel on the underside of the airplane below the copilot's seat. While access is more convenient using the panel under the airplane, the removal and installation of the bracket requires two people.

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Figure 201 : Sheet 1 : Fire Extinguisher



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COMMUNICATIONS - GENERAL

1. General

A. The Cessna 350 (LC42-550FG) utilizes either the Garmin AT Apollo™ SL15, Garmin GMA 340, or Garmin GMA1347 communications systems. There are variations of the aircraft communications and navigation systems. Refer to Chapter 34 for navigation system description. Refer to Chapter 34 and the Garmin AT and Garmin manuals for specific information on the actual Nav/Comm hardware. This chapter describes the system of providing communications output to the occupants and protection from precipitation static and lightning. For a specific listing of installed equipment, refer to Section 6, Appendix A, of the POH/AFM.

(1) **Description of Audio Panel** - Either the Apollo™ SL15MS Audio Selector Panel/Intercom System (ASPIS), Garmin GMA 340 audio panel, or Garmin GMA 1347 audio panel is installed in your aircraft. The primary purpose of the panel is to control communication and navigation audio selections, intercom functions, and the marker beacons. In addition, the unit has provisions for a stereo entertainment input. The audio panel is located in the top of the radio rack panel assembly.
The Garmin GMA 1347 audio panel is installed in your aircraft with the Garmin G1000 Option. Refer to Section 34-60-00, GMA 1347 Audio Panel for description, removal, and installation.

(2) **Description of Static Discharge System** - During use, the airplane is susceptible to a build-up of static electricity. An accumulated static electrical charge can render radio navigation and communication inoperative. The purpose of the static discharge system is to transfer the electrical charge off the airplane's surface and quietly discharge it into the air.

(3) **Description of Lightning Protection System** - The composite construction of the Cessna 350 has a very high resistance to electricity and very little conductivity. Having a high conductivity is critical to lightning protection because it is important that all parts of the airplane have the same electrical potential. To achieve high conductivity, aluminum and copper mesh was integrated with the composite construction of the airplane. The mesh varies from being 0.010 to 0.030 inches below the surface of the paint and encompasses most surfaces of the airplane. Various parts of the airplane are interconnected through use of metal fasteners inserted through several plies of mesh, mesh overlaps, and bonding straps. Refer to Section 20-70-00 for other lightning protection information.

CAUTION: Before any maintenance, removal, or installation of equipment is performed, make sure both the master and avionics switches are in the off position. Failure to do so could result in damage to these instruments.

2. Speech Communications

A. All communication radios are integrated with some form of navigational receiver, and it is impracticable to segregate the discussion of each function in two separate chapters. Accordingly, air-to-ground speech communications are covered in Chapter 34.

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AUDIO COMPONENTS - MAINTENANCE PRACTICES

1. Apollo SL15 Audio Panel

A. The SL15 Audio Panel acts as the audio selector for the airplane. The SL15 can switch between three transceivers (Com 1, Com 2, and Com 3) and six receivers (Nav 1, Nav 2, ADF, DME, MKR, and AUX). The SL15 also contains a voice activated intercom system, which allows the pilot, copilot, and passengers of the airplane to communicate among themselves. Refer to Figure 201 (SL15 Audio Panel). The intercom select switch can be set to three modes.

(1) In the ALL mode, the intercom is linked to all four-seat positions; everyone hears radio communications, and everyone hears music from input Entertainment No. 1.

(2) In the CREW mode, the pilot and front seat passenger are linked together and can receive radio communications. The rear seat passengers are linked together but cannot hear radio communications or the pilot or front passenger.

(3) In the ISOLATE mode, the pilot hears the radios, but is not connected to the intercom, while the front and rear passengers are on the same intercom loop but cannot hear radio communications.

B. All four headsets are hooked up to the intercom system. A volume knob on the left side of the SL15 controls volume for the headsets.

C. **Removal of SL15 Audio Panel**

(1) Insert a 3/32 inch Allen wrench into the hole on the faceplate. Refer to Figure 201.

(2) Turn the Allen wrench counterclockwise.

(3) The SL15 Audio Amplifier will slide out of the radio rack.

D. **Installation of SL15 Audio Panel**

(1) Install fittings into the back of the SL15.

(2) Slide the SL15 into its slot in the radio rack.

(3) Insert a 3/32 inch Allen wrench into the hole on the faceplate and turn clockwise. Refer to Figure 201.

2. Garmin Gma 340 Audio Panel

A. The GMA 340 Audio Panel acts as the audio selector for the airplane. The GMA 340 is a VHF Communication Transceiver/VOR/ILS Receiver/GPS Receiver. The system is a fully integrated panel-mounted instrument which contains audio and microphone switching and amplification, a marker beacon receiver, and an intercom system. The GMA 340 also contains a voice activated intercom system, which allows the pilot, copilot, and passengers of the airplane to communicate among themselves. Refer to Figure 202. The intercom select switch can be set to three modes.

(1) In the ALL mode, the intercom is linked to all four-seat positions; everyone hears radio communications, and everyone hears music from input Music 1.

(2) In the CREW mode, the pilot and front seat passenger are linked together and can receive radio communications. The rear seat passengers are linked together but cannot hear radio communications or the pilot or front passenger.

(3) In the PILOT mode, the pilot hears the radios, but is not connected to the intercom, while the front and rear passengers are on the same intercom loop but cannot hear radio communications or communicate with the pilot.

B. All four headsets are hooked up to the intercom system. There is a volume knob for the pilot and copilot's sides.

C. **Removal of GMA 340 Audio Panel**

(1) Insert a 3/32 inch Allen wrench into the jack screw on the faceplate. Refer to Figure 201, which shows the SL15 panel, but is similar to the GMA 340 installation.

(2) Turn the wrench counterclockwise until loose.

(3) Slide out the GMA 340 unit, and disconnect the connectors.

D. **Installation of GMA 340 Audio Panel**

(1) Install fittings into the back of the GMA 340.

(2) Slide the unit into the rack until the jack screw makes contact with the receptacle located in the back plate.

(3) Insert a 3/32 inch Allen wrench into the jackscrew access hole on the faceplate.

(4) Turn the Allen wrench clockwise until the unit is secured in the rack. Continue turning until tight, but do not over-tighten.

3. Speaker

A. **Removal of Speaker**

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(1) Remove the overhead center panel per instructions in Section 25-30-00.

(2) For Type A, remove the four screws connecting the stall warning buzzer mounting, speaker, loop clamp, and speaker mounting. For Type B, remove the four screws connecting the speaker to the adapter plate. Refer to Figure 203.

(3) Remove the stall warning buzzer mounting, as required.

(4) Disconnect the wires.

(5) Pull the speaker from the mounting.

B. **Installation of Speaker**

(1) Put the speaker on the speaker mounting.

(2) Connect the wires.

(3) Place the stall warning buzzer mounting on top of the speaker, as required.

(4) For Type A, tighten the four nuts connecting the stall warning buzzer mounting, speaker, loop clamp, and speaker mounting. For Type B, tighten the four nuts connecting the speaker to the adapter plate. Refer to Figure 203.

(5) Install the overhead center panel per instructions in Section 25-30-00.

4. Push-To-Talk Switch

A. **Location** - Each control stick has a push-to-talk switch located on the forward part of the stick where it is easily accessed by the index finger. Refer to Figure 204. The switch is press-fit into the stick and connected by wire on the inside of the stick assembly.

B. **Description** - The push-to-talk switch is a single-button switch that when pressed, makes a connection required to energize the Comm unit by way of the Audio Panel.

5. Troubleshooting Push-To-Talk Function

A. **Symptom** - Push-to-talk function not working properly.
Probable Cause - Broken wire or connection.

(1) Locate the defective connection:

(a) Remove the interior panel covering the control stick.

(b) Locate P-46 (pilots side) and P-47 (copilot's side), and disconnect the plug.

(2) Using a digital volt ohmmeter (DVOM), check for operation of the push-to-talk function switch. Place one probe on pin 1 and ground the other. The DVOM should show a closed circuit with the switch depressed (less than 2 ohms).

(3) Check the jack side of the connection. Pin 2 should be ground and Pin 1 should be for the audio panel.

(4) If the switch needs to be replaced, cut the tie-wraps at the base of the control stick to allow the switch to be gently pulled out of the wooden handle.

(5) Solder in a new switch, and pull the wires tight at the base of the control stick tube.

(6) Secure the wires with tie-wraps, and check for freedom of movement.

6. NAV/COM Bypass Switch

A. **Description** - With the Garmin G1000 system the Nav/Com bypass switch is not installed. With the Avidyne option the aircraft is equipped with a Nav/Com bypass switch located above the attitude indicator next to the clock. The switch is designed to be used in situations where the engine is not running. The switch is an alternate action device that activates when depressed and needs to be depressed again to cycle off. When activated, the switch turns on the No. 1 GPS, No. 1 Com, and No. 1 Nav by connecting them directly to the right battery, bypassing the power grid, circuit breaker panel, and associated wiring. Communications is only through the pilot's headset and activated with the pilot's PTT switch.

B. **Removal of Switch**

(1) Remove the glare shield per instructions in Section 25-11-00, Glare Shield Removal, to gain access to the back side of the instrument panel.

(2) Disconnect the connector at the back of the switch.

(3) Pull the press-fit switch out of its housing at the back of the instrument panel as shown in Figure 205.

C. **Installation of Switch** - Installation is a direct reversal of removal.

7. Headphone Jacks/Microphone Plugs

A. **Description** - The Cessna 350 (LC42-550FG) is designed to handle both standard headphone/microphone jacks as well

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as Bose™ Stereo headphone jacks.

B. **Location** - Each seat has a jack plug receptacle.

(1) **Front Seats** - The location of the front seat receptacles are at the lower part of the center console for the Avidyne option and at the front of the center console for the Garmin G1000 option. Refer to Figure 206 for Avidyne option and Section 25-12-00 for Garmin G1000 option.

(2) **Rear Seats** - The location of the rear seat receptacles are on the rear seat side panels. Refer to Figure 207.

CAUTION: Bose(TM) headphones require power, which is supplied by the Bose(TM) jack. Before removing the intercom jack, ensure the system master switch is in the off position.

C. **Removal of Rear Passenger Intercom Jacks**

(1) To gain access to the backside of the jacks, remove the interior sidewall panel per instructions in Chapter 25.

(2) Unscrew and remove the Bose™ jack cover. Loosen and remove the jam nuts securing the jacks to the interior panel. Refer to Figure 208.

(3) Label all wires according to where they were connected.

(4) Cut wire near the solder point at each jack.

D. **Installation of Rear Passenger Intercom Jacks**

(1) Remove all traces of old solder from the jacks.

(2) Reconnect and solder the wires to the jacks. Ensure all wires are soldered to the correct pins on the jacks.

(3) Secure the jacks to the sidewall panel using jam nuts.

(4) Install the Bose™ jack cover.

(5) Install the sidewall panel.

CAUTION: Bose™ headphones require power, which is supplied by the Bose™ jack. Before removing the intercom jack ensure the system master switch is in the off position.

E. **Removal of Pilot/Front Passenger Intercom Jacks/Microphone Plug**

(1) To gain access to the jacks, remove both sides of the center console panels. Only one panel in Garmin G1000 option.

(2) Unscrew and remove the Bose™ jack cover. Loosen and remove the jam nuts securing the jacks to the interior panel. Refer to Figure 208.

(3) Label all wires according to where they were connected.

(4) Cut wire near the solder point at each jack.

F. **Installation of Pilot/Front Passenger Intercom Jacks/Microphone Plug**

(1) Remove all traces of old solder from jacks.

(2) Reconnect and solder wires to the jacks. Ensure all wires are soldered to the correct pins on the jacks.

(3) Secure the jacks to the sidewall panel using jam nuts.

(4) Install the Bose jack cover.

(5) Install the sidewall panel.

8. Stereo Audio Input Jack, Optional

A. **Avidyne Option**

(1) **Removal of Stereo Audio Input Jack**

(a) To gain access to the jack, remove the center console access panel.

(b) Lift up the armrest and locate the jack at the forward end of the pencil box.

(c) Loosen and remove the jam nut securing the jack to the pencil box. Refer to Figure 209.

(d) Label all wires according to where they were connected.

(e) Cut the wire near the solder point at each jack.

(2) **Installation of Stereo Audio Input Jack**

(a) Remove all traces of old solder from the jack.

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(b) Reconnect and solder wires to the jack. Ensure all wires are soldered to the correct pins on the jack.

(c) Secure the jack to the pencil box using the jam nut.

(d) Install the center console access panel.

B. **Garmin G1000 Option** - The stereo audio input jack is located under the instrument panel to the right of the tower on the co-pilot's side.

(1) **Removal of Stereo Audio Input Jack**

(a) Loosen and remove the jam nut securing the jack to the instrument panel.

(b) Reach up behind the instrument panel and remove the jack.

(c) Label all wires according to where they were connected.

(d) Cut the wire near the solder point at each jack.

(2) **Installation of Stereo Audio Input Jack**

(a) Remove all traces of old solder from the jack.

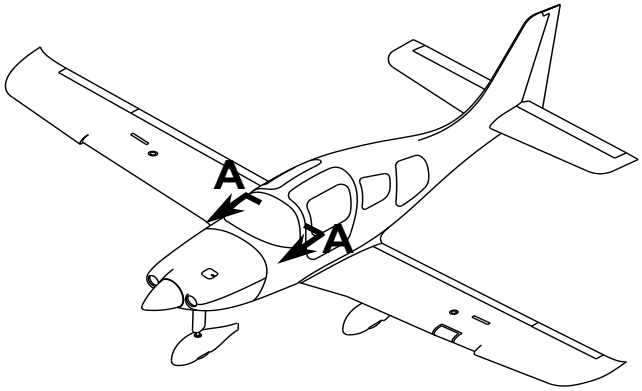
(b) Reconnect and solder wires to the jack. Ensure all wires are soldered to the correct pins on the jack.

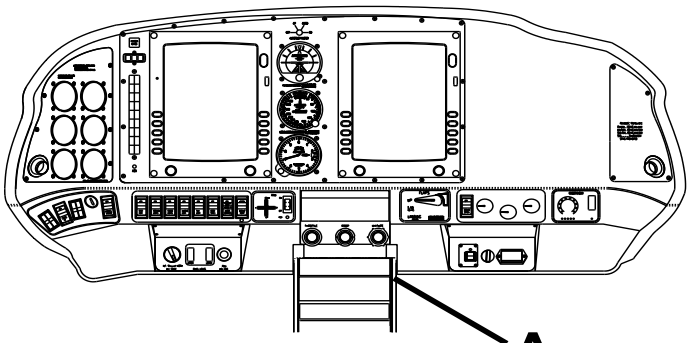
(c) Secure the jack to the instrument panel using the jam nut. Apply one drop of Loctite 243 to the threads and tighten snug (no looseness plus 1/4 turn).

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Figure 201 : Sheet 1 : SL15 Audio Panel

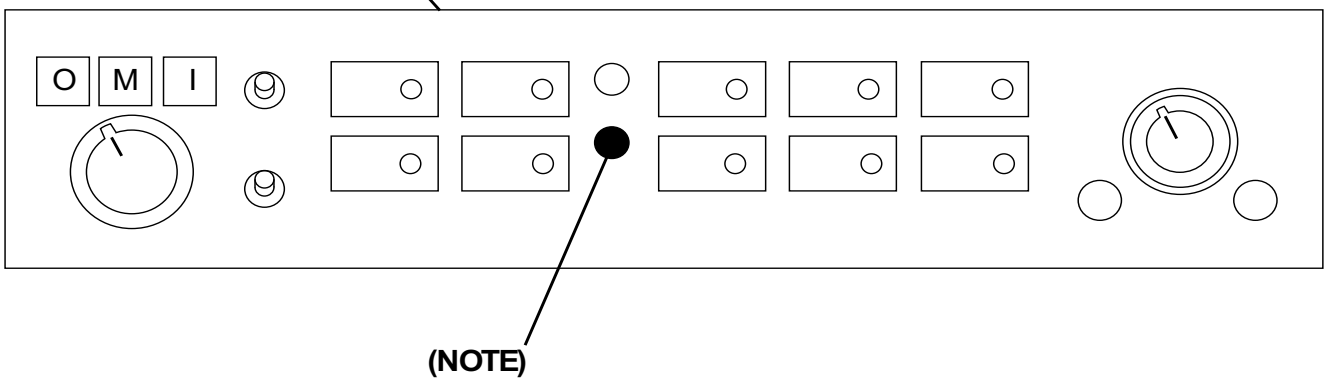
B11711





VIEW A-A

Apollo™ SL15 Audio Panel



(NOTE)

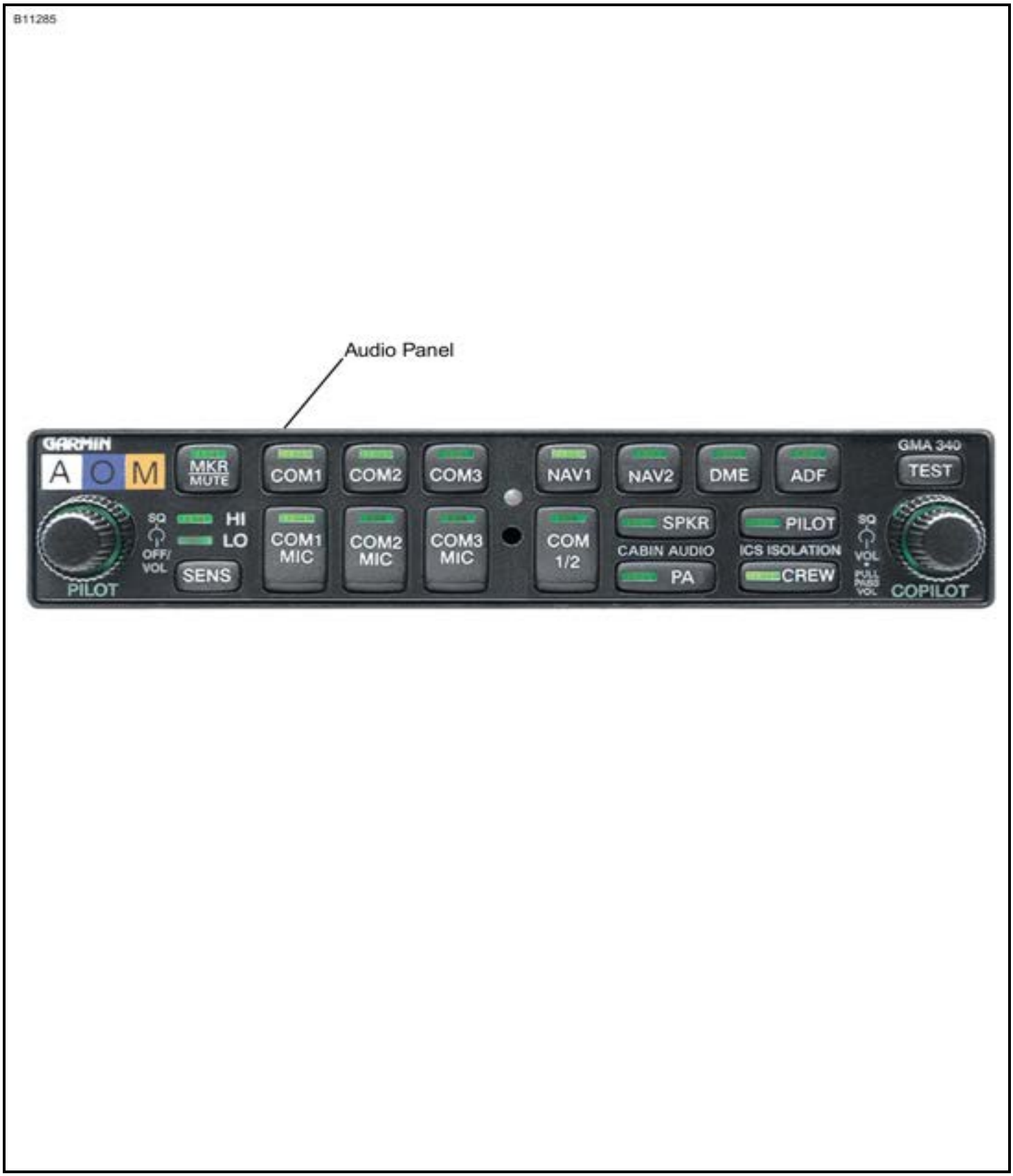
DETAIL A

NOTE: Insert Allen wrench here to rem ove/install panel.

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AALC53310000C

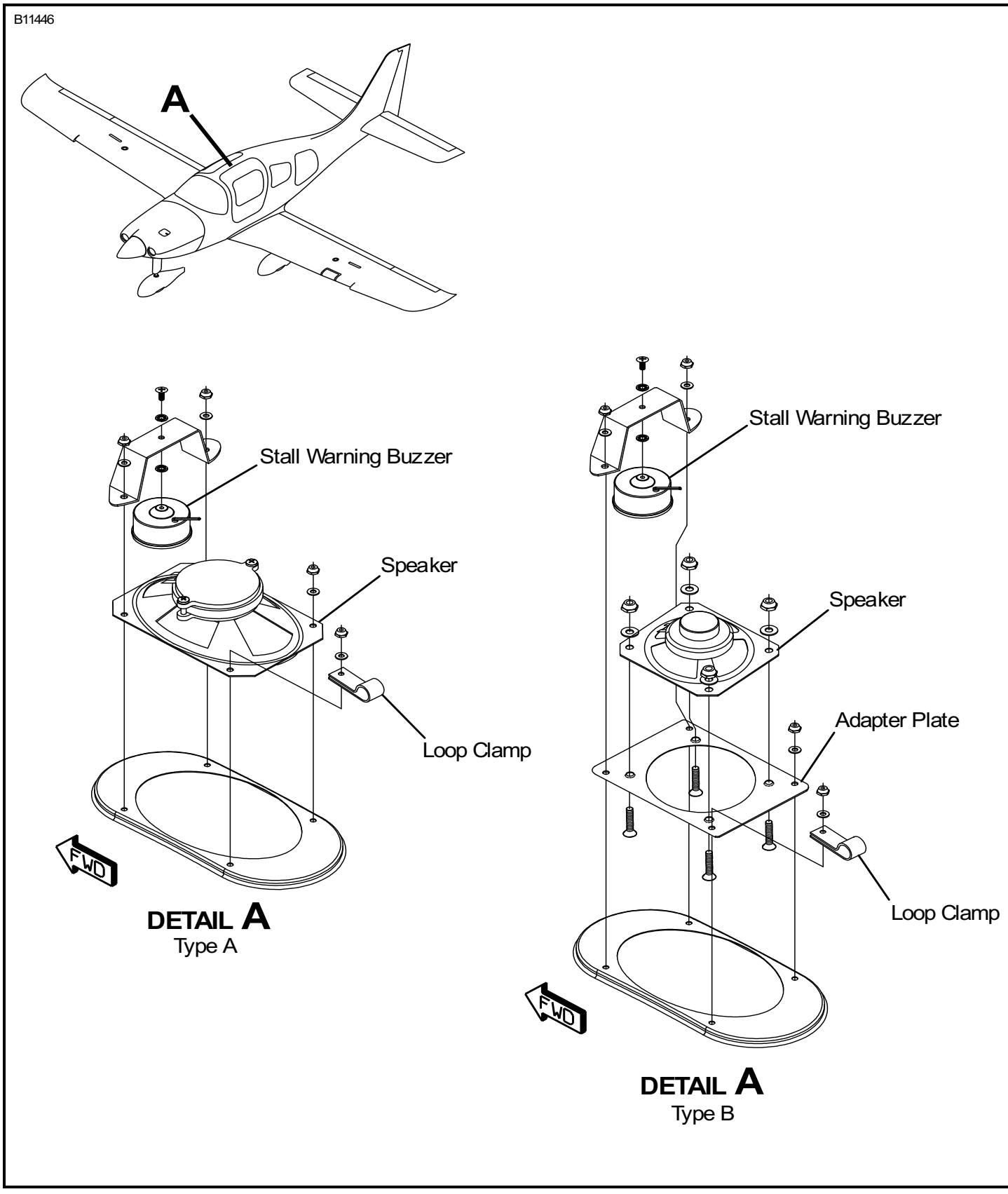
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Figure 202 : Sheet 1 : Garmin GMA 340 Audio Panel



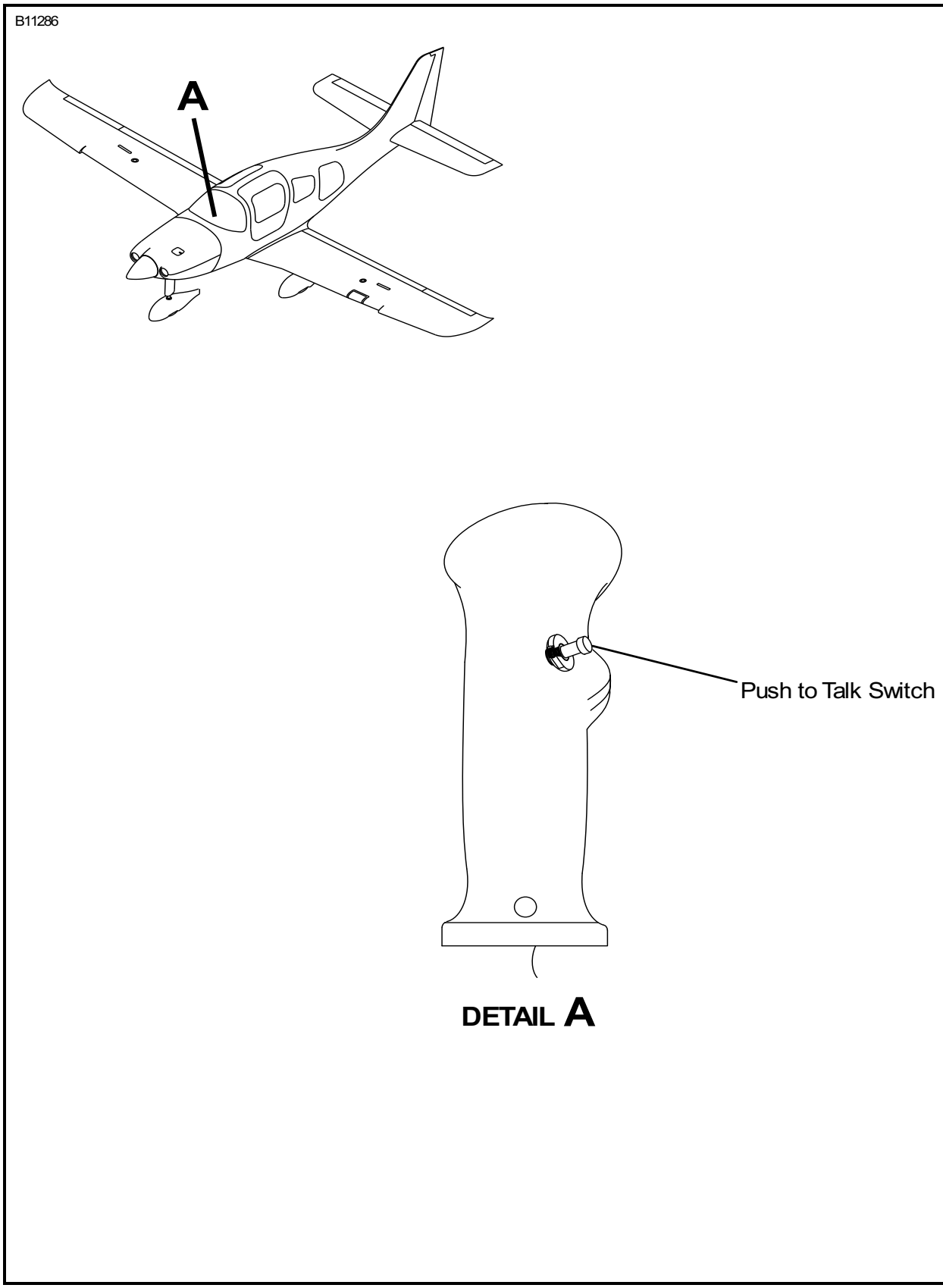
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Figure 203 : Sheet 1 : Overhead Speaker Installation



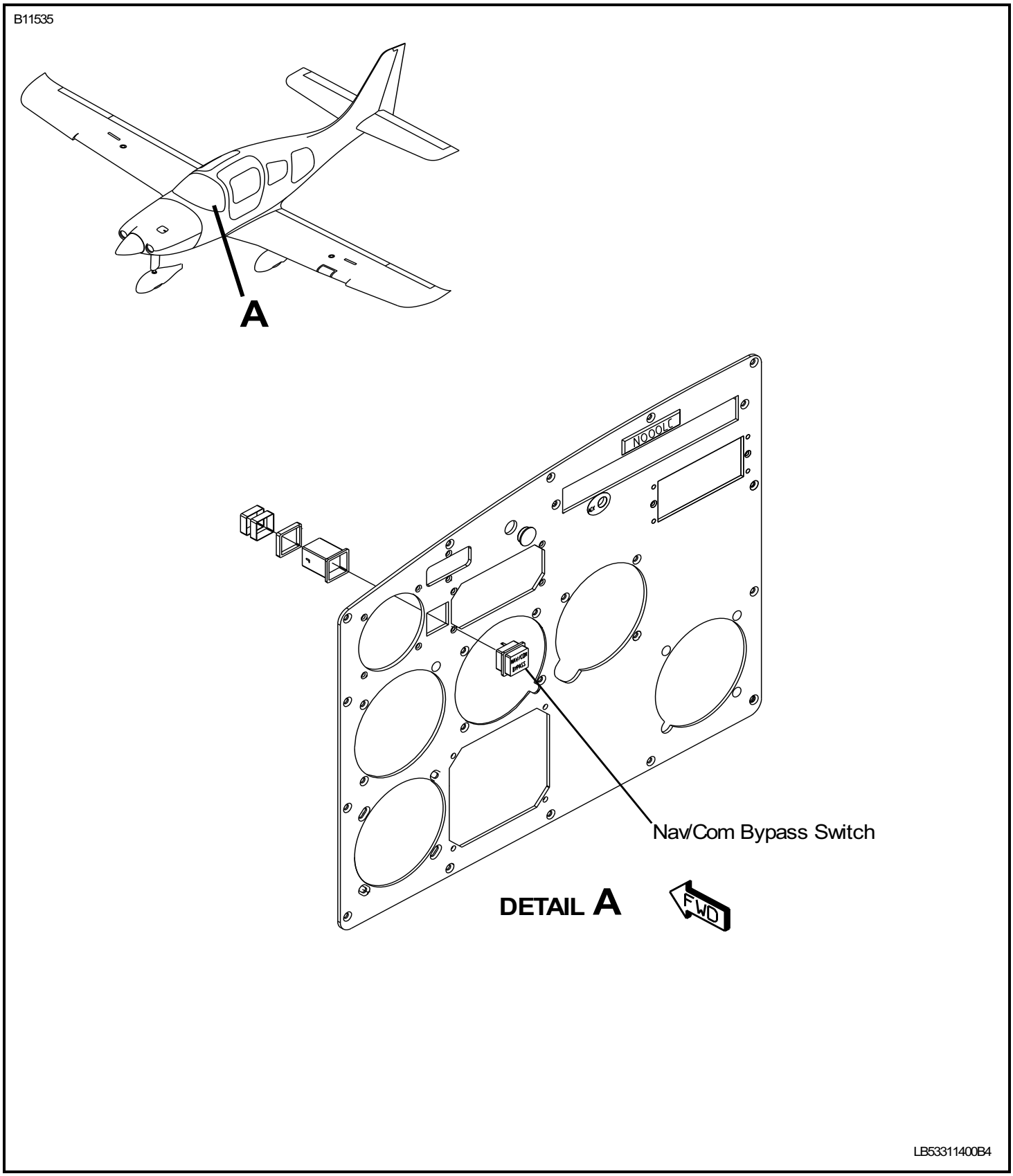
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Figure 204 : Sheet 1 : Push-To-Talk Switch Location



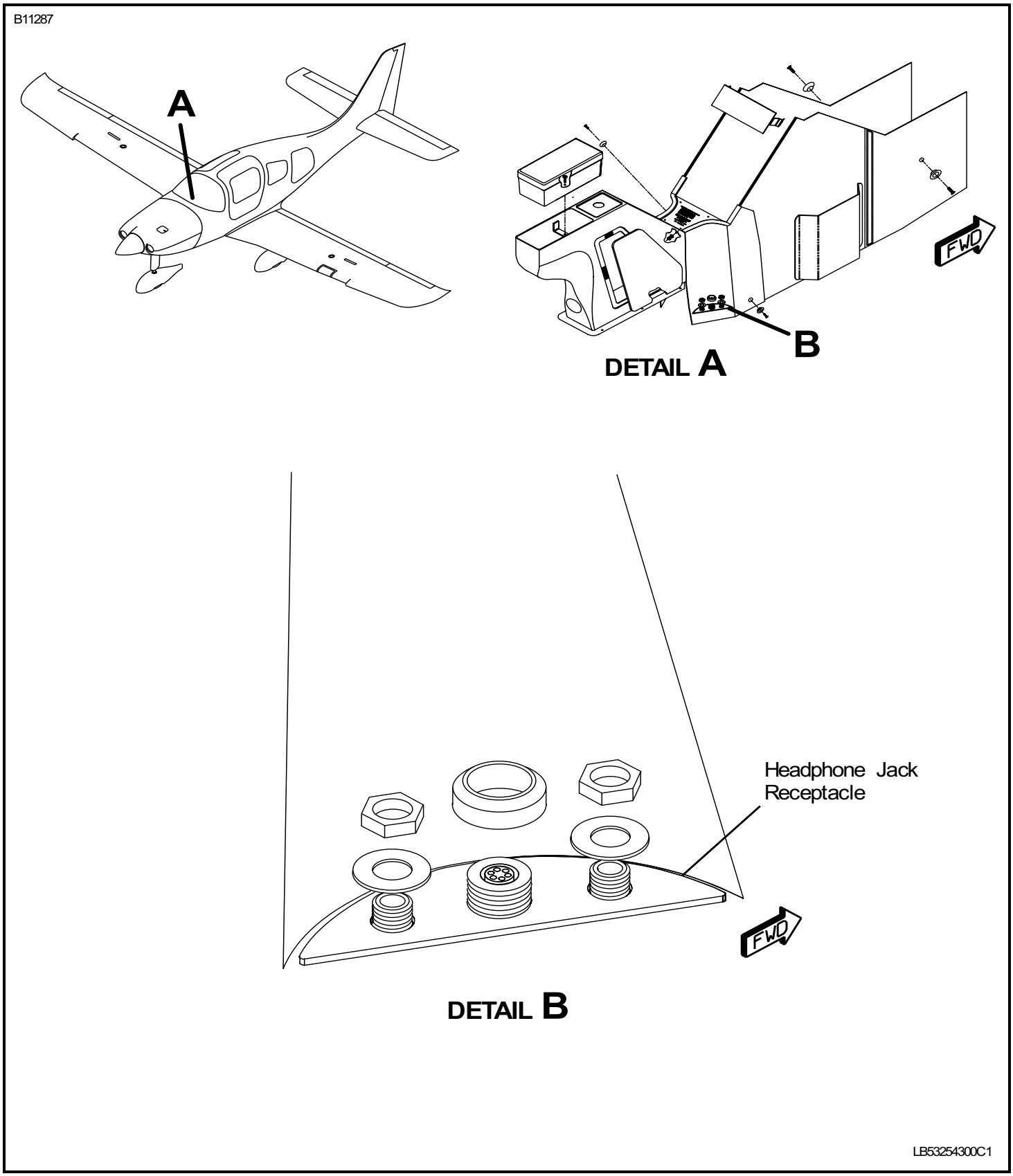
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Figure 205 : Sheet 1 : Nav/Com Bypass Switch



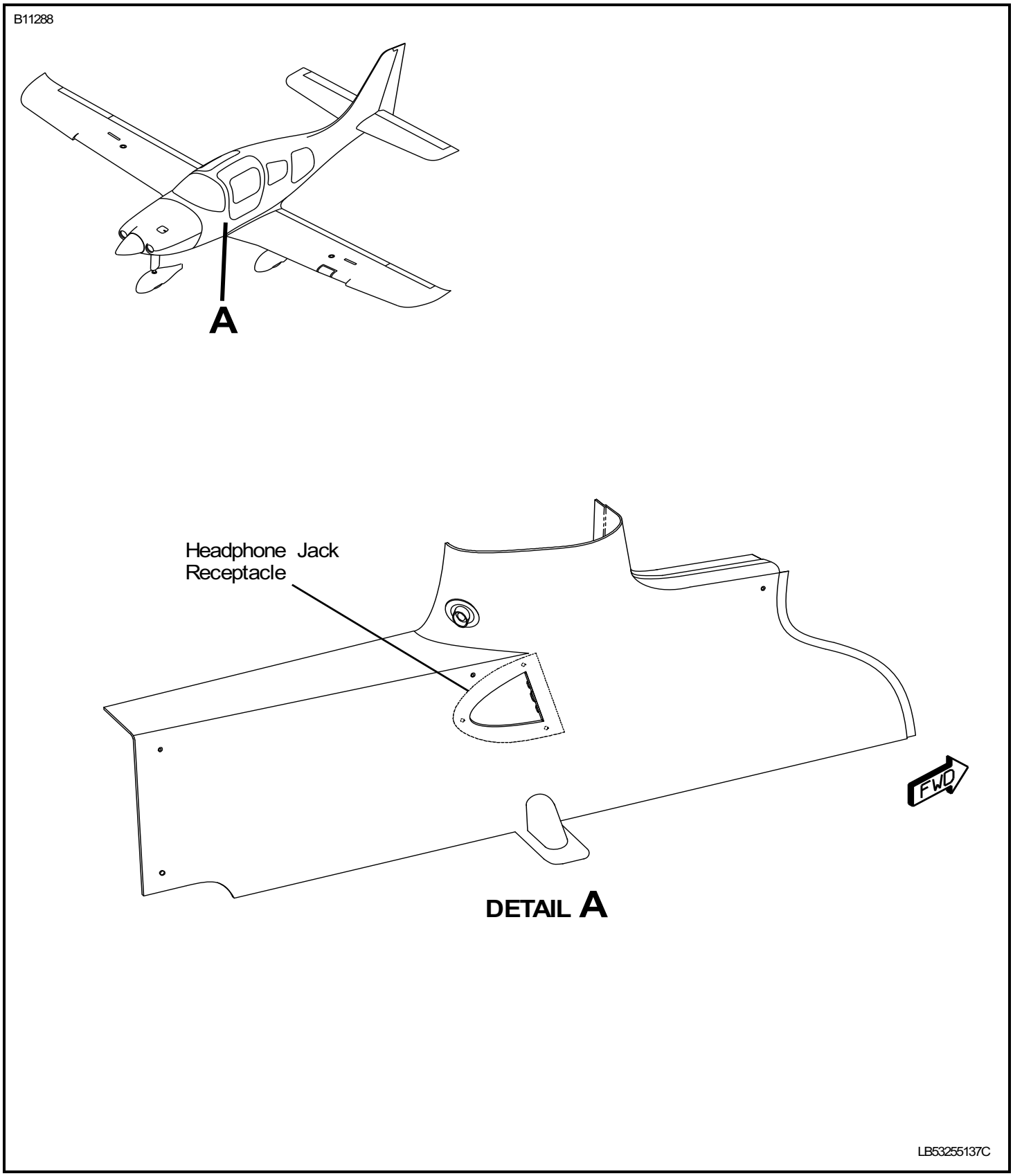
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Figure 206 : Sheet 1 : Front Seat Headphone Jack Receptacle



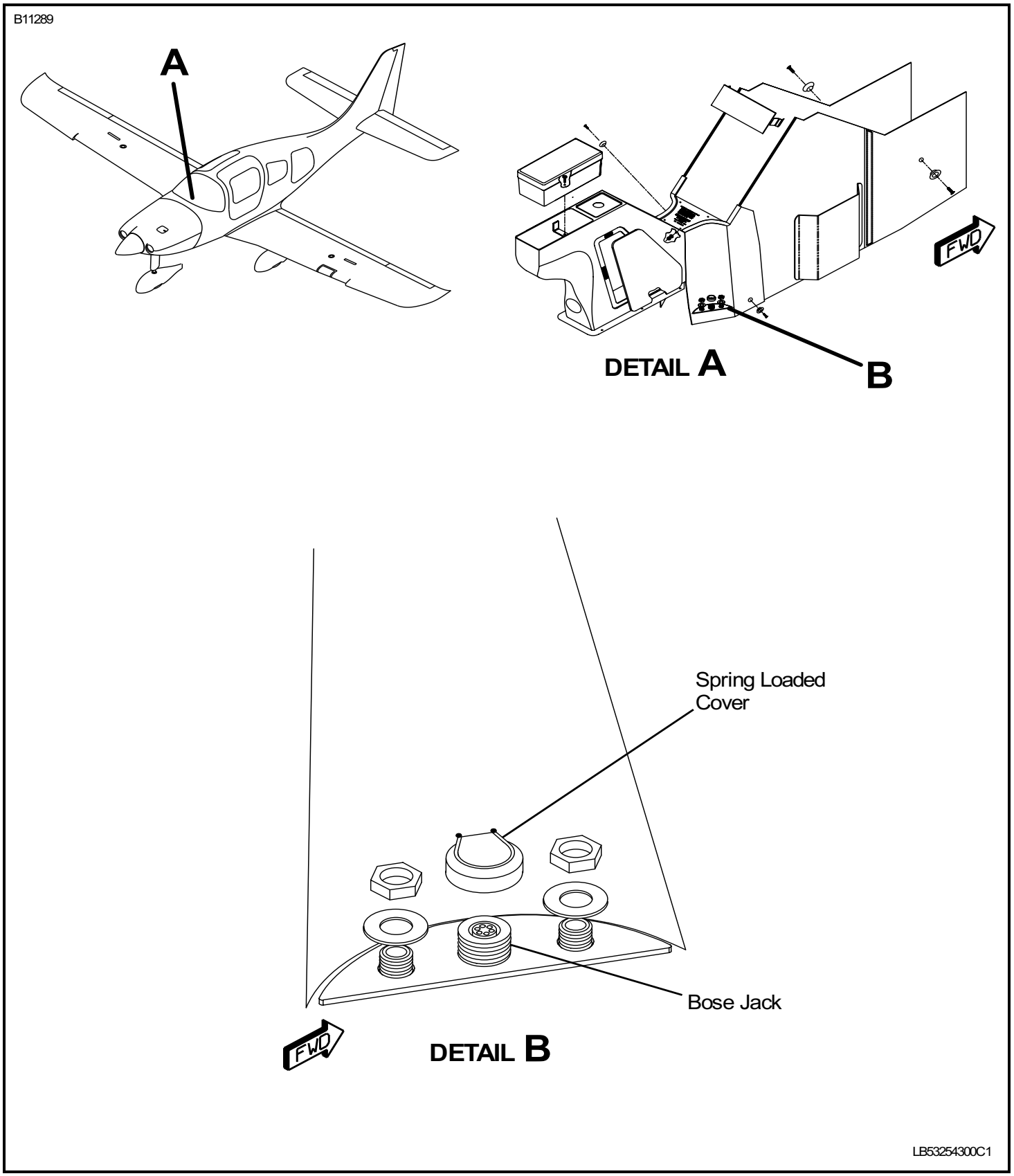
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Figure 207 : Sheet 1 : Rear Seat Headphone Jack Receptacle



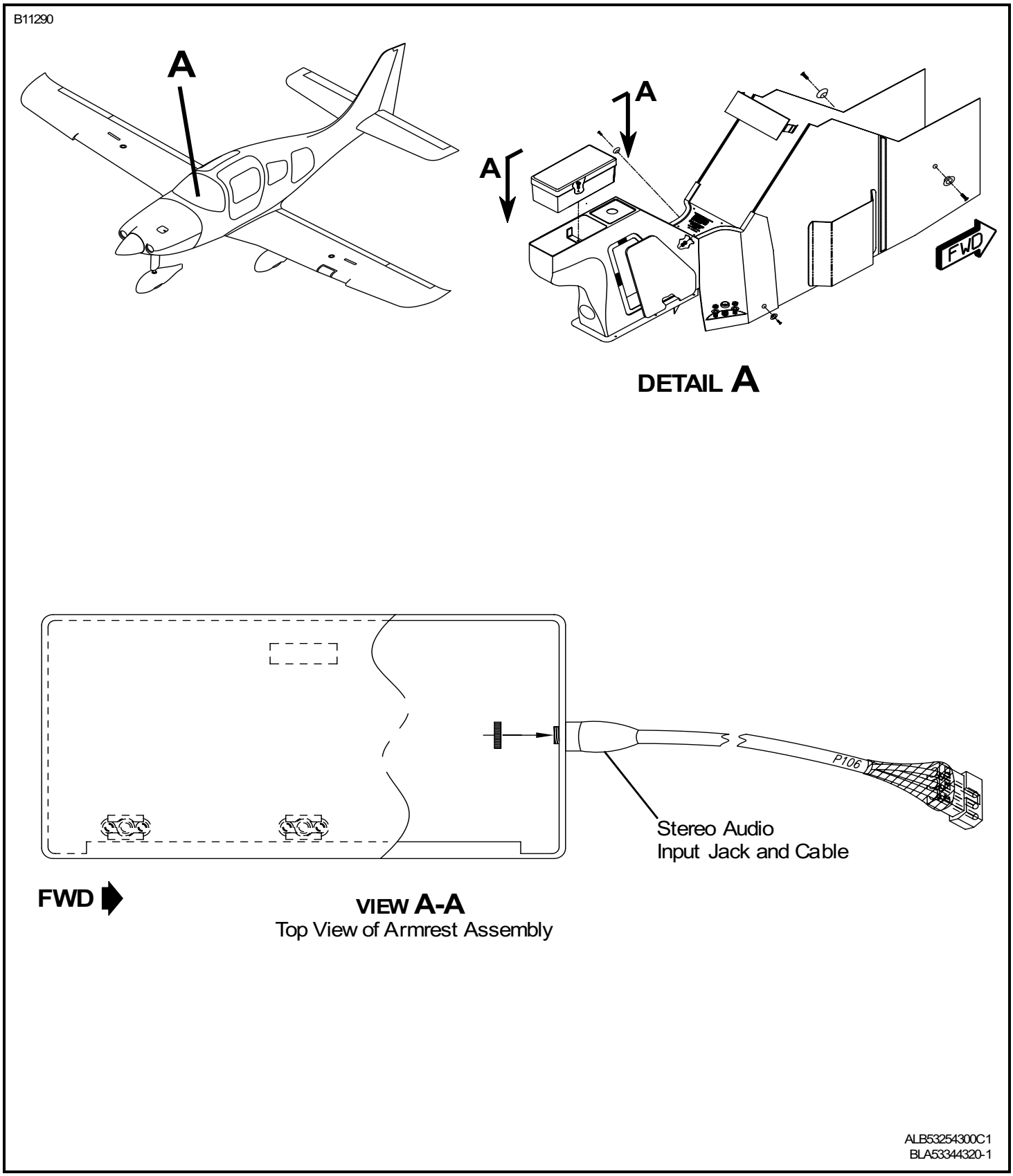
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Figure 208 : Sheet 1 : Headphone/Microphone Jack Installation



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Figure 209 : Sheet 1 : Stereo Audio Input Jack



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STATIC DISCHARGE WICKS - MAINTENANCE PRACTICES

1. Static Discharge Wicks

A. **Detection of a Broken Static Wick** - To check the wick's integrity, hold the trailing edge between the thumb and forefinger, and gently move it left and right approximately 2 inches. If the unit flexes at point A, as shown in Figure 201, the wick is broken and should be replaced.

B. **Removal of Static Wick** (Refer to Section 20-70-00, Repair of the Static Wick, for repair details)

(1) Unscrew the screw connecting the static wick to its housing.

(2) Remove the static wick from the housing.

CAUTION: When removing and handling a good static wick, do not bend it. The static wick can be broken without obvious exterior indications.

WARNING: If the wicks are removed and not reinstalled before flight, problems with navigation and communication equipment can occur.

NOTE: If the static wick does not slide out after removing the screw, use a pair of pliers and gently pull the static wick from its housing.

C. **Installation of Static Wick** (Refer to Section 20-70-00, Section 20-70-00, Repair of the Static Wick for repair details)

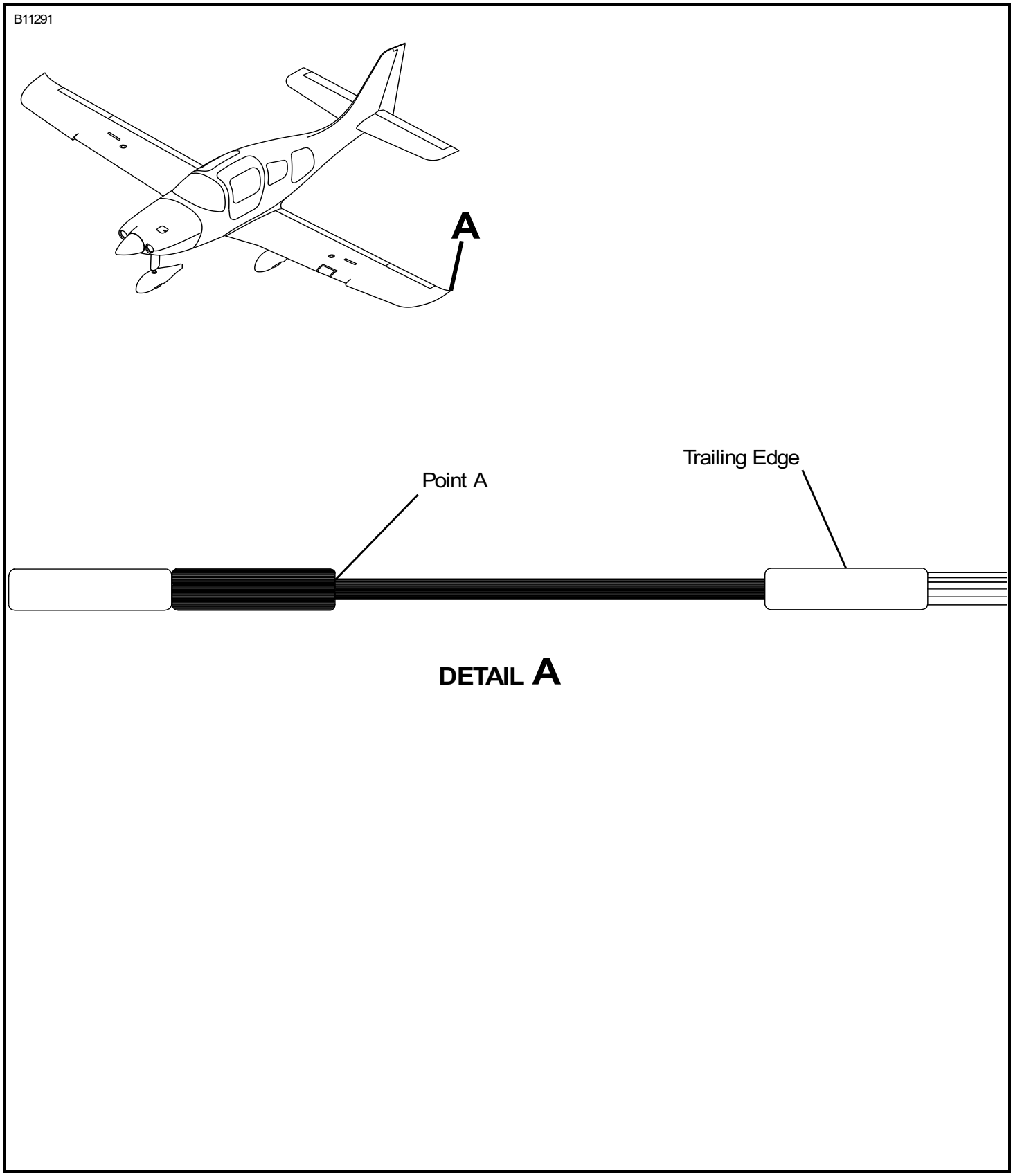
CAUTION: Before installing the static wick, perform the broken static wick test. A static wick can be broken without any obvious exterior indications.

(1) To install a static wick, place the wick in its housing.

(2) Screw the static wick and its housing together.

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Figure 201 : Sheet 1 : Static Wick



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LEVELING AND WEIGHING - GENERAL

1. General

A. To determine the empty weight and center of gravity of the airplane, the airplane must be in a level area and in a particular configuration.

2. Airplane Configuration (Empty Weight)

A. The airplane empty weight includes eight quarts of oil (dipstick reading), unusable fuel, hydraulic brake fluid, and installed equipment.

B. Defuel the airplane per instructions in Section 12-00-00 of this manual.

C. Ensure the oil sump is filled to eight quarts (cold engine). Check the reading on the dipstick and service as necessary.

D. Place the pilot's and front passenger's seat in the full aft position.

E. Retract the flaps to the cruise or 0-degree position.

F. Center the controls to the neutral static position.

G. Ensure all doors, including the baggage door, are closed when the airplane is weighed.

CAUTION: It is not recommended to weigh an airplane with full fuel and subtract the weight of the fuel to obtain empty weight because the weight of fuel varies with temperature. If this method of weight determination is used, fuel weight should be calculated conservatively. Use the specific weight of fuel at ambient temperature. Refer to Figure 1.

The following is offered as an example only. It is important to remember that the aircraft weight in the example does not apply to a specific airplane. Example:

Example Calculations: Aircraft Weight With Fuel

Unconservative Calculation

Conventionally used fuel specific weight (6 lbs./U. S. gal.)

Total Aircraft weight with fuel	= 3038 lbs.
Weight of fuel (98 gal. x 6 lbs./U. S. gal.)	= 588 lbs.
Airplane empty weight (3038 lbs - 588 lbs.)	= 2450 lbs.

Conservative Calculation

Fuel specific weight at 60° F (5.83 lbs./U. S. gal.)

Total Aircraft weight with fuel	= 3038 lbs.
Weight of fuel (98 gal. x 5.83 lbs./U. S. gal.)	= 571 lbs.
Airplane empty weight (3038 lbs - 571 lbs.)	= 2467 lbs.

3. Airplane Leveling

A. Since there are no perfectly level reference areas on the airplane and the use of digital inclinometers is not common, the airplane is leveled by use of a plumb bob suspended over a fixed reference point in the baggage compartment. Moreover, since the use of jacks with load cells is not prevalent, the wheel scales method is described in this manual. The following steps specify the procedures for installing the plumb bob and leveling the airplane. These steps must be completed before taking readings from the wheel scales.

(1) The airplane must be weighed in a level area.

(2) Remove the left rear seat cushion and place in the footwell. When the cushion is removed, a small washer, which is bonded to the bottom of the seat frame, will be exposed.

(3) Using a string with a plumb bob attached to it, run the string over the gas strut door flange between the flange ball and the point where the baggage door gas strut attaches to the ball, and tie the string off around the front seat belt bracket. Refer to Figure 2.

(4) The pointed end of the plumb bob, in a resting state, will be near a 3/16 inch washer bonded into the seat frame.

(5) Using the two-jack method (raising both wings) discussed in Chapter 7, position the two main tires and the nose tire of the airplane on three scales. Ensure the brakes are set before raising the airplane off the floor. When all of the airplane’s weight is on the three scales, move the jacks to a location that is not under the wings.

(6) It will be necessary to either deflate the nose tire or strut and/or main tires to center the plumb bob point over the

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washer. When the pointer of the plumb bob is over any part of the washer, the airplane is level.

(7) Once the airplane is level, release the brakes.

4. Using The Permanent Reference Point

A. To determine the empty weight center of gravity of the airplane, it is more convenient to work with the permanent reference. The permanent reference point on the airplane is located at the forward part of the wing bottom, in the center of the wing saddle, and is 97.05 inches aft of the datum. The location is shown in Figure 3. There is a pronounced seam at the point where the fuselage is attached to the wing, and the leading edge of the wing bottom is easy to identify.

B. Suspend a plumb bob from the permanent reference point in the exact center as shown in Figure 4.

C. Determine the center point on each tire and make a chalked reference mark near the bottom where the tire touches the floor. On the main gear tires, the mark should be on the inside, near where the arrows point in Figure 4.

D. Create a lateral reference line between the two main gear tires. This can be accomplished by stretching a string between the two chalk marked areas of the tires, snapping a chalk line between these two points, or laying a 7.3-foot board between the points.

5. Measurements

A. Measure the distance along the longitudinal axis from the permanent reference point (tip of the plumb bob) to the lateral reference line between the two tires of the main gear. This is Measurement B in Figure 4.

B. Measure the distance along the longitudinal axis between the plumb bob to the mark on nose tire. This is Measurement B in Figure 4.

6. Converting Measurements To Arms

A. To convert Measurement A and B distances to arms, use the formulas shown in Table 1 and Table 2, respectively.

Table 1. Determining Main Gear Arm

MAIN GEAR Measurement A Distance + 97.05 inches = Main Gear Arm

Table 2. Determining Nose Gear Arm

NOSE GEAR 97.05 inches - Measurement B Distance = Nose Gear Arm

7. Weights and Computations

A. The main gear scale should be capable of handling weight capacities of up to 1200 lbs., while the nose gear scale needs a capacity of at least 750 lbs.

B. Computing the total weight and moments requires seven steps or operations. These seven operations are shown below and also identified in Table 3.

Table 3. Determining Weight and Moments

	Operation No. 1	**Operation No. 2**	**Operation No. 3**		**Operation No. 4**		**Operation No. 5**
Scale Location	**Weight Reading (lbs.)**	**Tare or Scale Error**	**Corrected Weight (lbs.)**	**X**	**Arm (Inches)**	**=**	**Moments (lbs.-inches)**
Right Main Gear	Right Scale Reading	Scale Error	Right Scale Weight ± Error	X	Main Gear Arm	=	Right Gear Moment
Left Main Gear	Left Scale Reading	Scale Error	Left Sale Weight ± Error	X	Main Gear Arm	=	Left Gear Moment
Nose Gear	Nose Scale Reading	Scale Error	Nose Scale Weight ± Error	X	Nose Gear Arm	=	Nose Gear Moment
Total Empty Weight and Empty Moments			**Total Corrected Weight** Operation No. 6				**Total Moments** Operation No. 7

(1) **Operation No. 1** - Enter the weight for each scale into the second column.

(2) **Operation No. 2** - Next, enter the scale error. The scale error is sometimes referred to as the tare and is entered in the third column for each scale.

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(3) **Operation No. 3** - Add or subtract the respective error for the three scales and enter the result into the forth column. This is the correct weight.

(4) **Operation No. 4** - Using the formulas shown in Table 1 and Table 2, determine the arm for the main gear and nose gear. Enter this information into the fifth column.

(5) **Operation No. 5** - Multiply the corrected scale weights times their respective arms to determine the moments for each location. Enter the moments for each computation in the sixth column.

(6) **Operation Nos. 6 and 7** - Sum the weights in the fourth column and the moments in the sixth column.

NOTE: The areas of primary calculations are shaded in gray.

C. The final step, which is to determine the empty center of gravity, is to divide the total moments by the total corrected weight. A detailed example of this computation is shown in Table 5.

8. Example of Empty Center of Gravity (CG) Determination

A. The following is offered as an example problem to aid in understanding the computation process. It is important to remember that the weights, arms, and moments used in the example problem are for demonstration purposes only and **do not** apply to a specific airplane.

B. For the example problem, assume the following:

(1) Scale Weights

(a) Right Main Gear - 924 pounds

(b) Left Main Gear - 923 pounds

(c) Nose Gear - 490 pounds

(2) Scale Error (Tare)

(a) Right Main Gear Scale is -2 pounds

(b) Left Main Gear Scale is -1 pound

(c) Nose Gear Scale is +3 pounds

(3) Measurements

(a) Measurement Distance A is 24.05 inches

(b) Measurement Distance B is 56.15 inches

C. These uncorrected scale weights and tares are shown in Table 4. Note that after correcting for scale error, the right, left, and nose gear weights are 922.0, 922.0, and 493.0 pounds, respectively.

D. The arm for the main gear is computed as follows using the formula in Table 1.

Measurement distance A + 97.05 inches = Main Gear Arm (MGA) or 24.05 inches + 97.05 inches = 121.1 inches MGA

E. The arm for the nose gear is computed as follows using the formula in Table 2.

97.05 inches - Measurement Distance B = Nose Gear Arm (NGA) or 97.05 inches - 56.15 inches = 40.9 inches NGA

F. The main and nose gear arms, as computed, are shown in Table 4.

G. The corrected weights of 922 pounds are then multiplied with the 121.1 inch main gear arm, which produces total moments of 111,654.2 lb-in. In this example, the moments are the same for both the right and left gear since the weights are the same. However, it is not uncommon for the right and left gear weights to vary a few pounds.

H. Next, the corrected 493.0 nose gear weight is multiplied times its 40.9 inch arm, which produces a moment value of 20,163.7 lb-in.

I. Finally, the total moments and corrected weight are summed. In the example below, the total weight is 2,337.0 pounds and the total moment is 243,472.1 lb-in. All this information is summarized in Table 4. All required data for determining the empty center of gravity are now available.

Table 4. Sample Calculation of Weight and Moments

Scale Location	Weight Reading (lbs.)	Tare or Scale Error	Corrected Weight (lbs.)	X	Arm (Inches)	=	Moments (lbs.-inches)

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Right Main Gear	924	-2	922.0	X	121.1	=	111,654.2
Left Main Gear	923	-1	922.0	X	121.1	=	111,654.2
Nose Gear	490	+3	+493.0	X	40.9	=	+20,163.7
Total Empty Weight and Empty Moments			2,337.0				243,472.1

The formula for determining empty weight center of gravity is expressed as follows:

Table 5. Determining Center of Gravity

Total Moments	= Center of Gravity or	243,472.1 lb.-inches	= 104.2 inches
Empty Weight		2 337 lbs.	

J. In the above example the empty center of gravity of the airplane is at fuselage station (FS) 104.2.

9. Changes In The Airplane's Configuration

A. **Determining Location (FS) of Installed Equipment in Relation to the Datum** - If equipment is installed in the airplane, the weight and balance information must be updated. Individuals and companies who are involved with equipment installations and/or modifications are generally competent and conversant with weight and balance issues. These individuals or companies must be aware that the fixed datum is located at fuselage station (FS) 97.05. Refer to Figure 3 and the paragraph Using The Permanent Reference Point for more information.

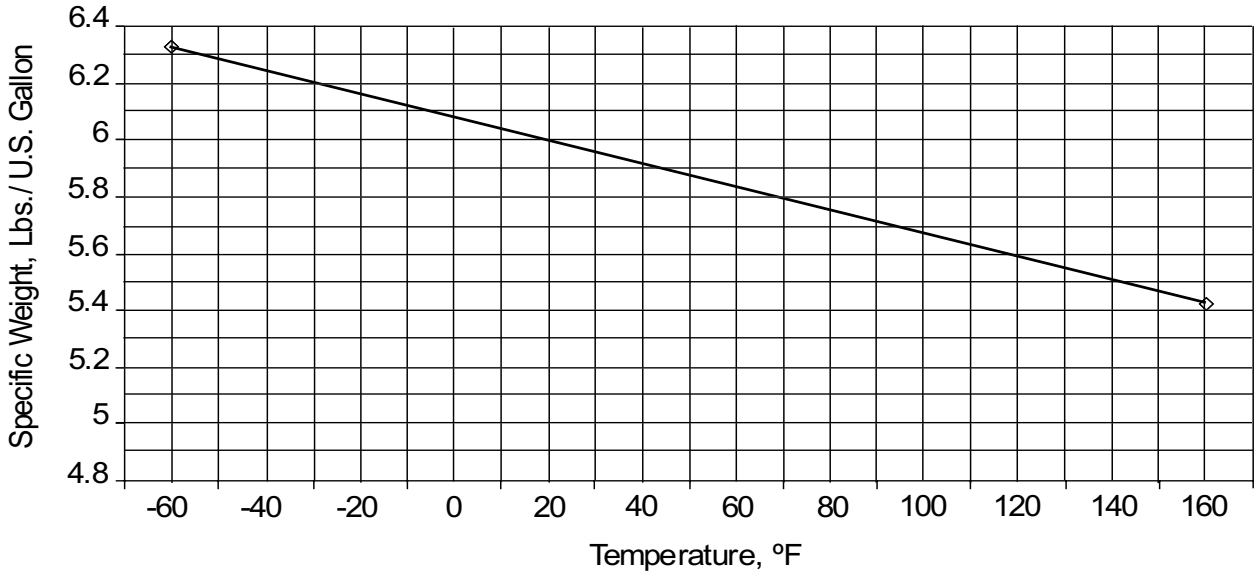
B. **Weight and Balance Forms** - There is a form in Appendix A of Chapter 6 of the *Pilot's Operating Handbook* that is used to track changes in the configuration of the airplane. When equipment is added or removed, these pages or an appropriate approved form must be updated. In either instance the required information is similar.

C. **Updating The Form** - Fill in the date the item is added or removed, a description of the item, the arm of the item, the weight of the item, and the moment of the item. Remember to multiply the weight times the arm of the item to obtain the moment. Finally, compute the new empty weight and empty moment by adjusting the running totals. If an item is removed, subtract the weight and moment of the item from the running totals. If an item is added, add the weight and moment of the item to the running totals.

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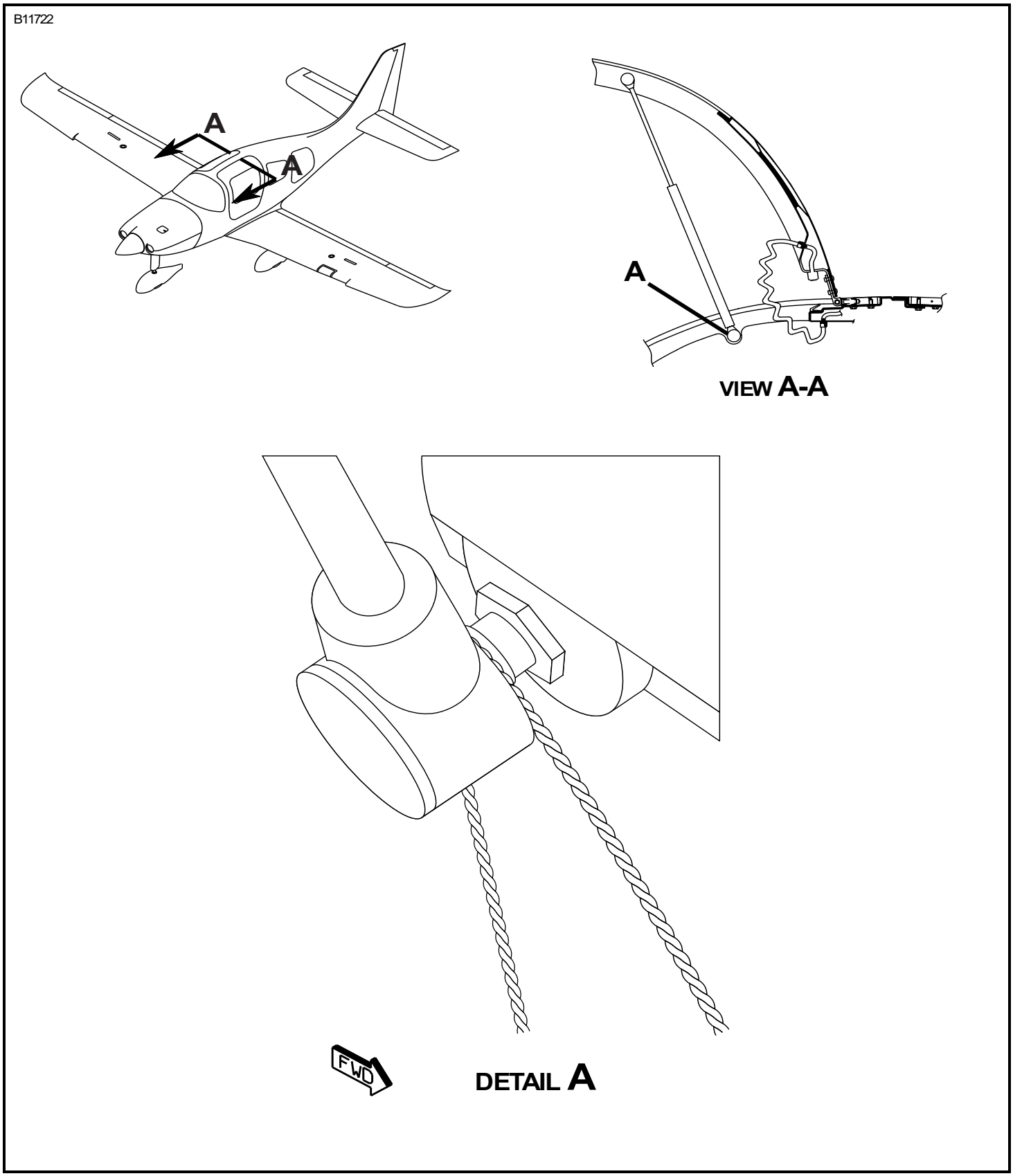
Figure 1 : Sheet 1 : Average Specific Weight of Aviation Gasoline (Mil-F-5572 Grade 100/130 Type) vs. Temperature

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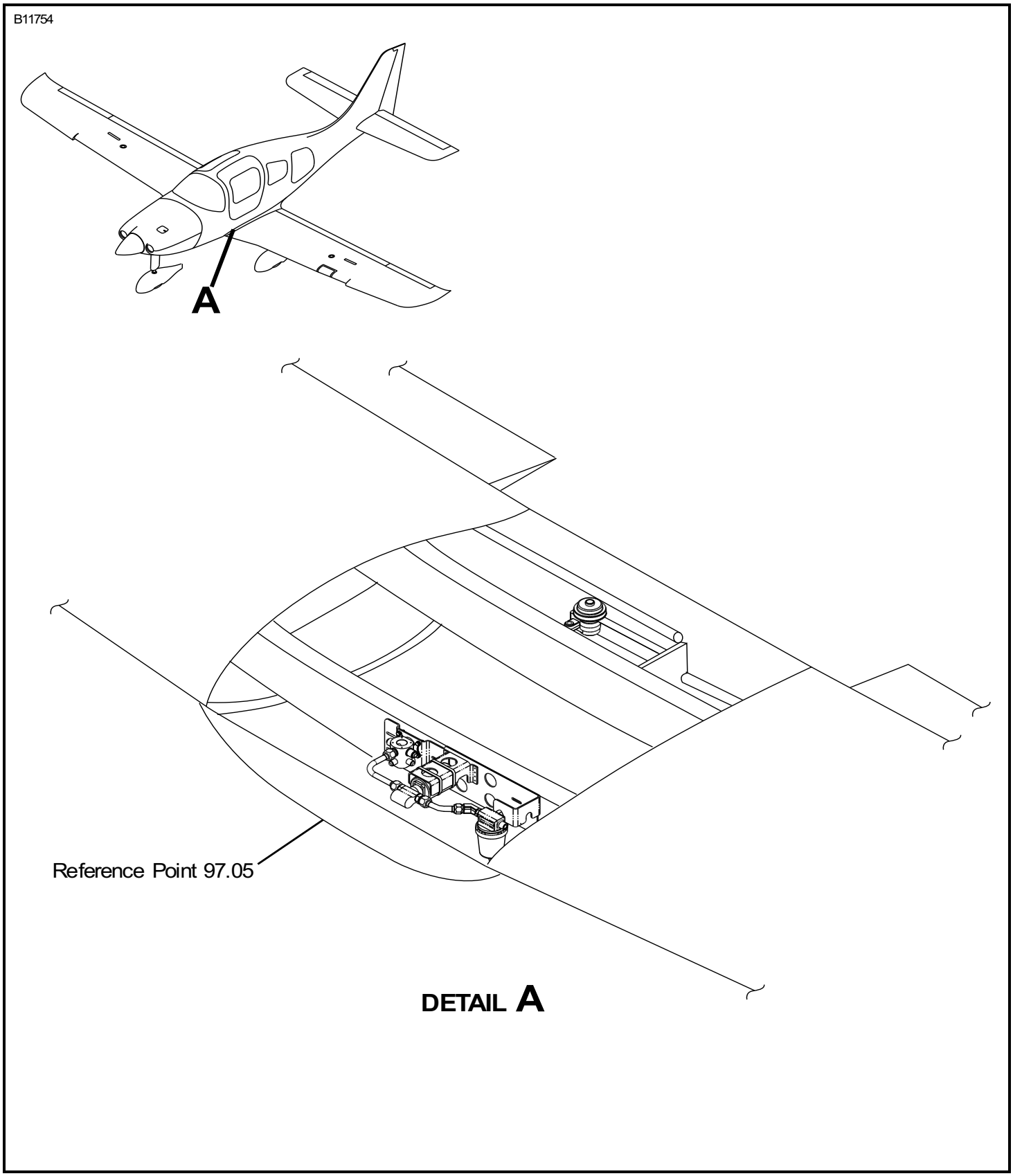
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Figure 2 : Sheet 1 : Attaching String to Gas Strut



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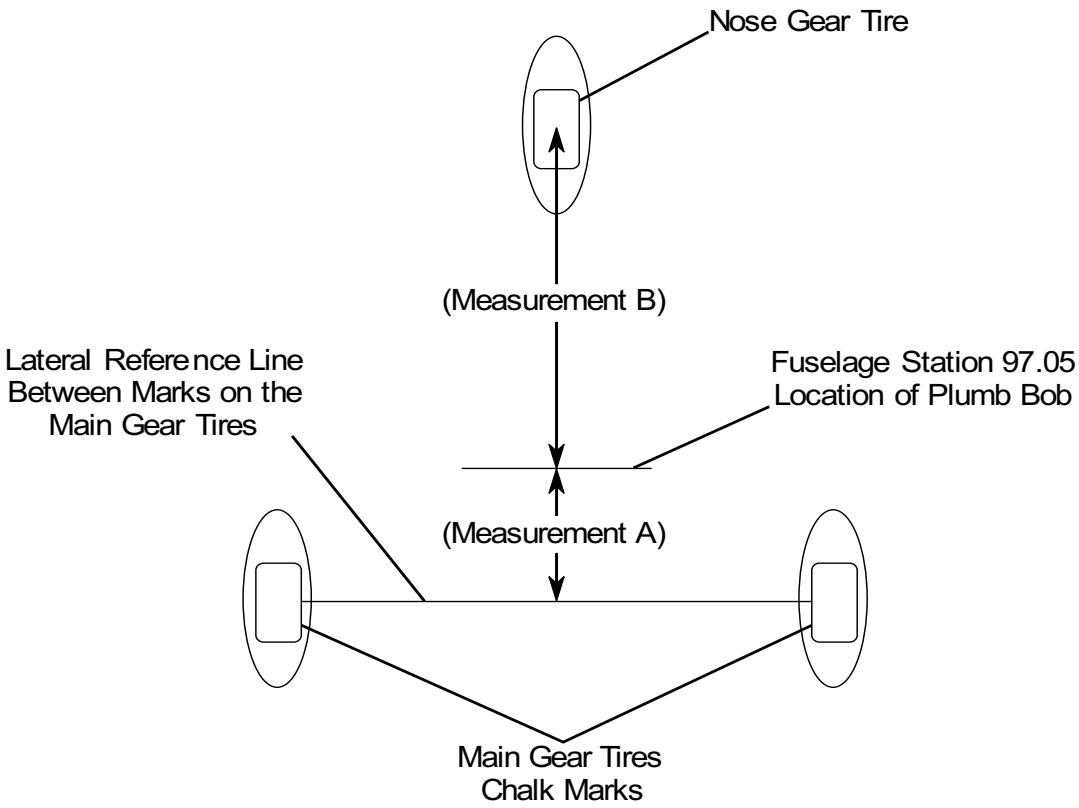
Figure 3 : Sheet 1 : Reference Point for Wing Saddle



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Figure 4 : Sheet 1 : Diagram of Measurements for Weighing

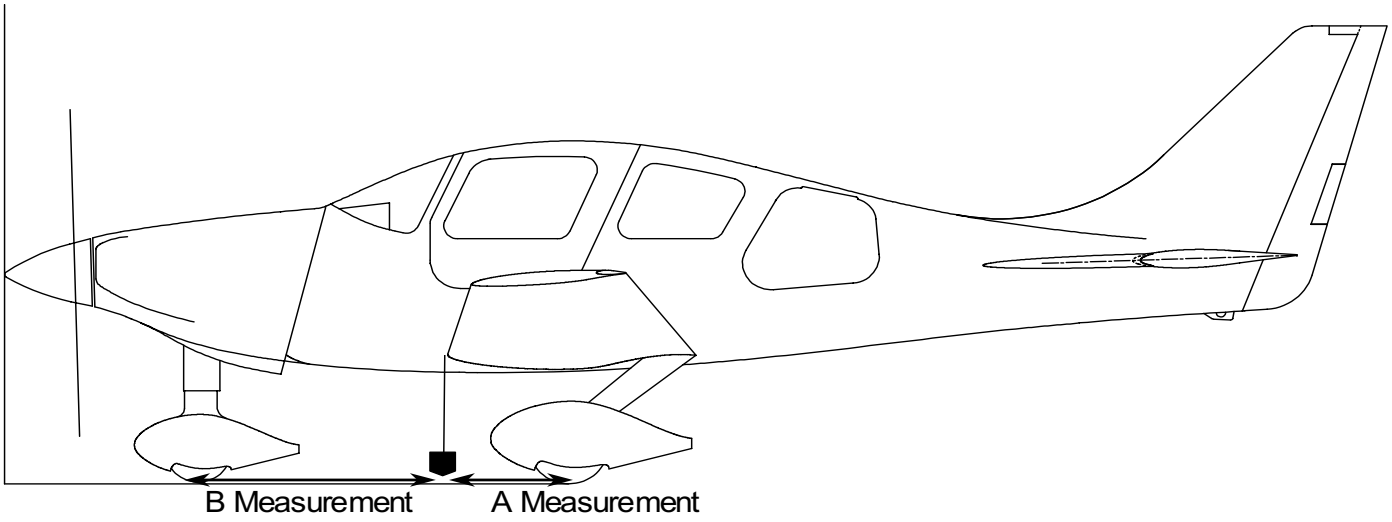
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Figure 4 : Sheet 2 : Diagram of Measurements for Weighing

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OXYGEN - GENERAL

1. General

A. The fixed oxygen system consists of three oxygen cylinders, high and low-pressure lines and fittings, a fill port, an overfill/overpressure burst valve, a regulator/valve assembly, an oxygen distribution manifold, a guarded emergency manual low-pressure oxygen shutoff, a logic and display panel, oxygen distribution devices, and flow meters. High-pressure lines connect the oxygen cylinders, fill port, overfill/overpressure burst valve and the regulator/valve assembly to the system. From the regulator/valve assembly, low-pressure oxygen lines are routed to the overhead console where a manual switch and distribution manifold are located. Attached to the low-pressure manifold and the regulator/valve assembly are pressure transducers that are connected electronically to the display/logic controller mounted in the instrument panel (Basic or Avidyne option) or to the MFD (Garmin G1000 option). Refer to Figure 1.

(1) **Oxygen Bottles** - There are three oxygen cylinders, each with 14 cu ft. capacity at 2000 psi. The oxygen cylinders used are manufactured to DOT-3AL/2015 with a 2015 psi working pressure, per 49 CFR Sect. 178.46. They are located in the right wing between the root rib and the WS 46 rib, and between the forward and aft spars. They are mounted to brackets attached to the lower wing skin. Stainless steel band clamps secure the cylinders to the brackets. The cylinders are fitted with brass discharge fittings that are connected by high-pressure stainless steel tubing and fittings.

(2) Tubing - Stainless steel lines are used throughout the system. The tubing routes high-pressure oxygen from the fill port to the high-pressure manifold on the regulator/valve assembly, and from there to the cylinders and the overfill/overpressure burst valve. All oxygen lines are marked with breathing oxygen identification tape, per MIL-STD-595B and MIL-STD-1247C, located within 3 inches of all fittings and at intervals along the lines not to exceed 48 inches. The ends of the tubing are flared at 37 degrees. The system uses stainless steel, 37 degree JIC fittings. Copper fitting seals (Seco Seals) have been incorporated at every flared tube connection to help ensure leak free connections.

(3) **Flexible Hose** - Behind the forward overhead light panel a flexible hose connects the cabin distribution manifold to the low-pressure side of the system. Flexible hose connection to tubing is accomplished using quick disconnect fittings.

(4) **Fill Port** - The oxygen system fill port is located just inside the baggage door on the baggage hat rack. It consists of a Schrader-type valve, silver soldered to a stainless line, which is routed to the high-pressure manifold. There is a placard cautioning "OXYGEN FILL PORT DO NOT EXCEED 2000 p.s.i." The valve includes a protective cap to screw over the valve when not in use.

(5) **Overfill/Overpressure Burst Valve Assembly** - The system incorporates an overfill/overpressure burst valve assembly that functions to discharge the system in the event it is overpressured. The valve is located in the center of the lower wing skin just under the rear footwell. Stainless steel tubing connects the assembly to the high-pressure manifold located on the root rib. A threaded cartridge bursts when oxygen pressure exceeds 2800 psi. The burst valve exhausts through a hole in the lower wing skin. Under normal conditions there is a pressed-in cap plug that closes the hole in the wing skin and protects the system from dust and dirt. In the event of an overpressure condition, the cap will be blown out of the hole from the internal pressure and act as an additional indication of the system overpressure protection being activated during preflight.

(6) **Oxygen Regulator/Valve** - The oxygen regulator/valve assembly includes the high pressure manifold, high pressure transducer, regulator, low-pressure valve, and valve actuating solenoid. The regulator/valve assembly is located on the inboard side of the root rib and is attached to the root rib with bolts, washers, and locknuts. The high-pressure manifold part of the assembly has three external ports for the fill line, the cylinder line and the overpressure burst valve line. The pressure regulator provides 22 to 33 psig to the cabin distribution manifold. The remote high-pressure control is an electrically operated on-off valve that is actuated by a latching solenoid connected to the aircraft essential electrical bus through the logic and display assembly. There is one low-pressure outlet port on the assembly. The assembly also contains a high-pressure transducer that sends a signal to the logic and display unit to show cylinder pressure. The cylinder pressure directly relates to the quantity of oxygen in the system.

(7) **Cabin Distribution Manifold** - The cabin distribution manifold is located in the overhead console, just behind the pilot's head. It has four quick-disconnect ports.

(8) **Manual Low-Pressure Oxygen Shutoff Valve** - The manual oxygen shutoff valve is located next to the distribution manifold. Its function is to turn off the flow of low pressure oxygen as necessary in case of a failure of the regulator/valve assembly. There is a guard around the switch to prevent inadvertent contact with the shutoff switch.

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(9) **Logic and Display Unit (Basic or Avidyne Option Only)** - The logic and display unit consists of two assemblies: the display located on the lower right of the instrument panel and the remote unit located behind the right kidney panel. The remote unit contains a circuit board, a display board assembly, and a dust cover. The display consists of a three-position switch (ON, OFF, and DISP) that controls the main high-pressure regulator, an LED lit oxygen quantity indicator, and an LED lit low-pressure indicator, and a fault light. The DISP position on the switch turns on the low-pressure and oxygen quantity indicators without energizing the regulator. The ON position turns power on to the display, and energizes the solenoid for a short duration to open the low-pressure valve and provide low-pressure oxygen to the cabin distribution manifold. The OFF position will disengage the solenoid and there will be no display. The logic controller also includes a fault indication for a short to ground, or an open wire for the solenoid wiring only. The system has built-in current protection in case the solenoid wires short each other.

(10) **Oxygen Flow Controls** - Four manually operated oxygen flow controls can be connected to the oxygen distribution manifold. The flow controls provide the means to distribute the appropriate amount of oxygen for the pressure altitude of flight and indicate the presence of flowing oxygen to the pilot or passenger(s). The flow controls can be either an A4 Flowmeter and Oxygen Conserving Cannulas or an A4 Flowmeter and Mask (Standard and Microphone).

(11) **Grounding and Bonding** - All lines are electrically bonded across fittings, and to the oxygen system components to prevent the build up of static electricity. The whole system is electrically bonded to the ground plane of the aircraft for both static electricity and lightning protection.

2. Safety Guidelines For Servicing The Oxygen System

A. In servicing and maintaining oxygen equipment, the greatest potential hazard is fire and explosion. Given conducive conditions, gaseous oxygen will cause combustibles like oil and grease to ignite and burn vigorously or explode. Other materials that do not burn in air will burn vigorously in an oxygen enriched environment. Oxygen equipment must be kept free of all types of oil and grease. Refer to this chapter and FAA Advisory Circular 43.13-1B/1 or subsequent prior to servicing the oxygen system.

(1) **Cleanliness of Tools and Equipment** - Servicing tools and equipment must be cleaned regularly with anhydrous ethyl alcohol, isopropyl alcohol, or other suitable oxygen-safe cleaning agents. After being cleaned and dried, tools and equipment must be protected with caps, plugs, or covers as applicable and stored in a clean storage area.

(2) **Cleanliness of Personnel** - All clothing must be free of oil, grease, fuel, hydraulic fluid, etc. Personnel must be clean without any lotions or lip balms. Thorough washing of hands and face should be adopted as standard practice before handling of any oxygen equipment or servicing tools.

(3) **Grounding** - Ground the system and servicing equipment before connecting to the fill port.

(4) **Electric Power** - Disconnect all unnecessary electrical power from the aircraft. When charging the oxygen system keep operation of electrical switches to the minimum required. Do not connect or disconnect the ground power supply.

(5) **Ignition Sources** - Avoid making sparks. Keep fire and burning cigarettes away from the vicinity of the aircraft during oxygen servicing.

(6) **Fueling** - Do not service or perform maintenance on the oxygen system when fueling, or servicing with other flammable fluid, of the aircraft is in progress.

(7) **Oxygen System Charging** - Do not charge the system too quickly. Rapid charging can generate heat and thus create a hazard to safety.

(8) **Oxygen System Pressure** - The oxygen system is designed to be filled to a maximum pressure of 2000 psi. Verify the pressure of oxygen filling equipment prior to servicing the aircraft to prevent over-pressurizing the system.

3. Oxygen System Servicing

A. Servicing of the oxygen system is as indicated in Chapter 12-10-00.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

4. Oxygen System Purging

A. The oxygen system should be purged anytime the system's pressure drops below 50 psig or the lines are left open. Removal of offensive odors in the system may also be accomplished by purging of the system. Use only MIL-O-27210 aviator's breathing oxygen for system purging.

WARNING: Do not use medical or industrial quality oxygen. They may contain excessive moisture that could

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freeze and cause malfunction of the oxygen system during flight.

(1) Open all aircraft doors to provide air circulation.

(2) Remove all unnecessary electrical power to the aircraft. Energize only the left bus of the aircraft electrical power (Basic or Avidyne option) or the avionics and the right bus (Garmin G1000 option).

(3) Locate the oxygen fill port in the baggage bulkhead.

(4) Ground the oxygen system by connecting a grounding cable from the oxygen fill port to the oxygen recharging cart, ensuring that the cart is properly grounded.

(5) Remove the fill port protective cap and attach the hose from the oxygen recharging cart to the fill port.

(6) Set the recharging cart to deliver 50 psig of oxygen to the system.

(7) Turn on the regulator/valve using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

(8) Open one of the ports in the cabin distribution manifold by using an oxygen mask line or equivalent and ensure that oxygen is flowing.

(9) Purge the system for one hour.

(10) If purging was performed to remove offensive odors, and odors still linger after one hour, purge the system for an additional hour. If offensive odors still persist, replace the supply cylinders, and repurge the system.

(11) Turn off the regulator/valve using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

(12) Turn off the recharging cart.

(13) Remove the oxygen mask line or equivalent from the cabin distribution manifold.

(14) After successful completion of system purging, service the system as described in Chapter 12.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

5. Oxygen System Depressurization And Evacuation

A. The oxygen system must be depressurized and the oxygen removed for any work done to the system that involves removal of component(s).

(1) **Using an Oxygen Cart Capable of Oxygen Recovery**

(a) Open all aircraft doors to allow for air circulation.

(b) Remove all unnecessary electrical power to the aircraft. Energize only the left bus of the aircraft electrical power (Basic or Avidyne option) or the avionics and the right bus (Garmin G1000 option).

(c) Locate the oxygen fill port in the baggage bulkhead.

(d) Ground the oxygen system by connecting a grounding cable from the oxygen fill port to the oxygen cart, ensuring that the cart is properly grounded.

(e) Remove the fill port protective cap and attach the hose from the oxygen cart to the fill port.

(f) Evacuate the oxygen system.

(g) Turn off the oxygen cart.

(2) **Using the Aircraft Cabin Distribution Manifold**

(a) Open all aircraft doors to allow for air circulation.

(b) Remove all unnecessary electrical power to the aircraft. Energize only the left bus of the aircraft electrical power (Basic or Avidyne option) or the avionics and the right bus (Garmin G1000 option).

(c) Turn on the regulator/valve using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

(d) Open one or more of the ports in the cabin distribution manifold by using an oxygen mask line or equivalent.

(e) Wait until all oxygen in the system has discharged.

(f) Remove the oxygen mask line or equivalent from the cabin distribution manifold.

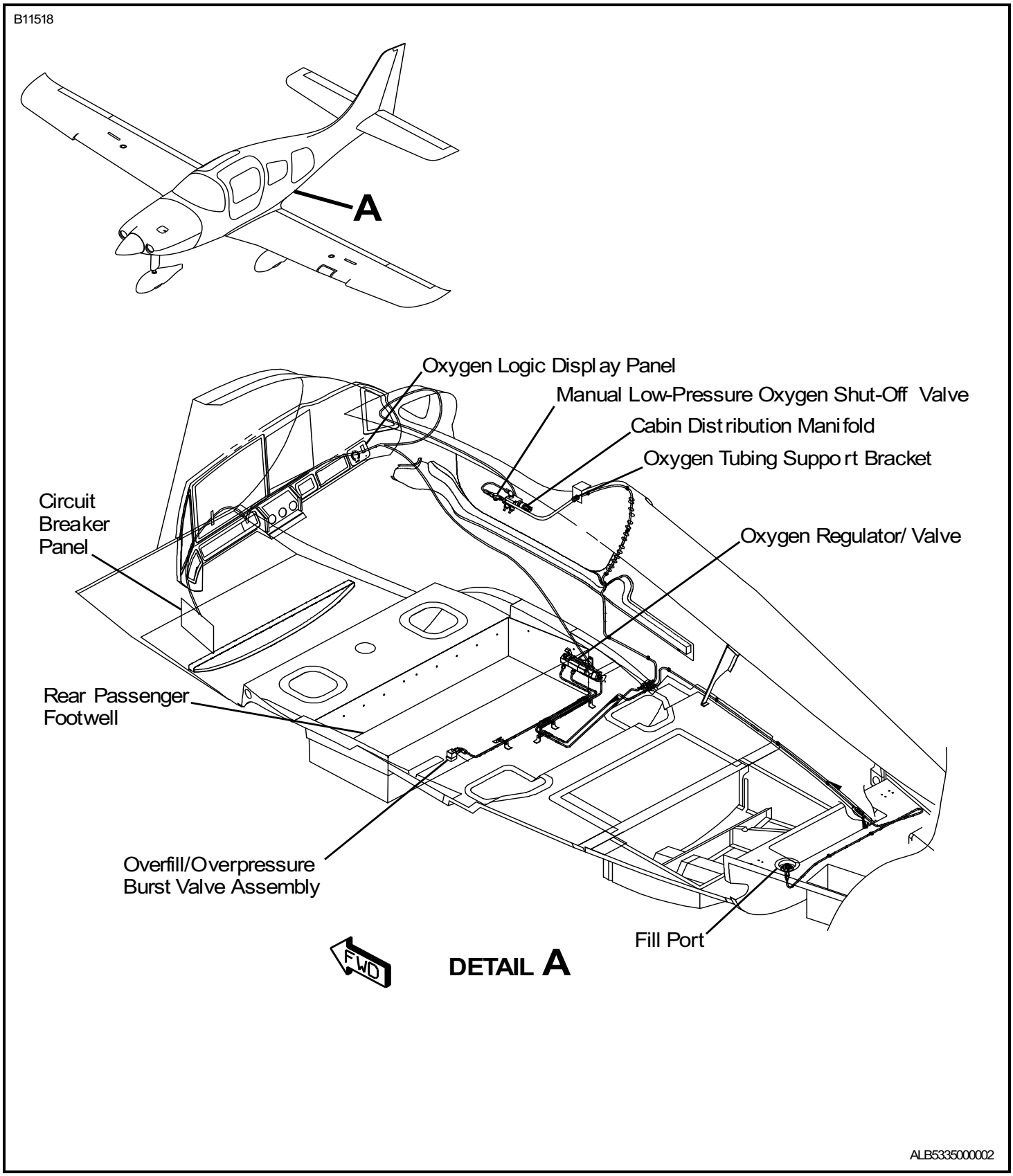
(g) Turn off the regulator/valve from the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

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WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

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Figure 1 : Sheet 1 : Oxygen System



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OXYGEN SYSTEM - MAINTENANCE PRACTICES

1. Oxygen Cylinder

A. Refer to Figure 201 for a diagram of the oxygen cylinders.

B. **Removal of Oxygen Cylinder(s)**

(1) Remove the oxygen regulator/valve as described in the Oxygen Regulator/Valve paragraph.

(2) Cap or plug the open ends of tubing and fittings to protect against contamination.

(3) Disconnect the grounding strap from the cylinder distribution manifolds.

(4) Disconnect tubing from the center cylinder distribution manifold. Discard Seco seals.

(5) Cap or plug the open ends of tubing and fittings to protect against contamination.

(6) Loosen the ring clamp that secures the center cylinder to the cylinder support cradle in the wing.

(7) Slide the center cylinder out through the ring clamp and through the opening in the footwell. Retain the ring clamp.

(8) Remove the cylinder.

(9) Disconnect tubing from the left cylinder distribution manifold and cap or plug the open ends of tubing and fittings to protect against contamination. Discard Seco Seals.

(10) Loosen the ring clamp that secures the cylinder to the cylinder support cradle in the wing.

(11) Slide the cylinder forward until its base clears the rear cradle. The cylinder may now be maneuvered into the center cylinder support cradle and out through the aircraft footwell. Retain the ring clamp.

(12) Repeat the disconnection instructions for the right cylinder.

C. **Installation of Oxygen Cylinder(s)**

(1) Insert the right cylinder, base first, through the opening in the aircraft footwell and maneuver it into the right cradle. Ensure the base of the cylinder seats into the rear of the cylinder cradle.

(2) Slide the ring clamp over the top of the cylinder and under the lip of the forward cylinder cradle.

(3) Insert the left cylinder, base first, through the opening in the aircraft footwell and maneuver it into the left cradle. Ensure the base of the cylinder seats into the rear of the cylinder cradle.

(4) Slide the ring clamp over the top of the cylinder and under the lip of the forward cylinder cradle.

(5) Insert the center cylinder, base first, through the opening in the aircraft footwell and maneuver it into the center cradle. Ensure the base of the cylinder seats into the rear of the cylinder cradle.

(6) Connect the tubing between the left cylinder distribution manifold and the center cylinder distribution manifold. Use new Seco seals.

(7) Connect tubing between the right cylinder distribution manifold and the center cylinder distribution manifold. Use new Seco seals.

(8) Install the oxygen regulator/valve as described in the Oxygen Regulator/Valve paragraph.

(9) Adjust the position of the cylinders within the cylinder cradle until the lower lip of the forward cradle is just aft of the sloped top of the cylinder.

(10) Tighten the ring clamps and torque 25 to 28 in-lbs.

(11) Connect the tubing between the center cylinder distribution manifold and the oxygen regulator/valve.

(12) Reconnect the grounding strap to the cylinder distribution manifolds.

(13) Charge and test the system for leakage per the Leakage Testing paragraph.

(14) If leakage is found, discharge the oxygen and reinstall the tubing and fittings until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary.

(15) Test the system for flow rate per the Flow Testing paragraph.

(16) Purge the system as described in the Oxygen System Purging paragraph.

(17) Replace the rear seat footwell and carpet. Refer to Chapter 25-96-00 for detailed instructions.

(18) Service the system as described in Chapter 12-10-00.

D. **Servicing and Maintenance** - The oxygen cylinders must be inspected whenever malfunction occurs, and hydrostatically

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tested every 5 years.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

(1) Concurrent with hydrostatic testing of the cylinders, all lines and connections should be carefully examined for leaks and abrasions, scratches, or any other damage due to vibration or looseness. Cylinder cradles and ring clamps should be examined for security and to ascertain that they do not damage the cylinder in any way. Check clamps for tightness and re-torque as necessary.

(2) The date of hydrostatic testing must be metal stamped on the shoulder of each cylinder. A cylinder should never be refilled if it is near the expiration point of the most recent date stamp. At the time of hydrostatic testing, all cylinders should be examined for evidence of scratches, abrasions, bulges, or mishandling per Compressed Gas Association Pamphlet, Standards for Visual Inspection of Compressed Gas Cylinders (CGA Pamphlet C-6). If cylinder damage is suspected, properly dispose of the cylinder and replace with new.

2. Tubing

A. **General** - All internal surfaces of tubing, hose assemblies, and fittings must be kept clean, and free of dust and petroleum contaminants. External and internal threads must be protected from contamination and damage prior to use in assembly. Regulators, gauges, and valves should have all open ports closed with oxygen clean and non-shredding polyethylene plugs or caps. Caps and plugs should be green or of a color to distinguish them from caps or plugs used on oily equipment. Where a cap or plug cannot be used, the opening should be sealed in a manner that insures protection.

(1) Masking or self-sealing tapes, paper, pressure sensitive materials, hot dip coatings, rubber, soft plastic or adhesives should not be used. High density plastic closure materials such as Teflon or polyethylene are acceptable.

(2) All items are to be shipped in clean plastic bags with nomenclature tags in or securely attached to the package and shipping container. Nomenclature tags must include title, part number, other information as required and include the following or similar statement: CLEANED FOR OXYGEN SERVICE. DO NOT ALLOW PETROLEUM CONTAMINANTS OF ANY KIND TO BE USED OR STORED ON OR ABOUT THIS CONTAINER. Nomenclature tags must be attached to the package or shipping container in such a way as to allow identification of the item without breaking the container or tag.

(3) No component of the oxygen system is to be internally exposed for more than 10 minutes. If exposed for more than 10 minutes the component must be cleaned.

(4) Any component of the oxygen system that requires cleaning should have contamination blown out using canned or dry filtered air and be cleaned with an oxygen system cleaning compound conforming to MIL-C-8638: Heavy Duty Degreaser C by Micro Care, trichloroethylene, acetone, or hexane. These solvents must be fresh, not having been used for any other purpose. Blow the tubing and fittings clean and dry with a steam of nitrogen (Heavy Duty Cleaner C does not require this step). Visually inspect immediately for cleanliness, and cover with clean protective plugs, caps and bags.

NOTE: All oxygen tubing flare fittings are to be installed with Christo-Lube MCG111 (MIL-PRF-27617) lubricant. All pipe thread joints, on tapered pipe threads only and not on flare fittings, must be wrapped with Teflon tape conforming to MIL-T- 27730A.

B. **O-Ring Installation** - Use only O-rings that conform to MS 28778 which specifies 90 durometer nitrile per MIL-P-5510.

(1) Prior to installation, visually inspect the O-ring for cuts, nicks, and excessive flashing. Most leaks are caused by damaged O-rings. No lubrication is allowed on the O-rings in the oxygen system. Protect O-rings from damage from threads during assembly with a sleeve temporarily installed over the threads.

(2) Ensure the O-ring port is free of contamination prior to assembly. If necessary, remove contaminants with oxygen system cleaning compound conforming to MIL-C- 8638. On tapered pipe thread joints only, wrap the threads with Teflon tape conforming to MIL-T-27730A. When installing O-ring sealed fittings, turn the fitting into the straight thread boss until the metal back-up washer contacts the face of the boss, forcing the O-ring into the port. For elbows and tees, back off the fitting one turn and position the fitting to proper angular direction. Hold the fitting in position while tightening the locknut.

(3) Torque steel to steel O-ring port fittings 95 to140 in.-lbs. (7.9 to 11.7 ft.-lbs.) dry torque value.

(4) Torque steel, aluminum, or brass to any combination of steel, aluminum, or brass O-ring port fittings 70 to 105 in.-lbs. (5.8 to 8.8 ft.-lbs.) dry torque value.

(5) When attaching stainless steel to stainless steel or stainless steel to aluminum, use Christo-Lube MCG111 (MIL-

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PRF-27617) lubricant on the fitting threads. To prevent contamination, do not apply the lubricant to the tapered seat. Use the lower torque value indicated above when lubricant is used.

C. **Removal of Tubing**

(1) Discharge the oxygen system per the Oxygen System Depressurization And Evacuation paragraph.

(2) Remove the rear seats, the center overhead light panel (COLP), aft overhead side panel right (AOSPR), middle lower side panel right (MLSPR), baggage lower side panel right (BLSPR), carpeting on floor and hat rack, and rear seat footwell as required. Refer to Chapter 25-96-00 for detailed instructions.

(3) Remove clamps and grounding strap from the portion of tubing to be removed. Retain hardware. Refer to Figure 202.

(4) Cut tie wraps securing tubing.

(5) Unscrew the flare nut at the end of the tubing and slide it away from the flared end of the fitting.

(6) Disconnect the tubing and remove the Seco seal. Discard the Seco seal.

(7) Temporarily cap or plug the end of the fitting so that contamination will not enter the oxygen system.

(8) Repeat the following steps for the other end of the tubing.

(a) Remove clamps and grounding strap from the portion of tubing to be removed. Retain hardware. Refer to Figure 202.

(b) Cut tie wraps securing tubing.

(c) Unscrew the flare nut at the end of the tubing and slide it away from the flared end of the fitting.

(d) Disconnect the tubing and remove the Seco seal. Discard the Seco seal.

(e) Temporarily cap or plug the end of the fitting so that contamination will not enter the oxygen system.

(9) Remove the tubing.

D. **Installation of Tubing**

(1) Apply Christo-Lube MCG111 (MIL-PRF-27617) lubricant to the male fitting threads. Do not apply compound to the first thread of the fitting or to the tapered seat of the tubing.

(2) Install the Seco seal (dry, no lubricant permitted) onto the male portion of fitting.

(3) Thread the flare nut onto the male fitting several turns with fingers until connection is snug. If the nut cannot be tightened with fingers, disassemble and correct the problem to prevent damage to the Seco seal.

(4) Torque the flare nut to the value in Table 201.

(5) Wait 15 seconds for the Seco seal to compress and deform.

(6) Do not loosen the flare nut. Torque the flare nut again to the value in Table 201.

(7) Charge and test the system for leakage per the Leakage Testing paragraph.

(8) If leakage is found, discharge the oxygen and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.

(9) Fittings equipped with O-rings are installed as indicated in the Tubing - O-Ring Installation paragraph. Remove old O-rings and replace with new.

(10) Test the system for flow rate per the Flow Testing paragraph.

(11) Secure the tubing to tie blocks with tie wrap.

(12) Reinstall the grounding strap across all affected fittings. Refer to Figure 202.

(13) Mark all tubing with breathing oxygen label tape, per MIL-STD-595B and MIL-STD- 1247C, located within 3 inches of all fittings and at intervals along the lines not to exceed 48 inches.

(14) Purge the system as described in the Oxygen System Purging paragraph.

(15) Reinstall the rear seat footwell, carpeting on the floor and the hat rack, and the interior panels. Refer to Chapter 25-96-00 for detailed instructions.

(16) Service the system as described in Chapter 12-10-00.

Table 201. Flare Nut Torque with Seco Seals

Tube Flare Nut Torque (with Seco seals)

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Size	Tube O.D.	Material and Torque Specification			
		Tube Material	Fitting Material	Threads PER MIL-S-8879	*Inch-Pounds
- 2	0.125	SST	Steel	.3125-24UNJF-3B	35-50
		Aluminum	Steel	.3125-24UNJF-3B	22-30
- 3	0.188	SST	Steel	.375-24UNJF-3B	75-100
		Aluminum	Steel	.375-24UNJF-3B	30-45
- 4	0.250	SST	Steel	.4375-20UNJF-3B	115-150
		Aluminum	Steel	.4375-20UNJF-3B	40-60

* Wet torque values shown

E. **Servicing and Maintenance** - The oxygen system tubing must be inspected every 5 years or whenever malfunction occurs.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

3. Fill Port

A. Refer to Figure 203 for a diagram of the fill port.

B. **Removal of Fill Port**

(1) Discharge the oxygen system per the Oxygen System Depressurization And Evacuation paragraph.

(2) Remove the hat rack carpet. Refer to Chapter 25-30-00 for detailed instructions.

(3) Remove the baggage bulkhead access panel by unscrewing the attachment screws.

(4) Disconnect the tubing.

(5) Temporarily seal the end of the fitting so that contamination will not enter the oxygen system.

(6) Remove the three pan head screws, nuts, and washers that secure the fill port to the baggage bulkhead.

(7) Remove the fill port, screw on cap, and bezel.

C. **Installation of Fill Port**

(1) Insert the fill port into the baggage bulkhead.

(2) Place the bezel over the fill port and align all screw holes with the holes in the baggage bulkhead.

(3) Install three pan head screws, washers, and nuts ensuring that the screw on cap chain and the grounding strap are attached to the fill port. Tighten the screws snug.

(4) Attach tubing.

(5) Charge and test the system for leakage as described in the Leakage Testing paragraph.

(6) If leakage is found, discharge the oxygen and reattach the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary.

(7) Purge the system as described in the Oxygen System Purging paragraph.

(8) Reinstall the baggage bulkhead access panel.

(9) Reinstall the hat rack carpet. Refer to Chapter 25-30-00 for detailed instructions.

(10) Service the system as described in Chapter 12-10-00.

D. **Servicing and Maintenance** - The fill port must be inspected every 5 years or whenever malfunction occurs.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

4. Overfill/Overpressure Burst Valve Assembly

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A. Refer to Figure 204.

B. **Removal and Installation of the Burst Cartridge** - Replacement of the burst cartridge is only required when the cartridge has burst due to an overfill or overpressure of the oxygen system.

(1) Locate the exit orifice of the burst valve assembly in the underside of the wing.

(2) Using a hex wrench inserted into the orifice, unscrew the burst cartridge from the burst valve assembly.

(3) Installation is the reverse of removal.

(4) Install a green plug to cap the burst valve at the opening in the underside of the wing.

(5) Purge the system per the Oxygen System Purging paragraph.

(6) Service the system per Chapter 12-10-00.

C. **Removal of Overfill/Overpressure Burst Valve Assembly**

(1) Discharge the oxygen system per the Oxygen System Depressurization And Evacuation paragraph.

(2) Remove the rear seat carpet and footwell. Refer to Chapter 25 for detailed instructions.

(3) Disconnect tubing and remove the Seco seals. Discard the Seco seals.

(4) Temporarily cap or plug the end of the tubing so that contamination will not enter the oxygen system.

(5) Remove the grounding strap from the burst valve. Retain hardware.

(6) If present, remove the green plug that caps the burst valve at the underside of the wing.

(7) From the underside of the wing, unscrew the four nylon screws that hold the burst valve to the fuselage.

(8) Remove the valve.

(9) Temporarily cap or plug the end of the fitting or tubing so that contamination will not enter the oxygen system.

D. **Installation of Overfill/Overpressure Burst Valve Assembly**

(1) Align all screw holes in the burst valve with the holes in the fuselage.

(2) With an assistant, install the four nylon screws and torque to 5 in-lbs.

(3) Attach tubing.

(4) Charge and test the system for leakage per the Leakage Testing paragraph.

(5) If leakage is found, discharge the oxygen and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.

(6) Test the system for flow rate per the Flow Testing paragraph.

(7) Reinstall the grounding strap to the burst valve.

(8) Purge the system per the Oxygen System Purging paragraph.

(9) Install the green plug to cap the burst valve at the opening in the underside of the wing.

(10) Reinstall the rear seat footwell and carpet. Refer to Chapter 25-96-00 for detailed instructions.

(11) Service the system per Chapter 12-10-00.

E. **Servicing and Maintenance** – The overfill/overpressure burst valve must be inspected every 5 years or whenever malfunction occurs.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

5. Oxygen Regulator/Valve

A. Refer to Figure 205.

B. **Removal of Regulator/Valve**

(1) Discharge the oxygen system per the Oxygen System Depressurization And Evacuation paragraph.

(2) Remove the rear seat carpet and footwell. Refer to Chapter 25-96-00 for detailed instructions.

(3) Remove the clamp and grounding strap from the tubing as required. Retain hardware.

(4) Disconnect the electrical connector.

(5) Disconnect all tubing from the regulator/valve.

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(6) Discard the Seco seals.

(7) Cap or plug the open ends of tubing and the regulator/valve inlets and outlets to protect against contamination.

(8) Remove the two hex head cap screws, washers, and nuts holding the regulator/valve to the wing rib.

(9) Remove the regulator/valve.

C. **Installation of Regulator/Valve**

(1) Align screw holes in the regulator/valve with the holes in the wing rib.

(2) Install the two hex head cap screws, washers, and nuts ensuring that the grounding strap is attached to the upper screw. Tighten 30 to 36 in-lbs.

(3) Attach fittings and tubing.

(4) Install the clamps and grounding strap.

(5) Fittings equipped with O-rings are installed as indicated in Tubing - O-Ring Installation. Remove the old O-ring and replace with new.

(6) Charge and test the system for leakage per the Leakage Testing paragraph.

(7) If leakage is found, discharge the oxygen, and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.

(8) Test the system for flow rate per the Flow Testing paragraph.

(9) Purge the system per the Oxygen System Purging paragraph.

(10) Replace the rear seat footwell and carpet. Refer to Chapter 25-96-00 for detailed instructions.

(11) Service the system as described in Chapter 12-10-00.

D. **Servicing and Maintenance** - The regulator/valve must be inspected whenever malfunction occurs, and overhauled every 5 years. During overhaul replace o-rings and verify the regulator pressure setting.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

6. Manual Low-Pressure Oxygen Shutoff Valve

A. Refer to Figure 206, Figure 207, and Figure 208.

B. **Removal of the Manual Low Pressure Oxygen Shutoff Valve**

(1) Turn off the low-pressure oxygen supply to the cabin distribution manifold using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

(2) Remove the forward center overhead light panel per Chapter 25-30-00.

(3) Remove the clamps and grounding strap as required. Retain hardware.

(4) Disconnect the flexible oxygen hose at the quick disconnect bulkhead connector. Refer to Figure 208.

(5) Remove the nuts and switch cover attaching the shutoff valve to the overhead light panel.

(6) Disconnect the tubing and remove the Seco seals. Discard the Seco seals.

(7) Temporarily cap or plug any open fittings or tubing so that contamination will not enter the oxygen system or the valve.

(8) Remove the shutoff valve.

C. **Installation of the Manual Low-Pressure Oxygen Shut-Off Valve**

(1) Install the valve with the toggle facing forward, and port nearest mounting threads on the forward side. It may be necessary to remove the toggle pivot pin and rotate the toggle 180º to have the valve in the open position with the toggle facing forward. When the valve toggle is placed in the neutral position (straight up) and released, it should default to the closed (aft) position.

(2) Place the switch cover over the shut-off valve toggle switch, and attach the assembly to the overhead light panel using the nuts. Apply Loctite 242 to the nuts before installation.

(3) Attach the tubing and the flexible oxygen hose to the shutoff valve.

(4) Connect the flexible oxygen hose to the quick disconnect bulkhead connector. Refer to Figure 208.

(5) Install the clamps and grounding strap.

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(6) Charge and test the system for leakage as described in paragraph 9.

(7) If leakage is found, reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.

(8) Test the system for flow rate per the Flow Testing paragraph.

(9) Reinstall the forward center overhead light panel per Chapter 25-30-00.

(10) Service the system as described in Chapter 12-10-00.

D. **Servicing and Maintenance** – The shutoff valve must be inspected every 5 years or whenever malfunction occurs.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

7. Cabin Distribution Manifold

A. Refer to Figure 207 and Figure 208.

B. **Removal of Cabin Distribution Manifold**

(1) Turn off the low-pressure oxygen supply to the cabin distribution manifold using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

(2) Remove the forward center overhead light panel per Chapter 25-30-00.

(3) Disconnect the electrical connector from the manifold.

(4) Remove the clamps and grounding strap as required. Retain hardware.

(5) Disconnect the flexible oxygen hose at the quick disconnect bulkhead connector. Refer to Figure 208.

(6) Disconnect the tubing from the manifold and remove the Seco seals. Discard the Seco seals.

(7) Temporarily cap or plug all open tubing and fittings so that contamination will not enter the oxygen system or the manifold.

(8) Remove the four pan head screws affixing the cover plate and the cabin distribution manifold support bracket to the overhead light panel.

(9) Remove the four cap screws and washers holding the manifold to the manifold support bracket.

(10) Remove the manifold.

C. **Installation of the Cabin Distribution Manifold**

(1) Align the screw holes in the manifold with the holes in the manifold support bracket and install the four cap screws and washers. Torque the screws 20 to 25 in.-lbs.

(2) Install the four pan head screws affixing the cover plate and the cabin distribution manifold support bracket to the overhead light panel. Tighten the screws snug.

(3) Attach the tubing.

(4) Connect the flexible oxygen hose to the quick disconnect bulkhead connector. Refer to Figure 208.

(5) Connect the electrical connector.

(6) Install the clamps and grounding strap.

(7) Charge and test the system for leakage as described in the Leakage Testing paragraph.

(8) If leakage is found, reinstall the tubing fittings until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.

(9) Fittings equipped with O-rings are installed as indicated in Tubing - O-Ring Installation. Remove the old O-rings and replace with new.

(10) Test the system for flow rate per the Flow Testing paragraph.

(11) Reinstall the forward center overhead light panel per Chapter 25-30-00.

(12) Service the system as described in Chapter 12-10-00.

D. **Servicing and Maintenance** - The cabin manifold must be inspected every 5 years or whenever malfunction occurs.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

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8. Logic and Display Unit (Basic or Avidyne Option Only)

A. Refer to Figure 209.

B. **Removal of the Logic and Display Unit**

(1) Remove the kidney panel as described in Chapter 25. This will give access to the Logic and Display Unit.

(2) Disconnect the 25 pin connector from the logic and display unit.

(3) The logic and display unit display panel has four threaded 6-32 studs that protrude through mating holes into the backside of the instrument panel. Four self-locking 6- 32 nuts with washers, accessible from the backside of the instrument panel, screw onto these studs to secure the display panel to the instrument panel. Removal of the nuts and washers allows the display panel to slide out for access to, and disconnection of, the bayonet plugs and ribbon cable connector. Make note or label each plug for reconnection to its proper location.

(4) Remove the three pan head screws and washers securing the Logic and Display Unit to the cross member at the back of the instrument panel and remove the unit.

C. **Installation of Logic and Display Unit** - Installation is the reverse of removal.

D. **Servicing and Maintenance** - The oxygen system logic and display unit requires no service or maintenance but must be inspected annually, or whenever malfunction occurs.

9. Leakage Testing

A. Any removal and replacement of components or tubing in the oxygen system requires that leakage testing be performed. Apply oxygen safe leak detection compound conforming to MIL-L-25567 to the affected fittings and connections. Observe the tested area for a minimum of 30 seconds. Any formation of bubbles or deformation of the leak detection compound constitutes failure of the test.

(1) Verify that the solenoid valve operates properly when the oxygen control panel is electrically switched on and off.

(2) If complete evacuation of oxygen from the system was required, perform the following tests:

(a) Charge the system to 50 psig with oxygen and verify there are no leaks.

(b) Increase the system pressure to 400 psig and verify there are no leaks.

(c) Increase the system pressure to 1,000 psig and verify there are no leaks.

(d) Increase the system pressure to 2,000 psig and verify there are no leaks.

(3) Any repair work to the low-pressure portion of the oxygen system from the regulator/valve to the cabin distribution manifold may not require complete evacuation of oxygen from the system. In this case test the system at its current pressure and verify there are no leaks.

(4) Remove all leak test compound from fittings and connections after successful completion of the leak test.

10. Flow Testing

A. Connect an oxygen flow meter to one of the overhead cabin manifold ports.

B. Toggle the oxygen logic and display panel switch to ON.

C. Immediately check the oxygen flow and verify that it is a minimum of 0.43 cu. ft. per minute. Flow of less than 0.43 cu. ft. per minute is indication of a constriction in the system or a malfunction of one or more system components.

D. Immediately switch the oxygen logic and display panel switch to OFF.

E. Disconnect the oxygen flow meter.

11. Disconnection And Reconnection For Wing Removal

A. Refer to Figure 210.

B. **Disconnection**

(1) Discharge the oxygen system as indicated in the Oxygen System Depressurization And Evacuation paragraph.

(2) Remove the clamps and grounding strap from the tubing at the wing to fuselage connection. Retain hardware.

(3) Cut the tie wrap securing the tubing to the fuselage stand-off.

(4) Disconnect the tubing at the wing to fuselage connection, and remove the Seco seals. Discard the Seco seals.

(5) Temporarily cap or plug the tubing and fittings so that contamination will not enter the oxygen system.

(6) Proceed with wing removal per Chapter 57-10-00.

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C. **Reconnection**

(1) Connect the tubing at the wing to fuselage connection.

(2) Charge and test the system for leakage as described in the Leakage Testing paragraph.

(3) If leakage is found, discharge the oxygen and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.

(4) Secure the tubing to fuselage stand-off with a tie wrap.

(5) Reinstall the grounding strap across the fittings at the wing to fuselage connection.

(6) Proceed with wing installation per Chapter 57-10-00.

(7) Service the oxygen system as described in Chapter 12-10-00.

WARNING: Refer to Safety Guidelines For Servicing The Oxygen System before attempting any servicing or maintenance of the oxygen system.

12. Oxygen System Troubleshooting

Table 202. Oxygen System Troubleshooting

Condition	Probable Cause	Remedy
Excessive leakage of system pressure.	Fill Port leaking.	Replace the fill port.
	Overfill/overpressure burst valve blown.	Replace the overfill/overpressure burst cartridge.
	Leak in lines.	Check tubing, fittings, and connections and repair or replace as necessary. Refer to Oxygen System Leaks below.
No low pressure reading at the logic and display panel (Basic or Avidyne option) or on the MFD (Garmin G1000 option)	System not charged.	Replenish the oxygen supply.
	Defective logic and display unit or panel.	Replace the logic and display unit or panel.
	Defective low pressure transducer in cabin distribution manifold.	Replace the low pressure transducer.
No system oxygen quantity reading at logic and display panel (Basic or Avidyne option) or on the MFD (Garmin G1000 option)	System not charged.	Replenish the oxygen supply.
	Defective logic and display unit or panel.	Replace the logic and display unit or panel.
	Defective high pressure transducer and switch in the regulator/valve.	Replace the regulator/valve.
Oxygen does not flow from the masks when the system is turned on.	System not charged.	Replenish oxygen supply.
	Oxygen mask defective.	Overhaul or replace mask.
	Defective low-pressure shutoff valve.	Replace the low-pressure shutoff valve.
	Defective high pressure transducer and switch in the regulator/valve.	Replace the regulator/valve.

A. **Oxygen System Leaks** – With the oxygen system turned off, leakage is most likely to occur in the high pressure side of the system.

(1) Remove the hat rack carpet and cushion, the aft baggage bulkhead access panel, the baggage lower side panel right (BLSPR), and the middle lower side panel right (MLSPR). Refer to Chapter 25 for specific instructions.

(2) Perform leakage testing per the Leakage Testing paragraph on the fill port, tubing, fittings, and connections accessible in this area of the aircraft.

(3) Repair any leaks and perform leakage testing. If no leaks are found in this area or the system is still losing pressure after leaks are repaired, the remainder of the high pressure side of the system must be exposed and tested as indicated below.

(4) Remove the flap and aileron push rod access panel on the underside of the plane, and the aft cabin footwell to gain

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access to the overfill/overpressure burst valve and the oxygen regulator/valve.

(5) Perform leakage testing per the Leakage Testing paragraph on the overfill/overpressure burst valve, oxygen regulator/valve, oxygen cylinders, tubing, fittings, and connections.

(6) Repair any leaks and perform leakage testing.

(7) Upon successful completion of leakage testing replace all access panels, the footwell, interior panels, hat rack and carpeting.

(8) Service the oxygen system as described in Chapter 12-10-00.

13. Annual Inspection with Garmin G1000 System

A. If the Garmin G1000 system is installed perform the following steps in addition to regular annual inspection of the oxygen system.

(1) Turn on both batteries, and the avionics to validate the Oxygen system.

(2) If the oxygen system is charged beyond 250 PSI no oxygen quantity annunciation should be displayed. Note: the oxygen system must have some charge in order to be verified by these procedures.

(3) Plug an oxygen mask with a flow indicator into the distribution manifold. On the MFD, full screen EIS page, press the oxygen soft key, and turn the oxygen switch at the distribution manifold to ON.

(4) The G1000 should issue and advisory for oxygen on (while on the ground), Check the oxygen outlet pressure is in the green band, and the mask is flowing oxygen.

(5) Press the oxygen soft key on the G1000, the oxygen outlet pressure should decay to zero as the system runs out of air, and the mask should stop flowing.

(6) Turn the oxygen switch at the distribution manifold to OFF, and then press the oxygen soft key on the G1000 to turn the oxygen on again.

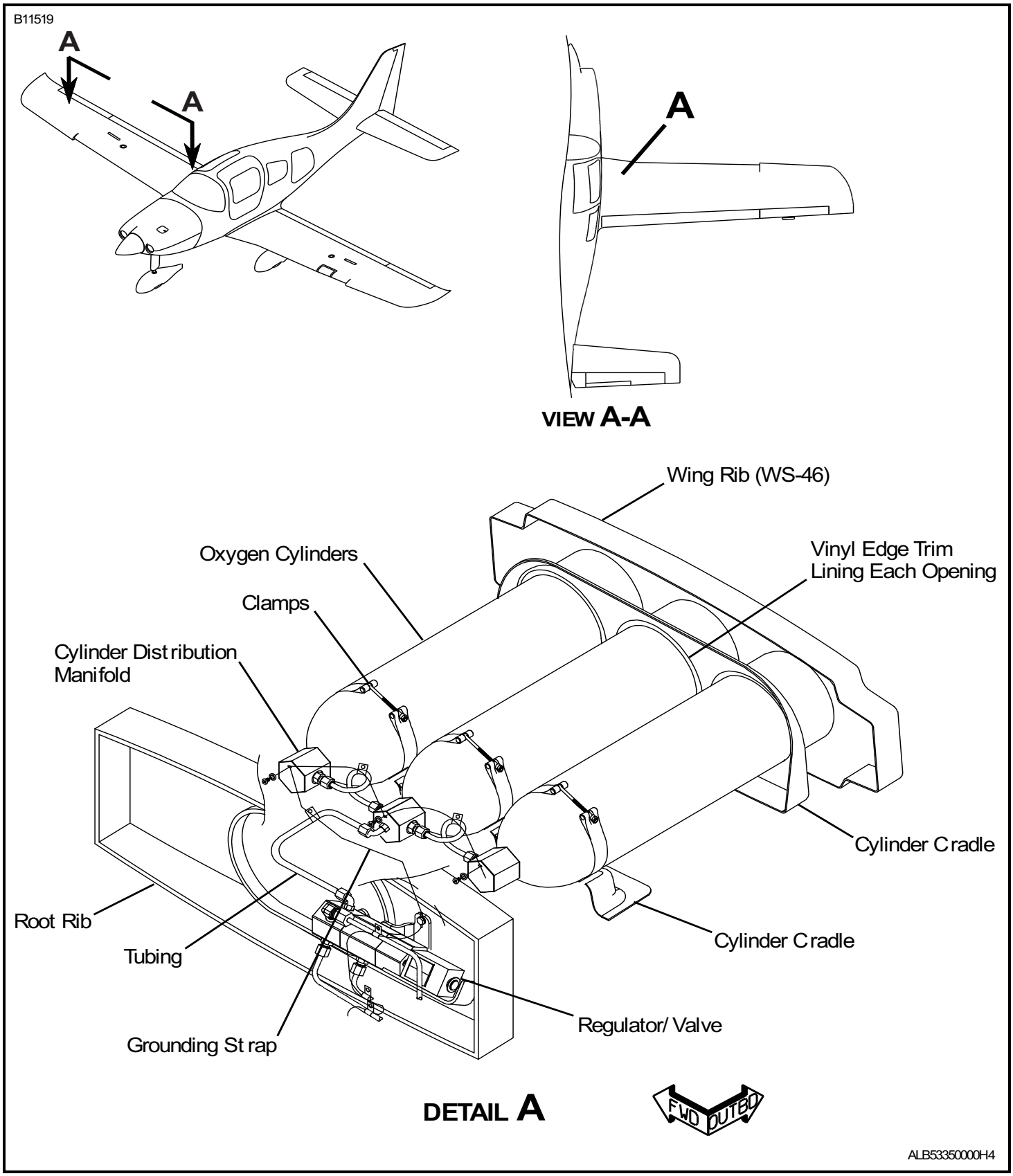
(7) Verify the oxygen outlet stays at zero, no oxygen is flowing to the mask and the G1000 annunciates oxygen pressure - no flow detected but the oxygen system is turned on.

(8) Turn the oxygen switch at the distribution manifold to ON and verify the outlet pressure increases and the oxygen pressure annunciation on the G1000 clears.

(9) Press the oxygen soft key, turning the oxygen system off, clearing all oxygen annunciations.

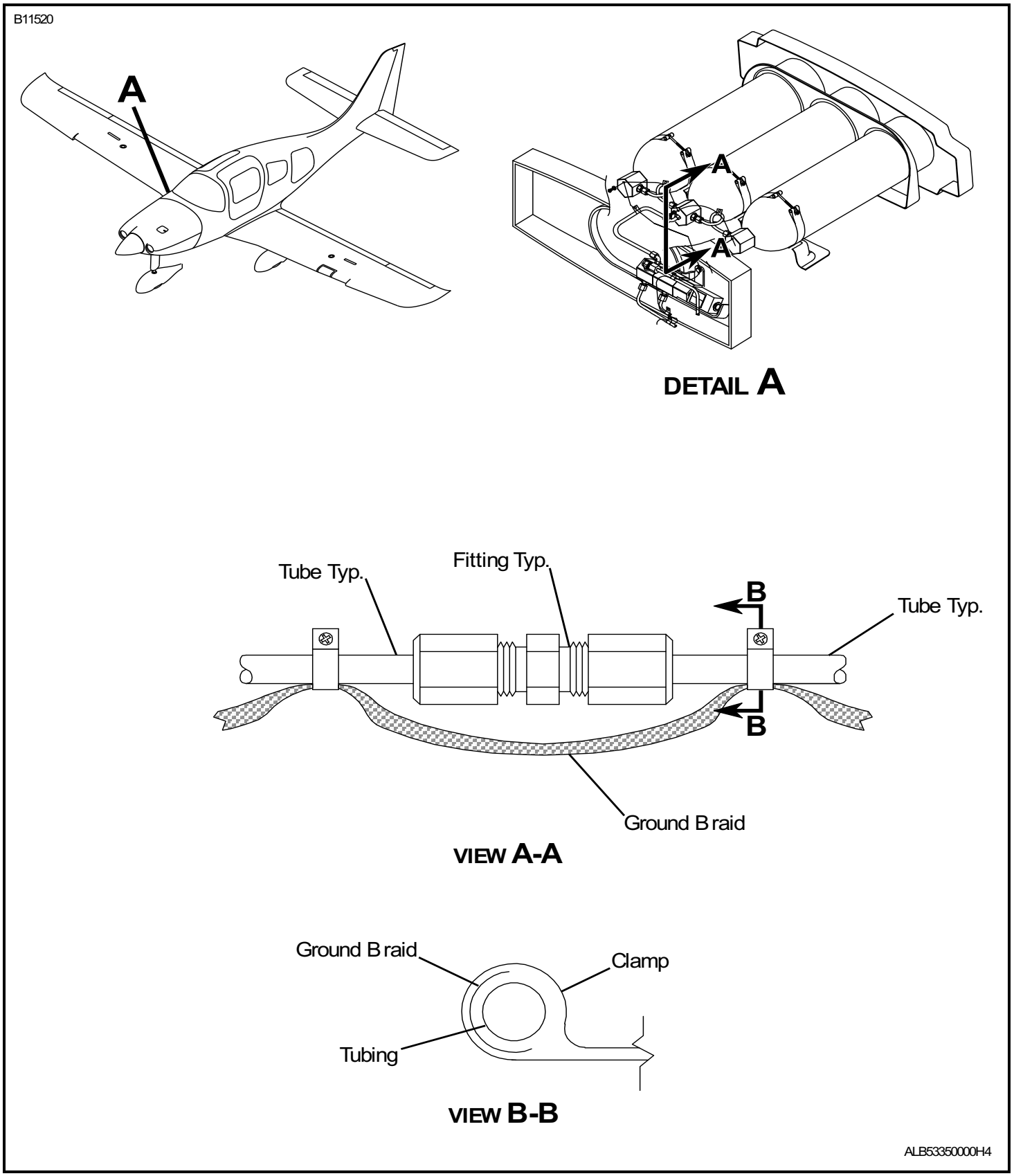
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Figure 201 : Sheet 1 : Oxygen Cylinders



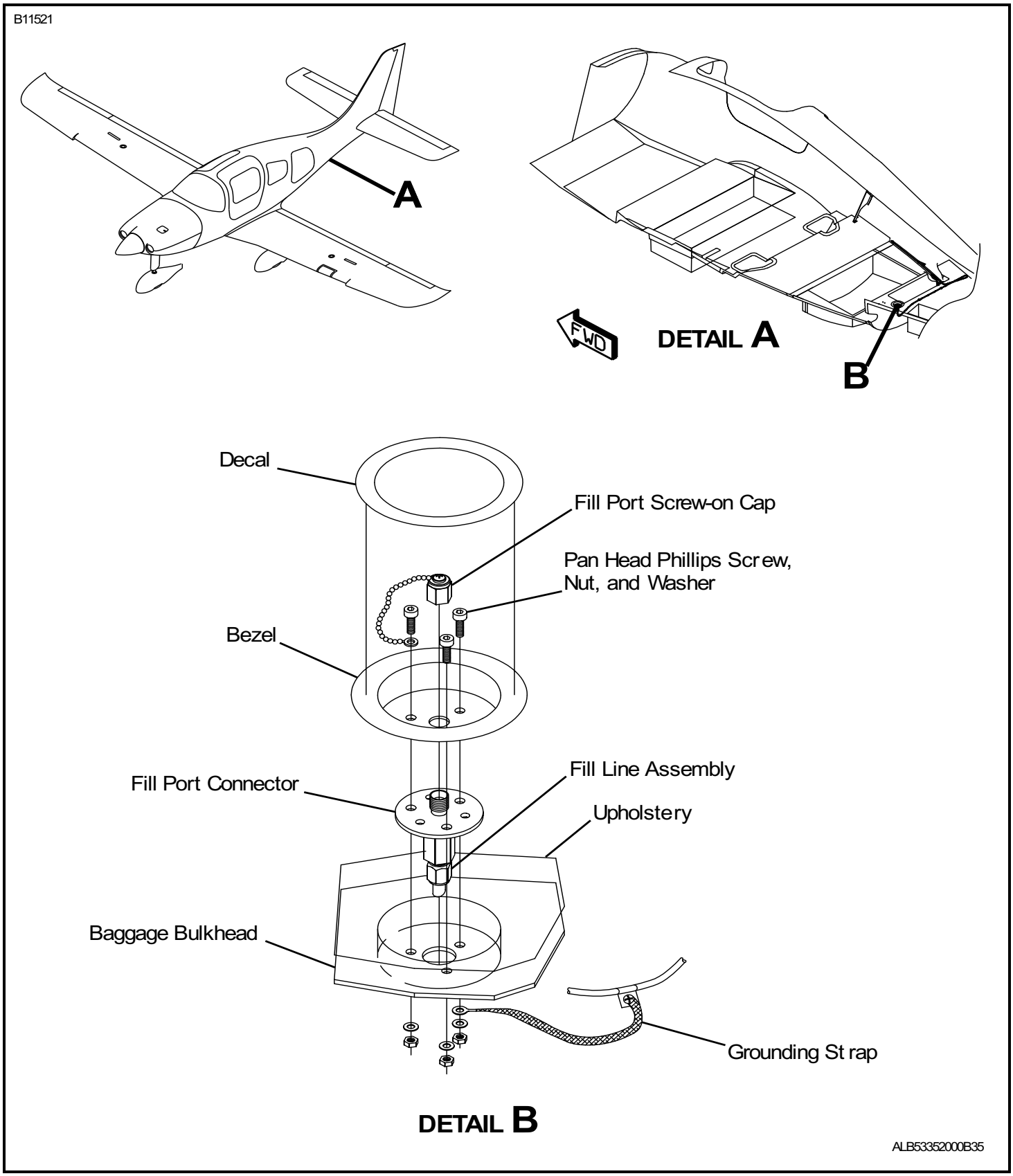
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Figure 202 : Sheet 1 : Typical Grounding Strap Installation



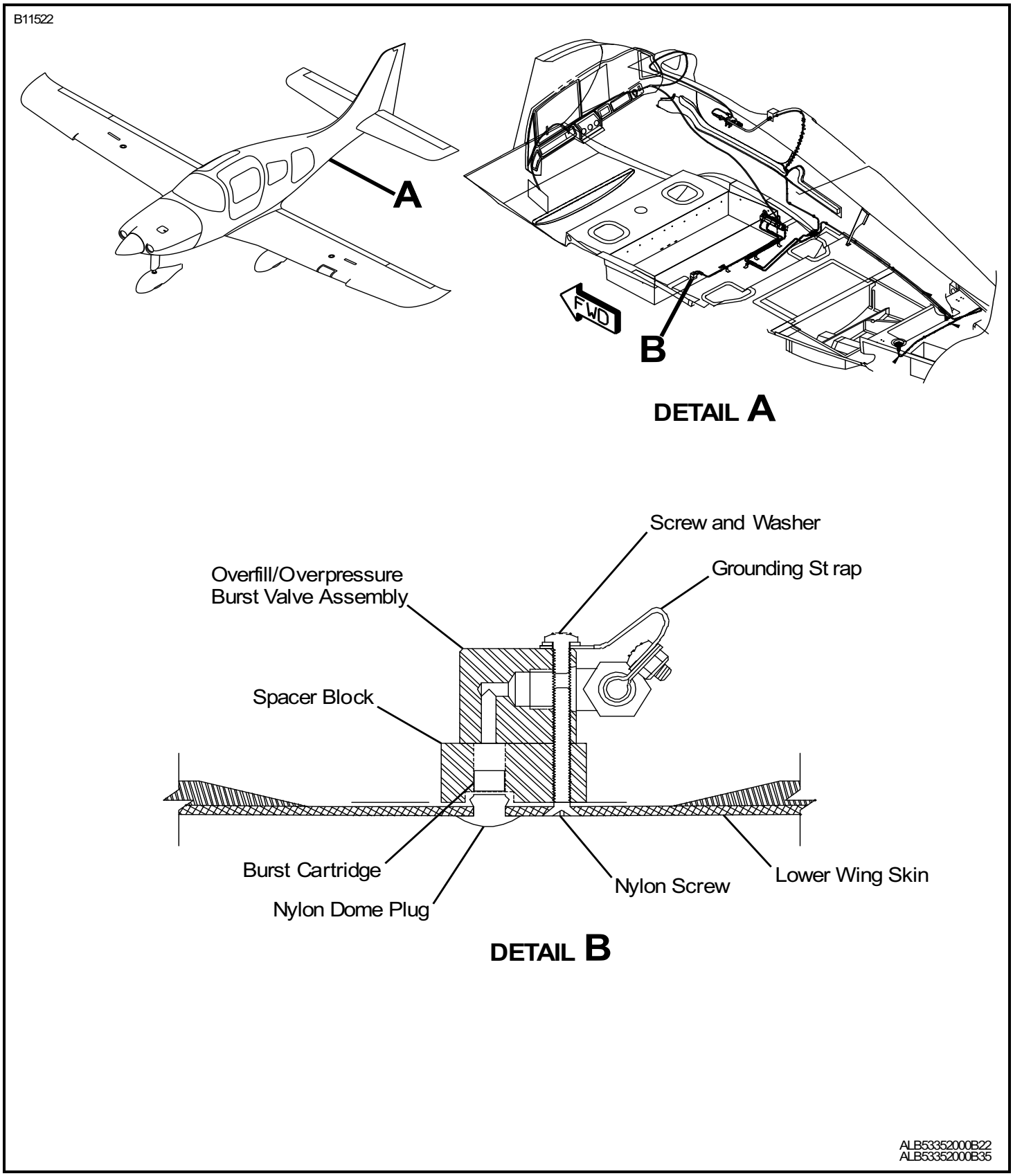
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Figure 203 : Sheet 1 : Fill Port



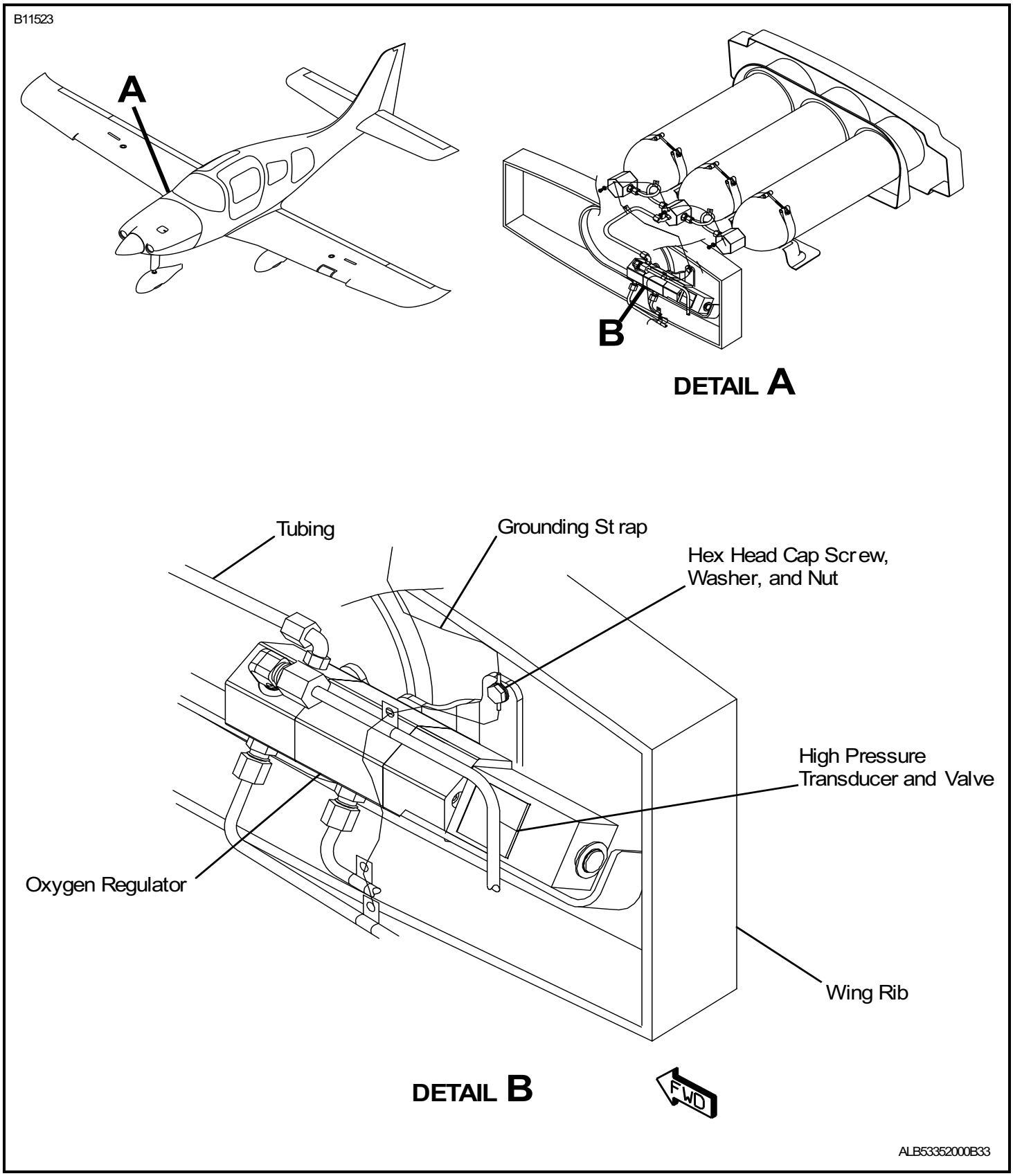
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Figure 204 : Sheet 1 : Overfill/Overpressure Burst Valve Assembly



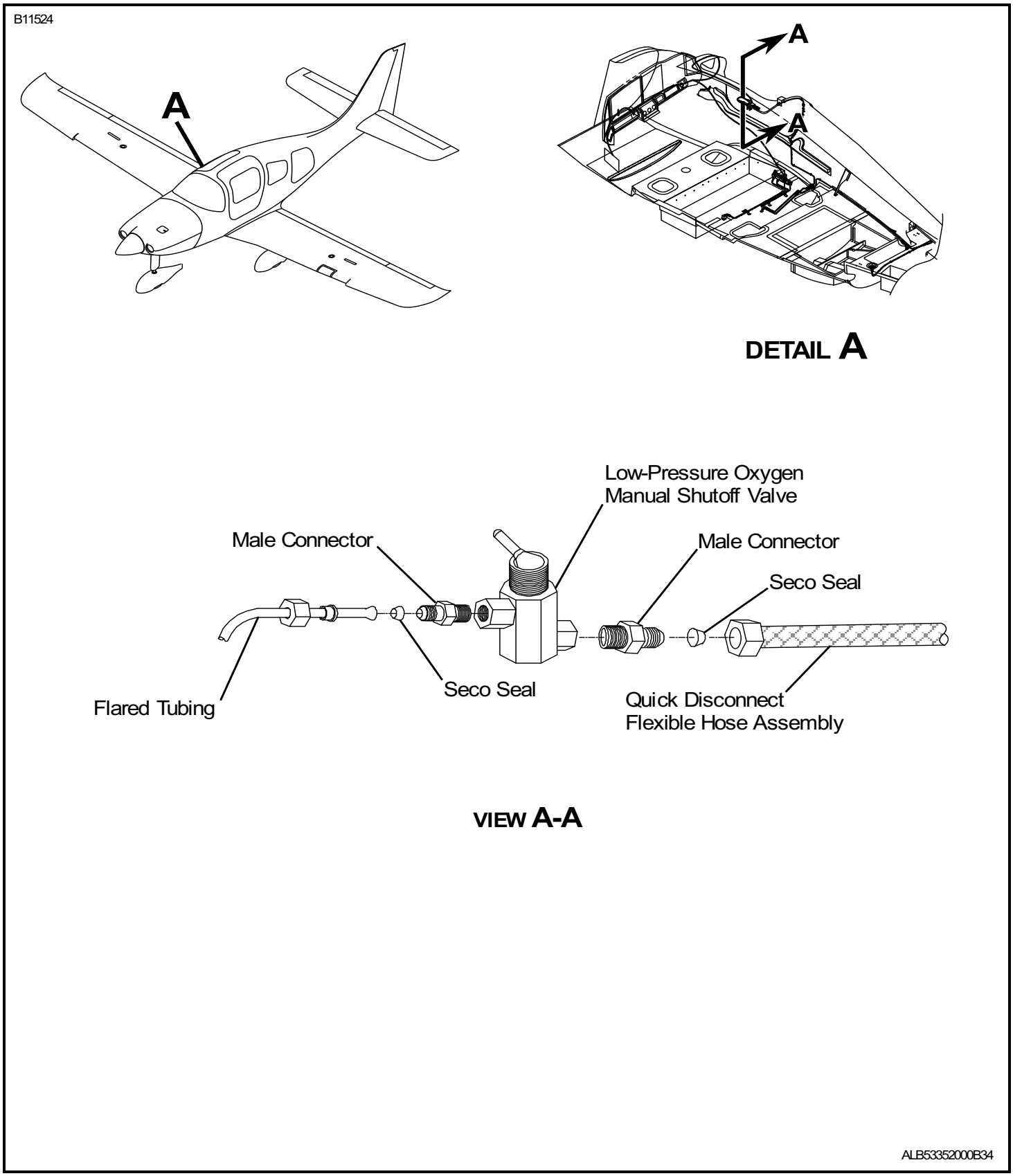
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Figure 205 : Sheet 1 : Regulator/Valve Assembly



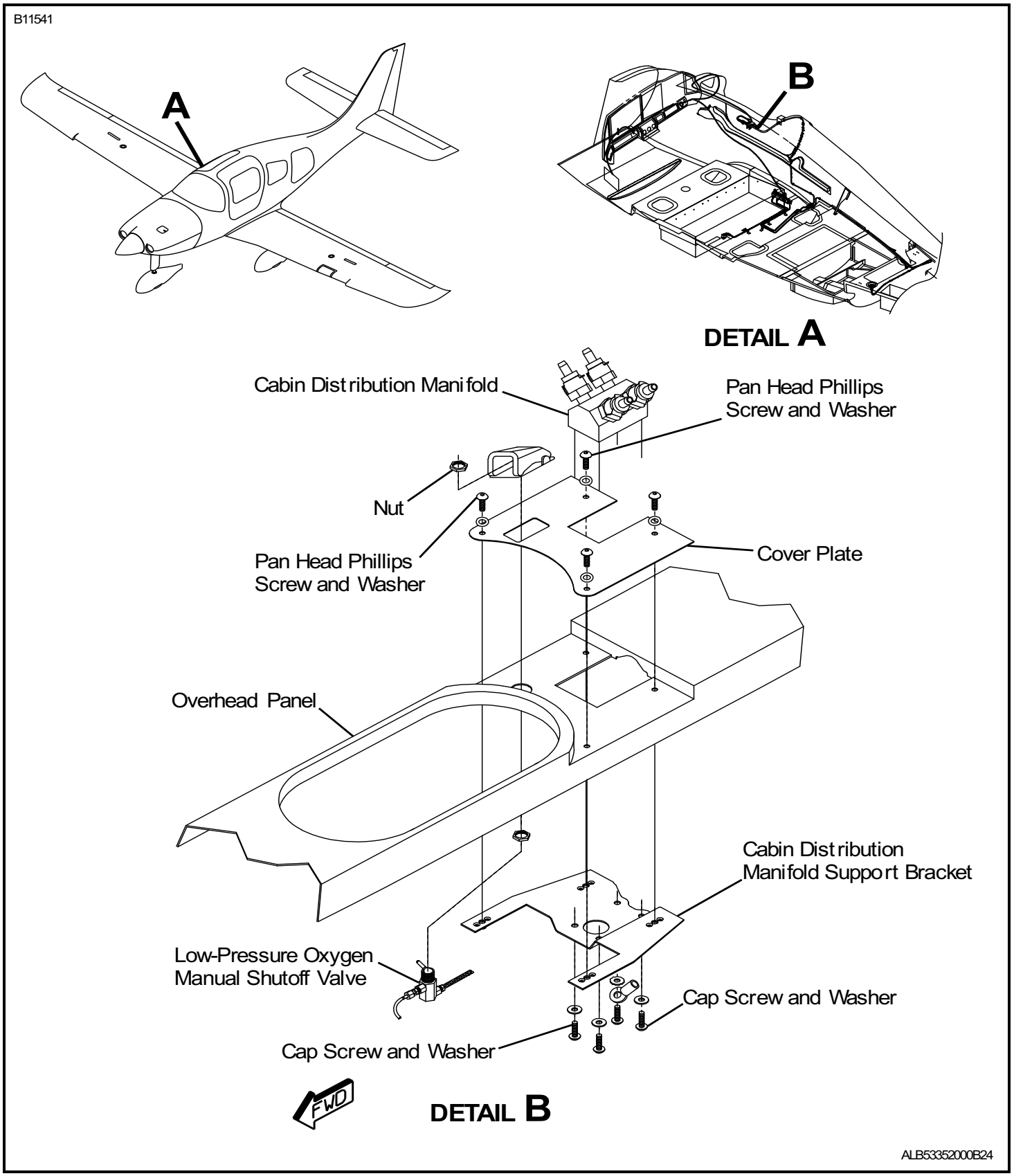
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Figure 206 : Sheet 1 : Manual Low-Pressure Oxygen Shutoff Valve



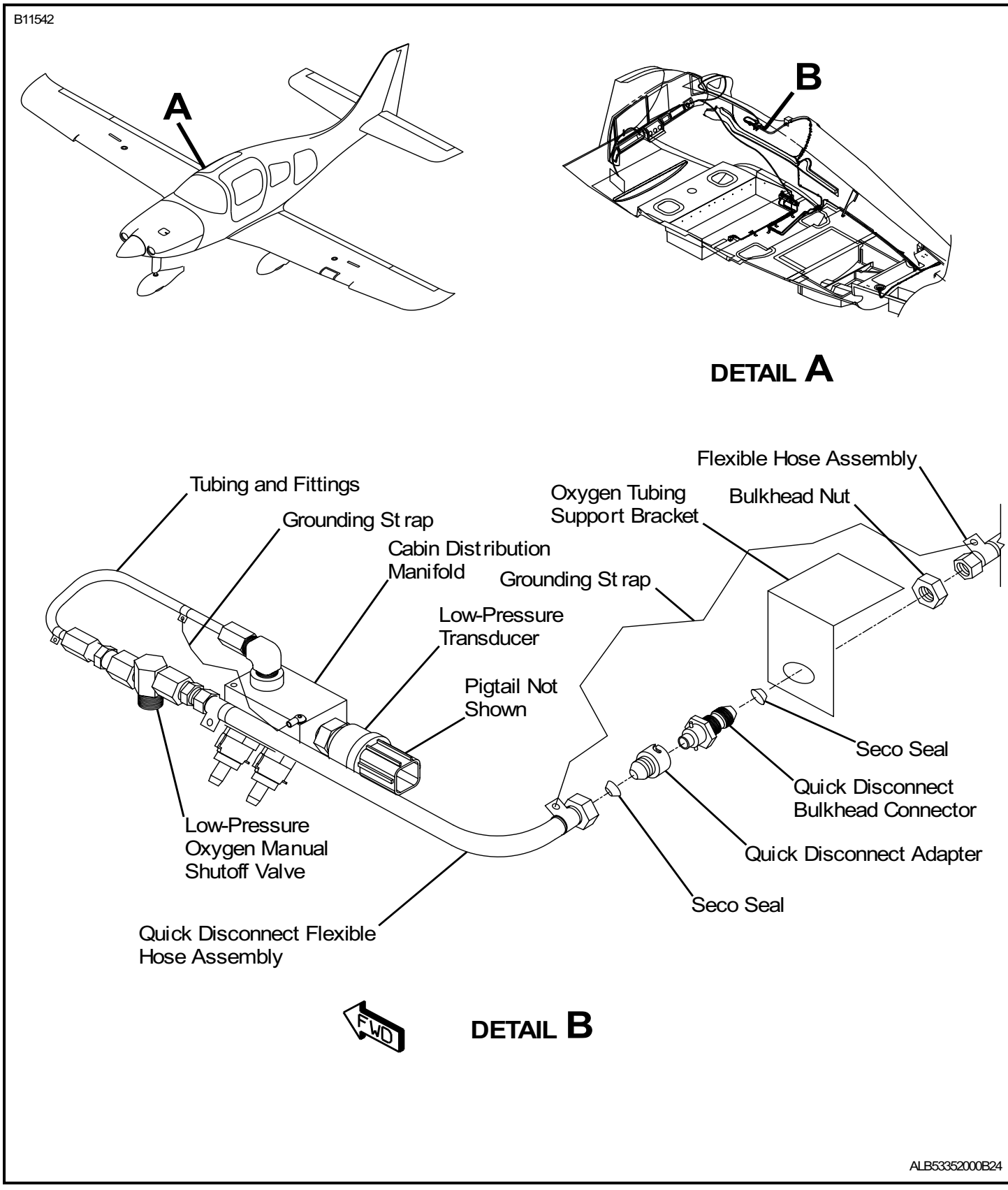
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Figure 207 : Sheet 1 : Cabin Distribution Manifold



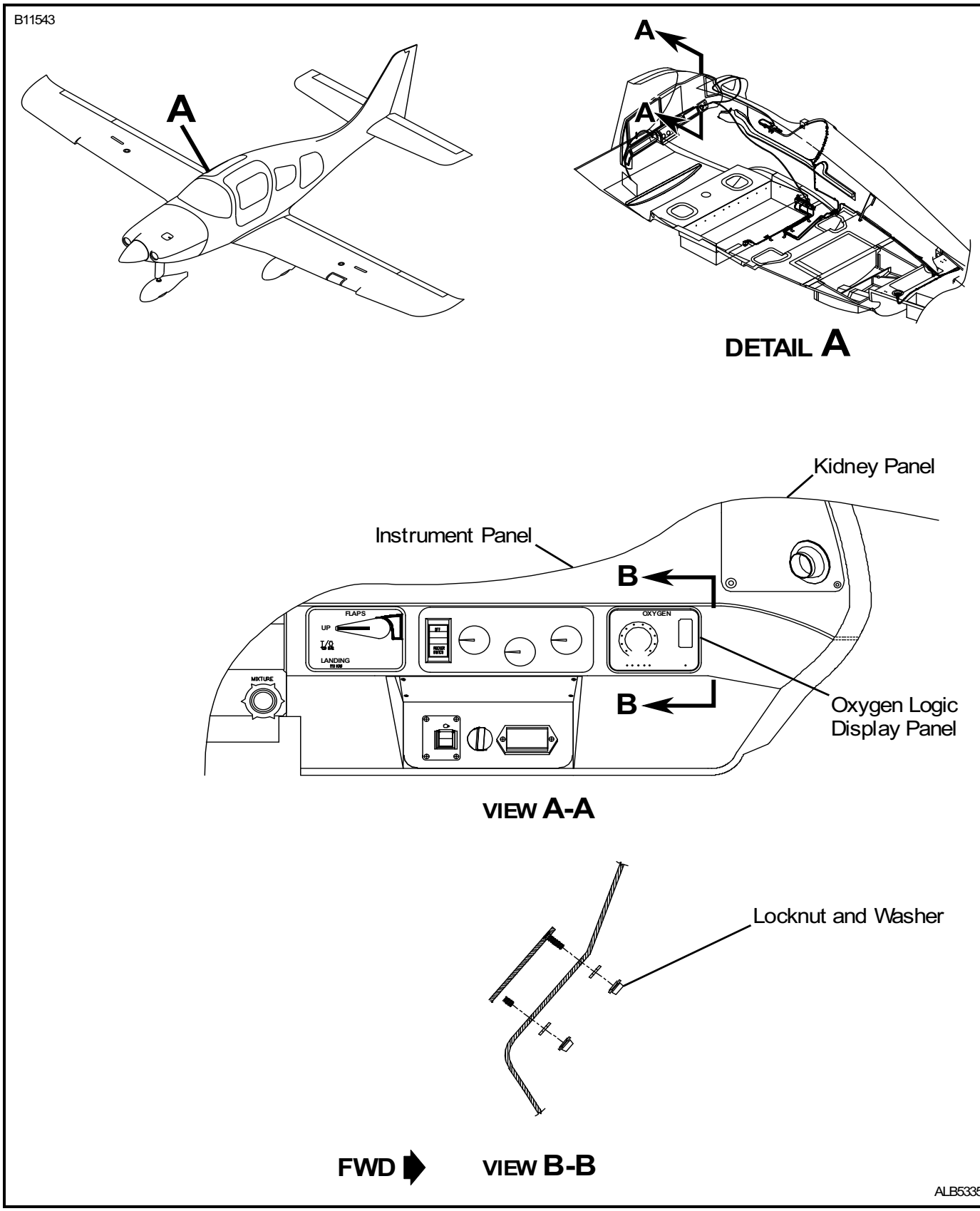
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Figure 208 : Sheet 1 : Cabin Distribution Manifold Assembly



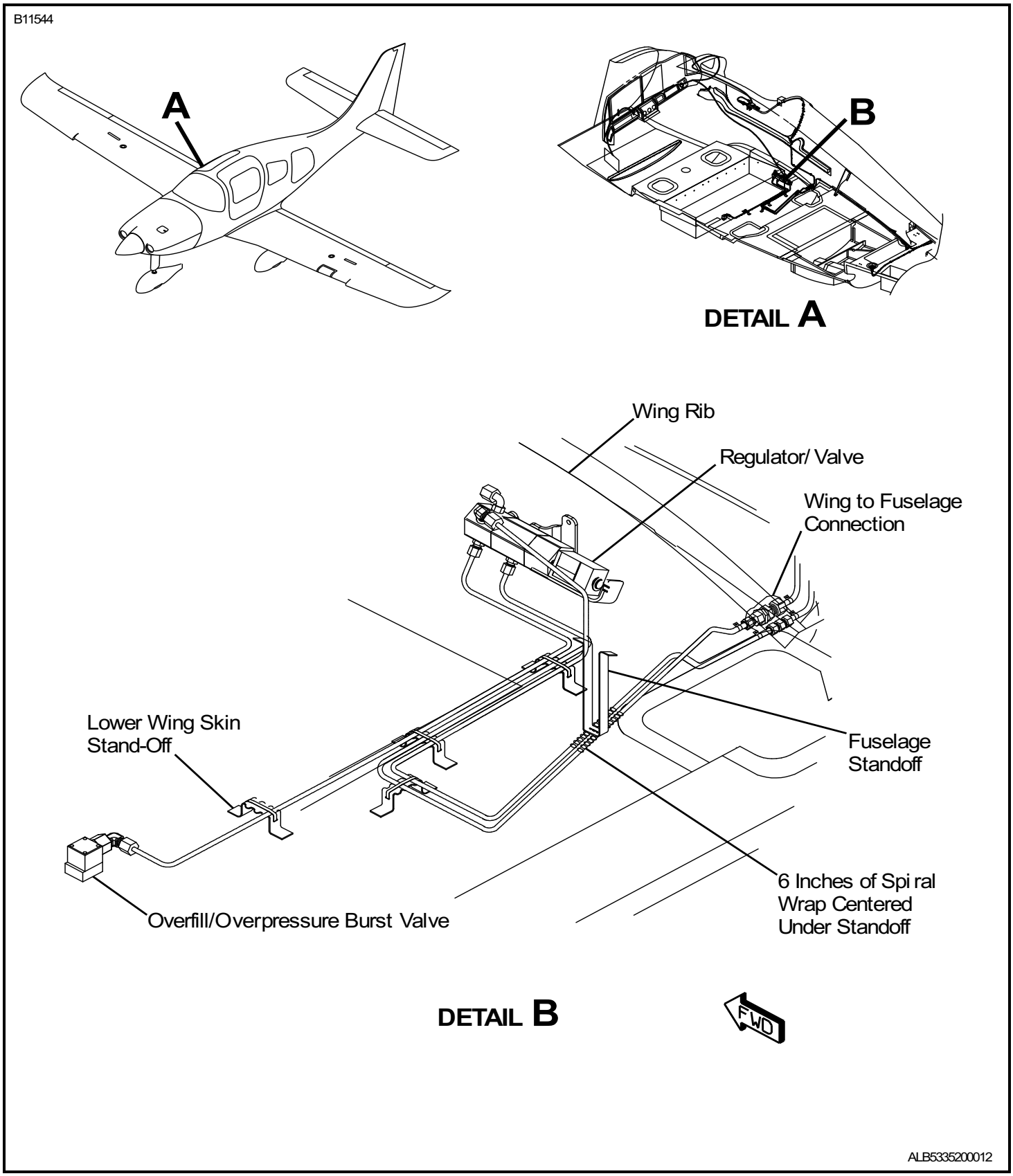
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Figure 209 : Sheet 1 : Logic Display Panel



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Figure 210 : Sheet 1 : Wing to Fuselage Connection



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SERVICING - GENERAL

1. Servicing

A. There are recommended and required servicing intervals specified in Section 05-10-00 and Section 05-20-00 of this maintenance manual. It is important to note that the servicing items for the recommended and required inspection intervals are based on typical operations. If the owner or operator of the airplane uses the airplane on an ongoing basis in an atypical environment, then the servicing procedures must be modified accordingly. These environmental conditions might include extremes in ambient temperature or humidity, operation in areas of high dust or sand, or unusual operating requirements, to mention a few.

B. In general, minimal servicing of the airplane is required between required maintenance inspection other than fuel, oil, and maintenance of proper tire pressures. In some instances, the nose strut may require unscheduled service, depending on how the airplane is used.

2. Procedures for Refueling and Defueling the Airplane

A. The high-speed characteristics of the airplane make generation of static electricity more likely, so it is important for the airplane to be grounded to the fuel source during refueling and defueling operations. Place the fuel source grounding clamp on the right or left exhaust stack of the airplane before touching the filler neck of the fuel tanks with metal parts of the ground refueling/defueling equipment. Remember that refueling is often done at the conclusion of a flight, and the exhaust stacks may still be hot, so care must be used when attaching the clamp.

B. Some defueling is possible using the defueling feature on the delivery system of the Avgas fuel supplier. This procedure is usually adequate for removing fuel when gross takeoff weight is an issue. To completely defuel the airplane use the following steps:

(1) Select a well-ventilated area for the defueling operation.

(2) Remove all fuel possible from the tank or tanks using the defueling feature on the Avgas delivery system.

(3) Disconnect the main fuel line downstream of the auxiliary fuel pump, at the firewall.

(4) Connect a flexible hose to the fuel fitting, and direct the hose into a fuel receptacle.

(5) Set the fuel selector to either the left or right tank, depending on which tank is to be defueled.

(6) Operate the auxiliary boost in low until the tank is drained. When the pump first starts to cavitate, turn it off, and attempt no further defueling with the auxiliary boost pump.

3. Procedures for Draining the Fuel System

A. Using the procedures described in Procedures For Refueling and Defueling the Airplane will leave approximately three to four gallons in the fuel tank. If further draining is required to work on a fuel tank or to establish the empty weight with unusable fuel, use the following procedure:

(1) Select a well-ventilated area for draining fuel from the tank.

(2) Create a small (2-inch diameter) dam around the wing tank drain with aluminum foil tape (sometime referred to as speed tape) or a similar substance.

(3) Place a container with a large opening under the fuel drain.

(4) Set the wing fuel drain to the locked-open position, and drain until fuel stops running.

(5) At this point, approximately 4 to 8 ounces of undrainable fuel will remain in each tank. To establish the airplane's empty weight with unusable fuel, ensure the fuel drains are closed, and then add three gallons of fuel to each tank.

(6) If all fuel is to be drained from the airplane, perform the above procedures for both tanks, and then drain the gascolator.

4. Fuel Drains

A. The inboard section of each tank contains a fuel drain near the lowest point in each tank. The fuel drain can be opened intermittently for a small sample, or it can be locked open to remove a large quantity of fuel. The gascolator, or fuel strainer, is located under the fuselage, on the left side, near the wing saddle. The gascolator is a conventional drain device that operates by pushing up on the valve stem. There is an internal bypass in the strainer that routes fuel around the filter if it becomes clogged.

5. Fuel Contamination

A. To test for fuel contamination, fuel samples must be taken from each of the wing drains and from the gascolator before each flight and after the airplane is refueled. Wait approximately 10 minutes before taking a sample after refueling to allow

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water and other contaminates to settle to the bottom of the tank. There are three types of contaminates that can inadvertently be introduced to the fuel system:

(1) Sediment, such as dirt and bacteria

(2) Water

(3) Improper grade of fuel

B. The accumulation of sediments is an inherent issue with most aircraft and can never be completely eliminated. Refueling the airplane at the conclusion of each flight and using fuel from a supplier who routinely maintains the filtration of the refueling equipment will lessen the problem somewhat. If specks are observed in the fuel sampler, continue the sampling operation until no debris is observed. Be sure the sampling device is clean before using it.

C. The two more common sources of water contamination are condensation of water from the air within a partially filled fuel tank and water-contaminated Avgas from a fuel supplier. Again, refueling after each flight and proper filtration of the fuel delivery system will mitigate water contamination. Water, which is heavier than Avgas, will collect near the bottom of the sampling device. If water is observed in the fuel sampler, take additional fuel samples until all the water is removed.

D. Aviation fuel is dyed according to its grade, and on new aircraft, like the Cessna 350 (LC42-550FG), the filler neck is sized to only accept fuel of the proper grade. Still, the color of the fuel must be verified according to the specifications in this paragraph, since the fuel truck might have been refilled improperly. If fuels of two different grades are mixed, the fuel sample will be clear. If an inferior, improper grade of fuel is noted, completely defuel and drain both tanks, and refuel with the proper grade of Avgas.

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REPLENISHING - SERVICING

1. Fuel Servicing and Recommended Fuel Grades

A. If possible and practicable, the airplane should be serviced at the conclusion of each flight. This procedure will significantly mitigate water accumulation in the fuel tanks. The following fuel grades are approved:

(1) 100LL Grade Aviation Fuel (Blue)

(2) 100 Grade Aviation Fuel (Green)

2. Fuel Capacities

A. Total Capacity: 106 Gallons US (401 L)

B. Total Capacity, each tank: 53 Gallons US (201 L)

C. Total Usable Fuel:

(1) S/N 42001 to 42567: 49 Gallons US (186 L)/tank, 98 Gallons US 371 L) total

(2) S/N 42568 and on:

(a) Standard: 43 Gallons US (163 L)/tank, 86 Gallons US (326 L) Total

(b) Long Range: 51 Gallons US (193 L)/tank, 102 Gallons US (386 L) Total

3. Approved Fuel Additives

A. Under certain ambient conditions of temperature and humidity, water can be supported in the fuel in sufficient quantities to create restrictive ice formation along various segments of the fuel system. To alleviate the possibility of this occurring, it is permissible to add isopropyl alcohol to the fuel supply, in quantities not to exceed 3 percent of the total. In addition, ethylene glycol monomethyl ether (EGME) and diethylene glycol monomethyl ether (DiEGME) compounds, in accordance with military specification MIL-I-27686E, may be added for this purpose. The EGME and DiEGME compounds must be carefully mixed, with fuel concentrations not to exceed 0.15 percent by volume.

B. It is important that the approved fuel additives are mixed in correct proportions. Consideration is required to ensure the appropriate concentration levels are achieved when the tank is filled. For example, adding 40 gallons of fuel with a 0.15 percent concentration of EGME to a tank with 10 gallons of untreated fuel will produce a mixture of something less than 0.15 percent. Consideration must also be made for the unusable fuel in the tank, since it will be combined with the total mixture.

C. The additives shall be added as the fuel is introduced to the fuel tank so that the mixture is properly combined. Alternatively, the additive can be mixed with a small amount of fuel in a separate container, such as a five-gallon can, and added to the fuel tank before normal fueling. Table 301 lists the number of ounces of each additive for a given fuel quantity.

4. Fuel Additive Mixture Table

WARNING: Proper mixing of EGME and DiEGME compounds is extremely important, because concentrations more than 0.15 percent by volume can have a harmful effect on engine components.

Table 301. Fuel Additive Mixture Table

Fuel(Gallons)	Isopropyl Alcohol, 3 percent (fl. oz.)	EGME and Di-EGME, 0.15 percent (fl. oz.)		Fuel(Gallons)	Isopropyl Alcohol, 3 percent (fl. oz.)	EGME and Di-EGME, 0.15 percent (fl. oz.)
1	3.8	0.2		27	103.7	5.2
2	7.7	0.4		28	107.5	5.4
3	11.5	0.6		29	111.4	5.6
4	15.4	0.8		30	115.2	5.8
5	19.2	1.0		31	119.0	6.0
6	23.0	1.2		32	122.9	6.1
7	26.9	1.3		33	126.7	6.3
8	30.7	1.5		34	130.6	6.5
9	34.6	1.7		35	134.4	6.7
10	38.4	1.9		36	138.2	6.9

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11	42.2	2.1		37	142.1	7.1
12	46.1	2.3		38	145.9	7.3
13	49.9	2.5		39	149.8	7.5
14	53.8	2.7		40	153.6	7.7
15	57.6	2.9		41	157.4	7.9
16	61.4	3.1		42	161.3	8.1
17	65.3	3.3		43	165.1	8.3
18	69.1	3.5		44	169.0	8.4
19	73.0	3.6		45	172.8	8.6
20	76.8	3.8		46	176.6	8.8
21	80.6	4.0		47	180.5	9.0
22	84.5	4.2		48	184.3	9.2
23	88.3	4.4		49	188.2	9.4
24	92.2	4.6		50	192.0	9.6
25	96.0	4.8		51	195.8	9.8
26	99.8	5.0		52	199.7	10.0

5. Engine Oil Capacity

A. The dipstick and oil filler cap access door are located on the top left engine cowl about two feet from the propeller hub. The engine must not be operated with less than six quarts of oil and must not be filled above eight quarts. For extended flights, the oil should be brought up to full capacity.

B. The engine drain-refill capacity is eight (8) quarts of oil. The airplane engine holds approximately 11 quarts of oil; however, about 3 quarts are for the oil cooler, oil filter, and propeller governor and cannot be drained from the system. Before checking the oil on a hot engine, ensure sufficient time has elapsed to allow the oil to drain back into the sump.

6. Recommended Oil for Engine Break-In Period

A. During the engine break-in period, straight mineral oil must be used for the first 25 hours unless the oil consumption has not stabilized. If the oil consumption has stabilized after the first 25 hours of the airplane's time in service, the oil and oil filter must be changed and a new supply of Teledyne Continental Motors specification MHS-24 (latest revision) ashless dispersant oil must be used. At 50 hours of time in service, the oil and oil filter shall be changed and the filter and discarded oil checked for evidence of metal particles. Thereafter, the oil and oil filter must be changed at every 100 hours of time in service.

B. If, after the initial 25 hour break-in period, the oil consumption has not stabilized, use the following procedure:

(1) Inspect and replace the oil filter.

(2) Drain and inspect the engine oil.

(3) Install a fresh supply of approved mineral oil.

(4) Continue to monitor oil consumption.

(5) When oil consumption stabilizes, remove, inspect, and replace the oil filter. Drain and inspect the engine oil, and add a new supply of Teledyne Continental Motors (TCM) Specification MHS-24 (latest revision) ashless dispersant oil.

C. In the extremely unlikely event the oil consumption has not stabilized at 50 hours of time in service, the cylinders may need de-glazing, or some other problem may exist. In either event, the issue should be referred to TCM.

7. Oil Servicing

A. The oil grades shown below in Recommended Oil Grades are recommended after the initial engine break-in period. Only lubricant oils conforming to Teledyne Continental Motors Specification MHS-24D (latest revision) can be used.

NOTE: The approval for use of MHS-25 synthetic oils has been removed.

Each time the oil and the oil filter are changed, the discarded filter and oil must be checked for evidence of metal particles. Refer to Chapter 6 of the Teledyne Continental Motors maintenance manual.

8. Oil Filter Servicing

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A. A full flow, spin on-type, 20-micron oil filter is used. As mentioned previously, the filter must be changed during the initial break-in period at 25 and 50 hours time in service. Thereafter, the filter must be replaced at every 100 hours time in service interval. If the airplane is predominately operated in conditions of high dust or other atmospheric debris, it is advisable to change the filter at 50 hour intervals. Refer to Chapter 6 of the Teledyne Continental Motors maintenance manual instructions for removal and replacement of the oil filter. Use a TCM filter or approved equivalent.

9. Recommended Oil Grades

A. The following oil grades are recommended for the two ambient temperature ranges shown below.

(1) Below 40° F: SAE30, 10W30, 15W50, or 20W50

(2) Above 40° F: SAE50, 15W50, 20W50, or 20W60

10. Oxygen System Servicing

A. General safety guidelines and servicing procedures must be followed and are as indicated in Chapter 35. Use only MIL-O-27210 Aviator's Breathing Oxygen for system recharging.

NOTE: This fixed oxygen system has a maximum bottle pressure of 2,000 psi.

WARNING: Refer to Safety Guidelines for Servicing the Oxygen System in Chapter 35 before attempting any servicing or maintenance of the oxygen system.

B. Basic or Avidyne Option:

(1) Open all aircraft doors to provide air circulation.

(2) Set the Left Bus switch to ON.

(3) Set the Oxygen Switch to DISP.

(4) Locate the Oxygen Fill Port in the baggage bulkhead.

(5) Remove the Oxygen Fill Port protective cap and connect the service line.

(6) Slowly fill the oxygen supply. By filling slowly, the temperature rise due to the compression of gas into the oxygen supply cylinders will be kept to a minimum.

(7) Fill the system until a pressure of 1950±50 psig is achieved. This value is valid at a steady state condition after the system has cooled to 70° F from the recharging heat buildup. System pressure may be increased 3.5 psig for each degree of increase in temperature above 70° F and decreased 3.5 psig for each degree decrease in temperature below 70° F.

(8) Shut off the oxygen supply to the aircraft.

(9) Set the Oxygen Switch to ON.

(10) Ensure that the Outlet Pressure Display green indication is illuminated.

(11) Set the Oxygen Switch to OFF.

(12) Set the Left Bus switch to OFF

(13) Disconnect the service line from the Fill Port and install the protective cap.

C. Garmin G1000 Option:

(1) Open all aircraft doors to provide air circulation.

(2) Set the Right Bus and Avionics Bus switches to ON.

(3) Turn on the MFD and access the Engine Indication System (EIS) page.

(4) Locate the Oxygen Fill Port in the baggage bulkhead.

(5) Remove the Oxygen Fill Port protective cap and connect the service line.

(6) Slowly fill the oxygen supply. By filling slowly, the temperature rise due to the compression of gas into the oxygen supply cylinders will be kept to a minimum.

(7) Fill the system until a pressure of 1950±50 psig is achieved. This value is valid at a steady state condition after the system has cooled to 70° F from the recharging heat buildup. System pressure may be increased 3.5 psig for each degree of increase in temperature above 70° F and decreased 3.5 psig for each degree decrease in temperature below 70° F.

(8) Shut off the oxygen supply to the aircraft.

(9) Set the Oxygen Switch to ON.

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(10) Ensure that the Outlet Pressure and Oxygen Quantity gauges confirm oxygen supply.

(11) Set the Oxygen Switch to OFF.

(12) Turn off the MFD.

(13) Set the Right Bus and Avionics Bus switches to OFF.

(14) Disconnect the service line from the Fill Port and install the protective cap.

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FILTER, STRUT, BRAKE AND BATTERY SERVICING - SERVICING

1. Induction Filter Servicing

A. The airplane has a disposable air filter, Brackett Aircraft Company, Inc. Part No. BA-9005. The rectangular filter element is a wettant-impregnated foam. The filter must be serviced annually, every 100 hours, or more frequently if operating in areas of high atmospheric debris. To service the filter, use the following procedure:

(1) Remove the air filter from the engine. Refer to Section 71-60-00, Air Filter Removal and Replacement, for instructions.

(2) Dispose of old filter element.

(3) Install a new filter element. Refer to Section 71-60-00, Air Filter Removal and Replacement, for instructions.

2. Tires and Wheels Servicing

A. Ensure tires are serviced to their proper pressures. Refer to Section 06-00-00, Landing Gear, for pressure specifications. The following items must also be inspected:

(1) Check tires for flat spots, splitting, dry rot, cuts, and any other evidence of damage.

(2) Check valve stem slip marks for evidence of slippage.

(3) Check wheels for damage.

(4) Check wheel bearings for condition and lubrication.

B. **Tire Considerations** - The airplane is normally delivered with Goodyear tires. These tires have a profile that provides about 3/8 inch clearance between the tire and wheel pants. Other brands of tires with similar specifications and TSOs may have slightly larger profiles. Tires with larger profiles are not recommended, since damage to the tire or wheel pant is possible, particularly during landing. If other brands of tires are used, the profile of the tire must be precisely measured and compared with the Goodyear tire.

CAUTION: The profile of replacement tires that are not a recommended brand should be measured precisely to ensure they are the same height and width. The use of tires that have slightly larger profiles can cause damage to the tire and to the wheel pant, particularly during landing operations.

3. Servicing the Nose Strut

A. The nose strut is normally serviced at required maintenance inspections. If less than three to four inches of chrome are visible on the nose strut or a low oil level is indicated, then unscheduled servicing is necessary. A low oil level in the nose wheel strut may be indicated by one or both of two symptoms. First, there will be a general deterioration in shimmy damping of the nose strut. Second, the nose will have a tendency to bounce noticeably while taxiing. If either or both of these conditions are experienced, a low oil level in the nose strut is the most likely cause.

B. **Cessna Nose Strut** (Refer to Figure 301 and Figure 302)

CAUTION: Always wear safety goggles when releasing gas pressure.

WARNING: Removal of the AN7-6A bolt without de-pressurizing the strut could cause serious injury.

(1) Remove the nose strut from the nose gear per instructions in Section 32-20-00, Removal of the Nose Strut.

(2) Remove the valve cap from the strut Schrader valve. Using a small flat-bladed screwdriver, gently press in on the Schrader valve core stem to slowly bleed off nitrogen gas pressure in the strut.

(3) Remove the AN7-6A bolt at the top of the strut.

(4) Pull the strut shaft to its fully extended position.

(5) Inflate the strut with nitrogen gas to a pressure of 50 psi.

(6) With the strut standing vertically, add MIL-H-5606F hydraulic oil through the AN7- 6A bolt hole until the chamber is full.

(7) Create an oil bleed device composed of a flexible hose inserted into a reservoir filled with MIL-H-5606F hydraulic oil. Ensure the hose is filled with oil and attach the free end to the AN7-6A bolt hole. Cycle the strut up, down, left, and right through its full range of motion until all air is removed.

(8) With the strut fully extended, remove the valve core from the Schrader valve and release the nitrogen gas pressure.

(9) Pinch shut the flexible hose on the oil bleed device and push the strut shaft up 0.75 inches from its fully-extended position to correctly position the internal piston.

(10) Un-pinch the flexible hose.

(11) Ensure the valve core is free from contaminates and reinstall it into the Schrader valve. Tighten snug.

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(12) Slowly pull the strut shaft to its fully extended position.

(13) Remove the bleed device and add additional hydraulic oil until the chamber is filled to the top.

(14) Replace the AN7-6A bolt and tighten snug.

(15) Inflate the strut with nitrogen gas until pressure reaches 250±25 psi. Best results are obtained using the following two techniques:

(a) Apply enough physical force on the nitrogen hose chuck to ensure a tight seal on the filler valve. The volume of area being filled is small and requires a lot of physical pressure to hold the chuck on the valve.

(b) When the strut is filled, remove the air hose chuck from the filler valve as quickly as possible. A persistent low nose gear strut is frequently the result of losing 50 pounds or so of pressure when the air chuck is removed. Over-pressurization will prevent full castoring movements of the nose wheel.

(16) With the strut firmly placed on the ground, attempt to compress the strut by hand. If the strut can be compressed, repeat the above instructions until all air is removed and the strut cannot be compressed.

(17) Torque the AN7-6A bolt at the top of the strut 80 to 100 in-lbs.

(18) Install the Schrader valve cap and torque 10 to 15 in-lbs.

(19) Install the nose strut into the nose gear per instructions in Section 32-20-00, Installation of the Nose Strut.

NOTE: Persistent and frequent loss of oil is an indication of more serious problems. The oil seals may need replacement, or there may be issues with the integrity of the cylinder. Refer to Section 32-20-00 for information.

C. **ESCO Nose Strut** (Refer to Figure 303)

(1) Remove the lower cowling, using procedures described in Section 71-00-00, Removal of the Engine Cowling, to expose the filler valve (Schrader valve).

(2) Remove all the weight from the nose strut by placing ballast on the tail, and let the strut settle for one hour.

(3) Release the nitrogen gas from the strut by pressing on the stem of the filler valve until most of the nitrogen is released. Hold a rag over the valve stem and use the hand to release the pressure to prevent a small amount of oil from being scavenged from the main cylinder as the nitrogen is released.

(4) Partially remove the valve core to evacuate the remaining gas in the cylinder.

(5) Remove the Schrader valve.

(6) Ensure that the strut is fully extended.

(7) Hydraulic oil that meets MIL-H-5606F can now be added slowly with a pump action oil can. Add oil until the level in the cylinder reaches the bottom of hole from which the Schrader valve was removed.

(8) Rotate the strut left and right throughout its full range of rotation (60 degrees each direction) to completely fill internal components of the strut with oil. When internal components are full of oil, resistance should be felt while rotating the strut. Keep adding oil while rotating until the oil drains back out of the filler valve hole and no more can be added.

(9) Refit the Schrader valve and torque 150 to 200 in-lbs.

(10) Ensure that the strut is fully extended. Insert the air hose chuck onto the valve stem. Fill until pressure reaches 250 psi. Best results are obtained using the following two techniques:

(a) Apply enough physical force on the nitrogen hose chuck to ensure a tight seal on the filler valve. The volume of area being filled is small and requires a lot of physical pressure to hold the chuck on the valve.

(b) When the strut is filled to 250 psi, remove the air hose chuck from the filler valve as quickly as possible. A persistent low nose gear strut is frequently the result of losing 50 pounds or so of pressure when the air chuck is removed.

(11) Repeat steps (3) through (6) in this paragraph and finish filling the strut with oil until oil drains back out the filler valve hole and no more can be added.

(12) Refit the Schrader valve and torque 150 to 200 in.-lbs. Service the strut with nitrogen per the instructions in step (10) above. Replace the Schrader valve cap and tighten snug. Over-pressurization will create a nose high attitude while taxiing and will prevent full castoring movements of the nose wheel.

(13) Lower the nose by progressively removing ballast from the tail. Pull down on the propeller to exert downward pressure on the nose gear. Verify that three to four inches of chrome are visible on the nose strut.

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(14) Replace the lower cowling per instructions in Section 71-00-00, Installation of the Engine Cowling.

NOTE: Persistent and frequent loss of oil is an indication of more serious problems. The oil seals may need replacement or there may be issues with the integrity of the cylinder. Refer to Chapter 32 for information.

4. Brake Fluids

A. Under most conditions, the airplane will not require servicing of the hydraulic system. There is a small hydraulic reservoir mounted on the upper portion of the firewall (engine side). The reservoir contains an Allen screw; however, this area is not normally used for servicing the airplane. If brake sponginess is experienced, this is an indication that there is air in the system. More important, it may be an indication of a hydraulic leak in the brake system, and the areas proximate to the hydraulic lines, master cylinders, and brake assembly must be inspected for evidence of hydraulic oil. If no leaking is detected, then the brakes should be bled. Section 32-40-00 contains instructions for bleeding the brakes. If hydraulic fluid is added, use MIL-H-5606F.

5. Battery Servicing

A. For 12V aircraft, the installed batteries are Concorde RG-1215; for 24V aircraft they are Teledyne Gill G230S, Teledyne Gill G248S, Concorde RG24-10, or Concorde RG24-16. The batteries are of valve regulated lead-acid (VRLA) design. Since the batteries have no electrolyte, there is no required servicing of the unit. The posts and cables are serviced at 100 hour inspections if applicable; otherwise, they are maintained during the annual inspection. It is recommended that the batteries be replaced every four years or sooner if poor overall performance is experienced.

6. Cleaning Aircraft

A. **Cleaning the Exterior** - The exterior painted surfaces are cleaned by washing with mild soap and water and drying with a soft towel or chamois. When operating in salt air environments, you will need to wash the airplane more frequently. The seal coats that are applied to the painted surface, in most instances, will provide adequate protection from moisture and the sun. Some additional protection is provided by waxing the painted surface and facilitates washing the airplane since bugs and dirt will not adhere as tightly to a waxed surface. A wax with a high concentration of carnauba is recommended. There are several commercial boat waxes available that are ideal for this use. Be sure to read the label with an eye for the percentage of carnauba in the compound.

NOTE: The exterior of the airplane should not be waxed within the first 90 days of the date of delivery. This period will ensure the painted surface has adequate time to cure. The exterior paint color on the upper fuselage area and the top of the wings has a good heat reflection index. This good index is required to ensure the continued bonding and integrity of the composite material. Only approved Cessna paint colors are permitted in these areas. Care must be taken to not lay dark, heat absorbing material on the top area of the wings and fuselage.

B. **Cleaning the Windshield and Windows** - The proper care of the windshield and windows (sometimes referred to as transparencies) is one of the more important exterior care items on the airplane. Never do anything that will scratch the surface of the acrylic plastic. The following points for cleaning and caring for the transparencies should be observed. Refer to Section 51-70-00, Transparency Repair, for approved cleaning materials and expanded procedures.

(1) First, when cleaning the windows, it is recommended that rings and watches be removed as they can cause deep scratches. Long-sleeved shirts should be turned up a few rolls to hide exposed buttons.

(2) When removing bugs and dirt, avoid touching the surface. If possible, remove most of the dirt by flushing the windows and windshield with water and a mild dish soap mixture. Allow the accumulation of dirt and/or bugs to soak for a few minutes. If rubbing is required, a bare hand is best. When all the debris on the surface of the window is loosened, apply a second water flush, and then dry with a 100 percent cotton cloth.

(3) Use a good quality, non-abrasive cleaner/polish specifically intended for acrylic windows, and apply per the manufacturer's instructions. Use up-and-down or side-to-side movements when polishing. Never use a circular movement, as this can cause glare rings.

(4) The best polishing cloth is the softest cotton available. One hundred percent cotton flannel is ideal and available in yard goods stores. **Never** use any type of paper product or synthetic material. In particular, never use shop rags or shop towels. Be sure the polishing cloth is clean and dry. Reserve polishing cloths should be stored in a plastic bag to limit dirt accumulation.

(5) Small scratches, the type that can be seen but cannot be felt with a fingernail, should be filled with a polishing compound that has scratch filling properties. A good quality, non-abrasive cleaner/polish specifically intended for acrylic windows frequently has scratch filling properties and is satisfactory for regular use. Some scratches cannot be

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filled with a scratch-filling product. While the scratches cannot be felt, they are still visible, particularly when flying into the sun. In this instance, a mildly abrasive scratch removal cream can be used per manufacturer's recommendations. Scratches of greater magnitude require the use of high abrasives and removal of some of the window's surface around the greatest depth of the scratch. This procedure requires considerable expertise and frequently makes areas where the scratch was removed more objectionable than the original scratch.

CAUTION: Do not use anything containing ammonia, aromatic solvents like methyl n-propyl ketone, acetone, lacquer thinner, paint stripper, gasoline, benzene, alcohol, anti-ice fluid, hydraulic fluid, fire extinguisher solutions, or window cleaner on the acrylic window surfaces. The use of these substances may cause the surface to craze.

NOTE: To remove difficult substance such as tape residue, oil, and grease, the safest solvents are 100 percent mineral spirits or kerosene. Some alcohols are safe, such as isopropyl alcohol.

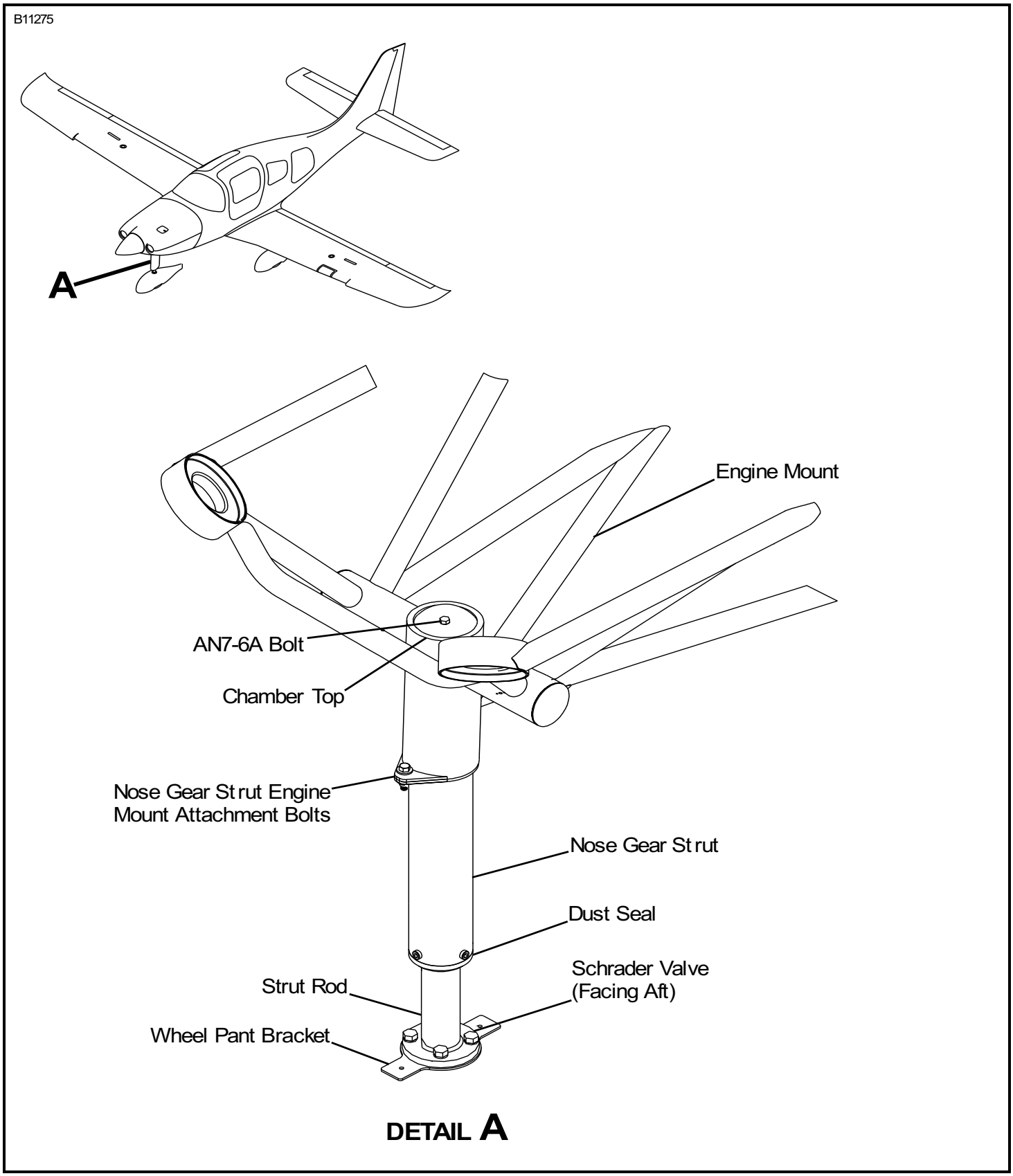
C. **Interior Cleaning and Care** - The useful life of the airplane's interior can be extended through proper care and cleaning. One of the major elements in the aging process is the interior's exposure to sunlight. If possible, the airplane should be hangared. Routine vacuuming is another item that helps extend the life of the airplane's interior, particularly the carpets. The leather seats, seatbacks, knee bolsters, glare shield and the like, should be routinely wiped with moist, soft cotton cloth. A general rule for spills is to blot the affected area with firm pressure for a few seconds. Never rub an area to remove a spill.

(1) Much of the airplane's interior is covered with high-quality leather. The leather is impregnated with polyvinyl chloride (PVC), and as treated, is impervious to most substances. The application of leather preservatives will provide little protection, and in some cases can cause problems. For difficult stains, such as an ink stain, remove with a mild soap and water. Do not use any type of chemical since these products tend to remove the impregnated PVC protection, and once removed it can never be reapplied.

(2) The carpet can be cleaned with a mild foam product, but care should be used not to oversaturate the carpet. Follow the manufacturer's instructions regarding use of the foam cleaner. Small spots can be cleaned with a commercial spot remover; however, this must be done with care. Again, follow the recommended procedure of the manufacturer, and try a test application in an area of limited exposure.

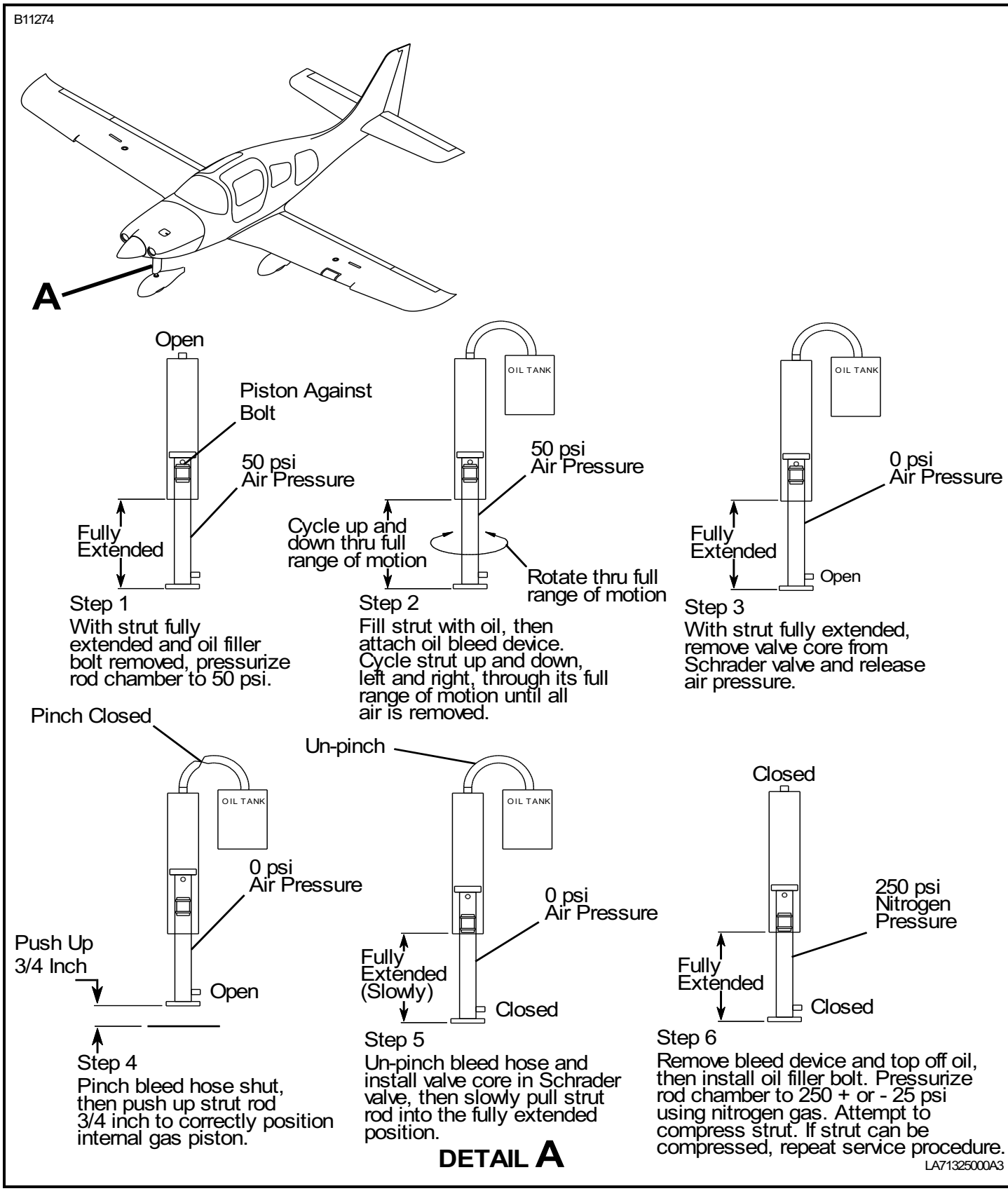
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Figure 301 : Sheet 1 : Cessna Nose Strut Assembly



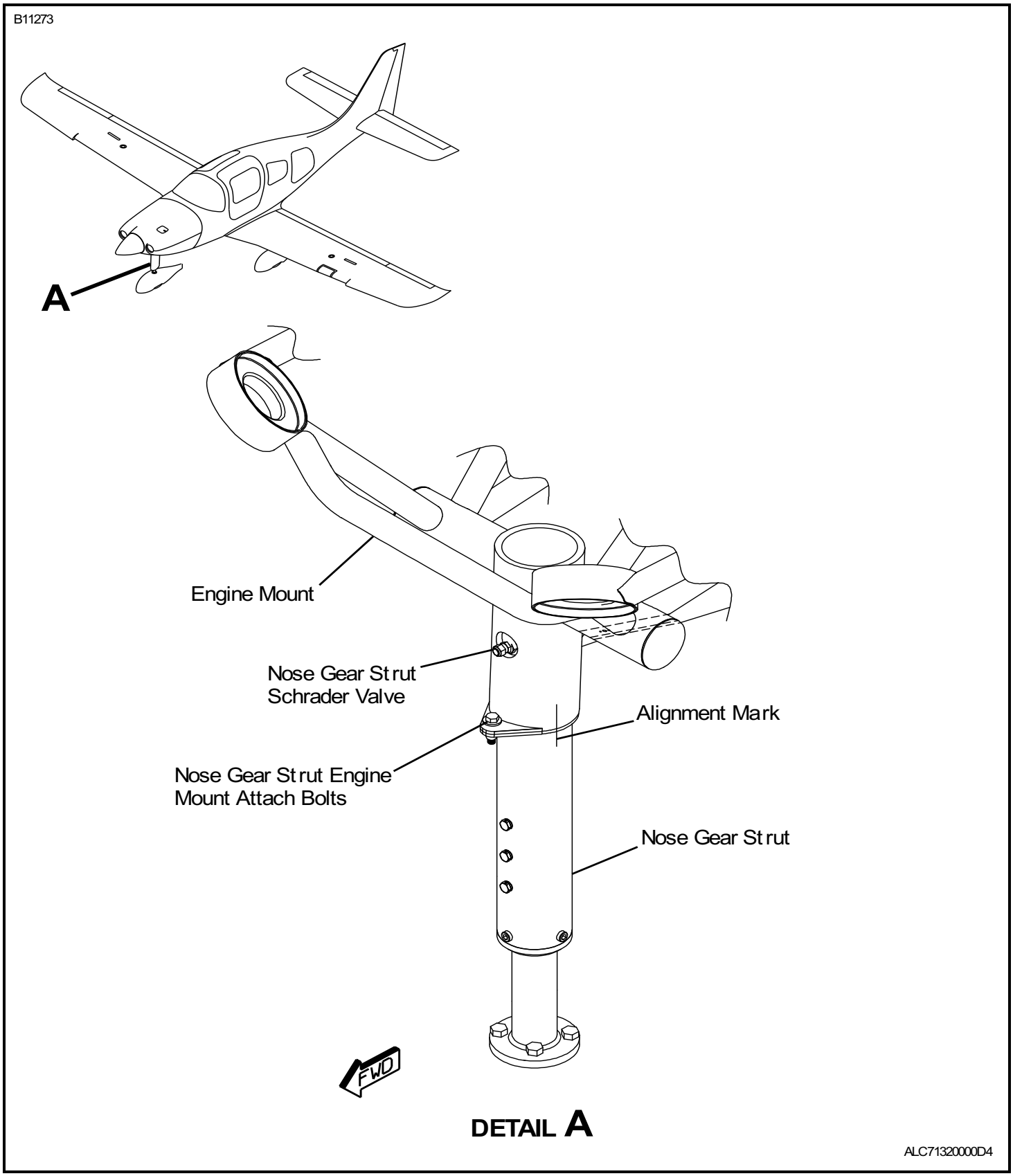
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Figure 302 : Sheet 1 : Cessna Nose Strut Service



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Figure 303 : Sheet 1 : ESCO Nose Strut Assembly



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PLACARDS AND MARKINGS

1. General

A. The various airplane placards on the interior and exterior of the airplane are required items for the continued airworthiness of the airplane. The placards must be inspected at the required inspection intervals for cleanliness, legibility, and security. The following pages contain the text and location of all required interior and exterior placards. In addition, procedures for removal, replacement, and care are included.

2. Replacing Worn, Damaged, or Missing Placards

A. If necessary, refer to Chapter 11 in *The Cessna 350 (LC42-550FG) Parts Catalog* to determine the part number for the affected placard.

B. Use the following steps to remove and replace a placard:

(1) Apply the placard in a relatively warm area. It is best if the ambient air temperature and the temperature of the surface to which the placard is applied is above 70° F.

(2) Make a few reference marks around the old placard to aid in positioning the new one. If the placard is missing, refer to Sections 11-20-00 and 11-30-00 for the location, and then apply reference marks.

(3) Carefully remove the existing placard by raising one corner and slowly peeling it back. **Do not pull out on the placard!** Pull in a manner that keeps the peeled portion near or parallel to the surface. It helps to apply primer remover between the glue on the back of the placard and the attachment surface.

(4) Do not remove the placard too quickly. A slow steady pull works best. Discard the placard once it is removed.

(5) Clean excess adhesive off of the attachment surface with PPG D837 Spirit Wipe.

(6) When the attachment surface is clean and dry, remove the paper backing on the new placard and apply it using the reference marks described in item (2) above.

(7) Attach one end of the placard to the attachment surface, and slowly roll it on in a manner that avoids air bubbles. Because the airplane's surface has no rivets, air bubbles are infrequent.

(8) If the placard has a protective covering, remove it at this point.

(9) If an air bubble does occur, prick the bubble with a small diameter needle. The number of holes depends on the size of the bubble. Five to ten needle holes are usually sufficient. Slowly work the placard until all air is removed.

3. Placard Care

A. Wash the placard with mild soap and water. Do not use petroleum-based, caustic, or abrasive products. Avoid applying wax to placards because it may contain certain amounts of petroleum-based substances.

B. If there is any question about the composition of the cleaner being used, apply a small amount to a test corner of the placard, and wait a few minutes to see if it produces adverse results.

C. If aviation fuel accidentally comes in contact with a placard, immediately rinse with water.

D. Never apply paint to, or paint over, a placard.

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EXTERIOR - PLACARDS

1. Exterior Placards

A. Figure 1 contains images of exterior placards, by location.

(1) Sheet 1: On Oil Filter Access Door, Near Pilot and Passenger Exterior Door Handles, On Main Wheel Fairing, On Nose Wheel Fairing, On Flaps Near Wing Root (Both Sides)

(2) Sheet 2: Near Fill Cap of Fuel Tank, Under Each Wing Near Fuel Drains, Under Left Side Wing Fillet Near Ground Power Supply Plug (S/N 42001 to 42500), Left Underside of Fuselage on Ground Power Supply Plug Cover (S/N 42501 and On), On Exterior of Gascolator Door (Underside of Fuselage)

(3) Sheet 3: On Interior of Gascolator Door, On Exterior of Fuselage - Forward of Wing on Copilot's Side, On Forward Portion of Nose Gear Fairing, On Nose Gear Wheel Pant

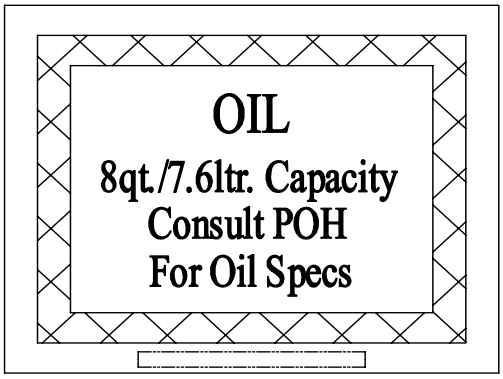
(4) Sheet 4: On Exterior of Fuselage - Forward of Wing on Pilot's Side

(5) Sheet 5: On the Baffling Behind the Cowl Inlet Hole on the Pilot's Side (if Winterization Kit Installed), On the Underside of the Wings Forward of the Jack Points and on Each Side of the Lower Fuselage Near the Rudder (S/N 421015 and On)

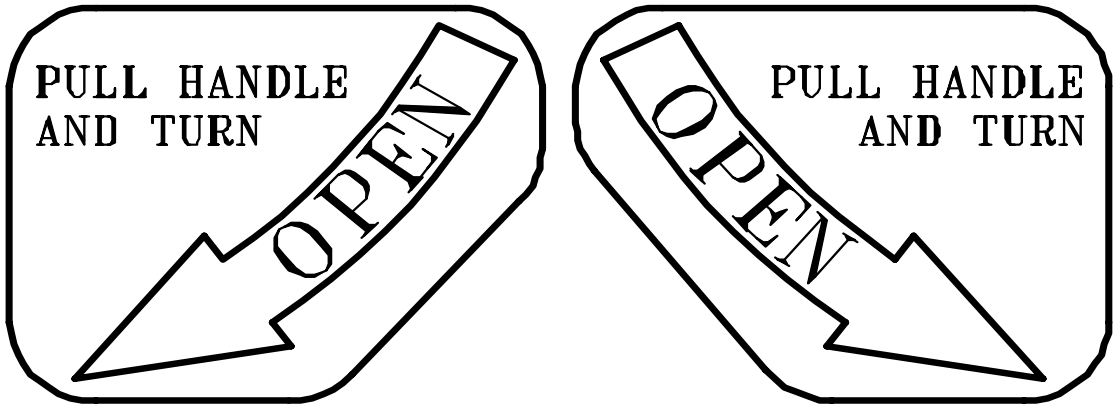
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Figure 1 : Sheet 1 : Exterior Placards

B11712



On Oil Filler Access Door



Near Pilot and Passenger Exterior Door Handles



On Main Wheel Fairing



On Nose Wheel Fairing



On Flaps Near Wing Root (Both Sides)

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Figure 1 : Sheet 2 : Exterior Placards

B11713

Near Fill Cap Of Fuel Tank

S/N 42001 to 42567

AVGAS ONLY
MIN FUEL GRADE 100 / 100LL
TOTAL USEABLE 49 GAL US / 185 LTR
TOTAL CAPACITY 53 GAL US / 201 LTR

S/N 42568 and on

AVGAS ONLY
MIN FUEL GRADE 100 / 100 LL
TOTAL STANDARD USEABLE 43 GAL US / 163 L
TOTAL LONG RANGE USEABLE 51 GAL US / 193 L
TOTAL CAPACITY 53 GAL US / 201 L

FOR DRAINING OF WING FUEL SUMP:
TO OPEN: PRESS CUP GENTLY INTO BOTTOM OF VALVE TO DRAIN REQUIRED AMOUNT OF FUEL.
TO CLOSE: REMOVE CUP AND VALVE WILL CLOSE

TO DRAIN WING TANKS: REFER TO MAIN TENANCE MANUAL .

Under Each Wing Near Fuel Drains

12 VDC POWER CONNECTION

Under Left Side Wing Fillet Near Ground Power Supply Plug (S/N 42001 to 42500)

24 VDC POWER CONNECTION

Left Underside of Fuselage on Ground Power Supply Plug Cover (S/N 42501 and on)

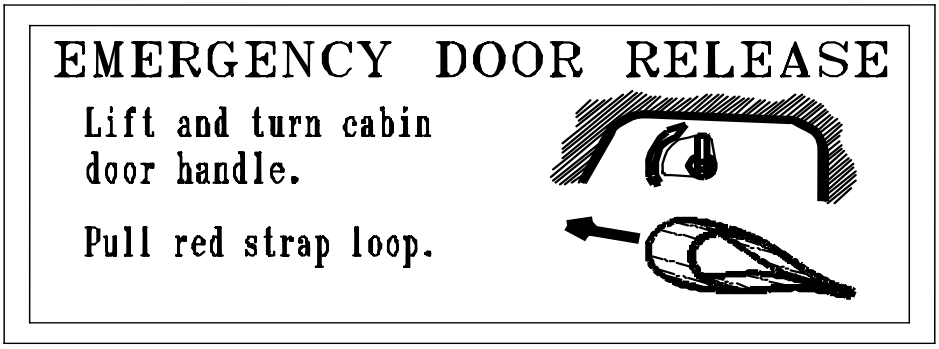
EMERGENCY
DOOR RELEASE
PRESS BUTTONS TO OPEN

On Exterior of Gascolator Door (Underside of Fuselage)

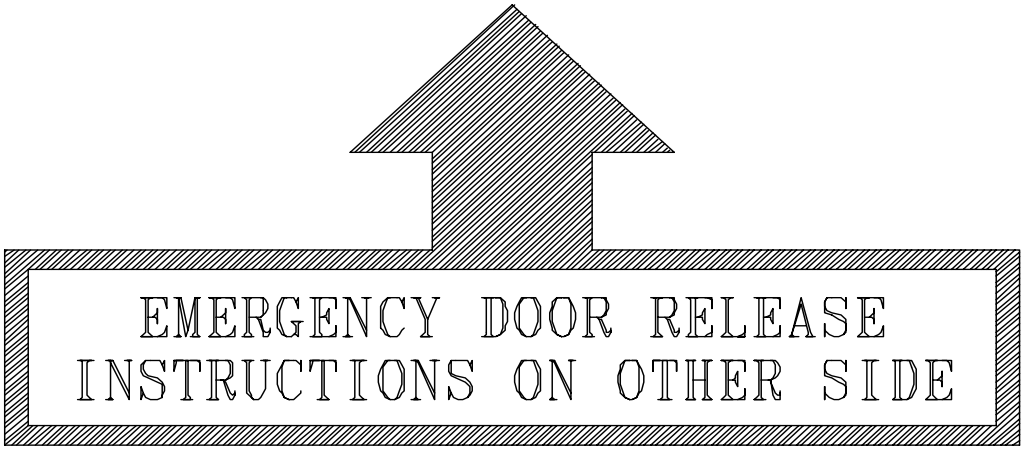
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Figure 1 : Sheet 3 : Exterior Placards

B11714



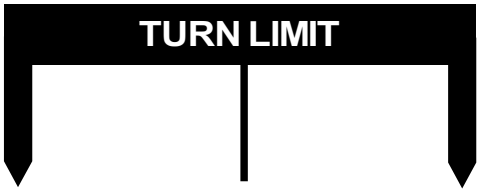
On Interior of Gascolator Door



On Exterior of Fuselage - Forward of Wing on Copilot's Side



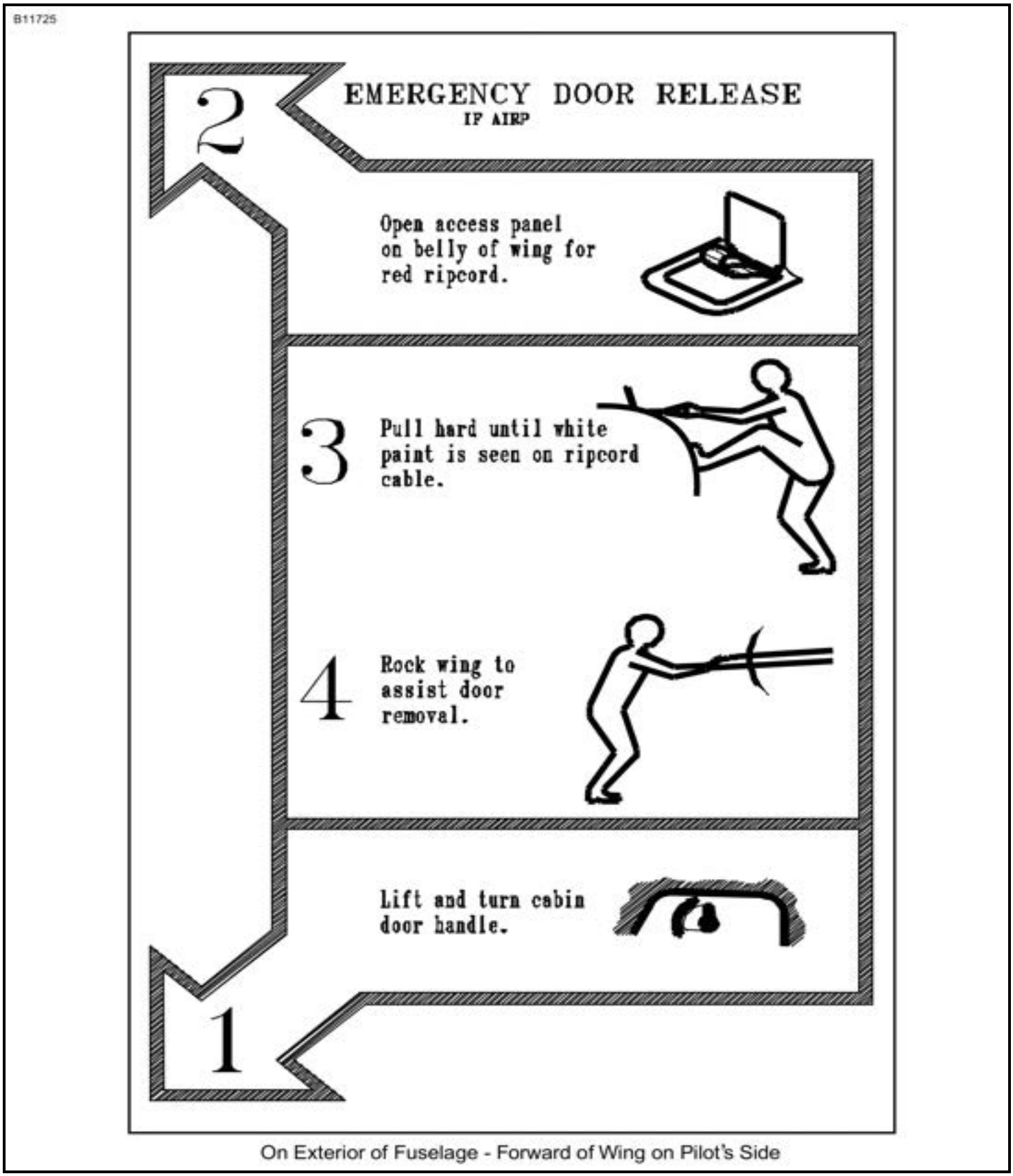
On Top of Nose Wheel Fairing (Pointing Aft)



On Forward Portion of Nose Gear Fairing

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Figure 1 : Sheet 4 : Exterior Placards



On Exterior of Fuselage - Forward of Wing on Pilot's Side

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Figure 1 : Sheet 5 : Exterior Placards

B12743

On the Baffling Behind the Cowl Inlet Hole on the Pilot's Side
(if Winterization Kit Installed)



On the Underside of the Wings Forward of the Jack Points and
on Each Side of the Lower Fuselage Near the Rudder
(S/N 421015 and On)



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INTERIOR - PLACARDS

1. Interior Placards

A. This section contains images of interior placards, by location.

(1) Sheet 1: On Center Console Below Radios, On Left Panel Behind Pilot's Control Stick (Garmin G1000 Option)

(2) Sheet 2: On Instrument Panel to Left of Backup Attitude Indicator (Garmin G1000 Option), Near Pilot and Copilot Interior Door Handles (S/N 42001 to 42005)

(3) Sheet 3: Near Pilot and Copilot Interior Door Handles (S/N 42006 and On), Near Interior Door Handle on Passenger Side,

(4) Sheet 4: On Bottom of Baggage Compartment Door Opening, On the Front of the Pilot's Seat Base on Escape Hatchet, In Aft Cabin on Aft Baggage Bulkhead, Under Left Rear Seat Next to Leveling Washer, Under All Seats, On Parking Brake Handle

(5) Sheet 5: Near Airspeed Indicator (Basic or Avidyne Option), Near Manifold Pressure Gauge (Basic or Avidyne Option), On Center Overhead Console (Basic or Avidyne Option)/On Compass (Garmin G1000 Option), Above Copilot's Fresh Air Vent (Basic or Avidyne Option)

(6) Sheet 6: On Right Panel Behind Copilot's Control Stick (Garmin G1000 Option), On Top Center of Engine Instrument Panel (when autopilot installed) (Basic or Avidyne Option), On Top Center of Flight Instrument Panel (when autopilot installed and FD Only Mode enabled) (Basic or Avidyne Option), On Top Center of Flight Instrument Panel (when autopilot installed and FD Only Mode disabled) (Basic or Avidyne Option)

(7) Sheet 7: Near the Left Dimmer Switch on the Pilot's Knee Bolster (Basic or Avidyne Option), On the Upper Left Side of the Tower Assembly (Garmin G1000 Option), On the Back Lower Portion of the Front Seat Headrests (S/N) 42001 to 42005), On Top Front of Center Console (Garmin G1000 Option)

(8) Sheet 8: On Engine Instrument Panel Above Fuel Gauge (Basic or Avidyne Option), On Engine Instrument Panel near EGT/CHT (when optional JPI digital engine scanner is installed), On Left Center of Engine Instrument Panel (when optional CO detector is installed) (Basic or Avidyne Option)

(9) Sheet 9: On Oxygen Distribution Manifold (if present) in Forward Overhead Panel, On Oxygen Fill Port (if present) set into Hat Shelf, On Air Conditioning System Bay Access Cover

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Figure 1 : Sheet 1 : Interior Placards

B11726

FIRE EXTINGUISHER LOCATED UNDER CO-PILOT'S SEAT

The markings and placards installed in this airplane contain operating limitations that must be complied with when operating this airplane in the Utility category. Other operating limitations that must be complied with when operating this airplane in this category are contained in the Airplane Flight Manual.

Utility Category – No acrobatic maneuvers approved, except those listed in the Pilot's Operating Handbook.

FLIGHT INTO KNOWN ICING PROHIBITED.

SPINS PROHIBITED.

APPROVED FOR DAY/NIGHT – VFR/IFR.

NO SMOKING

On Center Console Below Radios

FIRE EXTINGUISHER
LOCATED UNDER CO–PILOT'S SEAT

The markings and placards installed in this airplane contain operating limitations that must be complied with when operating this airplane in the Utility category. Other operating limitations that must be complied with when operating this airplane in this category are contained in the Airplane Flight Manual.

Utility Category· No acrobatic maneuvers approved, except those listed in the Pilot's Operating Handbook

FLIGHT INTO KNOWN ICING PROHIBITED.
SPINS PROHIBITED.
APPROVED FOR DAY/NIGHT· VFR/IFR

NO SMOKING

On Left Panel Behind Pilot's Control Stick (Garmin G1000 Option)

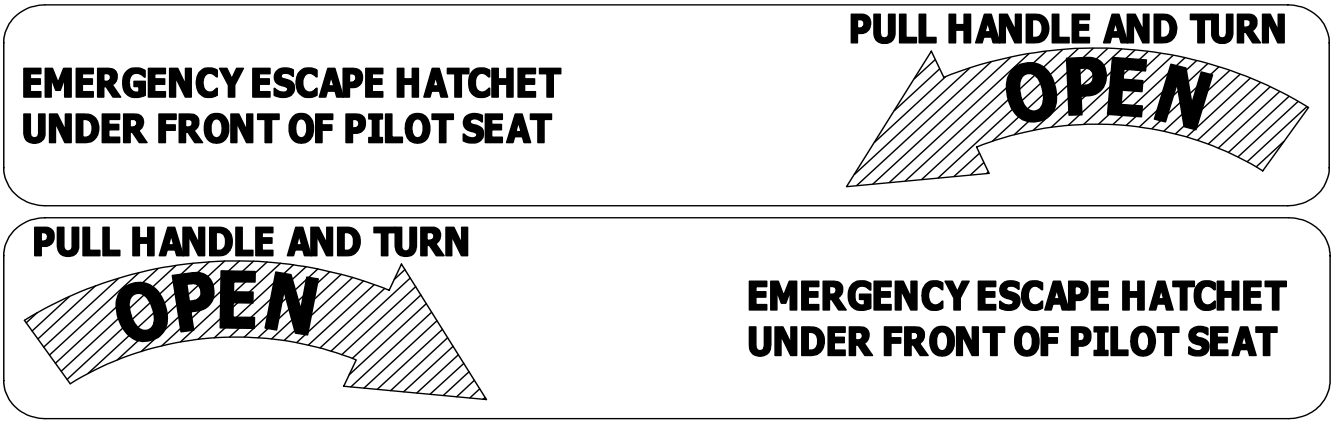
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Figure 1 : Sheet 2 : Interior Placards

B11727

MANEUVERING SPEED
148 KIAS (3400 LBS)
127 KIAS (2500 LBS)

On Instrument Panel to Left of Backup Attitude Indicator (Garmin G1000 Option)

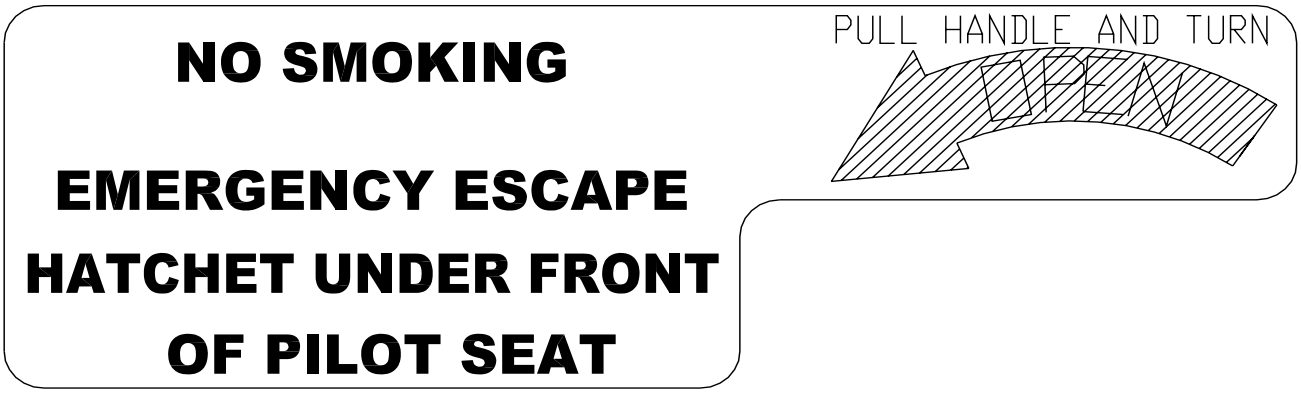


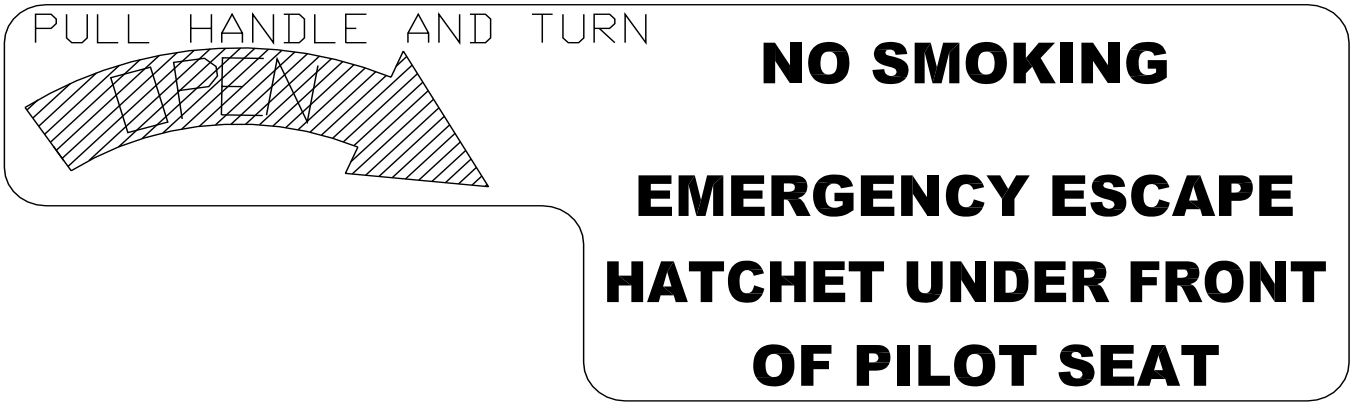
Near Pilot and Copilot Interior Door Handles (S/N 42001 to 42005)

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Figure 1 : Sheet 3 : Interior Placards

B11728





Near Pilot and Copilot Inte rior Door Handles (S/N 42006 and on)



Near Inte rior Door Handle on Passenger Side

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Figure 1 : Sheet 4 : Interior Placards

B11729

On Bottom of Baggage Compartment Door Opening

TOTAL BAGGAGE WEIGHT 120 LBS., SHELF WEIGHT 20 LBS. MAX.
SEE WEIGHT AND BALANCE DATA FOR ADDITIONAL LOADING INSTRUCTIONS

or

TOTAL BAGGAGE WEIGHT 120 LBS. (54 kg.), SHELF WEIGHT 20 LBS. (9 kg.), MAX.
SEE WEIGHT AND BALANCE DATA FOR ADDITIONAL LOADING INSTRUCTIONS

IN EMERGENCY
BREAK DOOR WINDOW

On the Front of the Pilot's Seat Base on Escape Hatchet

EMERGENCY LOCATION TRANSMITTER
LOCATED AFT OF THIS POINT. IT MUST
BE MAINTAINED IN ACCORDANCE WITH
THE FEDERAL AVIATION REGULATIONS

In Aft Cabin on Aft Baggage Bulkhead

AIRCRAFT LEVELING WASHER.
DO NOT REMOVE WASHER.

Under Left Rear Seat Next to Leveling Washer

SEAT BOTTOM CAVITY
MUST REMAIN CLEAR

Under All Seats
(not required under Oregon Aero seats)

BRAKE
ENGAGED

On Parking Brake Handle

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Figure 1 : Sheet 5 : Interior Placards

B11730

MANEUVERING SPEED: **148 KIAS (3400 LBS)**
127 KIAS (2500 LBS)

Near Airspeed Indicator (Basic or Avidyne Option)

ABOVE 12,000 FT (PA) REDUCE: V_{NE} **5 KIAS/1000 FT**
V_{NO} **4 KIAS/1000 FT**

Near Airspeed Indicator (Basic or Avidyne Option)

DO NOT EXCEED 20"
MANIFOL D PRESSURE
BELOW 2200 RPM

Near Manifold Pressure Gauge (Basic or Avidyne Option)

Without Electric A/C

For Steer	N	30	60	E	120	150
For Steer	S	210	240	W	300	330
Radios: On Strobes: On				DATE __/__/__		

With Electric A/C

Deviation > 10° with air conditioning on						
For Steer	N	30	60	E	120	150
For Steer	S	210	240	W	300	330
Radios: On Strobes: On				DATE		

The magenetic direction indicator is calibrated for level flight with the engine, radios, and strobes operating.

On Center Overhead Console (Basic or Avidyne Option)
On Compass (Garmin G1000 Option)

Acrobatic Maneuvers:
Chandelles.....150 KIAS entry speed
Lazy Eights....150 KIAS entry speed
Steep Turns....150 KIAS entry speed
Stalls.....Slow Deceleration
SPINS PROHIBITED

Above Copilot's Fresh Air Vent (Basic or Avidyne Option)

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Figure 1 : Sheet 6 : Interior Placards

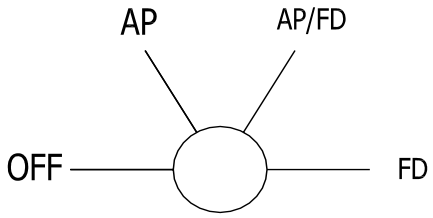
B11731

Acrobatic Maneuvers:
Chandelles.....150 KIAS entry speed
Lazy Eights....150 KIAS entry speed
Steep Turns....150 KIAS entry speed
Stalls.....Slow Deceleration
SPINS PROHIBITED

On Right Panel Behind Copilot 's Control Stick (Garmin G1000 Option)

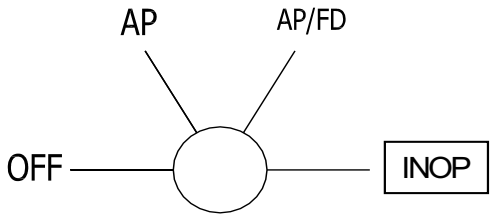
AUTOPILOT OPERATION - SPEEDS/ALTITUDE
MIN OPERATING: 90 KIAS
MAX OPERATING: 210 KIAS
MIN ALTITUDE IN APPROACH: 400 FT. AGL

On Top Center of Engine Inst rument Panel (when autopilot installed) (Basic or Avidyne Option)



AUTOPILOT MASTER

On Top Center of Flight Inst rument Panel (when autopilot installed and FD Only Mode ena bled) (Basic or Avidyne Option)



AUTOPILOT MASTER

On Top Center of Flight Inst rument Panel (when autopilot installed and FD Only Mode disa bled) (Basic or Avidyne Option)

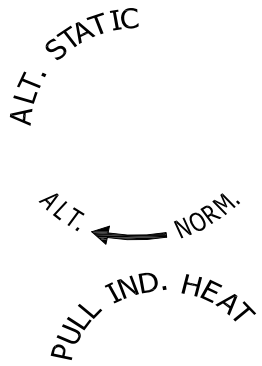
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Figure 1 : Sheet 7 : Interior Placards

B11732



Near the Left Dimmer Switch on the Pilot 's Knee Bolster (Basic or Avidyne Option)



On the Upper Left Side of the Tower Assembly (Garmin G1000 Option)

NO SMOKING

EMERGENCY ESCAPE HATCHET
UNDER FRONT OF PILOT SEAT

On the Back Lower Portion of the Front Seat Headreasts (S/N 42001 to 42005)
Embroidered into the leather with red stitching



On Top Front of Center Console (Ga rmin G1000 Option)

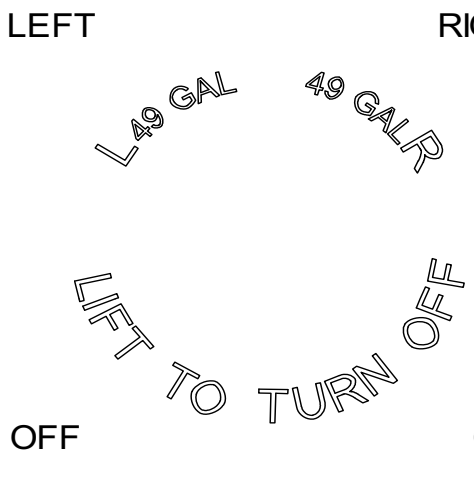
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Figure 1 : Sheet 8 : Interior Placards

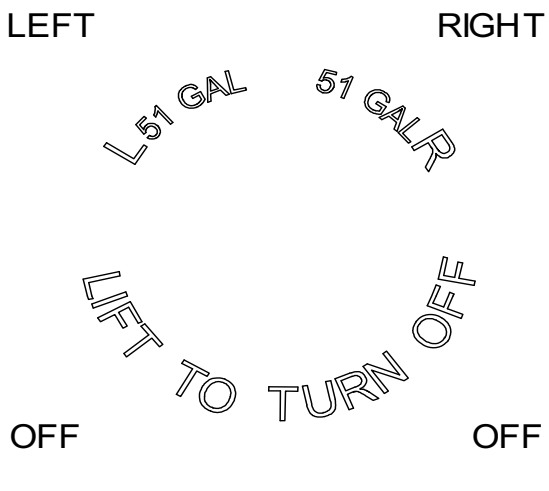
B11733

Engraved On Fuel Selector Knob and Upper Plate

S/N 42001 to 42567



S/N 42568 and on



MAXIMUM FUEL IMBALANCE
NOT TO EXCEED 10 GAL

On Engine Instrument Panel Above Fuel Gauge (Basic or Avidyne Option)

MAX
CHT
460°F

On Engine Instrument Panel Near EGT/CHT (when optional JPI digital engine scanner is installed)

CO
DETECT

TEST/RESET

On Left Center of Engine Instrument Panel (when optional CO detector is installed)
(Basic or Avidyne Option)

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Figure 1 : Sheet 9 : Interior Placards

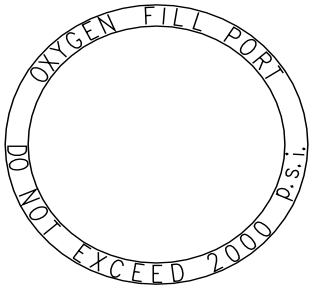
B11734

OFF

OXYGEN SHUTOFF

ON

On Oxygen Distribution Manifold (if present) in Forward Overhead Panel



On Oxygen Fill Port (if present) set into Hat Shelf

UPON REINSTALLATION ENSURE THIS ACCESS PANEL IS SEALED TO PREVENT CARBON MONOXIDE FROM ENTERING THE CABIN

On Air Conditioning System Bay Access Cover

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GENERAL - TIME LIMITS AND MAINTENANCE CHECKS

1. General

A. This chapter sets forth recommended scheduled maintenance checks and component overhaul/replacement time periods. These requirements are in addition to those stipulated in Chapter 4. An inspection checklist is included which should be photocopied and utilized during each scheduled inspection. The checklist includes the minimum items recommended by the manufacturer to be inspected and/or serviced. Also included is a list of special inspections and service items, which must be complied with in order to ensure the continued safe operation of this airplane.

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TIME LIMITS AND INSPECTIONS

1. Time Limits

A. The information in Table 1 identifies either component time limits and/or required special inspections. These inspections must be incorporated into a 100 hour or annual inspection as applicable. If it is anticipated that the component time limit and/or special inspection will occur before the next applicable required inspection, it must be incorporated in the current required inspection.

Table 1. Time Limits and Special Inspection Items

Special Inspection Item	Time Period	Chapter
1. Remove, inspect, clean, and reinstall the fuel nozzles. If ultrasonic cleaning is available, it is the preferred method to use. **Do not** insert anything into the fuel nozzle orifice. If cleaning will not remove internal deposits, the fuel nozzle must be replaced.	At first 100 hour inspection, and every 300 hours thereafter	28
2. Emergency Locator Transmitter (ELT) Battery	Every 2 years (Artex 200) or every 5 years (Artex ME406)	25
3. Altimeter and Static System Inspection - Refer to AC 43-6B, Appendix F of 14 CFR Part 43, and 14 CFR Part 91.	Every 24 calendar months	34
4. Automatic Pressure Altitude Encoding System and Transponder inspection and test - Refer to AC 43-6B, Appendix F of 14 CFR Part 43, and 14 CFR Part 91.	Every 24 calendar months	34
5. Perform a compass swing.	Every 24 calendar months	34
6. Repack new wheel bearings (nose and main landing gear) after first 100 hours, then every 400 hours thereafter. This should be done more frequently if experience indicates the need based on the type of operations conducted.	Every 400 hours	32
7. Inspect fuel selector valve.	At first 500 hour inspection, and every 1000 hours thereafter	28
8. Check fuel gauge accuracy.	Every 500 hours	28
9. Disassemble magnetos and inspect in accordance with TCM Manual X42002-1.	Every 500 hours	71
10. Overhaul magnetos in accordance with TCM Manual X42002-1.	Every 4 years or at engine overhaul, whichever occurs first	71
11. Remove and inspect oil pressure relief valve in accordance with the TCM Overhaul Manual. Check plunger for scraping and nicks. Check the conical face for roughing. Check limits in accordance with TCM specifications. Replace or reface parts as required. Refer to Chapter 18 of the TCM Overhaul Manual.	Every 500 hours	71
12. Inspect exhaust tone cone.	Every 500 hours	78
13. Check and record the control deflections on the ailerons, flaps, elevator, and rudder.	Every 1000 hours	27
14. Inspect aileron control rods for cracking.	Every 500 hours or biennially	27
15. Inspect aileron linear bearing for foreign object debris. Ensure there is no debris or adhesive attached to the aileron control rod, bearing housing, bearing race, or ball bearings.	Every 100 hours or annually	27
16. Inspect aileron trim tab friction device for proper sliding friction. Assure device is free from oils, grease, dirt, etc.	Every 500 hours or biennially	27
17. Inspect and service elevator trim motor.	Every 3000 hours	27

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18. If Garmin GFC 700 autopilot system is installed, visually inspect each Garmin GSM 85 servo mount and verify slip clutch settings.	Every 12 months	22
19. If the Garmin GFC 700 autopilot system is installed, visually inspect the GSA 81 servos every 1000 hours or 12 months (whichever occurs first), clean and apply grease to the output gear every 1000 hours or 3 years (whichever occurs first), and inspect the GTA 82 trim adapter.	Every 1000 hours, 12 months, or 3 years, as applicable	22
20. Inspect the alternators to ensure the bearings operate smoothly. If the bearings do not operate smoothly replace the alternator. If the No.1 (right) alternator brushes or the Kelly alternator brushes are more than 75 percent worn, replace the alternator. If the Plane Power alternator brushes extend less than 0.250 inches from the edge of the brush holder case, replace the brush assembly. For alternator No. 1 (right) ensure the clutch resistance is 140 in-lbs or greater of torque.	Every 1000 hours or 5 years	24
21. Remove rudder and check hinges.	Every 1000 hours	27
22. Check resistance of all static wicks between the wick base and the tip. The test requires use of a megohmmeter, sometimes referred to as a Megger. A properly functioning static wick should indicate 1 to 100 megohms of resistance.	Every 100 hours or annually	20
23. On each main gear, remove the wheel pants, wheels, axles, strut fairings, and underside gearbox access panels. Perform detailed inspection of the landing gear legs. Look for evidence of metal fatigue, corrosion, cracks, and deformation. Pay particular attention to the area where the strut is inserted into the gearbox.	Every 1000 hours	32
24. Service nose strut.	Every 200 hours or annually	12
25. Remove front and rear seats, carpeting, and interior side panels. Inspect the four SAFE strips in the lower fuselage interior for wear, tearing, adhesion, and overall general integrity.	Every 1000 hours	20
26. Conduct exhaust system pressure test.	Every 1000 hours or every 5 years, whichever occurs first	78
27. Inspect exterior emergency door release system. Activate exterior hinge release mechanism to ensure proper operation. Inspect hinge wires for corrosion, kinks, and wear. Replace as required. Move cable up and down to ensure ease of movement. Inspect overhead cabling for chaffing, binding, and frayed cables. To rearm the system, reinsert door hinge wires, and pull both the actuating cables back to their stops. Store the few inches of excess cable in a manner that prevents kinking or binding. Refer to Chapter 52.	Every 1000 hours or every 5 years, whichever occurs first	52
28. Inspect power grid.	Every 1000 hours	24
29. Clean door seal system pump filter.	Every 1000 hours	52
30. Purge the oxygen system. Inspect the oxygen lines and fitting for cracks, leaks, and damage. Overhaul the regulator/valve assembly, replace O-rings, and verify the regulator pressure setting. Replace the flexible oxygen lines on the breathing stations. Overhaul the A4/A5 flowmeters.	Every 5 years	35
31. Oxygen Cylinders - Hydrostatically test 5 years after date stamped on the cylinder, and every 5 years thereafter.	Every 5 years	35
32. Hard Landing Inspection - In the event of a hard landing, the aircraft should be inspected in accordance with Section 32-20-00, Hard Landing Inspection.	After Hard Landing	32

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33. Lightning Strike - In the event of a lightning strike follow inspection and repair procedures in Chapter 20.	After Lightning Strike	20
34. VCS Compressor - Inspect belt and adjust as required to 50 to 70 lbs tension.	First 25 hrs and every 100 hrs thereafter	21
35. Garmin GDC 74A - The static system and altimeter must be verified in accordance with 14 CFR sect. 91.411 and Appendix E of 14 CFR Part 43. Test procedure per the latest revision of the *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03.	Every 2 years	34
36. Rosen Sunvisors - Clean the lenses and adjust the pivot tensions on the visor assemblies.	Periodically	25
RECOMMENDED REPLACEMENTS		
37. Replace oxygen cannulas and/or non microphone oxygen masks.	Every 200 hours of use or every 3 years, whichever occurs first	35
38. Replace microphone oxygen masks.	Every 500 hours of use	35
39. Replace the O-rings in the CPC connector assembly identified on the breathing stations.	Every 3 years	35
40. Replace or rebuild the gyroscopic attitude indicator (artificial horizon).	Every 1200 hours	34
41. Replace or rebuild the gyroscopic directional indicator (DG).	Every 1200 hours	34
42. Replace or rebuild the turn coordinator.	Every 1800 hours	34
43. Replace clock battery. **NOTE:** **The Avidyne FlightMax Multi-Function Display (MFD) and the Garmin G1000 MFD include a clock function. If the MFD is installed in the aircraft there will be no separate clock requiring battery replacement.**	Every 2 years	31
44. Overhaul the propeller. **NOTE:** **The propeller must be overhauled at 2400 hours of time in service or 72 calender months, whichever occurs first by an authorized propeller overhaul facility.**	Every 2400 hours or 72 calendar months, whichever occurs first.	61
45. Overhaul the engine. **NOTE:** **The IO-550-N engine requires a complete overhaul every 2000 hours of time in service by an authorized engine repair facility. There are also overhaul limitations on the magnetos, which are discussed in this section. Inspect the engine mount bushings for corrosion; replace the bushing(s) if corroded.**	Every 2000 hours	71
46. Overhaul the propeller governor. **NOTE:** **The propeller governor requires a complete overhaul every 1800 hours of time in service or 60 calendar months, whichever occurs first, by an authorized propeller repair facility.**	Every 1800 hours or 60 calendar months, whichever occurs first.	61

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INSPECTION REQUIREMENTS

1. Inspection Requirements

A. **Annual Inspection** - Part 91.409 of the Federal Aviation Regulations (FARs) requires that an airplane must receive an annual inspection each 12 calendar months in accordance with part 43 of the FARs and be approved for return to service by a person authorized by the requirements of 43.7. For the purpose of the FARs, 12 calendar months extend from the particular day within the month the inspection is performed to midnight of the last day of the same month, one year later.

B. **100-Hour Inspection** - Part 91.409 of the FARs require that if an airplane is used to carry passengers for hire or give flight instruction for hire, the airplane must receive an inspection after 100 hours of time in service as specified in part 43 of the FARs. The 100 hour limitation may be exceeded by not more than 10 hours while en route to reach a place where the inspection can be done. The excess time used to reach a place where the inspection can be done must be included in computing the next 100 hours of time in service.

C. **50 Hour Inspection** - It is recommended that at each 50 hours of time in service the airplane receive a 50-Hour Inspection.

D. **Unscheduled Inspections** - If the limitations of the engine or airframe are exceeded such as a hard or overweight landing, the likely affected components and systems must be inspected as indicated in this chapter. Refer to Chapter 32 for hard landing inspection details. If a situation occurs which is not covered in this chapter, consult Cessna for technical support and recommendations.

E. **Progressive Inspection** - At some point in the future the outline for a progressive maintenance program will be published and included as a revision to the Cessna Maintenance Manual.

F. **Component Inspections** - Time limits and special inspection requirements may exist for equipment manufactured by other suppliers and installed on Cessna and should be followed to ensure continued airworthiness of that equipment and the aircraft as a whole.

2. Cessna 350 (LC42-550FG) Inspection Checklists

A. Pages 2 through 13 of this section contain the inspection items for the required and recommended inspections described in items A. through C. above. Prior to each 50 hour, 100 hour, or annual inspection, an engine run-up should be performed using the checklist in Figure 1. Depending on the specific types of operations, the owner or operator may need to modify and/or increase the frequency of some inspection and/or servicing items.

NOTE: All inspections and maintenance procedures must be in accordance with applicable Federal Aviation Regulations. Please refer to FAR parts 91.403, 91.207, 91.409, 91.411, and 91.413.

Table 1. Post Run-Up Checks

POST-RUNUP CHECKS	Each 50 hours	Each 100 hours	Annually
Engine cowling - Remove engine cowling.	[ ]	[ ]	[ ]
Engine oil - Drain within 30 minutes of engine shutdown. Oil sample should be collected within 15 minutes of engine shutdown. Replenish with appropriate grade of oil. This step may be deleted from the annual inspection, if desired, to keep oil changes on regular 50-hour intervals. However, regulations require that the oil filter still be cut open for inspection.	[ ]	[ ]	[ ]
Engine - Conduct engine differential compression check. Record on checklist coversheet. (Refer to latest TCM Service Bulletin for permissible drop and differential.)		[ ]	[ ]
Oil filter - Replace and torque 16 to 18 ft-lbs. Cut open and inspect old oil filter.	[ ]	[ ]	[ ]
Exterior lights - Check operation of taxi and landing lights, position lights, and strobe lights.	[ ]	[ ]	[ ]
Pitot heater - Check operation. **WARNING:** **Heater gets extremely hot!** Check within 60 seconds after switching on. Do not operate longer than 2 minutes on the ground.		[ ]	[ ]

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Stall warning system - Check operation.		[ ]	[ ]
Auxiliary lights - Check operation of courtesy entry lights, entry step lights, and baggage compartment light by opening one door at a time. Leave one door open to check operation of 10-minute lights-off timer.		[ ]	[ ]
Interior lights - Check operation of instrument flood bar, panel lights, instrument lights, overhead lights, electro-luminescent panels, and all dimming rheostats.		[ ]	[ ]
Annunciator lights - Check operation.		[ ]	[ ]
Aural warning - Check operation.		[ ]	[ ]
Flight controls - Check for smooth operation of all flight controls with flaps in retracted and extended positions. Leave flaps in extended position when checks are completed.		[ ]	[ ]
Door seals - Check operation of door seal air pump and inspect for air leaks in seals. (Note: Both doors must be latched for system to operate.)		[ ]	[ ]
Rudder limiter system - Check operation. Refer to Chapter 34 of the AFM/POH.	[ ]	[ ]	[ ]
ECS ventilation fan - Check operation.		[ ]	[ ]
Trim actuators - Check operation of all actuators and accuracy of position indicators. Check switches on trim panel (basic and Avidyne), right of the master switches in the overhead (Garmin G1000), and on pilot's control stick.		[ ]	[ ]
Emergency Locator Transmitter (ELT) - Check operation of remote switch. **NOTE:** **ELT may be tested only within the first 5 minutes after each hour and should be limited to 3 aural tone sweeps.**		[ ]	[ ]
Flight controls - Operate controls through full range of motion feeling for binding, ratcheting and sloppiness, and listening for unusual noises.		[ ]	[ ]
Brakes - Check for sponginess.	[ ]	[ ]	[ ]
Aircraft batteries - Fully charge, clean battery exterior surfaces, and clean battery cables. Check battery box for evidence of leakage.		[ ]	[ ]
Access panels, fairings, seats, carpets, covers, side panels, overhead panel (100 hour and annual inspection), and spinner - Remove for complete access for inspection. Check for missing or stripped screws. Do not remove nose wheel fairing at this point. Refer to Section 05-22-00 for a listing of the various access covers in the airplane. **WARNING:** **Do not remove fuel tank cover plates.**		[ ]	[ ]

Table 2. Aircraft Records

AIRCRAFT RECORDS	Each 50 hours	Each 100 hours	Annually
Aircraft logbooks - Determine total times, times since overhaul and times since last required or recommended maintenance checks and record on Inspection Coversheet.		[ ]	[ ]
Inoperative items - If any authorized inoperative items have been listed per FAR 91.213, inspect and maintain per FAR 91.405. Verify placard is installed per FAR 43.11. Refer to the equipment list of required items in Section 6, Appendix A, of the AFM/POH.		[ ]	[ ]
Airworthiness Directives (ADs) and Service Bulletins - Check for new and recurring ADs which may need to be complied with during this inspection. Also, check for applicable manufacturer's Service Bulletins.		[ ]	[ ]
Aircraft records - Check for presence and condition of aircraft federal registration form and airworthiness certificate.		[ ]	[ ]

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Approved Flight Manual/Pilot's Operating Handbook (AFM/POH) - Verify that all revisions have been installed and that the Equipment List and Weight and Balance forms are current and correct.		[ ]	[ ]

Table 3. Engine and Propeller

ENGINE AND PROPELLER (Refer to Chapters 30, 61, and 71 for specific details)	Each 50 hours	Each 100 hours	Annually
Cleaning - Clean engine. Protect magneto vents and cabin fresh air intakes.		[ ]	[ ]
Engine - Inspect for obvious signs of damage, oil or fuel leakage, loose components, and chafing or deterioration of hoses, lines, and wires.	[ ]	[ ]	[ ]
Cylinders, rocker box covers, and pushrod tubes - Inspect for cracks, broken fins, oil, fuel or exhaust leakage, and discoloration of cylinder head. Check hardware for signs of loss of torque.		[ ]	[ ]
Crankcase and sump - Inspect for cracks, oil leakage, corrosion and signs of loss of torque of hardware. Inspect engine mount supports for condition.		[ ]	[ ]
Induction system - Inspect the induction tubes for security and condition and loose or leaking couplers. Inspect manifold pressure transducer for security of installation and condition of wires. Check operation and positions of alternate air butterfly valves.		[ ]	[ ]
Induction air filter - Inspect for cleanliness and condition of sealing surfaces. Service per instructions in Chapter 12.	[ ]	[ ]	[ ]
Exhaust heater muff - Remove heat muff sleeve. Conduct pressure check of heat muff/muffler, and inspect for security of studs.		[ ]	[ ]
Exhaust system - Inspect entire system for leaks, cracks, and security. Inspect turbochargers for damage and security.	[ ]	[ ]	[ ]
Fuel metering unit - Inspect for security and presence of fuel leakage. Check operation of throttle and mixture controls. Ensure levers hit stops before cables reach end of travel. Lube throttle lever with LPS-2.		[ ]	[ ]
Fuel manifold valve and distribution lines - Inspect for security, presence of factory seal installed on safety wire, and evidence of fuel leakage. Inspect distribution lines for security, cracks, and signs of leakage.		[ ]	[ ]
Engine-driven fuel pump - Inspect for security, general condition, and evidence of fuel or oil leakage.		[ ]	[ ]
Fuel hoses - Inspect all flexible fuel hoses for routing, chafing, security, and signs of leakage.		[ ]	[ ]
Magnetos - Inspect for security and condition. Check timing to engine and record: L____ degrees BTDC, R____ degrees BTDC. Timing must be within 22±1 degrees BTDC. If either magneto must be adjusted, check logbooks for previous magneto adjustments. If the same magneto has already been "bumped up" twice without being removed to check internal timing, that magneto must be removed for internal timing check.		[ ]	[ ]
Spark plugs - Inspect firing end for condition. Clean, gap, reinspect, and test spark plugs following appropriate plug manufacturer's service manual. Reinstall with **new** gaskets and torque to specifications.		[ ]	[ ]
Ignition harness - Inspect for routing, broken shielding, chafing, and security. Inspect spark plug ends for condition and evidence of electrical arcing.		[ ]	[ ]

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Left Alternator - Inspect for security and general condition. Inspect electrical connections for security. Inspect belt for tension, fraying, and dry rot. Inspect pulleys for nicks, scratches, warpage, and alignment. Remove drive belt and turn alternator rotor to check bearings for abnormal noise or roughness.		[ ]	[ ]
Right Alternator - Inspect for security, general condition and signs of oil leakage. Inspect electrical connections for security. Check air blast tube for obstructions. Any evidence of alternator malfunction requires removal to conduct drive gear hub slippage inspection.		[ ]	[ ]
Electrical cables, wires, and connectors - Inspect for security, routing, chafing, signs of arcing, and proper termination.		[ ]	[ ]
Starter and starter solenoid - Inspect for security, general condition, and signs of oil leakage.		[ ]	[ ]
Oil cooler - Inspect for security of mounting, cracks, straightness of fins, and evidence of oil leakage around base and fin area. Clean debris from fins. Inspect condition and security of oil pressure switch and temperature bulb and associated wiring.		[ ]	[ ]
Engine accessory case - Inspect accessory case and any additional accessories for security, condition, and evidence of leakage or loss of torque of hardware.		[ ]	[ ]
Crankshaft - Inspect for evidence of oil leakage, corrosion, pitting, cracks, discoloration, or signs of overstressing.		[ ]	[ ]
Propeller governor - Inspect for security, evidence of oil leakage, and tightness of lever on governor shaft. Check operation of propeller control cable. Ensure high-RPM stop is hit before cable reaches end of travel.		[ ]	[ ]
Spinner and spinner bulkhead - Inspect for cracks, security to propeller, and security of any dynamic balance weights. Clean inside of spinner.		[ ]	[ ]
Propeller hub - Inspect for cracks, corrosion, leakage of oil, excessive loss of grease, security of static balance weights, and security of mounting to crankshaft. Re-torque **all** mounting nuts if loss of torque is suspected on any nut in accordance with Section 61-10-00). Service with appropriate grease.		[ ]	[ ]
Propeller blades - Inspect for play, cracks, nicks, dents, pitting, corrosion, and leading edge erosion. Dress propeller and paint back of blade. In operational areas of high humidity, wipe the metal surface with an oily rag.	[ ]	[ ]	[ ]
Propeller Heat/De-Ice System - Inspect the propeller de-ice boots, slip ring, brushes and brush block, control switch/annunciator, control module, and wiring.		[ ]	[ ]
Cooling baffles and seals - Inspect for cracks, excessive gaps, and security of mounting to engine. Check intercylinder baffles for security.		[ ]	[ ]
Engine cowls - Inspect for cracks, chafing, heat damage, delamination, evidence of exhaust leakage, condition of fastening system, presence of oil filler door placard, and condition of paint.		[ ]	[ ]
ECS air inlet scoop - Inspect for obstruction of inlets and condition of scoop and screen.		[ ]	[ ]
Firewall - Inspect for cracks, buckling, unsealed holes, and other signs of damage. Although the firewall is not structural, problems here may indicate damage to the composite bulkhead underneath and should be closely checked out. Inspect all firewall-mounted accessories for security of mounting.		[ ]	[ ]
Engine mount/NLG support - Raise NLG off of the ground and inspect for cracks, corrosion, loss of torque of mounting hardware, chafing by cables, wires, hoses, etc., and make sure that any flexing item is secured to the mount.		[ ]	[ ]
Engine isolation mounts - Inspect for general condition signs of loss of torque of engine mount bolts. If necessary, torque bolts in accordance with Section 71-00-00, Engine Removal, Cleaning, and Installation. Check safety wiring and security.		[ ]	[ ]

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Drain and vents lines - Inspect for security, condition, and obstructions.		[ ]	[ ]
Battery box - Inspect for security of mounting and condition.		[ ]	[ ]
Brake fluid reservoir - Inspect for condition, security, and fluid level. Service as necessary.	[ ]	[ ]	[ ]
Bonding ground straps - Inspect for condition.		[ ]	[ ]
Foreign objects - Check engine compartment for foreign objects.	[ ]	[ ]	[ ]

Table 4. Fuselage Exterior and Empennage

FUSELAGE EXTERIOR AND EMPENNAGE (Refer to Chapters 53 and 55 for specific details)	Each 50 hours	Each 100 hours	Annually
Fuselage exterior surface - Inspect for obvious signs of damage, including cracks, holes, buckling, and rippling. Check condition of paint and cleanliness. Check drain holes for obstructions.	[ ]	[ ]	[ ]
Exterior placards - Inspect for presence and condition. Refer to Chapter 11, *Placards and Markings*, for required placards.			[ ]
Windows - Inspect for cleanliness, condition, and sealing.	[ ]	[ ]	[ ]
Cabin doors - Inspect for security and fit. Inspect skin, hinges, gas struts, latching mechanisms, door seals, and upholstery. Lubricate hinges and all moving parts with a quality petroleum or silicone-based oil. Check that door seals inflate properly.		[ ]	[ ]
Baggage door - Inspect for security and fit. Inspect door skin, upholstery, seal, hinge, and latching mechanism. Inspect door strut for condition and security. Lubricate hinge and all moving parts.		[ ]	[ ]
Entry assist step - Inspect for security and condition.		[ ]	[ ]
Pitot port - Inspect for obstruction of pitot, signs of damage, which may affect proper airflow, and security of attachment. Remove the pitot port before attempting to clear any obstructions, making sure the port is not enlarged or damaged in any way. Test for proper operation after reinstallation.		[ ]	[ ]
Static port drain - Check static port drain for evidence of water. Remove cap from drain line and allow any water within the system to drain. Leak check after reinstallation. **CAUTION:** **Do not apply compressed air to the system, since this will result in damage to the static air flight instruments.**			[ ]
Antennas - Inspect for security and condition.		[ ]	[ ]
Aircraft identification plate - Inspect for security and legibility.			[ ]
Fuselage skin - Perform generalized inspection of fuselage for visible damage and evidence of latent damage.		[ ]	[ ]
Aft Fuselage - Perform a visual inspection and tap test of the exterior aft fuselage skin and a visual inspection of the interior aft fuselage skin in accordance with Section 53-70-00, Aft Fuselage Inspection.		[ ]	[ ]
Exterior emergency door release - Inspect exterior emergency door release system. Inspect hinge wires for corrosion, kinks, and wear. Replace as required. Inspect overhead cabling for chaffing, binding, and frayed cables. Store the few inches of excess cable in a manner that prevents kinking or binding. Inspect attachment screws of dynamic door hinge to door for tightness. Refer to Chapter 52.		[ ]	[ ]

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Vertical stabilizer - Perform generalized inspection of vertical stabilizer for visible damage and evidence of latent damage. Inspect hinge attach points for security and condition. Ensure leading edge tape is secure.		[ ]	[ ]
Rudder - Inspect for signs of damage, looseness, play in bearings, and condition of hinge attachments and rudder cable horns. Check security of static balance weight. Check for obstruction of drain holes. Inspect bonding strap and static wicks for condition. Lubricate hinges and cable-attach points. Inspect rudder crossover cable for proper cable tension. Inspect rudder crossover cable for wear, alignment, chafing, binding, and fraying especially where the cable rounds a pulley. Remove and inspect rudder crossover cable for wear and fraying at every 1000 hours and replace as required. Ensure rudder makes full contact with left and right rudder stops.		[ ]	[ ]
Horizontal stabilizer - Perform generalized inspection of horizontal stabilizer for visible damage and evidence of latent damage. Inspect hinge attach points for security and condition. Ensure leading edge tape is secure.		[ ]	[ ]
Elevators - Inspect for signs of surface damage, looseness or play in bearings, and condition of hinges and elevator horn. Check security of static balance weights. Inspect bonding straps and static wicks for condition. Check for obstruction of drain holes. Lubricate piano hinges **only** in accordance with Section 27-30-00, Elevator. Ensure elevator makes full contact with up and down elevator stops.		[ ]	[ ]
Elevator trim tab and actuator - Inspect for condition, corrosion, and security. If the drive rods show any signs of corrosion, replace with new rods. Lubricate hinge and clevis bolt in accordance with Section 27-30-00, Elevator Trim Tab. Check trim servo for smoothness of operation; free from binding. Measure and record total trim tab play: _____inches. **CAUTION:** **Use special care to ensure trim tab friction devices are not lubricated.**		[ ]	[ ]
Rudder Trim - Check for proper security.		[ ]	[ ]

Table 5. Wings

WINGS (Refer to Chapter 57 for specific details)	Each 50 hours	Each 100 hours	Annually
Wing saddle area - Inspect wing spar for cracks, debonding, and attachment to airframe. Check visible attaching hardware for loss of torque. If necessary, torque hardware in accordance with Section 57-10-00, Wing Installation. Inspect aileron bellcranks for cracks and corrosion.		[ ]	[ ]
Fuel tanks - Check that the airplane's fuel filler neck is grounded. Run a continuity check between each fuel filler neck and one of the exhaust stacks. Fuel cap resistance must be within 0 to 500 ohm. Also, check continuity of all static wicks or static wick attachment points to test the integrity of the lightning protection system.		[ ]	[ ]
Wing skins and exterior - Inspect for obvious signs of damage, including cracks, holes, buckling, and rippling. Check condition of paint and placards. Check drain holes for obstructions. Check fuel tank vents for obstructions. Ensure leading edge tape is secure.		[ ]	[ ]
Landing and taxi light assembly - Inspect for cracks, security of mounting, and cleanliness and condition of lens cover. Operate taxi and landing lights in a dark area and ensure that both lights are properly aimed. If lights are not properly aimed, adjust as required. Refer to Chapter 33 for detailed instructions.		[ ]	[ ]
Aileron and flap attach points - Inspect for security of attachment to wing.		[ ]	[ ]

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Aileron (left) servo tab and servo tab clevis - Inspect for damage, looseness, or play in attach bearings and Belleville washers, and condition of rod end attachment. Check security of static balance weights. Inspect bonding straps and static wicks for condition. Check for obstruction of drain holes. Inspect servo tab for proper operation and condition of push-pull rod, bearings, and hinge. If the rods show any signs of corrosion, replace with new rods. Lubricate only servo tab hinge and clevis bolt in accordance with Section 27-10-00, Aileron Servo Tab. Measure and record total servo tab play: _____inches. (Refer to Section 27-00-00, Table 1.)		[ ]	[ ]
Aileron, right - Inspect for damage, looseness, or play in attach bearings and Belleville washers, and condition of rod end attachment. Check security of static balance weights. Inspect bonding straps and static wicks for condition. Check for obstruction of drain holes. Inspect trim tab and actuator for condition and security. Lubricate only trim tab hinge and clevis bolt in accordance with Section 27-10-00, Aileron Trim Tab. Record total trim tab play: _____ inches. **CAUTION:** **Use special care to ensure trim tab friction devices are not lubricated.**		[ ]	[ ]
Aileron trim tab clevis - Inspect trim clevis and related hardware for evidence of corrosion and pitting. Check for proper security.		[ ]	[ ]
Aileron deflections - Ensure that both ailerons seat properly on their up and down stops.		[ ]	[ ]
Flaps - Inspect skins for condition and signs of debonding. Check hinges for play and attachment to wing and flap. Check hinges for corrosion. Check flap push-pull rods and rod end bearings for condition. Inspect condition of bonding strap.		[ ]	[ ]
Flap Motor and Flap Actuators - Clean and cycle flaps up and down to check for smooth operation.		[ ]	[ ]
Flap deflections - Ensure that flaps extend equally on each side of the airplane in both the takeoff and landing configurations. Measure the down deflection on each side using neutral ailerons as a reference point. The difference in static deflection should not be greater than 1/4 in.			[ ]
SpeedBrakes™ - Check Cap Strip Cover screws for security, if loose, remove screws and apply Loctite 242, retorque to 8 in-lbs. Check SpeedBrake top attachment screws for security, if loose, remove screws and apply Loctite 242, retorque to 8 in-lbs. Check drain tubes for debris.	[ ]		
SpeedBrakes™ - Remove cover plate. Clean and inspect unit for damage, corrosion, looseness and proper operation. Lubricate worm and worm gear with LUBRIPLATE. Install cover plate.			[ ]
Fuel Leaks - Check the security of the fuel tanks by inspecting the outer skin tank areas for evidence of fuel stains. Be sure to open the gascolator access panel and inspect all fuel line and fittings in this area.		[ ]	[ ]
Fuel filler and caps - Inspect for proper locking, condition of o-ring and receptacle, and presence and legibility of placards.		[ ]	[ ]
Wing interior - Inspect wing spar through outer access panel for signs of cracking or debonding. Inspect visible bonded areas of ribs and other structures.		[ ]	[ ]
Fuel Vent Hose - Check for proper connection of the fuel vent hoses to the outboard inspection panels. Check the condition of the hoses.		[ ]	[ ]
Flight controls - Inspect all push-pull rods, rod end bearings, and bellcranks for condition, play, and security of attachment. Ensure all rod end jamnuts are tight.		[ ]	[ ]
Contamination Test - Take fuel samples from both wings and fuel strainer. Inspect for contamination and proper grade of fuel. Clean strainer screen each 100 hours or annually, whichever occurs first.	[ ]	[ ]	[ ]

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Garmin GFC 700 Autopilot System - Inspect for condition and function. Refer to Chapter 22.			[ ]

Table 6. Landing Gear

LANDING GEAR (Refer to Chapter 32 for specific details)	Each 50 hours	Each 100 hours	Annually
Nose Gear and shimmy dampener - Place nose gear on a rotating table (shallow height - sometimes referred to as a "Grease Plate") with the normal static weight of the airplane applied to the table. A firm and consistent resistance to the steering rotation should be felt through the full ±60 degrees of wheel movement.		[ ]	[ ]
Lifting the Airplane - Install jacking pads and jack aircraft until all tires clear the floor. (Refer to Chapter 7 for jacking instructions.)		[ ]	[ ]
NLG - Remove nose gear wheel cover and strut fairing. Note that the four bolts attached to the nose gear fork must be removed to remove the nose wheel fairing.		[ ]	[ ]
NLG strut - Inspect from top to bottom for scratches, cracks, corrosion, and signs of overstress. Check for play in strut attaching brackets. Visually inspect for straightness along length of each strut.		[ ]	[ ]
MLG- Remove main landing gear wheel fairings.		[ ]	[ ]
Hydraulic Brake Lines - Inspect brake lines that are tie wrapped to the main gear strut. Check for security and evidence of chaffing. Check for leaks as evidenced by the presence of hydraulic fluid stains.		[ ]	[ ]
Brake calipers and brake lines - Clean and inspect for condition, fluid leakage, and security of components. Check caliper for small amount of free-floating motion in torque plate. Clean anchor bolts and torque plate if caliper does not free-float. Do not lubricate.		[ ]	[ ]
Brake linings - Inspect brake linings, and replace if the wear lines are not visible.		[ ]	[ ]
Wheels and brake discs - Inspect for cracks and corrosion. Inspect brake discs for cracks, pitting, and signs of overheating. Inspect all hardware for signs of loss of torque in accordance with Cleveland Maintenance Manual AWBCMM0001. Measure and record brake disc thickness: Left - _____inches, Right - _____inches.		[ ]	[ ]
MLG tires - Inspect for flat spots, wear, splitting, and dry-rotting. Check tire pressure, and service as necessary.	[ ]	[ ]	[ ]
MLG wheel pants - Inspect for condition. Clean interior.			[ ]
MLG wheel assemblies - Rotate and check for binding, unusual noises, excessive friction, play in bearings, and security of attachment of axles.		[ ]	[ ]
NLG assembly - Verify operation of NLG centering lock with strut fully extended. Remove four attaching bolts from bottom of NLG strut, remove wheel and fork assembly, and remove wheel pant. Remove fairing from strut assembly.		[ ]	[ ]
NLG strut and fairing - Inspect for cracks, corrosion, scratches on piston, fluid leakage, and signs of overstress. Check for security of attachment of strut to engine mount. Check for presence of steering limit markings and condition of strut fairing.		[ ]	[ ]
NLG fork assembly - Inspect for cracks, corrosion, signs of overstress, and sideloading. Inspect towing lug assemblies for cracks, distortion, and security of attachment.		[ ]	[ ]
Service Nose Strut - Check inflation pressure of nose gear, and service as required. Refer to Chapter 12 for servicing instructions.		[ ]	[ ]

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NLG wheel assembly - Inspect for cracks and corrosion. Check all hardware for signs of loss of torque in accordance with Section 32-20-00, Nose Gear - General and Nose Gear Wheel Pant And Fairing.		[ ]	[ ]
NLG tire - Inspect for flat spots, wear, splitting, and dry-rotting. Check tire pressure, and service as necessary.	[ ]	[ ]	[ ]
NLG wheel pant - Inspect for condition. Clean interior.		[ ]	[ ]
Wheel bearings (all) - Clean and inspect for damage, wear, and corrosion. Inspect bearing races for wear and damage. Repack with appropriate grease in accordance with the *Cleveland Maintenance Manual*, AWBCMM0001. **NOTE:** **Always wear rubber gloves when handling dry or repacked bearings!**		[ ]	[ ]
NLG assembly - Reinstall strut fairing and wheel pant, then reinstall wheel and fork assembly on strut. Torque four attaching bolts 140 to 180 in-lbs., and apply torque seal.		[ ]	[ ]
MLG - Reinstall outer and inner main gear wheel covers on each wheel.		[ ]	[ ]
Lowering airplane - Ensure all obstructions are clear from under aircraft and landing gear is ready for aircraft weight, then lower aircraft and remove jacking pads.		[ ]	[ ]

Table 7. Fuselage Interior

FUSELAGE INTERIOR (Refer to Chapters 21, 23, 25, 27, 28, 31, 35, 37, 56, 57 for specific details)	Each 50 hours	Each 100 hours	Annually
Fire extinguisher - Remove fire extinguisher and inspect. Refer to Chapter 26 for instructions.		[ ]	[ ]
Oxygen system - Inspect flexible lines for security of connections, kinks, or tube discoloration. Reattach or replace lines as necessary. If the Garmin G1000 system is installed refer to additional procedures in Chapter 35.	[ ]		[ ]
Inspection covers - Remove rear seat, rear seat inspection plates, ELT access cover, right and left console covers, and glare shield.		[ ]	[ ]
Flight controls - Inspect for nicks, scratches and dents in push-pull rods, play-in rod end and linear bearings, security of linear bearing mounts and rudder cable guide tubes to fuselage.		[ ]	[ ]
Seats - Inspect all seat structures for cracks, corrosion, and general condition. Check seat controls for positive locking. Inspect cushions and upholstery for condition.		[ ]	[ ]
Seat tracks and stops - Inspect for cracks, wear of latching holes and tracks, and security of tracks and stops.		[ ]	[ ]
Restraint systems - Inspect belts for cuts, fraying, broken stitching, and presence of TSO tags. Check all buckles for proper locking and release. Check inertia reels for proper operation. Check attach points to structure for all belts.		[ ]	[ ]
Upholstery and trim - Inspect for general condition, attachment, and cleanliness.		[ ]	[ ]
Placards - Inspect for presence and condition of all required interior placards. Refer to Chapter 11 for required placards.		[ ]	[ ]
Instruments - Check security of instruments in panel and legibility of markings. Ensure that no magnetic tools are used during this procedure.		[ ]	[ ]
Avionics - Check security of radios, GPS display, indicators and controls on panel and center console, legibility of markings and operation of knobs. Ensure that no magnetic tools are used during this procedure.		[ ]	[ ]

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Instrument panel, radio rack, and center console - Inspect for general condition, security of attachment, and cleanliness. Ensure that no magnetic tools are used during this procedure.		[ ]	[ ]
Nav/Com Bypass (Basic and Avidyne Option Only) - Check operation. Check operation of light.		[ ]	[ ]
Garmin G1000 System - Verify operation. Refer to Chapter 34 for annual inspection procedure.			[ ]
Magnetic compass - Inspect for security, clarity of glass, and signs of oil leakage. Inspect compass correction card for presence and legibility of all headings. Ensure that no magnetic tools are used during this procedure.		[ ]	[ ]
Fuel selector valve - Operate through full range and verify that valve locks into each detent. Ensure that green LEDs in fuel gauge (Basic or Avidyne option) or blue lights in the fuel gauge on the MFD (Garmin G1000 option) illuminate.		[ ]	[ ]
Fuel selector valve - Inspect fuel valve for evidence of fuel leakage.		[ ]	[ ]
Environmental Control System (ECS) controls - Check operation of manual controls.		[ ]	[ ]
ECS vents - Check all vents for proper operation of manual controls.		[ ]	[ ]
Instrument panel backside - Remove glare shield and inspect all wires, lines, hoses, control cables, instruments, etc., for any interference, chafing, and loose or stressed connections. Inspect structure of forward bulkhead, engine mount support structures, etc., for cracks, debonding, and general condition. Inspect soundproofing material for condition and adhesion to forward bulkhead.		[ ]	[ ]
Reinstall glare shield - Check security of attachment and condition. The glare shield for the Garmin G1000 option has foil attached and must be grounded correctly with a "U" clip. Check pull-out shield (Avidyne option only) for freedom of movement, condition, and security.		[ ]	[ ]
ECS ducts - Inspect for cracks, holes, routing, and chafing.		[ ]	[ ]
Grounding rails - Inspect for cracks and security along forward length. Pay special attention to bonding straps connecting sections of rails at formers and to engine mount bolts.		[ ]	[ ]
Rudder pedals - Inspect for security, cracks, and play. Check for leakage from brake cylinders. Lubricate rudder cable clevis bolt and brake cylinder attach points. **Do not lubricate rudder pedal torque tube pillow blocks.**		[ ]	[ ]
Parking brake valve - Inspect for security of mounting and signs of leakage.		[ ]	[ ]
Seat box structure - Inspect for cracks, debonding, and any signs of obvious damage.		[ ]	[ ]
Wing spar - Inspect for cracks, debonding, and security of attachment to fuselage.		[ ]	[ ]
MLG box support structure - Inspect for cracks, debonding, and security of hardware. If paint has been removed, metal is exposed, or corrosion is present, clean the surface and apply paint as needed.		[ ]	[ ]
Floor and supporting structure - Inspect for cracks, holes, and security of attachment.		[ ]	[ ]
Flight controls (forward fuselage through baggage compartment) - Inspect for nicks, scratches, and dents in push-pull rods, play in rod end and linear bearings, security of linear bearing mounts, and attachment of rudder cable guide tubes to fuselage.		[ ]	[ ]
Baggage compartment floor - Inspect for cracks, holes, and security.		[ ]	[ ]
Hat shelf - Inspect for cracks, holes, and security.		[ ]	[ ]

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Emergency Locator Transmitter (ELT) - Remove from tray and remove battery cover. Inspect for battery corrosion and any obvious internal or external damage to housing. Verify replacement date on battery matches date on housing placard. Reinstall battery cover. Test operation of G-switch in strict accordance with instructions in Chapter 25. Record battery replacement due date: _____		[ ]	[ ]
ELT installation and support shelf - Inspect ELT wiring and antenna cable for security, routing, and chafing. Check connectors for security of pins and proper connection. Inspect ELT tray and support shelf for cracks and security. Replace tray if any cracks are found.		[ ]	[ ]
Empennage interior structure - Inspect for cracks, debonding, or other signs of damage. Verify that all drain holes are clear of debris.		[ ]	[ ]
Access panels and fairings - Inspect for condition. Check fasteners and receptacles for condition.		[ ]	[ ]
Automatic Climate Control System (ACCS) controls - Check operation of system and of the system control head.		[ ]	[ ]
ACCS air conditioning compressor drive belt - Check belt condition and tension. Replace the belt whenever there is cracking, fraying, or signs of excessive wear.		[ ]	[ ]
ACCS air conditioning compressor drive belt pulleys - Check pulleys for security, tightness, rotational smoothness, and freedom.		[ ]	[ ]
ACCS air conditioning refrigerant hoses - Inspect hoses for chafing, leads, and security.		[ ]	[ ]
ACCS air ducting - Inspect for intake and outlet obstructions, damage, cracks, holes, routing, and chafing.		[ ]	[ ]
ACCS/ECS vents - Check all vents for proper operation of manual controls. Inspect for intake and outlet obstructions, and damage.		[ ]	[ ]

Table 8. Inspection Wrap-Up

INSPECTION WRAP-UP	Each 50 hours	Each 100 hours	Annually
Fuselage and wings - Verify aircraft is free of any tools, parts, and debris, and reinstall all access fairings, panels, carpeting, seats, etc. removed for this inspection.	[ ]	[ ]	[ ]
Engine - Verify oil sump is properly serviced with oil and engine compartment is free of tools, rags, and debris, and install lower cowl.	[ ]	[ ]	[ ]
Engine - Run engine for no more than two minutes at 1000 to 1200 RPM. After shutdown, check for leaks at oil filter base, fuel nozzles, and any other components removed during this inspection. If no leaks are noted, install top cowling.	[ ]	[ ]	[ ]
Aircraft - Operate engine at 1000 to 1500 RPM to warm engine. Operate all aircraft systems to verify proper operation. As engine warms, operate engine systems at appropriate engine speeds and complete all checks listed on Inspection Coversheet.	[ ]	[ ]	[ ]
Aircraft records - Complete entries in logbooks, AD compliance lists, and any other required records and work orders.	[ ]	[ ]	[ ]

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Figure 1 : Sheet 1 : Inspection and Run-up Checklist

B11739

Reg. number	Serial number:	Owner's name:
Date started	Date completed	Work Order #

Inspection
50-hr ☐ 100-hr ☐ Annual ☐

Hobbs	Eng TSN	Prop TSN	**Mechanic's Name**
TT	Eng TSO	Prop TSO	

ELT battery due:	**RUNUP CHECKS**		
Left Battery due:	☐ - check (4) if okay. For all others, write actual indication.		
Right Battery due:			
Altimeter/Transponder test due:	**System**	**Pre-inspection**	**Post-inspection**
O2 bottle hydro. test due:	Boost/primer pump	☐	☐
Fire extinguisher insp. due:	High/Lo boost pump cut-off	☐	☐
Magneto 500-hr insp. due:	Starter	☐	☐
Flexible hose rplcmt. due:	Oil pressure (cool)	psi	psi
Misc.:	Charging system, Left (1200 RPM)	Amps volts	Amps volts
Misc.:	Charging system, Right (1200 RPM)	Amps volts	Amps volts
Oil sample? Yes ☐ No ☐	Left brake	☐	☐
Comments	Right brake	☐	☐
Differential compression check: Taken: Hot ☐ Cold ☐	Parking brake	☐	☐
	Comm/Nav	☐	☐
6 - /80 5 - /80	Ignition ground test (idle)	☐	☐
4 - /80 3 - /80	L. Mag drop (1700 RPM)	RPM Drop	RPM Drop
2- /80 1 - /80	R. Mag drop (1700 RPM)	RPM Drop	RPM Drop
Discrepancies Noted During Runup (Additional space available on last page)	Prop response (1200 RPM)	☐	☐
	Oil temperature	°F	°F
	CAUTION: Ensure CHT and oil temp. are in the green range.		
	Static RPM	RPM	RPM
	Fuel flow at static RPM	gph	gph
	Induct. Heat (1700 RPM)	☐	☐
	Defrost	☐	☐
	Cabin heat	☐	☐
	Idle RPM	RPM	RPM
	CAUTION: Allow engine to cool to 300°F (CHT) before shutdown.		
	All annunciator lights are off	☐	☐
	Check for fuel odors in cabin	☐	☐
	Check for fuel stains on underside of aircraft	☐	☐
☐ See last page for additional comments.	Check fuel valve *off* function	☐	☐
	Idle cutoff RPM rise	RPM Rise	RPM Rise

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LIST OF INSPECTION AREAS

1. List of Inspection Areas

Table 1. Inspection Access Areas

CESSNA 350 (LC42-550FG) INSPECTION ACCESS AREAS			
Area	**Type of Access**	**Location**	**Inspection and Maintenance Items**
Left Wing	Access Panel	Underside, inboard, below fuel tank	Fuel tank, two fuel sending units, slosh box, fuel low annunciator switch, vapor return lines
	Access Panel	Underside, middle, below fuel tank	Fuel tank, one fuel sending unit
	Access Panel	Underside, outboard	Fuel vent
	Access Panel	Underside trailing edge, near right side of aileron	Final aileron drive, mounting bolts to aileron, bellcrank assembly
	Access Panel	Underside towards leading edge, forward of inboard fuel tank access panel	Fuel lines and fuel lines to fuel tank connection fittings
	Access Panel	Aft of inboard fuel tank cover	Final drive for flap bellcranks
	Access Panel	Underside trailing edge, near middle of aileron	Aileron control rod linear bearing
Right Wing	Access Panel	Underside, inboard, below fuel tank	Fuel tank, two fuel sending units, slosh box, fuel low annunciator switch, vapor return lines
	Access Panel	Underside, middle, below fuel tank	Fuel tank, one fuel sending unit
	Access Panel	Underside, outboard	Fuel vent
	Access Panel	Underside trailing edge, near left side of aileron	Final aileron drive, mounting bolts to aileron, bell crank assembly
	Access Panel	Underside towards leading edge, forward of inboard fuel tank access panel	Fuel lines and fuel lines/fuel tank connection fittings, outside air temperature probe
	Access Panel	Aft of inboard fuel tank cover	Final drive for flap bellcranks
	Access Panel	Underside trailing edge, near middle of aileron	Aileron control rod linear bearing
Fuselage Exterior	Access Panel	Underside center near trailing edge of the flaps	Aileron bellcranks and crossover assembly, flap motor, marker beacon tuning slug
	Access Panel	Underside near left wing root	Auxiliary fuel pump, gascolator, pitot drain, wires and plugs for wing electrical (position/strobe lights, landing/taxi lights, stall horn, fuel quantity, and fuel low annunciator)
	Access Panel	Tail cone right side	Air vent, elevator push-pull rods
	Access Panel	Vertical stabilizer, left side	Antenna connection junction
	Engine Cowling - Top and Bottom	Forward of fuselage firewall	All engine and propeller parts and components.
	Access Panel	Under front seat bench	Fuel and fuel vent lines

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Fuselage Interior	Access Panels (2)	Right side of center console (arm rests)	Fuel selector valve, environmental control system (ECS) heated air ducts
	Cover Panel	Rear seat footwell	Wing attachment bolts
	Access Panels (2)	Under rear seat back	Gear box
	Access Panels (2)	Baggage compartment floor	Gear box
	Access Panel	Floor area of baggage compartment (Avionics Bay)	Special optional avionics equipment (DME, HSI slaving transmitter, remote stormscope sensor, altitude hold computer for system 30 autopilot)
	Access Panel	Floor area of baggage compartment hat rack	Power plug, pitch servo
	Access Panel	Aft bulkhead of baggage shelf	ELT, elevator push-pull rods
	Cabin, Side-Trim Panels	Pilot's side - runs from control stick to rear seat back	Aileron and elevator push-pull rods, circuit breaker panel, control stick switches wiring, fuel lines, ECS fresh air ducts
	Cabin, Side-Trim Panels	Copilot's side - runs from control stick to rear seat back	Aileron and elevator push-pull tubes, communication wiring, control stick switches wiring, common wire bundle, ECS fresh air ducts
	Baggage Compartment Trim Panels (2)	Left and right sides of baggage compartment	Elevator push-pull rods
Controls	Access Panel	Right elevator, underside, middle	Trim tab servo motor
	Access Panel	Underside, right aileron	Trim tab servo motor

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UNSCHEDULED MAINTENANCE CHECKS

1. General

A. During operation, the airplane can have forces put on it that make completion of unscheduled maintenance necessary. This section describes the conditions or events that will result in the need to complete an unscheduled maintenance procedure.

B. When the flight or maintenance crew reports that they have experienced of any conditions listed below, you must complete the unscheduled maintenance check requirements for that condition or event.

2. Unscheduled Maintenance Checks Defined and Areas of Inspection

A. Severe vibration or oscillatory yawing motion due to excessive braking or tire skidding.

(1) If the pilot reports that the airplane experienced severe vibration or oscillatory yawing motion due to excessive braking or tire skidding, complete the Aft Fuselage Inspection procedure found in Section 53-70-00, Fuselage Components - Adjustment/Test.

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PARKING, MOORING, STORAGE, AND RETURN TO SERVICE - DESCRIPTION AND OPERATION

1. Parking

A. During parking operations, it is best to head the airplane into the wind if possible. Normally, setting the parking brake is recommended, however, there are two situations where doing so is not a good idea:

(1) Do not set the parking brake if the brakes are overheated, which might result from a short field landing or extensive taxiing. It is best to not set the parking brake until the brakes have had a sufficient cooling period. A brake pad clamped to a hot chrome disk can cause uneven cooling of the brake disc, which could potentially warp it.

(2) Do not set the parking brake in cold weather. Accumulations of freezing rain, ice, and snow can freeze-weld the brake pad to the disk. Landing or taxiing in standing water at near-freezing temperatures can cause similar problems if the parking brake is set when the airplane is parked.

2. Securing the Airplane

A. In any event, whether the brakes are set or not set, the airplane should be chocked and the following items should be accomplished to secure the airplane.

B. Install the control lock.

C. Chock the main gear tires with chocks on both sides of each tire.

D. Attach a rope or chain to each tie-down point, and secure the rope or chain to a ramp tie-down point. There are three tie-down points, one on each wing and one on the tail. The ropes or chains should have a tensile strength of at least 750 lbs.

E. Install the pitot tube cover.

3. Storage

A. **Overview** - The storage of an airplane mostly deals with engine related items. Very little needs to or can be done to preserve the airframe, particularly for flyable and temporary storage. The best protection for the exterior is, of course, to hangar the airplane. If the airplane cannot be hangared, then a coat of wax using the material and techniques described in Section 8 of the *Pilot's Operating Handbook* should be applied to all exterior surfaces. In addition, all typical items associated with securing the airplane should be done. These include:

(1) Installing the pitot tube cover

(2) Chocking all wheels and tying the airplane down with the parking brakes released

(3) Installing the control lock

(4) Topping off the fuel tanks

(5) Cleaning the bolts and nuts on the brakes and applying a non-stick preservative like graphite or silicone

(6) Installing other owner-option protection devices

There are three types of storage categories, flyable, temporary, and indefinite. The time period and applicable storage procedure for each type is discussed below.

B. **Flyable Storage (7 to 30 days)** - If the airplane is to be maintained in flyable storage, then it should be flown for a minimum of 30 minutes every 30 days; ground running the engine is not a substitute for flying the airplane. During flyable storage, the propeller should be rotated by hand every seven days. This operation should include at least six complete revolutions of the engine. Stop the propeller 45 to 90 degrees from its original position. For maximum safety, use the following procedures:

(1) Ensure that the ignition switch is set to the OFF position.

(2) Set the throttle to the CLOSED position.

(3) Set the mixture to IDLE CUT OFF.

(4) Set the parking brake and chock the wheels.

(5) Ensure that airplane tie-downs are secure.

(6) Open the cabin door on the pilot's side of the airplane.

(7) Always assume the propeller could start when moving it manually and use an appropriate technique for hand turning the propeller.

(8) Release the parking brake when the operation is completed.

WARNING: Always assume that the engine could start when rotating the propeller by hand. Remain clear of the arc of the propeller blades at all times.

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C. **Temporary Storage (up to 90 days)** - Use the following procedures to preserve the engine for temporary storage. Refer to Airframe Preservation for Temporary or Indefinite Storage below for airframe preservation items.

(1) Remove the top spark plug from each of the six cylinders and apply an atomized injection of preservation oil, MIL-L-46002, Grade 1. As the oil is injected into each cylinder, the piston should be near bottom dead center, and the preservation operation should be done at room temperature.

(2) When Step 1 is complete, and with none of the pistons at dead center, re-spray each cylinder thoroughly making sure to cover all interior surfaces.

(3) Install spark plugs.

(4) Spray approximately two ounces of preservation oil through the oil filler tube.

(5) Seal all engine openings exposed to the atmosphere with suitable plugs or moisture resistant tape.

(6) Tag engine, cowling, and other appropriate areas with the statement, "Do not turn propeller, engine preserved."

D. **Indefinite Storage (Over 90 Days)** - If the airplane is to be stored for a long period, follow the procedures listed below to preserve the engine. Refer to Airframe Preservation for Temporary or Indefinite Storage below for airframe preservation items.

(1) Drain the engine oil and refill with MIL-C-6529 Type II preservation oil. Start the engine and operate until normal temperature ranges are achieved. Fly the airplane for about 30 minutes and then allow the engine to cool to the ambient temperature.

(2) Remove the top spark plug from each of the six cylinders and apply an atomized injection of preservation oil, MIL-L-46002, Grade 1. As the oil is injected into each cylinder, the piston should be near bottom dead center, and the preservation operation should be done at room temperature.

(3) When Step 1 is complete, and with none of the pistons at dead center, re-spray each cylinder thoroughly making sure to cover all interior surfaces.

(4) Spray approximately two ounces of preservation oil through the oil filler tube.

(5) Install dehydrator plugs MS27215-1 or -2, in each of the top spark plug holes. Ensure the dehydrator plug is blue when installed. Protect and support the spark plug leads with AN-4060 protectors.

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

(7) Seal the induction system with moisture resistant tape.

(8) Seal the engine breather by taping a dehydrator plug, M527215-2, in the lower end. Seal the "whistle hole" vent in the breather tube with moisture resistant tape.

(9) Tag engine, cowling, and other appropriate areas with the statement, "Do not turn propeller, engine preserved."

(10) Install plugs in engine cowl inlets, wing leading edge air inlets, and all other openings. **Do not plug or seal tank vents on the bottom of each wing.**

NOTE: During the storage period, FAA Airworthiness Directives and manufacturer's service bulletins may apply which require action based on calendar dates, not operating hours. These items must still be complied with even though the airplane is in storage.

NOTE: The dehydrator plugs must be visually checked every 30 days to verify that the color has not changed. Bad dehydrator plugs should be replaced. If more than half of the plugs change color, the bad plugs and all the desiccant bags on the engine should be replaced.

(11) Every six months the dehydrator plugs should be replaced and the cylinders re-sprayed with preservation oil. When removing the plugs, check the cylinder interior. If rust stains are noted, spray the cylinder with preservation oil, turn the prop through six revolutions, and then re-spray all cylinders.

E. **Airframe Preservation for Temporary and Indefinite Storage** - If the airplane is to be stored for over 30 days, some or all the procedures below may be applicable, depending on the anticipated storage time period:

(1) Ensure the tires are free of grease, oil, tar, and gasoline. The presence of these items accelerates the aging process. Sunlight and static electricity convert oxygen to ozone, a substance that accelerates the aging process. Special tire covers can be installed which retard the erosion process.

(2) It is best if the weight of the airplane is removed from the tires to prevent flat spots. If the airplane cannot be blocked or set on jacks, then every 30 days each wheel should be rotated about 90 degrees to expose a new tire pressure point.

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(3) If the airplane does not have a recent coat of wax, a new coat should be applied as discussed in Chapter 12 of this manual.

(4) Lubricate exposed exterior metal fittings, hinges, push rods, etc. Use plugs or moisture resistant tape to seal all openings except fuel vent holes and drain holes.

(5) Remove the battery and store in a cool, dry location. The battery may need periodic servicing and recharging depending on the storage period.

(6) Prominently tag areas where tape and plugs are installed.

4. Inspections During Temporary Storage

A. The following inspections should be performed while the airplane is in temporary storage:

(1) Check the cleanliness of the airframe as frequently as possible and remove any dust that has collected.

(2) Check the condition and durability of the protective wax coat and renew as required.

(3) Every 30 days, check the interior of at least one cylinder for evidence of corrosion.

5. Inspections During Indefinite Storage

A. The following inspections should be performed while the airplane is in indefinite storage:

(1) Check the condition of the dehydrator plugs ever 30 days to verify that the color has not changed. Bad dehydrator plugs should be replaced. If more than half of the plugs change color, the bad plugs and all the desiccant bags on the engine should be replaced.

(2) Every six months the dehydrator plugs should be replaced and the cylinders re-sprayed with preservation oil. When removing the plugs, check the cylinder interior. If rust stains are noted, spray the cylinder with preservation oil, turn the prop through six revolutions, and then re-spray all cylinders.

NOTE: When an airplane has been in storage for a long period, the date of the required annual inspection may have passed. There is no requirement to perform this inspection during the temporary or indefinite storage period. However, the inspection must be completed before the airplane is returned to service.

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RESTORING - MAINTENANCE PRACTICES

1. Return to Service from Temporary Storage

A. To return an airplane that has been in temporary storage to active service, perform the following steps:

(1) Remove seal plugs, tape, and all methods of tagging the airplane, including items tagged on the airframe.

(2) Remove the bottom spark plug from each of the six cylinders and rotate the propeller several times to remove the preservation oil.

(3) Reinstall the spark plugs according to manufacturer's recommendations.

(4) Conduct a normal engine start, and idle the engine for several minutes until oil temperature is within normal limits. Monitor engine instruments to ensure they are within normal operating ranges.

(5) Stop the engine and inspect the entire airplane before test flying.

(6) Test fly the airplane.

2. Return to Service from Indefinite Storage

A. To return an airplane that has been in indefinite storage to active service, perform the following steps:

(1) Remove all dehydrator plugs, seal plugs, tape, and all methods of tagging the airplane including items tagged on the airframe.

(2) Drain the preservation oil, and service the airplane engine with the recommended lubricating oil.

(3) Remove the bottom spark plugs from each of the six cylinders, and rotate the propeller several times to remove the preservation oil.

(4) Re-install the spark plugs, and carefully rotate the propeller by hand several times to check for possible liquid lock.

(5) Conduct a normal engine start, and idle the airplane for several minutes until the oil temperature is within normal limits. Monitor all engine instruments to ensure they are within normal operating ranges.

(6) Stop the engine and inspect the entire airplane before test flying.

(7) Test fly the airplane.

3. Return to Service Airframe Items

A. To return the airframe portion of an airplane that has been in temporary or indefinite storage to active service, perform the following steps, as applicable:

(1) Remove all methods of tagging and sealing the airplane including any items on or in the engine area.

(2) Remove tire covers or other protective devices. Check the condition of the tires, and service to proper pressures. Cracked, deformed, and desiccated tires should be replaced.

(3) Thoroughly clean the exterior of the airplane, including the transparencies. If necessary, renew the protective wax coat. Refer to Section 12-30-00, Cleaning Aircraft, for instructions on care of the airframe.

(4) Check the condition and charge of the battery. If the battery is still serviceable, reinstall it in the airplane; otherwise, install a new battery.

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LIST OF REVISIONS

1. Revisions

A. This Maintenance Manual includes the original issue and the revisions listed in Table 1. Put revisions in this manual when they are received. This will make sure that this manual has the most current maintenance and inspection procedures.

Table 1. Original Issue - Feb 1/2009

Revision Number	Date	Revision Number	Date
1	Jan 1/2012	2	Jul 1/2012
3	Mar 1/2013	4	Sep 1/2015
5	Sep 1/2016		

B. FAA Approved Airworthiness Limitations are incorporated in this maintenance manual as Chapter 4. Revisions to Chapter 4 are dated as approved by the FAA Regional Manager, Aircraft Certification Office. To make sure that the maintenance information required under Title 14 of the Code of Federal Regulations, Parts 43.16 and 91.409 is current, the revisions listed in Table 2 must be incorporated in Chapter 4 as they are issued.

Table 2. Original Issue - Feb 1/2009

Revision Number	Date	Revision Number	Date
1	Mar 1/2012		

2. Export Compliance

A. This publication contains technical data and is subject to U.S. export regulations. This information has been exported from the United States in accordance with export administration regulations. Diversion contrary to U.S. law is prohibited. ECCN: 9E991

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INTRODUCTION

1. General

A. The information in this publication used the data available at the time of publication and is updated, supplemented, and changed by service letters, service bulletins, publication revisions, reissues, and temporary revisions, which are supplied by subscription services available from Cessna Customer Service. All of these changes become part of and are specifically included in this publication. The latest changes to this publication are given through the Cessna Customer Service subscription services and/or Cessna Service Centers and Cessna Service Stations.

WARNING: The purchaser is warned not to use the data in Cessna's maintenance/service manuals and parts catalogs when parts are designed, manufactured, remanufactured, overhauled, and/or approved by entities, other than Cessna, are installed. when non-Cessna parts are used, the data in Cessna's maintenance/service manuals and parts catalogs are no longer applicable. All of the inspection intervals, replacement time limits, overhaul time limits, the method of inspection, life limits, cycle limits, etc., Cessna recommends are given when new, remanufactured, or overhauled Cessna approved parts are installed. All inspection intervals, replacement time limits, overhaul time limits, the methods of inspection, life limits, cycle limits, etc., for non-Cessna parts must come from the manufacturer and/or seller of the non-Cessna parts.

B. Inspection, maintenance and parts requirements for Supplemental Type Certificate (STC) installations are not given in this manual. When the airplane has an STC installation, those parts of the airplane that the installation has an effect on, must be examined in accordance with the inspection program published by the owner of the STC. Cessna-supplied inspection criteria may not be valid for airplanes that have STC installations because they may change the systems interface, operating characteristics and component loads or stresses on adjacent structures.

2. Coverage

A. The Cessna Model 350 Maintenance Manual is prepared with information given by the ATA (Air Transport Association) Specification Number 2200 for Manufacturer's Technical Data.

B. The information in this maintenance manual is applicable to US and foreign certified Model 350 airplanes. Information unique to a particular country is identified in the affected chapter(s).

C. This manual gives the necessary information required to help maintenance personnel service, examine, troubleshoot, remove and replace components or repair systems on the Model 350 airplanes.

D. Additional information not given in this manual can be found in the Cessna Aircraft Company Model 350 Electrical Manual and Illustrated Parts Catalog. For technical aid applicable to this manual, contact Cessna Customer Service.

E. Use the technical publications available from manufacturers of the different airplane components and systems, which are not covered in this manual, as necessary for maintenance of those components and systems.

3. Temporary Revisions (Paper Version of this Manual Only)

A. Additional information, when it is available, may be supplied by a temporary revision. This service is used to give, without delay, new information, which will help maintain safe flight and ground operations. Temporary revisions are numbered consecutively in the ATA chapter assignment. Page numbering uses the three-element number, which matches the maintenance manual. Temporary revisions are incorporated into the maintenance manual at the next regularly scheduled revision.

4. Serialization

A. All Model 350 airplanes are assigned a serial number. Serial numbers are assigned one after the other to each airplane as it is made and will stay with the airplane during its service life. The serial number is shown on the airplane identification plate.

B. This maintenance manual includes Model 350 airplanes Serial Numbers 42001 and On. The airplane serial number may be shown in the text or in an illustration to identify maintenance instructions specified for an airplane. Instructions and illustrations that are not identified by airplane serial number(s) are standard or optional and they are applicable to airplanes with the system(s) or component(s) installed.

5. Compact Disc-Read Only Memory (CD-ROM)

A. Model 350 publications are available in paper and as electronic media CD-ROM. The paper publications are available as individual publications, such as Maintenance Manual, Illustrated Parts Catalog, etc., while the electronic service publications are all supplied on the CD-ROM. These publications are kept up to date through subsequent revisions.

6. Service Bulletins

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A. Service bulletins prepared for the Model 350 airplanes are recorded in the Service Bulletin List. The Service Bulletin List gives service bulletin information.

(1) Service Bulletin Number - Service bulletins are given numbers by ATA chapter number, followed by a number that is in sequence that has at least two digits, and starts with the numbers zero and one (01).

(2) Service Bulletin Date - The service bulletin date is the date the service bulletin was printed.

(3) Manual Incorporation Date - This shows how the service bulletin affects the manual.

(a) Date - This shows when the service bulletin information was put into the manual.

(b) No Effect - This shows the service bulletin does not affect the manual.

(c) Other self-explanatory statements can show other replacement data.

(4) Title - The title identifies the service bulletin by nomenclature. It is the same title shown on Page 1 of the service bulletin.

7. Maintenance Manual Use

A. Division of Subject Matter.

(1) The Model 350 Maintenance Manual is divided into four major sections. The major sections are separated into chapters. Each chapter has its own List of Effectivity Page and Contents page. The manual divisions are as follows:

(a) Major Section 1 - Airplane General

Chapter	Title
4	Airworthiness Limitations
5	Time Limits and Maintenance Checks
6	Dimensions and Areas
7	Lifting and Shoring
8	Leveling and Weighing
9	Towing and Taxiing
10	Parking, Mooring, Storage and Return to Service
11	Placards and Markings
12	Servicing

(b) Major Section 2 - Airframe Systems

Chapter	Title
20	Standard Practices - Airframe
21	Air Conditioning/Environmental Control System
22	Auto Flight
23	Communications
24	Electrical Power
25	Equipment/Furnishings
26	Fire Protection
27	Flight Controls
28	Fuel
30	Ice and Rain Protection
31	Indicating/Recording Systems
32	Landing Gear and Brakes
33	Lights

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34	Navigation
35	Oxygen
37	Vacuum
39	Electrical Panels

(c) Major Section 3 - Structures

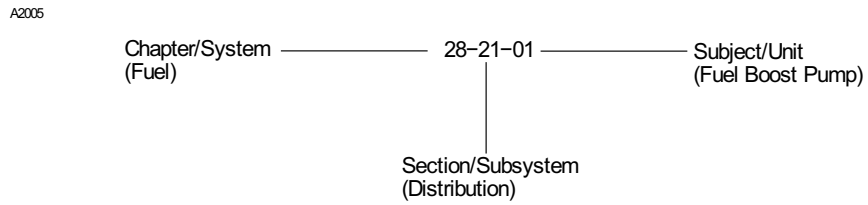
Chapter	Title
51	Standard Practices and Structures
52	Doors
53	Fuselage
55	Stabilizers
56	Windows
57	Wings

(d) Major Section 4 - Power Plant

Chapter	Title
61	Propeller
71	Powerplant
73	Engine Fuel System
74	Ignition
76	Engine Controls
77	Engine Indication
78	Exhaust
79	Oil
80	Starting

B. Page Number System.

(1) The page number system used in the Cessna Model 350 Maintenance Manual has three-element numbers that are separated by dashes. The three-element number is found at the bottom right corner of the page, to the left to the page number. The date is found below the page number.



(2) When the chapter/system element number is followed with zeros in the section/subsystem and subject/unit element number (28-00-00), the information is applicable to the entire system.

(3) When the section/subsystem element number is followed with zeros in the subject/unit element number (28-21-00), the information is applicable to the subsystems in the system.

(4) The subject/unit element number is used to identify information applicable to units in the subsystems. The subject/unit element number continues in sequence from the number -01- with the number of subsystem units in which maintenance information is necessary.

(5) All system/subsystem/unit (chapter/section/subject) maintenance data is separated into specified types of

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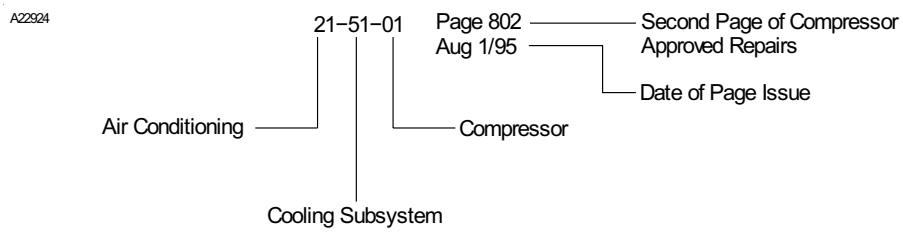
information: Description and Operation, Troubleshooting, Maintenance Practices, etc. Blocks of page numbers that are in sequence are used to identify the type of information:

(a) Description and Operation or Troubleshooting information may not be included if the procedure is easy. When subtopics are short, they may be put together into the Maintenance Practices section. Maintenance Practices can have a mix of subtopics that includes information to service, remove, install, adjust, test, clean, paint or do approved repairs.

(b) Longer procedures that are not as easy to do may be included in a specified section.

Page 1 through 99 - Description and Operation
Page 101 through 199 - Troubleshooting
Page 201 through 299 - Maintenance Practices
Page 301 through 399 - Servicing
Page 401 through 499 - Removal/Installation
Page 501 through 599 - Adjustment/Test
Page 601 through 699 - Inspection/Check
Page 701 through 799 - Cleaning/Painting
Page 801 through 899 - Approved Repairs

(6) A typical page number:



(7) Illustrations use the same figure numbers as the page block in which they appear. For example, Figure 202 would be the second figure in a Maintenance Practices section.

8. List of Effective Pages (Paper Version of this Manual Only)

A. A List of Effective Pages is supplied at the start of each maintenance manual chapter. All pages in the specified chapter are recorded in numerical sequence on the List of Effective Pages with the date of issue for each page.

9. Revision Filing Instructions (Paper Version of this Manual Only)

A. Regular Revision.

(1) Pages to be removed or put in the maintenance manual are given in sequence by their three-element number (chapter/section/subject) and page number on the List of Effective Pages. When two pages show the same three-element number and page number, the page with the most recent date of page revision date must be put in the maintenance manual.

(2) The date column on the List of Effective Pages will help you make sure the correct page(s) are in the manual.

(a) When a page is added to the chapter, the word "Added" will be put adjacent to the related page number on the List of Effective Pages.

(b) When a page is removed from the chapter, the word "Deleted" will be put adjacent to the related page number on the List of Effective Pages.

B. Temporary Revision.

(1) Put temporary revisions in the chapter where they apply. Refer to the filing instructions that are supplied on the cover page of the temporary revision.

(2) Remove a temporary revision when the data has been included in the maintenance manual or when it has been replaced by a different temporary revision. A Record of Temporary Revisions is found in each chapter, immediately after the List of Effective Pages. A Date Removed column on the Record of Temporary Revisions will show the date the temporary revision data was included in the manual. The temporary revision can be removed at that time.

10. Identifying Revised Material

A. Additions or revisions to text is in a section will be identified by a revision bar adjacent to the change in the left margin of the page.

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B. Chapter 5 has revision bars to show revised text. Inspection items are also shown as REVISED, ADDED or DELETED and include the date the item code was revised. Also, a revision date can be found in the lower right corner of the page, below the page number.

C. When large technical changes are made to text in a section, a revision bar will be shown the full length of the text.

D. The illustrations for placards in Chapter 11, Placards and Markings, are supplied in the Model 350 Illustrated Parts Catalog.

11. Dimensions and Units

A. Dimensions and Units.

(1) All dimensions and units are given in both English and metric (International System of Units [SI]) units in the text and the illustrations. The English value is given and then the metric unit is given in parenthesis.

(2) Metric units are given in standard abbreviations.

(3) Station lines and locations are not given in metric units

12. Abbreviation

A. Abbreviations are not often given in this manual; however, abbreviated identification for switches, circuit breakers and annunciator panel legends are shown as they are seen on the placards or electroluminescent panels.

13. Warnings, Cautions and Notes

A. Warnings, cautions and notes, related to the procedures being done, are used. Each gives important information.

(1) WARNING - Tells maintenance personnel how to use the materials, processes, methods, procedures or limits, which must be followed accurately to prevent injury or death to persons.

(2) CAUTION - Tells maintenance personnel how to use the materials, processes, methods, procedures or limits, which must be followed accurately to prevent damage to equipment.

(3) NOTE - Tells maintenance personnel how to use methods, which will make the work easier.

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GENERAL DESCRIPTION

1. General

A. The Cessna 350 (LC42-550FG) is a pre-molded composite built, semi-monocoque, four seat, single engine, low wing, tricycle gear design airplane. The airplane is certified in the utility category and is used primarily for transportation and related general aviation uses. On the following few pages, a brief overview of the airplane and its basic systems is provided, as well as a three view drawing of the airplane.

B. General Dimensions - Refer to Table 1 (Specifications) and Figure 1 (Three-View Drawing of the Airplane) for details.

Table 1. Specifications of the Cessna 350 (LC42-550FG)

Dimensions		Weight Data	Speed Data		Specifications
Wing Area	141.2 sq. ft.	Empty Weight:2400± lbs	Stall Speed	57 KCAS	Engine - 310HP IO-550-N Continental
Wing Span	35.8 ft		Maneuvering Speed	158 KCAS	
Length	25.2 ft	Gross Weight: 3400 lbs	Cruising Speed	190 KCAS	Propeller - 78 in Hartzell/McCauley
Height	9 ft		Never Exceed Speed	235 KCAS	

2. Basic Construction Techniques

A. The construction process used to build the shell or outer surfaces of the fuselage, wing, and most control surfaces involves creating a honeycomb sandwich. The sandwich consists of two outer layers of pre-preg fiberglass around a honeycomb interior. The term "pre-preg fiberglass" means the fibrous material is impregnated with catalyzed epoxy resin by the manufacturer of the fiberglass. This process ensures consistency in surface thickness and strength. The honeycomb sandwich is assembled in molds of the wing, fuselage, and control surfaces. A vacuum is used during the heat curing procedure to ensure proper compaction. Other structural components of the airplane, like ribs, bulkheads, and spars, are constructed in the same manner. In areas where added structural strength and stiffness is needed, such as the wing spars, carbon material is added to the honeycomb sandwich.

B. **Fuselage** - The fuselage is built in two halves, the left and right sides; each side contains the area from the firewall back to, and including, the vertical stabilizer. The bulkheads are inserted into the right side of the fuselage through a process known as secondary bonding. The two fuselage halves are bonded together, and the floors are bonded in after the fuselage halves are joined. Before the fuselage is assembled into one unit, cables, control actuating systems, and conduits are added at this point because of the ease in access. To prevent damage to the leading edge of the vertical stabilizer, anti-erosion tape may be installed.

C. **Wings and Fuel Tanks** - The bottom wing skin and two spars are one continuous piece. The spars are placed in the bottom wing and bonded to the bottom inside surface. Next, the ribs are inserted and bonded to the inside surfaces of the bottom wing and to the spars. Finally, after wires, conduits, and control tubes are inserted, the two top wing halves are bonded to the bottom wing and all the spars and ribs. The airplane has integral fuel tanks, commonly referred to as a "wet wing." The ribs, spars, and wing surfaces are the containment walls of the fuel tanks. All interior seams and surfaces within the fuel tanks are sealed with a fuel impervious substance. To prevent damage to the leading edge of the wing, anti-erosion tape may be installed.

D. **Horizontal Stabilizer** - The horizontal stabilizer is two separate halves bonded to two horizontal tubes that are bonded to the fuselage. The shear webs and ribs are bonded into the inside surface of the lower skin and the upper skin is then bonded to the lower assembly. To prevent damage to the leading edge of the horizontal stabilizer, anti-erosion tape may be installed.

3. Flight Controls

A. **Ailerons and Elevator** - The ailerons and elevator are of one-piece construction with most of the stresses carried by the control surface. Additional structural support is provided by the end caps and the drive rib which is used to mount the control's actuating hardware. The aileron and elevator control systems are operated through a series of actuating rods and bellcranks that run between the control surface and the control stick in the cockpit. (Refer to Chapter 27.)

B. **Rudder** - The rudder is of one-piece construction with most of the stresses carried by the control surface. Additional structural support is provided by the drive rib which is used to mount the control's actuating hardware. The rudder control

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system is operated through a series of cables and mechanical linkages that run between the control surface and the rudder pedals in the cockpit.

4. Rudder Limiter

A. The rudder limiter, which is an integral part of the stall system, is designed to limit the normal full left rudder deflection of 17 degrees to only 11.5 degrees. The rudder limiter system is automatically armed in a relaxed position when the aircraft's electrical power is turned on. The system is activated when two conditions exist, (1) the airplane's stall warning is active, and (2) the engine manifold pressure is more than 12 in. of Hg. When the system is activated, a solenoid near the left rudder pedal moves a cam that physically limits the travel of the left rudder pedal. There is a time delay of approximately one second before the system is activated. This delay feature prevents inadvertent activation of the rudder limiter in turbulent air. A light located in the annunciator panel, triggered by a magnetic sensor located next to the rudder limiter cam acts as a visual indication of when the rudder limiter is engaged. (Refer to Chapter 27.)

B. Two points need to be emphasized regarding the operation of the rudder limiter. First, if a left rudder deflection of greater than 11.5 degrees exists before the stall warning is active with a throttle setting greater than 12 inches of Hg, the cam cannot engage. In addition, if a left rudder deflection of greater than 12 degrees exists while the stall warning is active and before the throttle setting is greater than 12 in. of Hg, then the cam cannot engage when the throttle is advanced beyond 12 in. of Hg. Second, if the rudder limiter is activated and pressure is applied to the left rudder pedal so that the rudder limiter cam is engaged and then the conditions which caused the rudder limiter to activate cease to exist so that the rudder limiter action is no longer needed, then the pressure on the left rudder pedal must be released in order for the rudder limiter cam to disengage. In either of these two conditions the cam actuation does not override the rudder input. It should also be noted that should the manifold pressure gauge itself or the stall warning horn become inoperative, the rudder limiter will still be functional provided that the stall warning vane is still operative.

5. Trim System

A. The airplane is equipped with a two-axes trimming system. The elevator trim tab is located on the right side of the elevator and the aileron trim tab is on the aileron of the right wing. The aileron and elevator trim tabs are electrically controlled by two switches in the trim assembly panel above the engine controls (basic and Avidyne option) or by the A/P Trim switch to the right of the master switches in the overhead (Garmin G1000 option). The two trim motors are protected by one-amp circuit breakers located in the circuit breaker assembly panel. (Refer to Chapter 27.)

6. Wing Flaps

A. The airplane is equipped with Fowler flaps. During flap extension, the flaps move out from the trailing edge of the wing which increases both the camber and surface area of the wing. The flaps are powered by a 32-watt motor located under the front passenger's seat and protected by a 10-amp circuit breaker located in the circuit breaker panel. (Refer to Chapter 27.)

7. Door Seal System

A. The airplane is equipped with a pneumatic door seal system that limits air leakage and improves soundproofing. An inflatable gasket around each main door expands when the door seal system is turned on. An electrical motor near the pilot's rudder pedals operates the system, which maintains a differential pressure of 12 to 15 psi. The system is activated by a switch in the rocker switch panel labeled "Door Seal" and is protected by a five-amp circuit breaker. The door seal system should be turned on before takeoff and turned off after landing. However, a limit switch will turn off the pump when the door locking mechanism is disengaged. Similarly, if the door locking mechanism is not properly engaged, the pneumatic door seal system will not operate. (Refer to Chapter 52.)

8. Landing Gear

A. The airplane has tricycle landing gear with the two main wheels located behind the center of gravity (CG) and a nose wheel forward of the CG point. The main gear is made from high quality rod steel that has been gun-drilled. The main gear is attached to a tubular steel gearbox that is bolted to the bottom of the fuselage, just aft of the wing saddle. There are 15x6.00-6 tires (tire width and rim diameter in inches) which are inflated to 55 psi and mounted to the gear with Cleveland disk brakes. Composite wheel fairings are mounted over each main gear tire to reduce drag.

B. The nose gear has a nitrogen and oil filled oleo type strut that is bolted to the engine mount and serves as a shock absorber. The landing or vertical impact is absorbed by forcing oil through orifices in the piston and an internal plug or barrier. A rotation key or vane working within an oil filled pocket contains rotational movements (shimmy dampening). Both of these movements, vertical and rotational, are fully contained within the main cylinder body and under normal usage will require little maintenance. (Refer to Chapter 32.)

9. Engine and Propeller

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A. The airplane engine is a Teledyne Continental Motors Aircraft Engine Model IO-550-N. It is a horizontally opposed, six-cylinder, fuel injected, air-cooled engine that uses a high-pressure wet-type oil sump for lubrication. There is also a full flow, spin-on, disposable oil filter. The engine has top air induction, an engine mounted throttle body, and a bottom exhaust system. On the front of the engine, accessories include a hydraulically operated propeller governor, a gear driven alternator, and a belt driven alternator. Rear engine accessories include a starter, gear driven oil pump, gear driven fuel pump, and dual gear driven magnetos. The airplane is equipped with a Hartzell three-bladed constant speed propeller with a McCauley governor. (Refer to Chapter 61.)

10. Fuel System

A. The fuel system has two tanks that gravity feed to a three-position (Left, Right, and Off) fuel selector valve located between the pilot and copilot seats. The fuel flows from the selected tank to the auxiliary fuel pump and then to the strainer. From this point it goes to the engine-driven pump where, under pressure, it is sent to the throttle/mixture control unit and then on to the fuel manifold valve for distribution to the cylinders. Unused fuel from the continuous flow is returned to the selected fuel tank. A pressure gauge on the metered side of the fuel manifold valve measures system pressure and displays both the fuel pressure and the equivalent fuel flow reading on the same gauge. (Refer to Chapter 28.)

11. Electrical System

A. The electrical system in this aircraft consists of two independent buses, which are referred to as the left bus and right bus. (Refer to Chapter 24.) Two 60-amp (continuous output) alternators provide charging power for the batteries, as well as system power. The batteries will also provide additional power in the event of an over demand situation where the requirements on the system are greater than what can be provided by the alternator. The left and right buses in turn feed the avionics and essential buses.

B. Five current limiters protect the alternators and bus outputs. In addition, the left and right buses are physically isolated inside a covered area mounted to the firewall. Left and right bus controls, grounds, and outputs are routed through separate holes, connectors, and cable runs so any failure on one bus will not affect the operation of the other bus.

C. Control of the buses is via the master switch panel located on the lower left portion of the instrument panel. There is also a crosstie switch on this panel, which will restore power in the event of failure of the right or left systems. For example, if the alternator or some other component on the left side should fail, the crosstie switch will restore power to the electrical items on the left bus by connecting the left bus to the right bus.

D. As its name may suggest, power to the essential bus is never affected, provided power from at least one bus (left or right) is available. The essential bus is diode fed, i.e., current will only flow in one direction, from both the right bus and the left bus allowing the essential equipment to have two sources of power.

E. There is an alternator switch in the cockpit area that disconnects the alternator and stops the excitation. A red "Alt Out" light in the annunciator panel illuminates (Basic or Avidyne option) or an annunciation message displays on the PFD (Garmin G1000 option) if the alternator becomes inoperative. At 16 volts (S/N 42001 to 42500) or 32 volts (S/N 42501 and on) output, an overvoltage control will stop the excitation to the alternator. The airplane is equipped with a voltmeter that measures bus voltage and an ammeter that measures the charging or discharging of the battery.

12. Environmental Control System

A. The environmental control system (ECS) incorporates the use of an air-to-air heat exchanger, ram intake air, and an electric fan to distribute heated and outside air to various outlets within the cabin. The ECS essentially consists of two subsystems, heated air and fresh air. Heated air is sent to a floor vent system and defroster, and fresh air is ducted through the eyeball vents. The system demand affects the volume of flow to a particular vent. As more vents are opened, the airflow to each vent is decreased. (Refer to Chapter 21.)

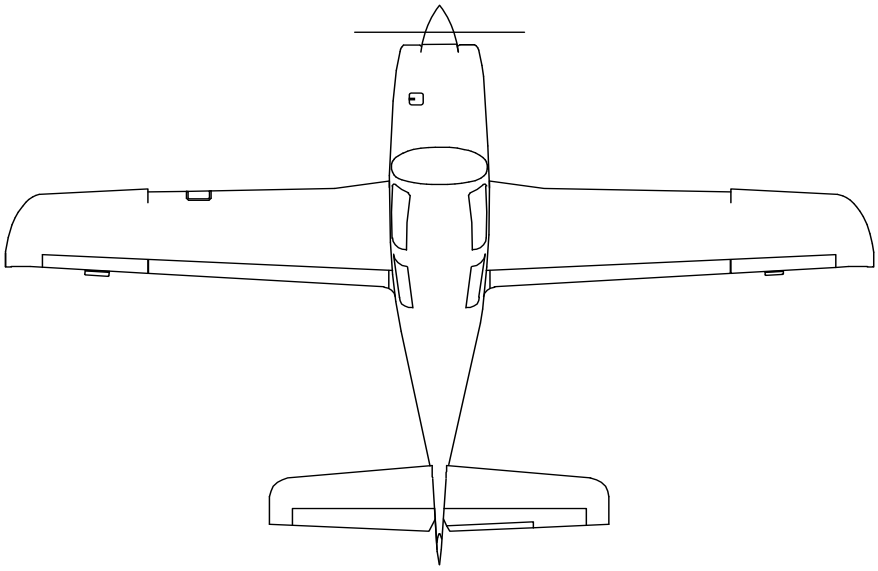
13. Flight Instruments

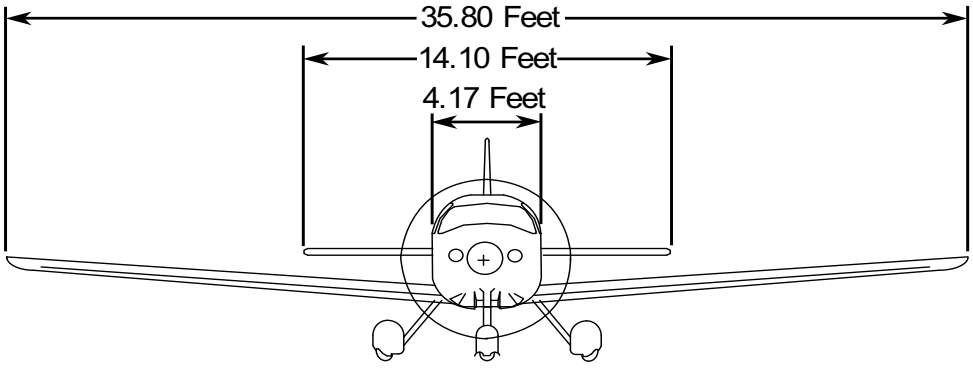
A. The airplane is equipped with electrically-driven attitude and heading indicators. The pitot-static system measures pressure differences in ram air from the pitot tube and ambient air from the static air vent. Static air is also used as the pressure source for the vertical speed indicator and the altimeter. (Refer to Chapter 34.)

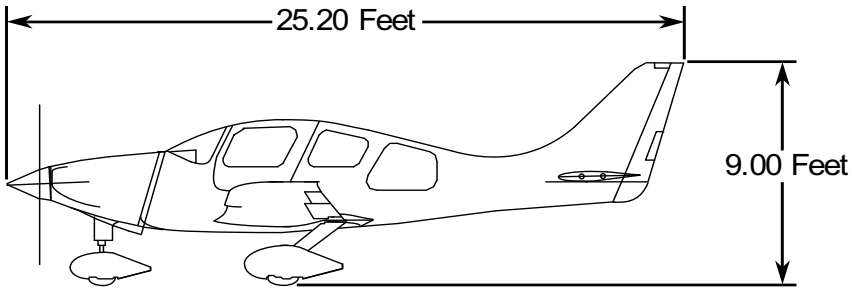
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Figure 1 : Sheet 1 : Cessna 350 (LC42-550FG)

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LC57605000A1

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ICA SUPPLEMENT LIST

ICA Supplement Number	Title	ICA Supplement Date	Manual Incorporation Date
ICA-T240-27-00001	Rudder Cable Tube Insert and Clamp Installation	May 07/2010	Incorporation Pending

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AIRWORTHINESS LIMITATIONS

1. General

A. The Airworthiness Limitations section (Chapter 4) is FAA approved and specifies maintenance required under Sections 43.16 and 91.403 of the Federal Aviation Regulations, unless an alternative program has been FAA approved.

B. **Inspections** - This chapter contains mandatory maintenance tasks including inspections (Refer to Table 1) and life-limited item replacement requirements (Refer to Table 2). An airframe inspection, in accordance with Table 1, is required every 3000 hours. This 3000-hour period is known as an Airframe Service Cycle (ASC). Table 1 identifies and describes the required maintenance tasks for each ASC.

NOTE: These items are in addition to the items required to maintain the continued airworthiness of the airplane. After 25,200 TIS hours, the FAA requires a Major Airframe Inspection (MAI). Table 2 contains the MAI items.

C. **Grace Period** - The required inspections listed in the above paragraph must be performed within 100 hours of the specified interval or at the next annual inspection, whichever occurs first.

D. **Airframe Service Cycle Consistency** - If work is accomplished during the grace period, then the next inspection/part replacement is due at the time in service determined by adding the required interval to the BEGINNING of the grace period. For example, if the first ASC is accomplished during the grace period at 3099 hours TIS, then the next ASC is due at 6000 hours TIS (or within the grace period following 6000 hours). If work is accomplished early, then the next inspection/part replacement is due at the time in service determined by adding the required interval to the TIS at which the work was last accomplished. For example, if the first MAI was accomplished at 24,000 hours TIS, then the next MAI is due at 49,200 hours TIS (or within the grace period following 49,200 hours).

2. Airframe Service Cycle Inspection (Every 3000 Hours)

Table 1. Airframe Service Cycle Inspection (Every 3000 Hours)

INSPECTION AREA	MAINTENANCE TASK	CHAPTER NUMBER
Fuselage and Vertical Stabilizer		
Bonding of the two fuselage halves	**Visual inspection of the bonding seams:** Tap test seam area around fuselage halves in accordance with Section 20-95-00, Tap Testing. This test detects sun-bonded areas by the change in audible response when the surface is systematically tapped.	53
Aft Fuselage	Complete the Aft Fuselage Inspection in accordance with the instructions given in Chapter 53, Fuselage Components - Adjustment/Test. **NOTE:** **This inspection must be completed by a composite technician that has been trained in accordance with the requirements found in the LC-550FG Series Structural Repair Manual, Chapter 51-00-01.**	53
Wing Attachment	Replace the wing attachment bolts and nuts with new bolts and nuts. There are a total of 30 bolts, 15 on each side of the wing saddle, that connect the fuselage to the wing as shown in Section 57-10-00, Figure 202. The bolts vary in size from AN4-16 to AN4-20. When a bolt is removed, replace it with one that is the same size. The bolts must be replaced one at a time using a systematic process. It is recommended that the left forward bolt be removed and replaced first, and then continuing aft on the left side of the wing saddle. When the left side is complete, start on the right forward position and move aft.	53 and 57
CAUTION: **The right aft wing bolt is reversed from the other bolts. That is, the nut is on top rather than the bolt head. This is the correct method of installation and ensures clearance of the bolt from the flap actuating system.**		

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Wing Attachment (continued)	Use the following steps to remove and install the wing bolts: 1. Remove carpeting and footwell panel in rear seat area, exposing area between front and rear spars. 2. Remove flap assembly panel plate under the aft wing area in the fuselage saddle, exposing area aft of rear spar. 3. Remove inspection cover in the fuel assembly panel, which is located in the left forward saddle area, exposing area forward of main spar. 4. Replace all bolts and nuts. The bolts must be removed and replaced one at a time to prevent misalignment of the wing. Refer to Chapter 57 for torque values. 5. When a bolt is removed, visually inspect the area adjacent to the bolt hole for evidence of damage, failure, and cracking. Refer to Chapter 57 regarding bolt length and torque requirements. 6. The bolts and nuts cannot be reused for any purpose. All bolts and nuts that are removed must be permanently discarded or destroyed, regardless of the time in service.	53 and 57
Cockpit Frame	Leave rear seat footwell exposed for this inspection. 1. Remove front and rear seats and front floor carpeting. 2. Remove carpeting from baggage floor and shelf. 3. Inspect the exposed area for signs of cracks, delamination, and fatigue.	25
Airframe Paint	Perform a detailed inspection of the airplane's clear-coat and paint. All paint must conform to the latest FAA approved revision of Cessna Document SX512701. Contact the manufacturer for information if paint repair is necessary. **NOTE:** The airplane paint has a clear-coat cover that provides protection from UV rays. If the clear-coat is damaged, degradation of the paint and underlying surface is possible. The airframe paint does not have a specific life-limit. Its serviceability depends on the proper care of the airplane as discussed in Section 8 of the AFM/POH and the operating environment. For example, if the airplane is washed and waxed infrequently or the airframe is subject to a hail or dust storm, erosion and/or damage to the clear-coat is possible.	51
Transparencies	Check all transparencies for excessive crazing, UV yellowing, and evidence of cracking. Replace as required.	56
Doors	Remove left and right doors with door hinge blocks and inspect hinge block attachment holes for evidence of damage. Inspect attachment screws of dynamic hinge to the door for tightness.	52
Door Openings	Check door openings, particularly the corner of the door openings (including baggage compartment door), for evidence of cracking. **NOTE:** Sometimes the paint may crack around a door opening without the underlying fiberglass surface being affected. If paint cracks are observed, the painted surface must be removed by sanding to verify there are no cracks in the fiberglass.	52
Wings		
Bonding of the Top and Bottom Wing	Visually inspect the bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-95-00. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped.	57

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Bonding of Wing Skin to Ribs and Spars	Tap test areas where the top and bottom of the skin are bonded to the spars and ribs in accordance with Section 20-95-00. If the skin is separated, a sponginess will be felt or a dullness in tone will be heard when tapped.	57
Flight Control Systems and Systems		
Wing Flaps	Visually inspect bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-95-00. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped. Check for cracks on drive rib.	27
Ailerons	Visually inspect bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-95-00. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped. Check for cracks and buckling in area of drive rib.	27
Rudder	Remove rudder and perform visual inspection of the bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-95-00. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped. Check condition of hardware for cracks, corrosion, and wear.	27
Wing Flap Actuator	Disassemble flap assembly and grease motor and gear train in accordance with Section 27-50-00. Check hardware for cracks, corrosion, and wear.	27
Aileron Push Tubes and Hardware	Check for wear and cracks. Inspect tubes for signs of corrosion. Check bell cranks and attachment hardware for signs of cracks and corrosion.	27
Elevator Push Tubes and Hardware	Check for wear and cracks. Inspect tubes for signs of corrosion. Check bell cranks and attachment hardware for signs of cracks and corrosion.	27
Elevator Trim Tab	Remove elevator trim tab and repack servo motor in accordance with Section 27-30-00, Elevator Trim Tab and Elevator Trim Tab Servo. Inspect hardware for cracks, corrosion, and wear.	27
Elevator	Remove elevator and inspect piano hinge for wear and corrosion.	27
Horizontal Stabilizer	Remove horizontal stabilizer and check stabilizer tube and bolts for cracks, corrosion, and wear. Replace as required.	27
Fuel System		
Boost Pump	Overhaul per Duke Inc. instructions included in overhaul kit, or replace the boost pump.	28
Landing Gear		
Teflon Bushings In Gear Box	Remove landing gear and check Teflon bushings in gear box for each main gear leg for condition and wear. Refer to Chapter 32 for specific details about the landing gear.	32
Gear Box Attachment Bolts	Remove one attachment bolt from each side of the gear box and check for corrosion. If corrosion is noted, replace all attachment bolts in the gear box. If no corrosion is noted, replace bolts that were removed and check the torque on all bolts in the gear box in accordance with Section 32-10-00, Gearbox. Refer to Chapter 32 for specific details about the landing gear.	32
Landing Gear Alignment	Once landing gear is reinstalled, check camber and toe-in. Refer to Chapter 32 for specific details about the landing gear.	32

3. Life-Limited Parts

A. The items listed below are to be replaced with new items at the times/intervals shown.

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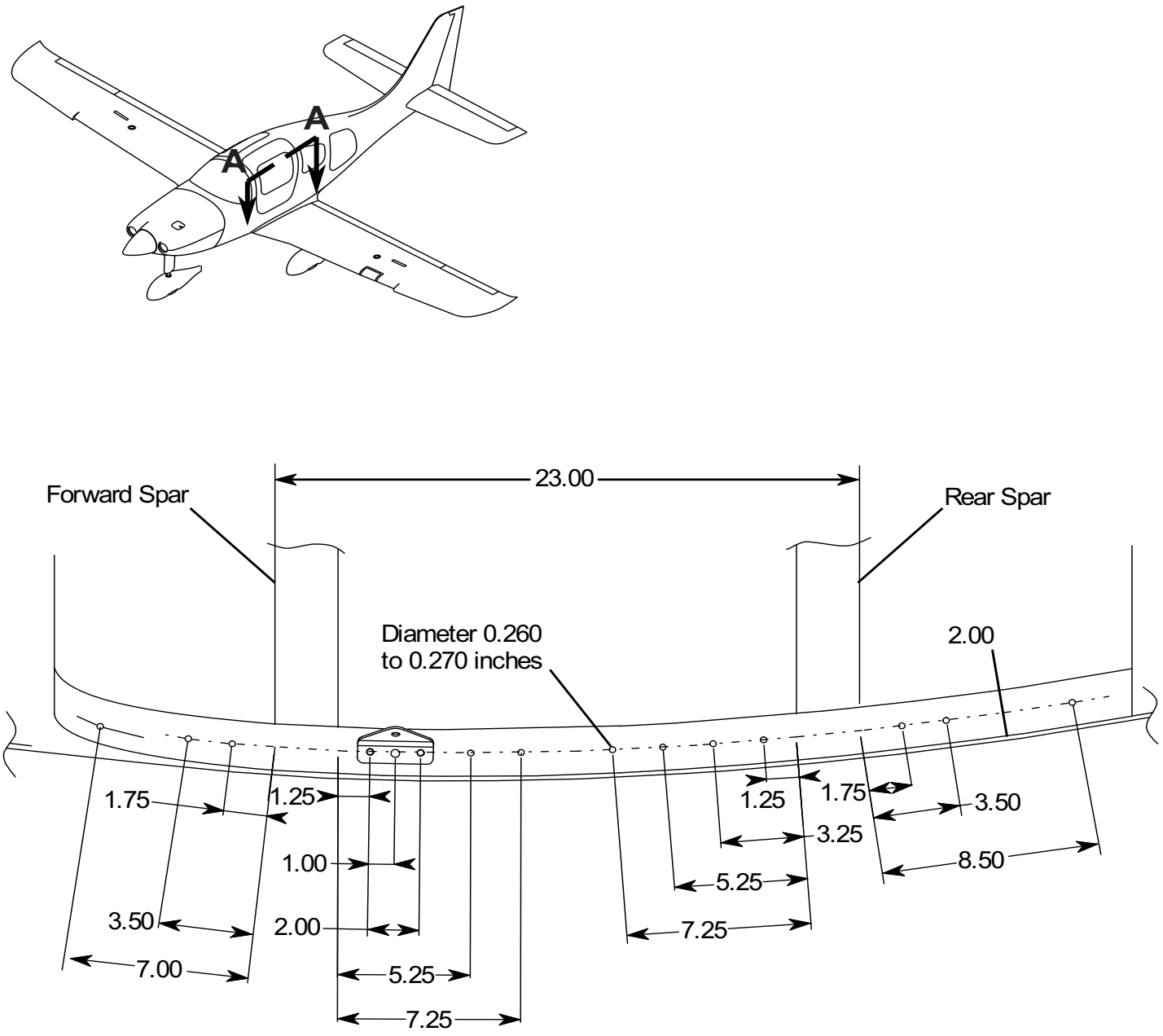
Table 2. Life-Limited Parts

Item	Location	Replacement Time/Interval
Emergency Locator Transmitter (ELT) Battery	Aft of the baggage compartment hat rack - Refer to Chapter 25 for more information.	Every two years (Artex 200), every five years (Artex ME406), or when the battery has been used for more than one hour or 50 percent of its life
3-volt Lithium Battery in Avidyne Multi-Function Display (MFD)	On an internal processor board inside the MFD - Refer to Chapter 34 for more information.	Every ten years or when the CMOS memory of the MFD fails to retain configuration data
System - Dry Sealed Lead-Acid Type Battery (Two)	Just forward of the firewall on the Co-pilot's side (S/N 42001 to 42022) or on the Pilot's side (S/N 42023 and On).	Alternate replacement every two years - First replace the right battery, after two more years replace the left battery. Alternate replacement every two years. However, if the battery fails to hold a charge, it must be replaced
No. 1 (Right) Alternator	Just rear of the propeller on the copilot's side.	Overhaul or replace alternator every 2000 hours
No. 2 (Left) Alternator	Just rear of the propeller on the Pilot's side.	Overhaul or replace alternator every 2000 hours. Check belt condition and tension every 100 hours or annually. Belt tension range is 13 to 17 lbs-ft measured as described in Chapter 24. Replace the belt whenever there is cracking, fraying, or signs of excessive wear, every 5 years, or 2000 hours whichever comes first
Front and Rear Seat Restraint Devices	At front and rear seats.	Every 3000 hours or 10 years, whichever occurs first.
Wing Attachment Bolts	Wing saddle	Every 3000 hours
SpeedBrakes	Outboard top of wing, each side.	Every 1000 hours return SpeedBrake cartridges to Precise Flight Inc. for clutch lubrication and spring replacement.
SpeedBrakes	Outboard top of wing, each side.	Every 5000 hours return SpeedBrake cartridges to Precise Flight Inc. for drive assembly replacement
Carbon Monoxide Detector	Mounted to the gusset on the firewall behind the Primary Flight Display (PFD).	Every 7 calendar years return to CO Guardian LLC for CO Sensor replacement and calibration
Airplane		25,200 flight hours

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Figure 202 : Sheet 1 : Wing to Fuselage Attachment Bolt Locations

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VIEW A-A

NOTE: All measurements represented in inche s.

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INSPECTION TIME LIMITS

1. Major Airframe Inspection Checklist (Every 25,200 Hours)

Several pages will be added here that describe all the inspection items that must be performed on the Major Airframe Inspection. This checklist will be added to the manual as part of a revision level. There is currently a life limit on the airplane of 25,200 hours until Section 04-10-00 is completed and approved by the FAA.

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TOWING AND TAXIING - GENERAL

1. General

A. The following pages contain instructions for towing and taxiing the airplane. It is important to remember that the airplane has a free-castoring nose wheel with stops at the 60-degree left and right positions. Personnel unfamiliar with the techniques required when taxiing an airplane with a free-castoring nose wheel should be given appropriate instruction or a taxi checkout.

2. Ground Handling

A. Towing - A locking, hand-held tow bar is provided with the airplane and stored in the baggage compartment. The tow bar is inserted into two small holes in the nose wheel fairing, forward of the nose wheel axle. The tow bar must be locked in place before attempting to move the airplane. The tow bar is collapsible for storage by removing the locking pin. With the pin removed, slide the handle in and reinsert the locking pin. It is recommended that the airplane only be maneuvered during towing by use of the handheld tow bar. If it is necessary to tow with a vehicle, extreme care is required to ensure the rotation limits of the nose wheel (60 degrees left and right) are not exceeded. Since the rotation of the nose gear is limited by physical stops, rotating the gear beyond 60 degrees will damage the airplane.

B. It is a good idea to have another person serve as a spotter when towing the airplane, particularly if a vehicle is used. Remember that the airplane has vertical limitations as well as horizontal restrictions. The vertical stabilizer is frequently overlooked as an airplane is being pushed into a hanger with most of the attention directed towards the wingtips. When moving the airplane over uneven surfaces, remember that small up and down oscillations of the nose strut result in amplified movement of the vertical stabilizer. Finally, keep in mind that the height of the vertical stabilizer is affected by inflation levels of both the nose tire and strut. A flat tire or low nose strut will increase the height of the vertical stabilizer.

CAUTION: Do not attempt to move the airplane by pushing or pulling on the propeller. This a common practice for airplanes with fixed pitch propellers; however, it is not recommended for constant speed propellers, since pressures applied to the propeller blades are transmitted to moving parts within the propeller hub. Over time, these forces could cause damage to the propeller.

3. Taxiing

A. When taxiing, minimize the use of the brakes. Since the airplane has a free-castoring nose wheel, steering is accomplished with light breaking. Avoid the tendency to ride the brakes by making light steering corrections as required and then allowing the feet to slide off the brakes and the heels to touch the floor.

B. Avoid taxiing in areas of loose gravel, small rocks, etc., since it can cause abrasion and damage to the propeller. If it is necessary to taxi in these areas, maintain low propeller speeds. If taxiing from a hard surface through a small area of gravel, obtain momentum before reaching the gravel.

C. The airplane is stable on the ground. The low wing design minimizes the tipping tendency from strong winds while taxiing. Still, the proper positioning of control surfaces during taxiing will improve ground stability in high wind conditions. Table 1 summarizes control positions that should be maintained for a given wind component.

Table 1. Control Positions for Various Wind Components

Wind Component	Aileron Position	Elevator Position
Left Quartering Headwind	Left Wing Aileron Up(Move Aileron Control to the Left)	Neutral(Hold Elevator Control in Neutral Position)
Right Quartering Headwind	Right Wing Aileron Up(Move Aileron Control to the Right)	Neutral(Hold Elevator Control in Neutral Position)
Left Quartering Tailwind	Left Wing Aileron Down(Move Aileron Control to the Right)	Down Elevator(Move Elevator Control Forward)
Right Quartering Tailwind	Right Wing Aileron Down(Move Aileron Control to the Left)	Down Elevator(Move Elevator Control Forward)

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LIGHTING - GENERAL

1. Interior Lighting System - General

A. The interior lighting system of the Cessna 350 is one of the more sophisticated systems available for small airplanes. There are six different lighting systems within the interior of the airplane. For a summary of these lighting systems, refer to Table 1 or Table 2. For information about trim, annunciators, and flap position LEDs, refer to Chapter 25-11-00. For information about the fuel tank LEDs, refer to Chapter 28-40-00.

B. **Flip and Access Lights** - The flip-lights are rectangular shaped fixtures located in the middle of the overhead panel and in the baggage compartment. The lights bypass the system master switch and operate without turning on power to the system. The two flip-lights are turned on by rotating or flipping the lens right or left. In the center position, they are used as part of the airplane's access lighting system.

(1) When either main entrance door is unlatched, a switch in the door latching mechanism activates the two flip-lights and the two lights that illuminate each entrance step. The access lights are on a ten-minute timer and turn off automatically unless reset by activating both main door latching mechanisms when all the doors are closed.

(2) This design has two advantageous features. First, opening either of the main cabin doors provides an immediate light source for preflight operations, passenger access, and baggage loading. Second, the flip-lights, when rotated either left or right, serve as emergency lighting in situations which necessitate turning off the master switch. The only disadvantage is that the flip-lights can inadvertently be left on, depleting battery power. To prevent this from happening, ensure the flip-lights are in the centered or flush position when securing the airplane at the end of a flight.

C. **Overhead Reading Lights** Each light is on a swivel that can be adjusted to an infinite number of positions. The intensity of these lights is adjusted by moving the left thumbwheel (basic or Avidyne option) or slide-type (Garmin G1000 option) dimmer switch in the center of the overhead panel, near the windshield. The dimmer has an on-off switch at the extreme aft position. Rotating the thumbwheel, or sliding the slide, forward increases the light intensity. The airplane's position lights must be on for the overhead reading lights to operate.

(1) For S/N 42001 to 42003 there are three overhead reading lights, two in the forward overhead panel between the front seats and one in the aft overhead panel between the two backseats. Each light has its own switch.

(2) For S/N 42004 and on there are four overhead reading lights, two in the forward overhead panel between the front seats and two in the aft overhead panel between the two backseats.

D. **Instrument Flood Bar** - There is a tube array of either color-corrected lights or LEDs inserted under the glare shield. This indirect light source complements the backlighting in each instrument and facilitates the use of adjustable instruments such as the true airspeed indicator, directional gyro, and navigational instruments. The intensity of the lights can be adjusted by moving the right wheel-type (basic or Avidyne option) or slide-type (Garmin G1000 option) dimmer switch in the center of the overhead panel, near the windshield. The dimmer has an on-off switch at the extreme aft position. Rotating the thumbwheel, or sliding the slide, forward increases the light intensity. The airplane's position lights must be on for the instrument flood bar lights to operate.

E. **Upper Instrument and Engine Panels**

(1) **Basic or Avidyne Option** - The instruments in the flight and engine instrument panels have backlighting, i.e., a small light within each instrument case that illuminates its dial. The right thumbwheel switch between the pilot's legs on the knee bolster, controls the dimmer for these lights. The dimmer has an on-off switch at the extreme down position of its rotation. Rotating the thumbwheel up increases the light intensity. The airplane's position lights must be on for the upper instrument and engine panel lights to operate.

(2) **Garmin G1000 Option** - The brightness of backlighting for the PFD, MFD, audio panel, and keypad are controlled by photo cells located on the devices. The brightness of backlighting for the backup flight instruments is controlled by the left slide dimmer switch at the front of the center console. The dimmer has an on-off switch at the extreme up position, and moving the slide down increases the light intensity.

F. **Lower Instrument Panel, Circuit Breaker Panel, and Rocker Switches**

(1) **Basic or Avidyne Option** - The lower instrument, circuit breaker, and rocker switch panels contain switches and controls that have backlighting. The lighting illuminates the placards on or next to the breaker, switch, or control and the internally lighted engraved rocker switches. The dimmer for these lights is controlled by the left thumbwheel switch between the pilot's legs on the knee bolster. The dimmer has an on-off switch at the extreme down position of its rotation. Rotating the thumbwheel up increases the light intensity.

(2) **Garmin G1000 Option** - Backlighting of the 5 pack (pitot heat, door seals, and optional equipment) switches, flap

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panel, lighted position bar, slide dimmer labels, master switches and circuit breaker panel is controlled by the right slide switch at the front of the center console. Backlighting of the fuel pump "armed" light is controlled by the position lights switch. The backlighting illuminates the placards on or next to the breaker, switch or control, and the internally lighted switches. The dimmer has an on-off switch at the extreme up position, and moving the slide down increases the light intensity. Backlighting of the pitot heat, door seals, and optional items switches will dim down to a preset value while all other lighting controlled by this switch will dim to zero.

Table 1. Lighting Systems (Basic or Avidyne Option)

LIGHT	LOCATION OF LIGHTS	LOCATION OF SWITCH	REMARKS
Courtesy Lights	Front and rear flip lights in overhead console	If all doors are latched, flip light is activated by flipping the lens from the neutral position.	Door switch activates timer that turns off access lights after 10 minutes.
	Exterior lights near the right and left entrance steps	If a door is unlatched, a switch activates flip lights when the lens is in the neutral position.	Operates with master switch on or off
		Step lights are turned on with door switch.	
Overhead Swivel Lights	Two overhead swivel lights in the front seat area	The left thumb wheel dimmer switch is in the overhead panel.	Master switch and navigation lights must be on for the system to operate.
	S/N 42001 to 42003 - One centered swivel light in the rear seat area	S/N 42001 to 42003 - Individual switch at each light.	
	S/N 42004 and on - Two centered swivel lights in the rear seat area	S/N 42004 and on - No individual switch at each light.	
Glare Shield Flood Bar	Color corrected flood bar under the glare shield that lights the flight instruments and front panel areas	The right thumb wheel dimmer switch is in the overhead panel.	Master switch and navigation lights must be on for the system to operate.
			Part of the standby battery system
Upper Instrument Panel	Provides backlighting for engine and flight instruments	The right thumb wheel dimmer switch is in the knee bolster on the pilot's side.	Master switch and navigation lights must be on for the system to operate.
Lower Instrument and Circuit Breaker Panels	Provides backlighting for radios, switches, or placards next to switches, circuit breakers, and controls	Left thumb wheel dimmer switch is in the knee bolster on the pilot's side	Master switch and navigation lights must be on for the system to operate.
Trim, Flaps, Fuel Tank and Annunciator Panel	The trim position LEDs are in the trim panel.	The day/night and test features for these items are located in the trim panel, just above the engine control.	Master switch and navigation lights must be on for the system to operate.
	The flap position LEDs are in the flap panel.		
	The fuel tank LEDs are on the fuel quantity gauge.		
	Annunciator LEDs are in the annunciator panel.		

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Table 2. Lighting Systems (Garmin G1000 Option)

LIGHT	LOCATION OF LIGHTS	LOCATION OF SWITCH	REMARKS
Courtesy Lights	Front and rear flip lights in overhead console	If all doors are latched, fliplight is activated by flipping the lens from the neutral position.	Door switch activates timer that turns off access lights after 10 minutes.
	Exterior lights near the right and left entrance steps	If a door is unlatched, a switch activates flip-lights when the lens is in the neutral position.	Operates with master switch on or off
Overhead Swivel Lights	Two overhead swivel lights in the front seat area	The left slide-type dimmer switch is in the overhead panel.	Master switch must be on for the system to operate.
	Two overhead swivel lights in the rear seat area		
Glare Shield Flood Bar	Flood bar under the glare shield which lights the flight instruments and front panel areas	The right slide-type dimmer switch is in the overhead panel.	Master switch must be on for the system to operate.
Upper Instrument Panel	Provides backlighting for the backup flight instruments.	The left slide-type dimmer switch is in the front of the center console.	Master switch must be on for the system to operate.
Lower Instrument and Circuit Breaker Panels	Provides backlighting for switches, or placards next to switches, circuit breakers, and controls	The right slide-type dimmer switch is in the front of the center console.	Master switch must be on for the system to operate.

2. Exterior Lighting System - General

A. **Description of System** - The exterior lighting system includes the position lights, the strobe or anti-collision lights, the landing light, and the taxi light. These lights are activated through use of switches in the rocker switch panel. The light system is protected by circuit breakers in the circuit breaker panel.

B. **Position and Anti-Collision Lights** - The left and right position lights (red and green) are mounted on each wing tip. Each wing position light contains the required aft or rearward projecting white lights. The anti-collision lights are on each wing tip and contained within the same light fixture as the position lights.

C. **Taxi and Landing Lights** - The taxi and landing lights are contained in the leading edge of the left wing. The outboard bulb in the light housing is the taxi light that provides a diffused light in the immediate area of the airplane. The inboard bulb is the landing light, which has a spot presentation with a slight downward focus. The taxi and landing lights are sized for continuous duty and can be left on for operations in high-density traffic areas.

CAUTION: Before any maintenance, removal, or installation of equipment or light bulbs is performed, make sure both the master and avionics switches are in the off position. Before removing the light bulbs, be sure that they have not been in use for several minutes and have cooled sufficiently.

3. Troubleshooting Light Problems

A. **Light Bulbs** - As a general rule, light bulbs that have a short life have good accessibility and can be exchanged easily. For example, most engine and flight instrument lights can be changed in 10 to 15 minutes depending on the location of the instrument. Light bulbs that have a long life require more time for access and replacement.

B. **Check For The Easy Solution** - If none of the lights in the flood bar work, it is unlikely that all 20 bulbs burned out at the same time. In this situation, there is clearly a problem with the supply to the flood bar, and the problem should be diagnosed from this perspective. Conversely, if the altimeter light is inoperative, but all other flight instrument lights are normal, the problem is probably a light bulb.

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INTERIOR LIGHTING - MAINTENANCE PRACTICES

1. Flight Compartment Lighting

A. **Courtesy Lights (General)** - There are four courtesy lights. Two courtesy lights are located in the airplane's interior on the overhead console, and two are on the airplane's exterior (commonly referred to as step lights). The entrance step lights are located above and forward of the steps on both sides of the aircraft, on the underside of the wing-to-fuselage fairing. Each are easily accessible in the event of bulb burnout.

B. **Exterior Courtesy/Step Lights Removal**

(1) Remove the light assembly by gently prying it with a flat-head screwdriver. It should slide out with very little effort. Refer to Figure 201.

(2) Gently pull the step light assembly out of the hole, and unplug it at the connector.

C. **Exterior Courtesy/Step Lights Installation or Replacement**

(1) Reconnect the connector and feed excess wire into the fuselage.

(2) Insert the light fixture into the hole, and push it in until it is flush with the surface.

(3) The exterior step light bulb is not replaceable. If a light fails, replacement of the whole assembly is required.

CAUTION: Before doing any work on a system, ensure that all power and light switches are in the off position and the light that is to be replaced has not been in use for at least five minutes.

D. **Removing the Interior Courtesy Lights**

(1) The interior light housing can be removed by prying it off the panel with a screwdriver. Refer to Figure 202.

(2) Disconnect the three wires to the light housing.

(3) To remove the light bulb, push in the two tabs located to the left and right of the bulb.

(4) Push the bulb out of its slot to remove it.

E. **Installing the Interior Courtesy Lights**

(1) To install the interior light, place a new light bulb in the slot, and push it into place.

(2) Push the housing and the light cover together so that the two tabs click into place.

(3) Reconnect the three wires, and push the housing back into its slot in the panel. Refer to Figure 202.

F. **Troubleshooting the Interior Courtesy Lights**

(1) If the interior courtesy lights are not working, remove the housing from the panel, and make sure that all three wires are connected properly.

(2) Check each light bulb for a failure.

(3) Check the circuit protection for a blown fuse. The courtesy light system is connected directly to the battery (bypasses the master switch), and the fuse block is located next to the battery in the engine compartment. All fuses are labeled.

2. Overhead Swivel Lights

A. For S/N 42001 to 42003, there are three overhead swivel lights. Two are located in the front section of the cabin in the center of the overhead console. The third is located in the rear section of the cabin in the middle of the overhead console. For S/N 42004 and on there are four overhead swivel lights. Two are located in the front section of the cabin in the center of the overhead console and two are located in the rear section of the cabin in the middle of the overhead console These lights are also referred to as spot lights, map lights, or reading lights. Refer to Figure 202 for the removal and installation of the overhead swivel lights.

CAUTION: Before doing any work on a system, ensure that all power and light switches are in the off position and the light that is to be replaced has not been in use for at least five minutes.

B. **Removing the Overhead Swivel Lights**

(1) S/N 42001 to 42003

(a) Remove the eyebrow cover from the swivel light by pulling it out of the light assembly. The eyebrow is pressure fitted into the assembly to limit side glare.

(b) The swivel lights can be removed from the panel by gently prying them with a screwdriver.

(c) Disconnect the two wires to the light housing. Push in the three plastic tabs that hold the light bulb housing together.

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(d) To access the light bulb, push down on the housing, and turn it to remove the light bulb cover.

(e) Gently pull the light bulb out of the metal tabs to remove it.

(2) S/N 42004 and On

(a) Unscrew three screws holding the swivel light in place.

(b) Remove the swivel light from the panel.

(c) Unplug the connector.

C. **Installing the Overhead Swivel Lights**

(1) S/N 42001 to 42003

(a) To install the swivel light, place a new light bulb in the slot, and gently push it in until the tabs hold the light bulb in place.

(b) Put the cover over the light bulb, push it down, and twist it until the cover and the housing latch together.

(c) Push the housing down into the frame until the three tabs click into place.

(d) Reconnect the two wires to the housing, and push the light back into its mounting hole in the panel.

(e) Reinstall the eyebrow cover.

(2) S/N 42004 and On

(a) Plug in the connector.

(b) Push the light into it's mounting hole in the panel.

(c) Screw in three screws to hold the swivel light in place.

(d) The overhead swivel light bulb is not replaceable. If a light fails, replacement of the whole assembly is required.

D. **Troubleshooting the Overhead Swivel Lights**

(1) S/N 42001 to 42003

(a) If the overhead swivel lights are not functioning properly, make sure that both of the wires are hooked up properly.

(b) Check the light bulb to be sure it is not burned out.

(c) Check the light dimmer assembly.

(2) S/N 42004 and On

(a) If the overhead swivel lights are not functioning properly, make sure the connector is plugged in correctly.

(b) Check the light to be sure it is not burned out.

(c) Check the light dimmer assembly.

3. Glare Shield Flood Bar

A. The glare shield flood bar is located under the glare shield in the cabin. The flood bar is composed of light-emitting diodes (LEDs) spaced at 1/2 in. intervals. If more than eight LEDs burn out, a new flood bar must be installed if the airplane is to be used for IFR operations. Contact the factory for ordering a replacement flood bar assembly.

B. **Flood Bar Inspection/Airplane Restrictions** - With the flood bar turned on, use a mirror to count all the burned out LEDs. If 8 or more LEDs are burned out, then the flood bar must be replaced, or the airplane must be restricted to VFR operations. The flood bar is an integral component in the airplane's IFR certification. Refer to the minimum equipment list in Appendix A, Section 6, of the POH/AFM.

C. **Removing the Glare Shield Flood Bar**

(1) Remove the glare shield per instructions in Chapter 25-11-00, and place it inverted on a soft work bench.

(2) Pull the flood bar to remove it from attachment clips and to break the adhesive from the glare shield.

(3) Clean excess adhesive off the glare shield.

D. **Replacing the Glare Shield Flood Bar**

(1) Glare shield should be inverted on a work bench.

(2) Attach the flood bar to the underside of the glare shield at the front lip by snapping onto the attachment clips.

(3) Use several strips of RTV sealant along the wiring to secure it in place.

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(4) Note that the wiring tail must hang from the left (pilot) side of the assembly.

(5) Reinstall the glare shield per instructions in Chapter 25-11-00.

4. Upper Instrument Panel

A. **Field Repair Instrument Lights** - The field repair instrument lights are located in the engine and flight instruments. The upper instrument lights are broken into two groups - flight instruments and engine instruments. Flight instruments include the air speed indicator, altimeter, vertical speed indicator, turn coordinator, directional gyro, and artificial horizon. Engine instruments include fuel quantity, manifold pressure/fuel flow, vacuum/ammeter, oil pressure/temperature, and cylinder head/exhaust gas temperature gauges. The magnetic compass is also included with the upper panel discussion.

B. **Factory Repair Instrument Lights** - There are a few flight, engine, and navigational instruments that have sealed lights, i.e., lights that cannot be accessed in the field. These instruments include the flight director, horizontal situation indicator (HSI), navigational indicator (OBS), and the tachometer. The lights in these instruments are designed to last several thousand hours and, under normal conditions, should never require replacement. If a light does become inoperable, and it is confirmed that there is power to the instrument, then the instrument must be removed and sent to an instrument shop for bulb replacement.

C. **Removal and Replacement of Flight Instrument Lights** - These instruments use a light tray for lighting. To change a bulb use the following steps:

(1) Remove the glare shield, and partially remove the instrument panel to gain access to the desired flight instrument. Refer to Chapter 25-11-00 for details.

(2) If necessary, remove the flight instrument panel by removing the screws in the instrument panel.

(3) Unscrew the two screws holding the light tray to the top of the instrument. The whole light tray must be replaced since the bulb is soldered to it. Refer to Figure 203.

(4) Once the light tray is replaced, reverse the above steps for installation of the instrument. The screws holding the instrument to the instrument panel do not have torque values and should be snug tight.

D. **Removal and Replacement of Engine Instrument Lights (Basic or Avidyne Option Only)**

(1) Remove the engine instrument panel per instructions in Chapter 25-11-00.

(2) If necessary, remove the engine instrument by removing the screws in the instrument panel.

(3) To change the light bulb, unscrew the lights on top of the instrument 1/4 turn, and screw in a new light. Refer to Figure 204.

(4) Once the engine instrument light is replaced, reverse the above steps for reinstallation of the instrument. The screws holding the instrument to the instrument panel do not have torque values and should be snug tight.

E. **Magnetic Compass** - The compass light is an integral part of the compass and cannot be replaced. If the light is inoperative, replacement of the entire compass assembly is required.

5. Lower Instrument Panel

A. **Lower Instrument Lights** - The lower instrument panel lights include switches in the ignition panel, rocker panel, ECS panel, and circuit breaker panel.

B. **Ignition, Rocker, and ECS Panels** - There are two rocker switches in the ignition panel, eight switches in the rocker panel, and one switch in the ECS panel. The light bulb in these switches is soldered into the unit, which means the switch must be replaced if the bulb in the switch burns out.

C. **Rocker Switch Removal and Installation** - Use the following steps to remove and install a switch. Refer to Figure 205.

(1) Remove the panel per instructions in Chapter 25-11-00.

(2) Identify the rocker switch that is to be removed, and label each wire in relationship to its spade connector.

(3) After each wire is labeled, remove it from the switch.

(4) To remove the switch, push it out from the back of the panel. To do this, simultaneously apply pressure to the top and bottom (usually the thumb and index finger) across the width of the switch to compress the four retaining clips.

(5) Install the switch by compressing the retaining clips and inserting it into the panel.

(6) Connect the wires to the spade connectors, and with the master switch on, test the switch for proper operation.

D. **Circuit Breaker Panel Lights (Overview)** - The circuit breaker panel is located on the left side of the airplane, just forward of the pilot's seat. The panel contains five tiered rows. Above each row is a small overhang that spans the width of

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the row. Refer to Figure 206. Within each overhang there are light elements, which are epoxied into holes in the overhead row. The number of lamps in each row depends on the length of the row. Rows 1-3 have three lamps, row 4 has four lamps, and row 5 has five. The bulbs are soldered into place in series, in pairs of two. This means that if one bulb in a pair burns out, the other bulb in series will not function.

E. **Removal and Replacement of Circuit Breaker Panel Lighting**

(1) Remove the circuit breaker panel per instructions in Chapter 25-11-00.

(2) Remove the two large bus cables, disconnect the three black connectors, and remove the circuit breaker panel. Be sure to label each cable and connector location before disconnecting it.

(3) To gain access to the bulbs in rows 2-5, remove the required circuit in the row one number lower than the row with the malfunctioning light. For example, if the lights in row three are inoperative, remove the required circuit breakers (for access to the lights) in row two.

(4) Break the burned out light bulb and the companion series bulb out of their epoxied holes, and remove excess epoxy and fragments.

(5) Solder replacement bulbs.

(6) Place each bulb in its hole and, while holding it in, place with a screwdriver, add a BB-sized drop of epoxy on to the bulb and panel, and hold in place for about 30 seconds.

(7) Reconnect all plugs to the circuit breaker panel and, before reinstallation of the panels, turn on the power to the system to check the bulbs for proper operation.

6. Trim, Flaps, Fuel, And Annunciator LEDs

A. **14 V Basic or Avidyne Avionics**

(1) The trim, flap, fuel, and annunciator panel LEDs are located in the trim and flap panels, the fuel quantity gauge, and the annunciator panel. These LEDs are not field repairable. Contact a factory representative for referral to a qualified technician to replace the LEDs.

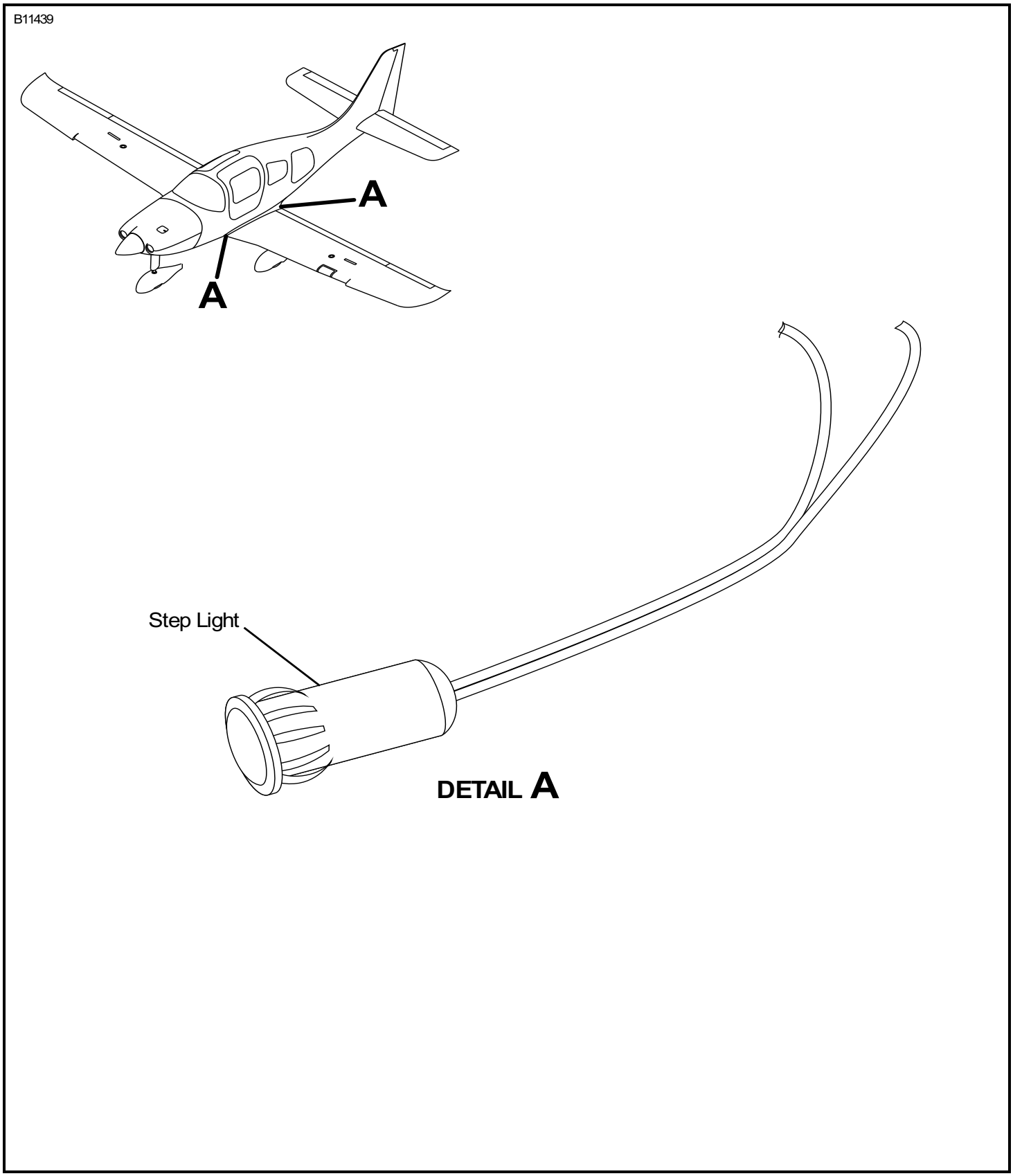
(2) **Electroluminescent Lamps (EL) Lighted Panels** - There are four EL panel overlays in the airplane. These include: the ignition panel, trim panel, flap panel, and ECS panel. The EL portion of each panel is AC powered, and through a so-called cool chemical reaction process, illuminates the letters on the panel. The inverter (AC power source) for the flap panel is located on the back of the flap assembly. The inverter for the ignition, trim, and ECS panel is located on the back of the trim assembly. Therefore, if the trim panel inverter fails, all three panels are affected.

(3) **Panel Replacement**- The panel overlays are durable, and the light source has a very high mean failure cycle and should be serviceable through four airframe service cycles, or about 12,000 hours. Should a panel fail to function, contact a factory representative for referral to a qualified technician that can replace the overlay or the inverter, depending on the problem. In some instances, particularly for airplanes with a high number of hours, it might be more cost effective to replace the entire panel assembly.

B. **28 V Basic or Avidyne Avionics and Garmin G1000 system** - All panels are LED surface lit. These LEDs are not field repairable. Contact a factory representative for referral to a qualified technician to replace the LEDs.

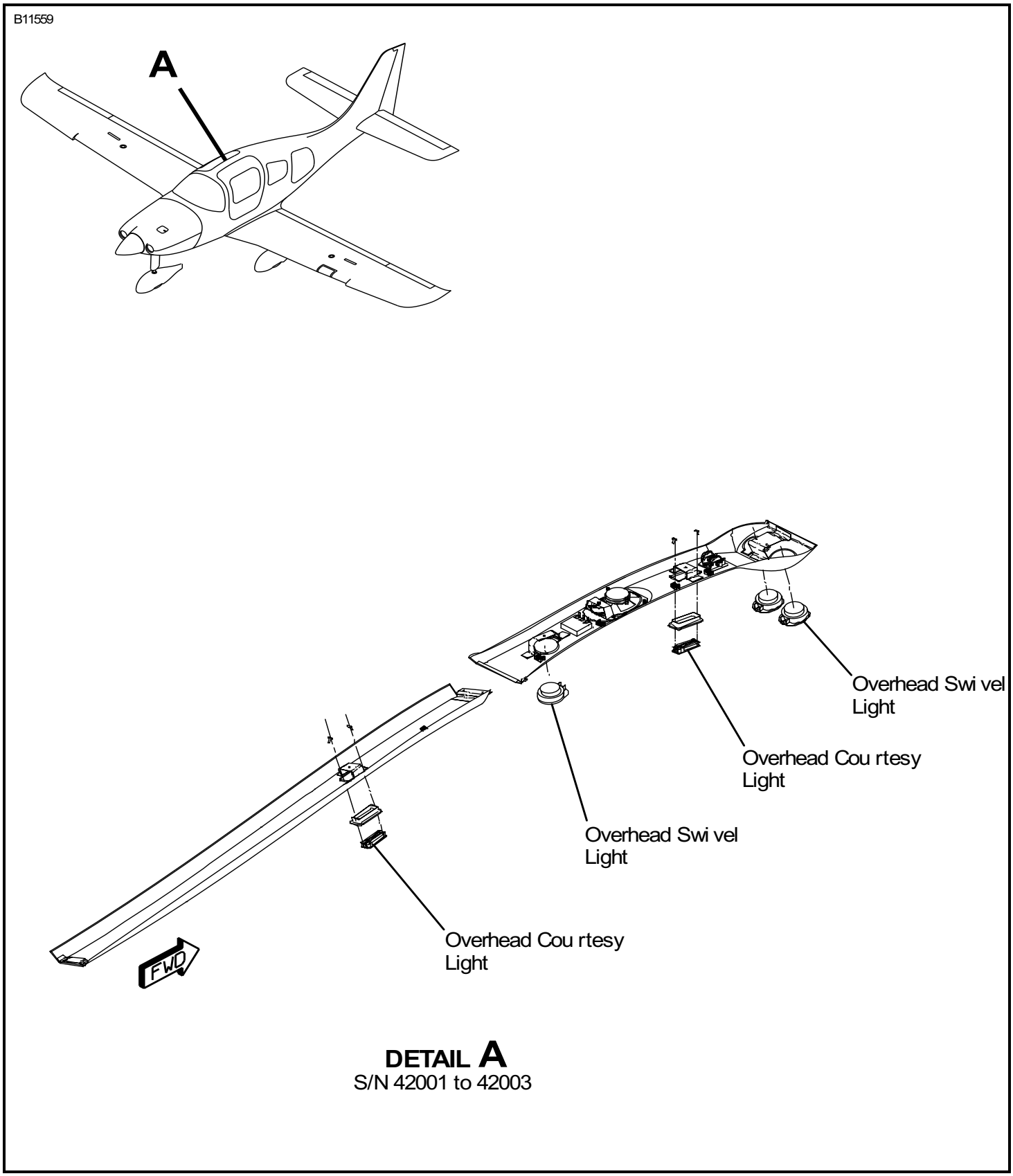
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Figure 201 : Sheet 1 : Step Light Assembly



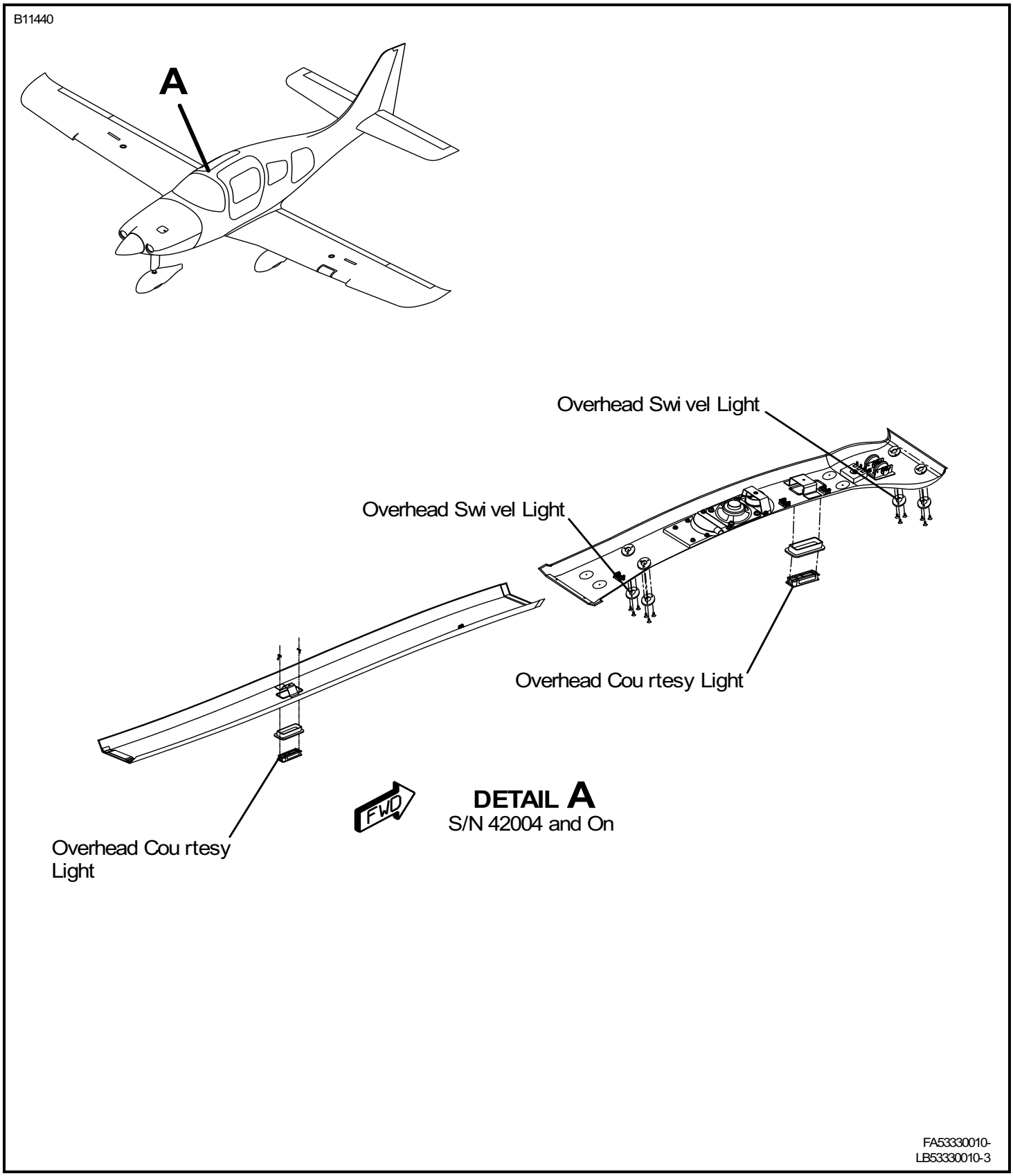
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Figure 202 : Sheet 1 : Overhead Lights



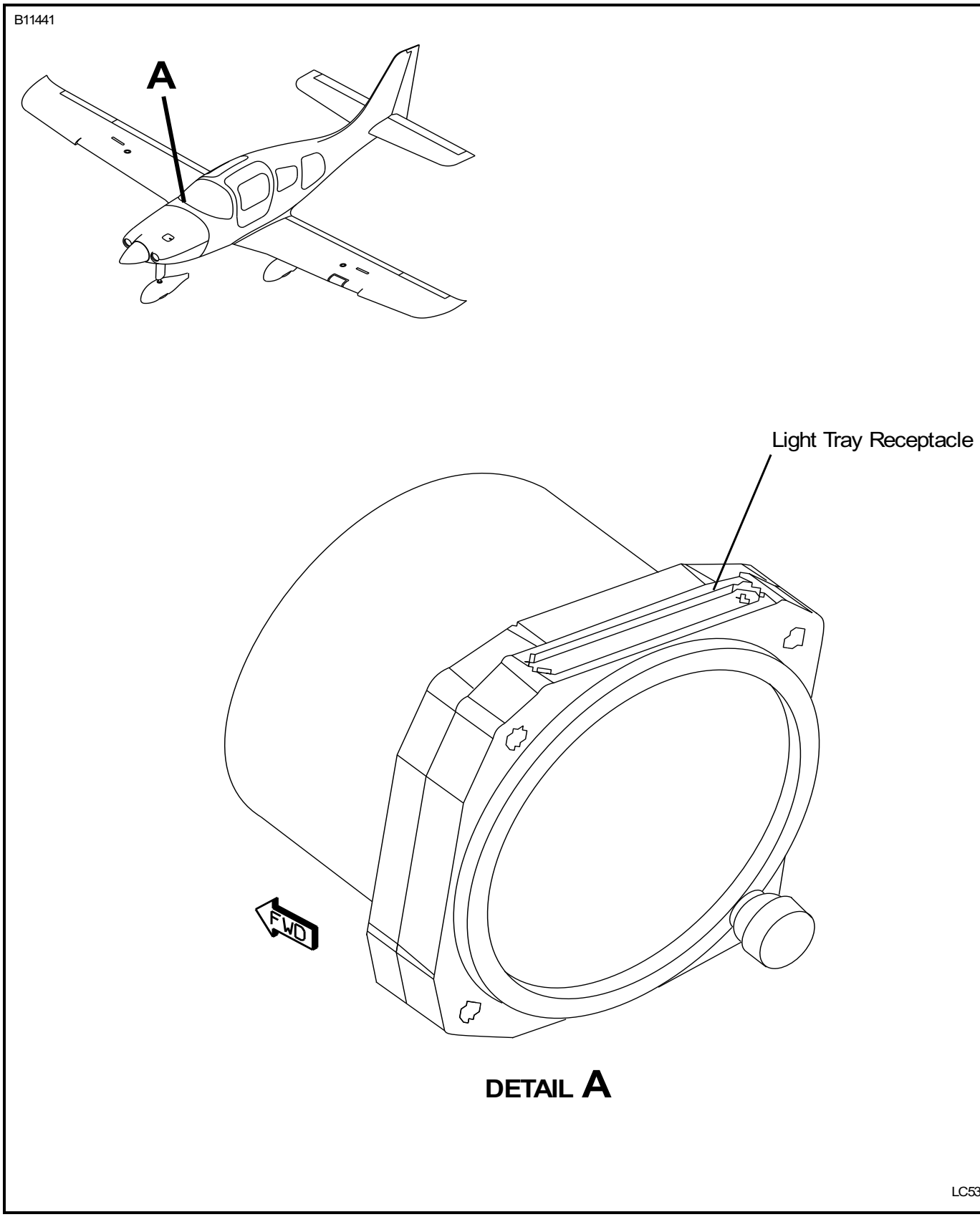
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Figure 202 : Sheet 2 : Overhead Lights



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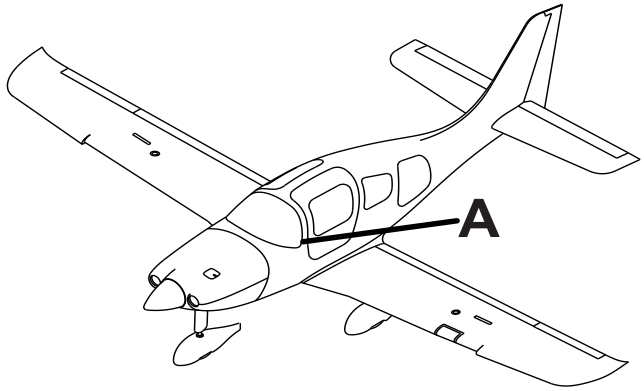
Figure 203 : Sheet 1 : Light Tray-Type of Instrument

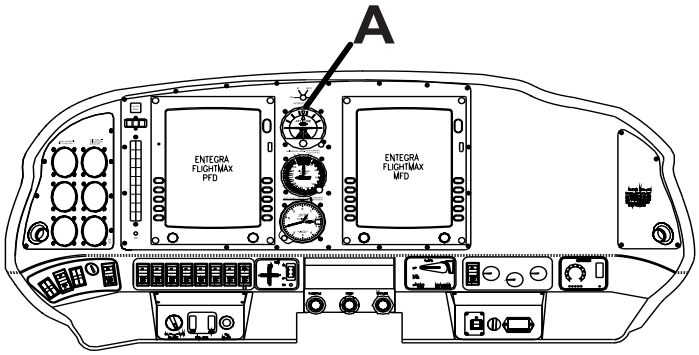


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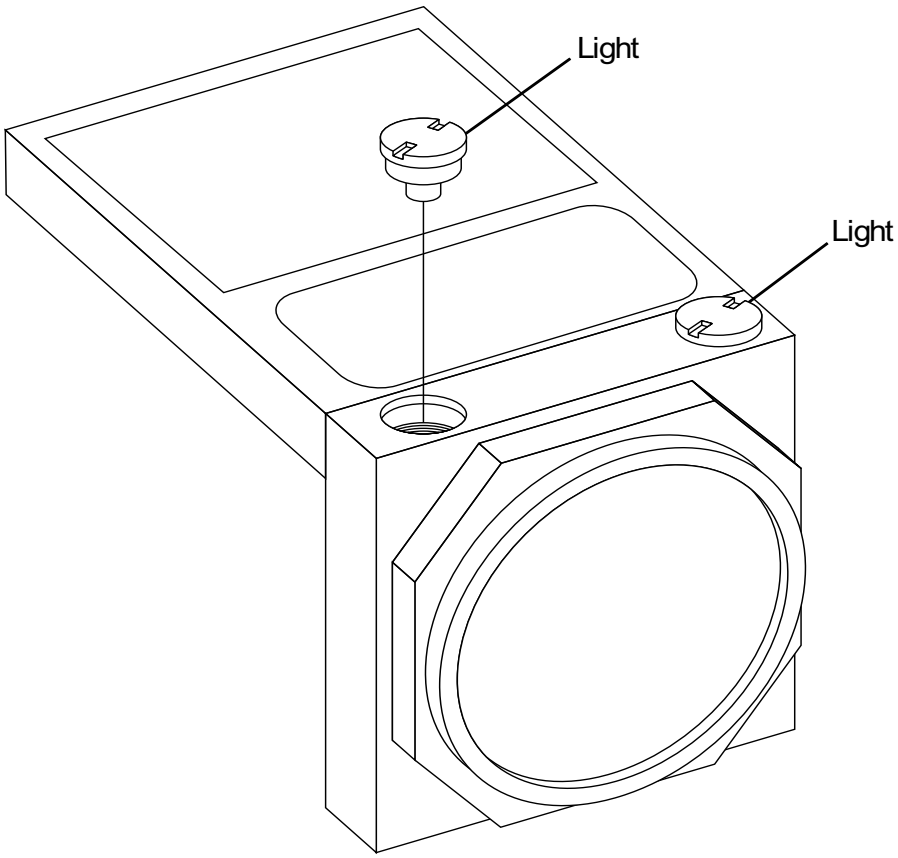
Figure 204 : Sheet 1 : Engine Instrument

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VIEW **A-A**

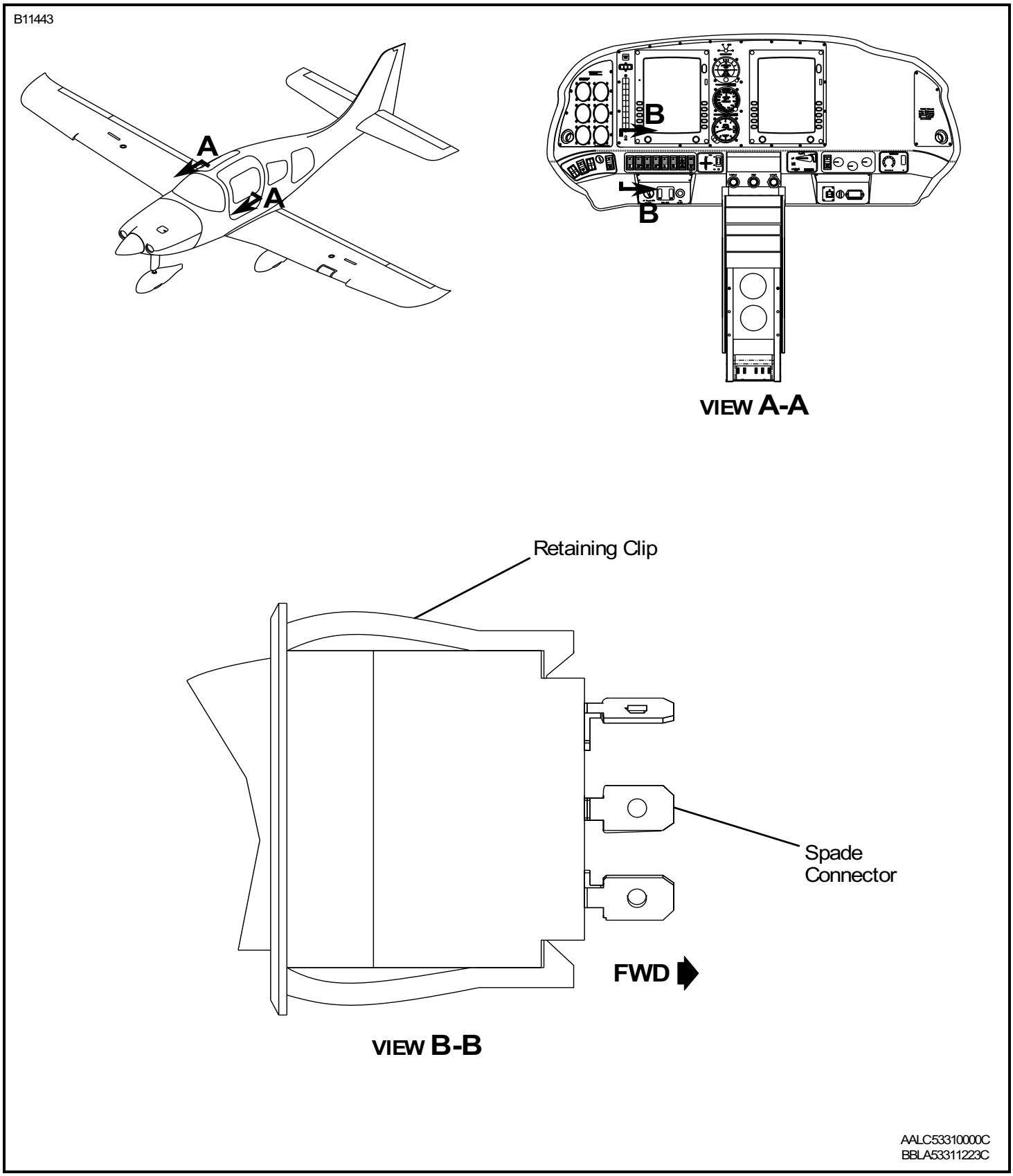




DETAIL A

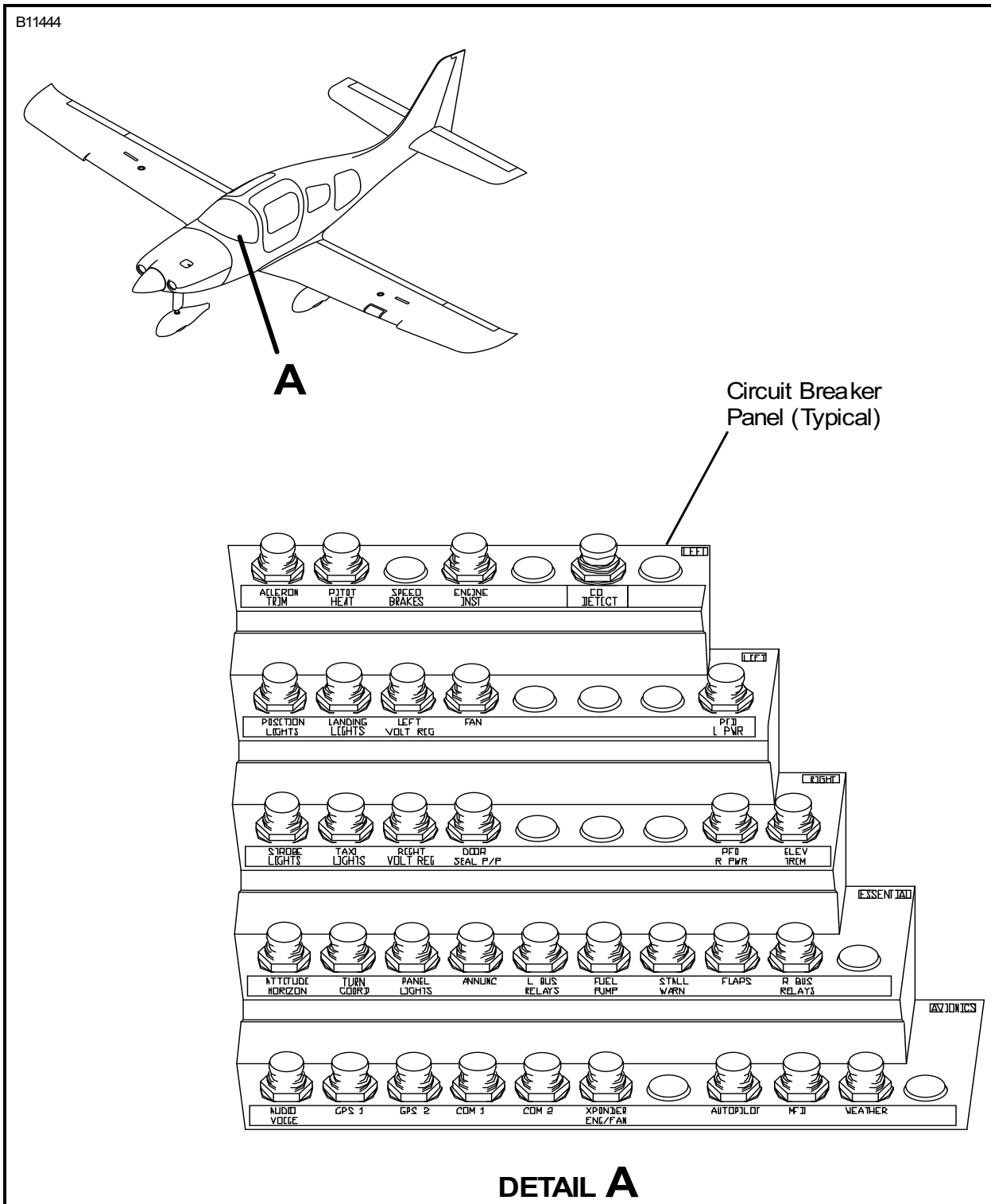
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Figure 205 : Sheet 1 : Rocker Switch



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Figure 206 : Sheet 1 : Circuit Breaker Panel



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EXTERIOR LIGHTING - MAINTENANCE PRACTICES

1. Position Lights

A. There are a total of four position lights located on the airplane. Each wing tip contains a housing that holds a colored lens forward and a clear lens aft. The position light housing also contains the strobe lights. Refer to Figure 201. All four bulbs are clear. The forward lens on the left side is red and the forward lens on the right side is green.

CAUTION: If lights have been in use, use protective gloves when removing hot bulbs.

B. **Removal of Position Light Lamp**

(1) Remove the screws securing the lens shield to the wing tip.

(2) Remove the lens from the position light assembly.

(3) Grasp the lamp, depress slightly, and turn counterclockwise to release and remove the lamp from the bayonet mount.

C. **Installation of Position Light Lamp**

(1) Place the lamp in the bayonet socket, depress, and gently turn clockwise until the lamp seats in the socket.

(2) Position the lens and gasket in place.

(3) Secure the lens by installing the lens shield to the wing tip with screws.

D. **Maintenance of Position Light** - General maintenance is limited to lamp replacement and cleaning of ozone buildup inside the position light lenses. The lenses may be cleaned with dish soap and water using a soft cotton cloth.

2. Strobe Lights

A. The strobe lights are contained in the left and right wing tip in the same housing as the position lights. Refer to Figure 201. The strobe lights and lenses are both clear. Switching the strobe rocker switch to the on position activates the strobe lights. This position supplies power to the to the strobe power supply which alternates the power between the left and right strobe light.

B. **Strobe Tube Assembly Removal (Left or Right)**

(1) Remove the screws securing the lens shield to the wing tip.

(2) Remove the lens from the front of the strobe tube assembly.

CAUTION: If the same strobe tube assembly is to be reinstalled after removal, use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the strobe tube assembly.

(3) Carefully pull the strobe light assembly wiring to retrieve the strobe connector (P17/P18) from inside the wing tip. Disconnect the strobe connector.

(4) Remove the strobe tube assembly from the base assembly, and remove from the wing tip.

CAUTION: Use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the strobe tube assembly.

C. **Installation of Strobe Tube Assembly**

(1) Position the strobe tube assembly, and connect the strobe connector.

(2) Feed the strobe connector and the strobe light assembly wiring into the wing tip.

(3) Position the spacer, and install the strobe tube assembly to the base assembly.

(4) Position the lens at the strobe tube assembly, and place the ground wire inside of the lens.

(5) Place the lens on the strobe tube assembly.

(6) Secure the lens by installing the lens shield with screws.

D. **Maintenance of Strobe Tube Assembly** - General maintenance is limited to lamp/tube replacement, removal and installation of the strobe light power supply, and cleaning of ozone buildup inside the strobe lenses. The lenses may be cleaned with dish soap and water using a soft cotton cloth.

CAUTION: Use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the strobe tube assembly.

E. **Removal of Strobe Power Supply**

(1) Ensure the system master and strobe rocker switches are in the off position.

(2) To access the strobe power supply, remove the center console access panels (basic or Avidyne Options) or remove the tower (Garmin G1000 option). Refer to Chapter 25-30-00.

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(3) Disconnect the strobe connector and strobe power supply connector. Be sure to label the location of each wire.

(4) Remove the screws securing the strobe power supply to the cabin floor, or to the tower, and disconnect the mode jumper.

(5) Remove the strobe power supply from the airplane.

F. **Installation of Strobe Power Supply**

(1) Position the strobe power supply to the nutplates in the cabin floor, or in the tower.

(2) Secure all required screws.

(3) Connect the ground wire with the screw used to secure the strobe power supply.

(4) Connect the strobe connectors, strobe power supply connector, and mode jumper.

(5) Install the left and right center console access panels, or the tower, as applicable.

(6) Set the system master and strobe rocker switches to the on position, and check strobes for proper operation.

3. Landing And Taxi Lights

A. The taxi and landing lights are contained in the leading edge of the left wing. The outboard bulb in the light housing is the taxi light that provides a diffused light in the immediate area of the airplane. The inboard bulb is the landing light, which has a spot presentation with a slight downward focus. Refer to Figure 202. Xenon taxi lights are only available on 28V aircraft.

CAUTION: The landing and taxi lights become extremely hot in a short period of time. Do not touch the lens or any part of the lights under any circumstances as they can cause serious burns.

CAUTION: Use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the landing or taxi light lens.

B. **Landing and Taxi Light Removal**

(1) Remove four screws on top and four screws on bottom of the landing and taxi light shielding.

(2) Remove the mounting screws on each corner of the light.

(3) Pull the light out and disconnect.

C. **Landing and Taxi Light Installation**

(1) Install the light connector.

(2) Place the light in the slot and install the mounting screws.

(3) Reinstall the landing and taxi light shielding with four screws on top and four screws on bottom. As required, fill the gap between the edge of the joggle and the edge of the shielding with RTV 162 white silicone caulk; do not block the 0.25 inch relief channel in the gasket.

D. **Xenon Landing and Taxi Light Ballast Removal and Installation** - (Refer to Figure 203)

(1) Remove the nine 6-32 flat head screws from the outboard access panel on the left wing and remove the panel.

(2) Disconnect the electrical connections from the ballast.

(3) Remove the locknuts, washers, and rubber spacers (as applicable) holding the ballast to the wing and remove the ballast.

(4) Installation is the reverse of removal. Tighten the nuts hand tight plus 1/4 turn. Apply Permatex No. 2 or Loctite 222 threadlocker to each screw and install the access panel. Torque the access panel screws 12 to 15 in.-lbs.

4. Taxi and Landing Light Adjustment

CAUTION: The landing and taxi lights become extremely hot in a short period of time. Do not touch the lens or any part of the lights under any circumstances as they can cause serious burns.

A. The taxi and landing lights are aimed in a manner that maximizes coverage for each respective flight operation. Over time, hard landings and rough field operations, to mention a few reasons, may cause the lights to lose their optimum alignment. Since both lights have a 100 percent duty cycle (can be used 100 percent of the time), their luminosity is somewhat reduced. For this reason it is important that the lights be kept in proper alignment. The following paragraphs describe the method for checking the alignment of both the taxi and landing lights and, if necessary, procedures for adjusting each lens.

B. **Reference Card** - To determine how the taxi and landing lights are aimed, a simple reference card must be constructed. The card should be approximately four feet high and six feet wide; corrugated cardboard works quite well and is easy to handle. Draw two horizontal, parallel lines and one vertical line on the cardboard. The horizontal lines are drawn on the 6 ft.

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length up from the bottom of the card: 25 inches for incandescent lights or 29 inches for xenon lights, and 35 inches The vertical line bisects the 6 ft. length. The horizontal line 35 inches from the bottom is the taxi light reference, and the horizontal line 25 inches or 29 in from the bottom is the landing light reference. Refer to Figure 204 for a drawing of the reference card.

C. **Landing/Taxi Light Cover Removal** - It is best to remove the outer light cover before performing the test to facilitate a more accurate measurement and allow easy access to the adjustment screws for re-aiming both of the lights. The acrylic plastic cover for the light assembly is secured with eight screws, four on the top and four on the bottom. Remove all eight screws, and pull forward on the cover to remove it. If the light lenses are dirty, they should be cleaned at this time.

D. **Ambient Light** - The test must be performed in an area of reduced ambient light such as a hanger with the doors closed and part of the overhead lighting off. Adequate lighting conditions are also available outside about one hour after sunset.

E. **Placement of Reference Card** - Place the reference card in front of the airplane so that the card is 9 ft. forward of the landing/taxi lights, and the vertical line on the card is 10 ft., 3 inches to the left (when facing forward) of the propeller spinner tip. Refer to Figure 205.

F. **Checking the Alignment** - Each light is operated and checked separately and should produce the displays discussed below. An approximate representation of the light dispersion patterns is shown in Figure 206.

   (1) Taxi Light, and Precise Flight Landing and Taxi Lights - The brightest concentration of the taxi light should focus laterally on the line 35 inches from the bottom and the pattern should be equally dispersed on the left and right side of the vertical line. Towards the edges of the reference card, the concentration will be somewhat above the line on the left side and below the line on the right side.

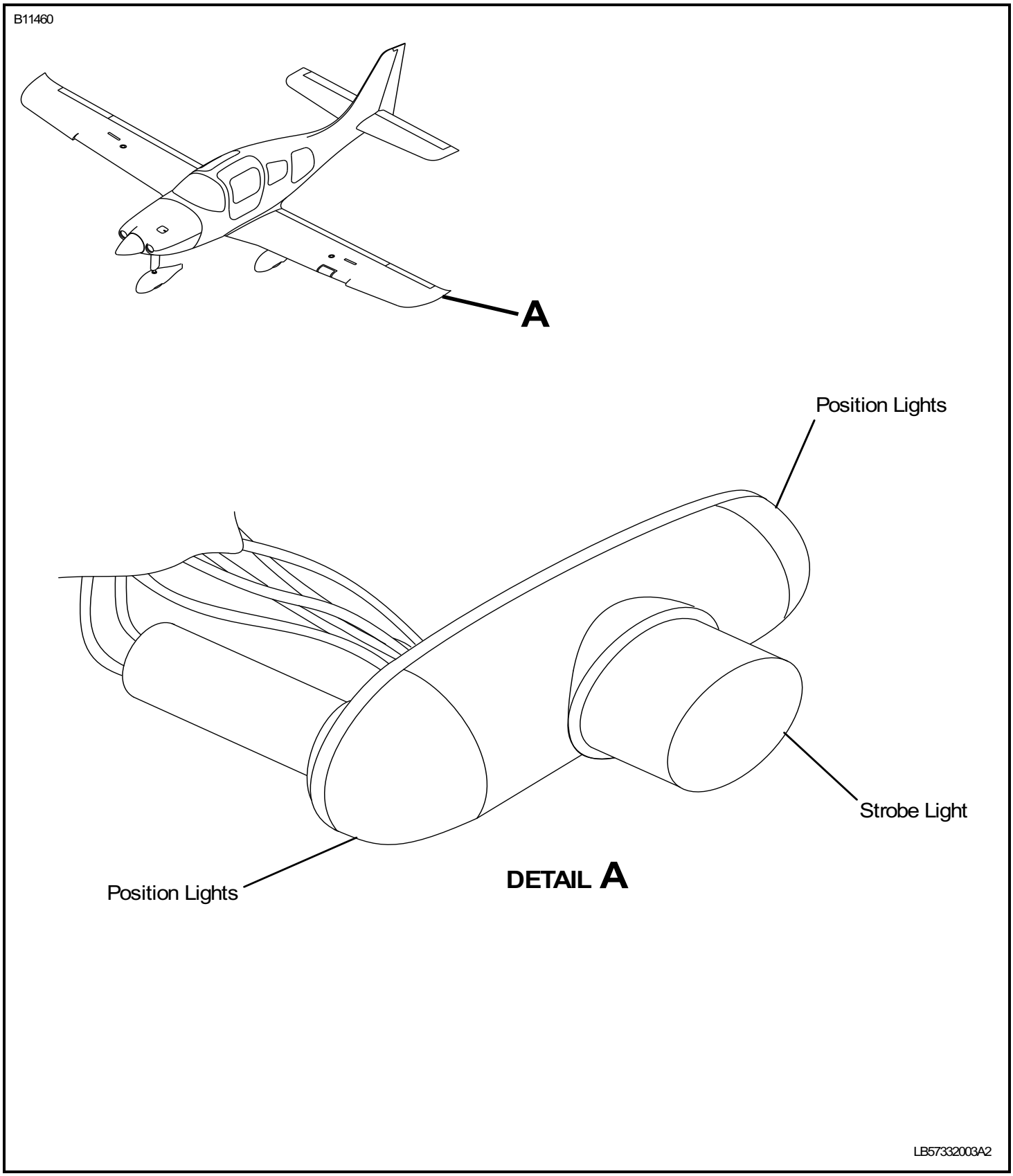
   (2) Landing Light (except Precise Flight) - The center of this light should be focused on the intersection of the vertical line and the line 25 inches for incandescent lights or 29 inches for xenon lights from the bottom.

G. **Aiming or Adjusting the Lights** - If the lights are not properly aimed, there are adjustment screws on the face of each lens that can be turned so that each lens is properly aimed. Turning the screw on the face of the lens clockwise will cause the lens to move. The amount of adjustment to each screw is limited by physical stops, so it works best to adjust both sides of the lens in small increments. For example, a full clockwise turn on a right screw should be followed by a full counterclockwise turn on the left screw.

H. **Lens Assembly** - Once the lenses are properly aimed, ensure both lights and the master switches are off. Clean the acrylic plastic housing cover, and check the condition of the rubber gasket on the inside edge of the cover. Replace or re-glue the gasket as required. After the lights have cooled sufficiently, install the cover, and tighten the eight screws. Torque screws 10 to 12 in.-lbs.

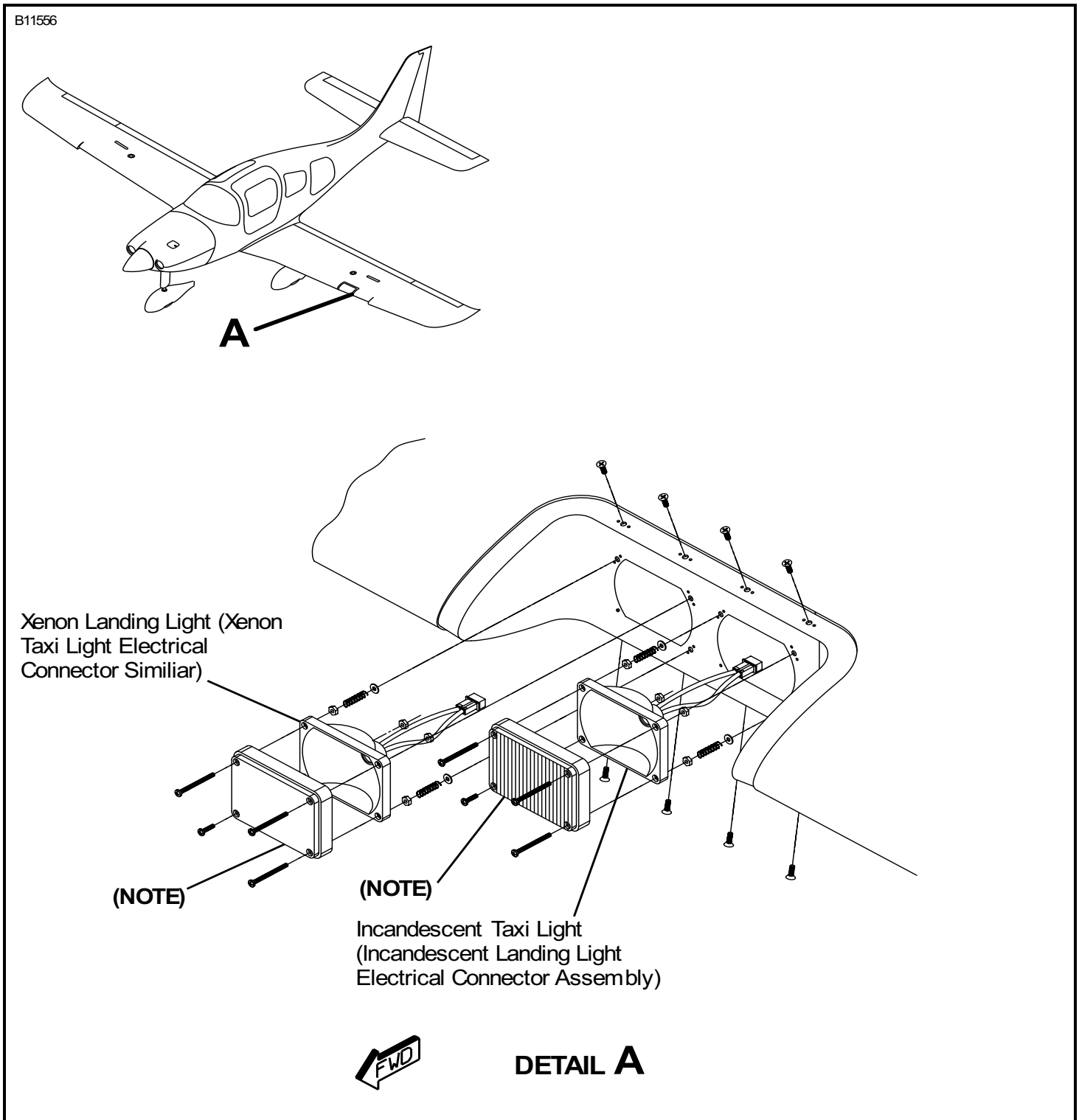
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Figure 201 : Sheet 1 : Position and Strobe Lights



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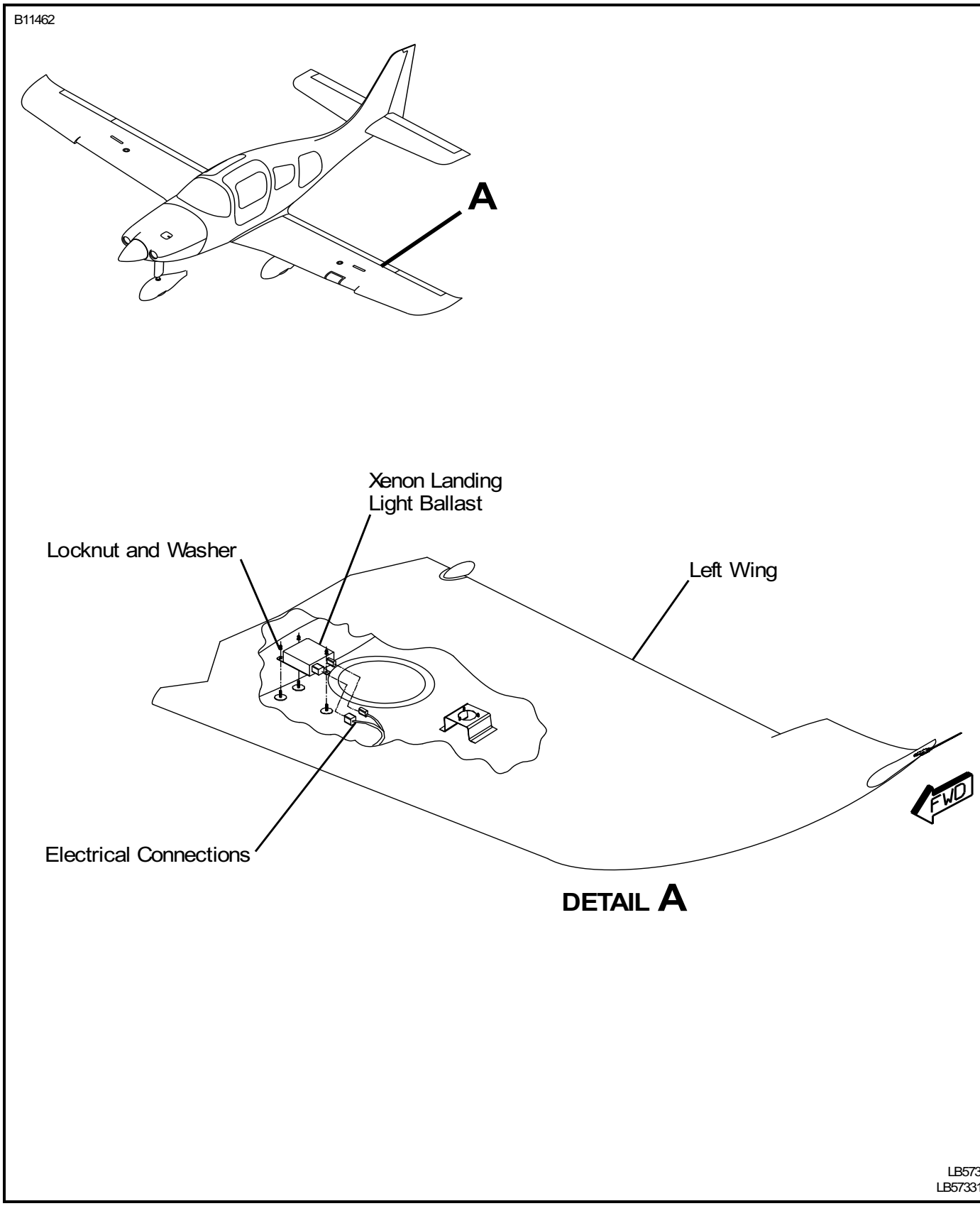
Figure 202 : Sheet 1 : Landing and Taxi Light Assembly



NOTE: Precise flight lights have integral lenses.

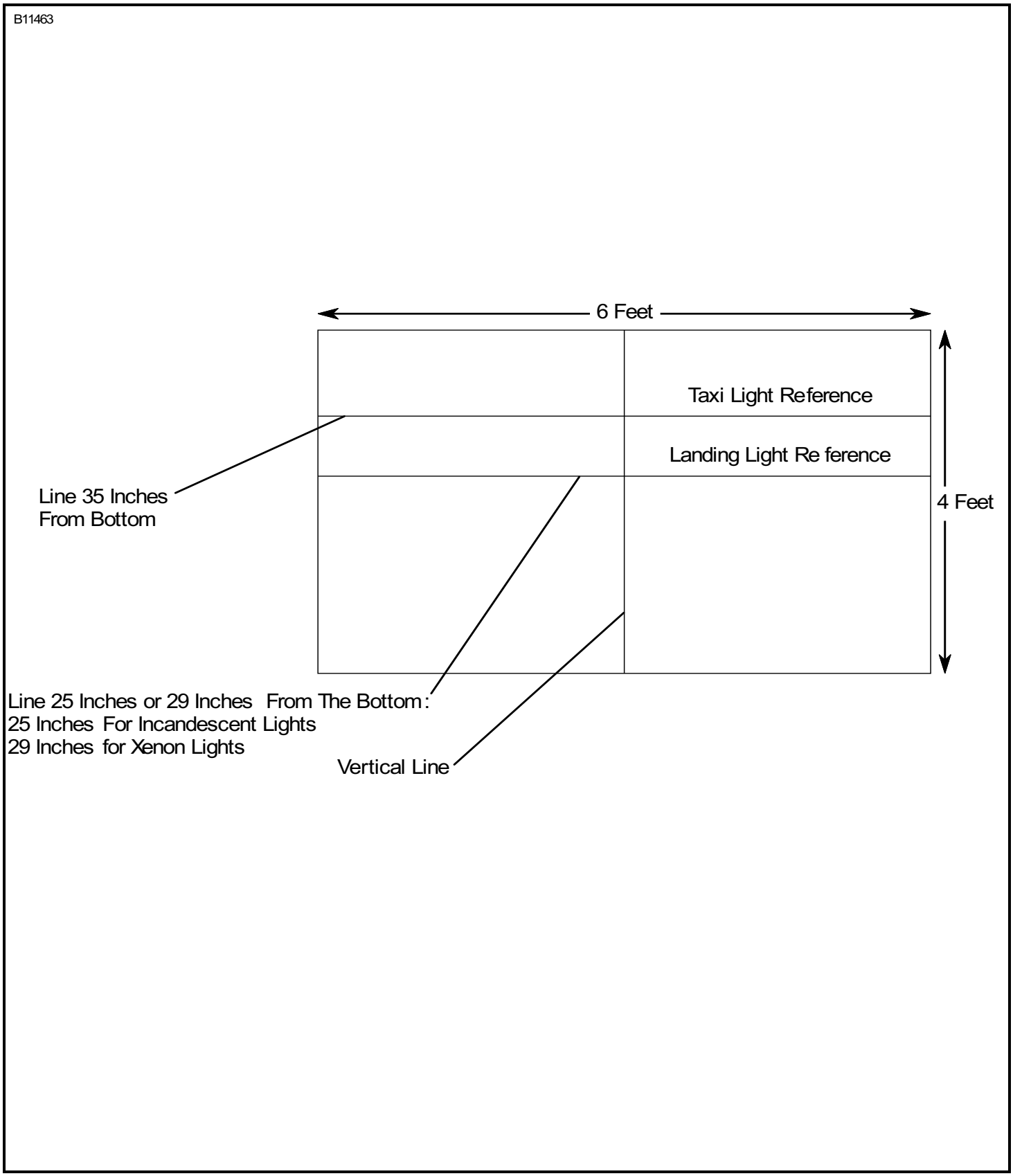
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Figure 203 : Sheet 1 : Xenon Landing Light Ballast



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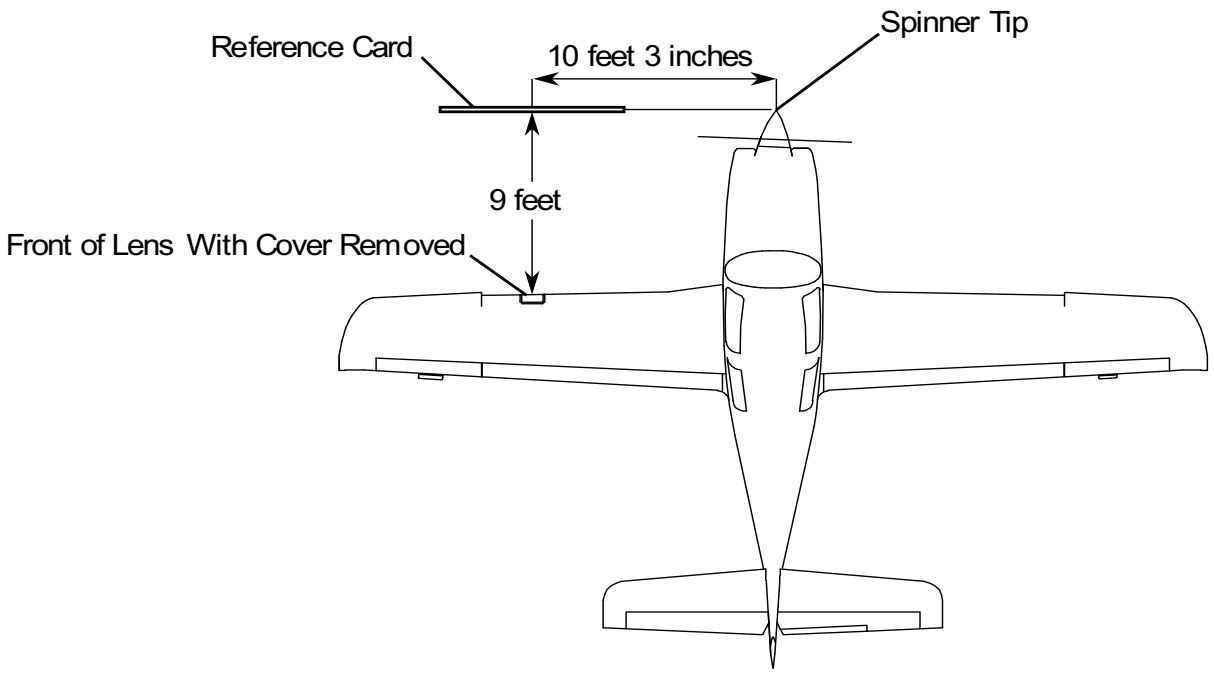
Figure 204 : Sheet 1 : Light Adjustment Reference Card



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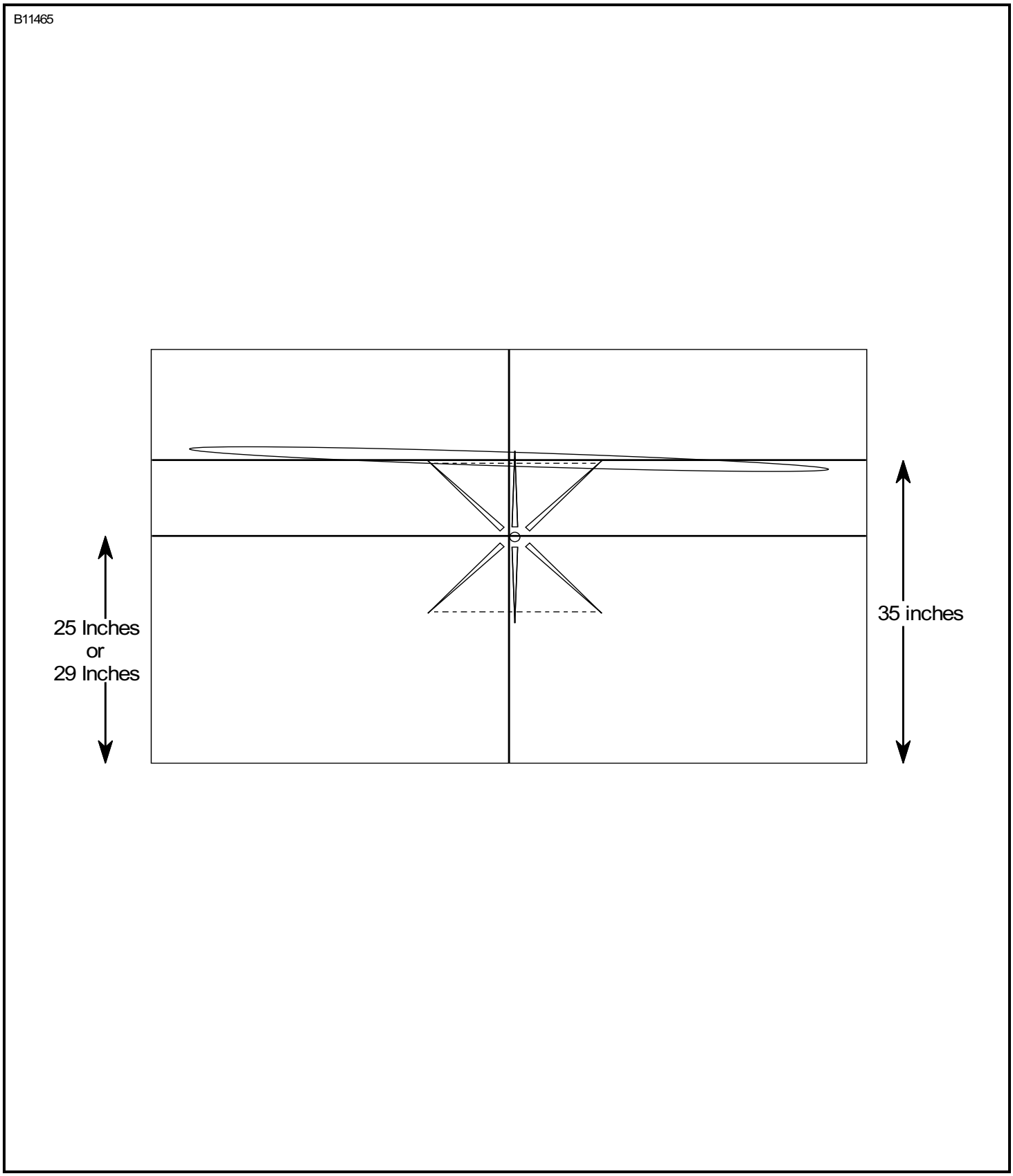
Figure 205 : Sheet 1 : Placement of Light Adjustment Reference Card

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Figure 206 : Sheet 1 : Light Adjustment



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LIFTING AND SHORING - GENERAL

1. Jacking the Airplane

A. **Lifting Points** - There are two jack points under each wing proximate to the wing saddle. The points are near the center of mass of the longitudinal axis and may also be identified by the 5/16 inch holes in those areas. Great care must be used when jacking the airplane. The tailskid is used as a third point of stabilization. Refer to Figure 1.

B. **Raising One Wing Only** - If only one wing is raised, as when changing a single tire, the airplane can be safely jacked by one person using the following steps:

(1) Perform operation in a level area, such as an airplane hangar.

(2) Set the parking brake and chock the nose tire and the main gear tire that is not raised.

(3) Place 50 pounds of ballast (usually sandbags) on the engine cowling, near the propeller.

(4) Place a jack under the jack point of the wing to be lifted and raise the jack up to the wing jack point. Take extra precaution to ensure the jack is properly stabilized, the base is locked in position, and the jack is lifting vertically. Be sure the raising point of the jack is properly inserted into the jack point on the wing.

(5) Slowly raise the jack until the desired ground clearance is achieved. The clearance between the bottom of the tire and lifting surface (ground or hangar floor) must not exceed three inches.

C. **Raising Both Wings** - If the airplane is simultaneously lifted by both jacks, then the procedures established below should be followed:

(1) Perform operation in a level area, such as an airplane hangar.

(2) Use a minimum of three trained individuals when jacking the airplane, one person for each jack and one observer.

(3) Because the jack points are near the airplane's center of gravity, 50 pounds of ballast must be placed on the horizontal stabilizer. Sandbags are perfect for this operation since they distribute pressure evenly and will not mar the surface.

(4) Attach the tail to a stabilizing stand.

(5) Place the sandbags or other ballast on the horizontal stabilizer as close as possible to the longitudinal axis.

(6) Place a jack under each jack point of the wing. Raise the jacks up to the wing jack point. Take extra precaution to ensure the jacks are properly stabilized, the bases are locked in position, and the jacks are lifting vertically. Be sure the raising point of the jacks are properly inserted into the jack points on the wing.

(7) Slowly raise the jacks until the desired ground clearance is achieved. The observer must ensure the jacks are lifting vertically and there is no tail wobble. As the airplane is lifted off the surface, the observer must connect the stabilizing stand to the tail.

D. **Raising with Two Jacks**

CAUTION: Several points must be considered when raising the airplane with two jacks:

(1) It is best to jack the airplane in an enclosed area. Wind gusts will destabilize the airplane and cause it to fall off the jacks.

(2) The airplane must be jacked in a level area on a concrete surface.

(3) Raise the jacks simultaneously in short increments to ensure the airplane stays as level as possible.

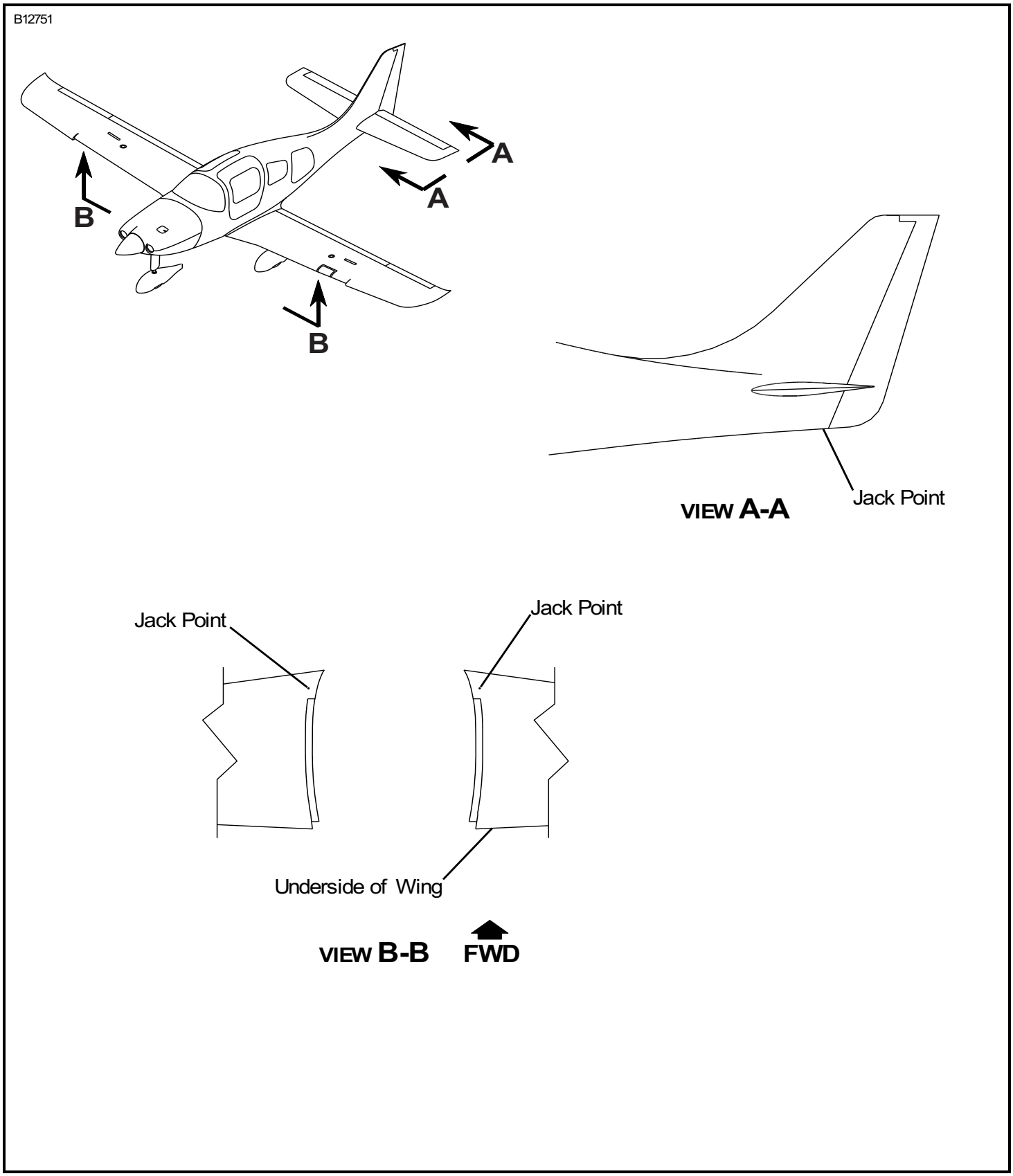
(4) If the jacks have locking pins, they must be inserted once the desired jacking height is achieved. Otherwise, the airplane must be continuously observed to ensure the jacks do not bleed off pressure. If the jack loses pressure while the tail is attached to the tail stand, damage to the tail will result.

(5) Once the airplane is on jacks, do not attempt to enter the airplane.

(6) When lowering the airplane, remove the tail stand support and bleed pressure evenly from jacks in short increments.

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Figure 1 : Sheet 1 : Jack Point Locations



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LANDING GEAR AND BRAKES - GENERAL

1. General - Scope and Definition

A. The Cessna 350 (LC42-550FG) has a fixed tricycle landing gear configuration. The main gear is constructed of high strength tubular steel and mounted to the fuselage via a welded steel gearbox. The main gearbox is bolted directly to the fuselage structure. The airplane has brakes located on the main wheels. The brakes are used for steering the airplane via differential braking. The nose gear is an oleo strut which casters for steering purposes. The oleo strut has a self-centering device to lock the nose wheel in a fixed position during flight. Aerodynamic fairings are used on the airplane to cover the landing gear and help reduce drag during flight. This chapter covers the removal and installation of the main gear, nose gear, wheel pants, and fairings. Standard maintenance procedures are also covered.

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MAIN LANDING GEAR - MAINTENANCE PRACTICES

1. Main Landing Gear - General

A. The main landing gear consists of the gearbox (located inside the fuselage), the gear legs, the axle mounting foot, axle, wheel, tire and tube, upper fuselage fairing, gear leg fairing, wheel pant, and wheel pant access panel.

2. Main Gear Wheel Pants and Fairings

A. **Removal and Installation of the Main Gear Wheel Pants and Fairings** – The removal and installation instructions below are for the left main gear. Removal and installation practices for the right side are typical unless otherwise noted.

(1) Remove the wheel pant access panel from the inside face of the wheel pant by removing the attachment screws.

(2) Remove the screws attaching the leg fairing to the wheel pant.

(3) Remove the two bolts attaching the wheel pant to the support bracket.

(4) Remove the bolt and washer attaching the wheel pant to the standoff shown in Figure 201 and Figure 202.

(5) Slide the wheel pant off by lifting and rotating it to clear the wheel and tire assembly.

(6) Remove the brake caliper per the instructions in the Servicing and Maintenance of the Brake System section.

(7) Jack the left side of the airplane such that adequate clearance is achieved for removing the main gear wheels and leg fairings.

(8) Place a small pan or bucket under the brake caliper, and disconnect the brake line from the brake caliper. The pan will help catch any excess brake fluid that may leak from the brake line once it is removed. Once the brake line is removed, cap the end to prevent any excess brake fluid from draining out of the system.

(9) Remove the main gear axle foot from the gear leg by removing the two AN5-22A bolts and two MS20365-524C all-metal lock nuts (refer to Figure 203) and sliding it straight down along the axis of the gear leg.

(10) Slide the gear leg fairing straight down along the axis of the gear leg to remove it. Jack the airplane as necessary to remove the leg fairing from the leg.

(11) Remove the attachment screws holding the upper leg fairing to the fuselage belly.

(12) Slide the fairing straight down the gear leg to remove it.

(13) Reinstall in the reverse order. Torque all fairing attachment screws 10 to 12 in.-lbs., and torque the MS20365 nuts 140 to 180 in.-lbs.

(14) When installing the standoff and attaching hardware, seat the standoff in the end of the axle on the inside of the wheel pant as shown in Figure 201 and Figure 202. Up to two washers may be glued to the inside of the wheel pant where the pant attaches to the standoff to compensate for the variation in wheel pant size. The washer should not exceed a 1-inch diameter. Use Loctite 243 on the bolt and torque 23 ft.-lbs., using care not to cause compression damage to the pant. Torque two bolts attaching the pant to the support bracket 30 to 36 in.-lbs.

B. **Servicing and Maintenance of the Main Gear Wheel Pants and Fairings** - The main gear wheel pants and fairings are a glass fiber composite construction and do not require any servicing or maintenance. The wheel pants and fairings should be inspected periodically for cracking, chips, etc. and repaired or replaced as necessary. Refer to Chapter 57 for removal and installation of optional anti-chafe tape. Contact the manufacturer for replacement parts. The interior of the wheel pant should be cleaned of all foreign debris.

3. Main Gear Assembly

A. **Removal and Installation of the Main Gear Assembly** - Use the following procedures for removal and installation of the main gear assembly.

(1) Remove the main gear wheel pant per the Main Gear Wheel Pants and Fairings Removal and Installation procedures.

(2) Raise the airplane on jacks such that adequate room is achieved for removing the main gear wheel assembly.

(3) Remove the two AN5-22A bolts and two MS20365-524C lock nuts from the main gear axle foot, and slide the wheel assembly straight down along the axis of the gear leg. Refer to Figure 203.

(4) Disassemble the main wheel assembly by removing the cotter pin and axle nut from the outboard end of the axle. Refer to Figure 204.

(5) Remove the wheel from the axle by sliding it straight off.

(6) Cut and remove the lockwire from the upper axle attachment bolts.

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(7) Remove the upper and lower axle attachment bolts to disassemble the remaining parts. Note order of assembly for reinstallation.

(8) Remove the access panels from the aft cabin floor to gain access to the upper end of the gear legs.

(9) Remove the gear leg attachment bolt located at the top end of the gearbox.

(10) Remove the gear legs from the gearbox by pulling the leg straight out.

CAUTION: Take care when removing the gear legs so as not to cause nicks, scratches, or other damage to the surface of the leg.

(11) Installation is in the reverse order. Re-safety wire the upper axle attachment bolts prior to installing the wheel.

(12) Torque the gear leg attachment bolt 80 to 85 ft.-lbs. Torque the axle attachment bolts 240 to 280 in.-lbs., and torque the axle foot attach bolts 140 to 180 in.-lbs. Rotate the wheel while tightening the axle nut 150 to 200 in.-lbs. to seat the bearing, back off the axle nut to zero torque, tighten the axle nut to 30 to 40 in.-lbs. while rotating the wheel, rotate the axle nut to the nearest slot and cotter pin hole, and install the cotter pin.

(13) Assure that the wheel rotates freely without perceptible play.

B. **Wheel Toe-in/Toe-out and Camber** - (Refer to Figure 205 and Figure 207) - During wheel installation it is important to determine the wheel toe-in/toe-out and camber. Wheel toe-in/toe-out and camber is established by inserting one or more shims between the axle block and spacer, if necessary, as shown in Figure 204. It is unlikely that the shims will need to be changed from what was initially manufactured, however, if the proper toe-in/toe-out and camber is not achieved, contact the manufacturer for the required shim(s).

(1) **Toe-in/Toe-out** - Wheel toe-in/toe-out is the angle between the aircraft centerline and the wheel. Toe-out is set with all wheels on the ground, on a low-friction surface. Set toe-out, to the value indicated in the toe-out vs. weight chart shown in Figure 206. No more than 4 shims may be used with a maximum total thickness of 0.254 inches.

(2) **Camber** - The camber is the angle from horizontal to the centerline of the axle. Camber is set with all wheels on the ground, on a low-friction surface. Set camber to the value ndicated in the camber vs. weight chart shown in Figure 208. No more than 4 shims may be used with a maximum total thickness of 0.254 inches.

C. **Servicing and Maintenance** - The landing gear legs should be inspected every 100 hours or annually. Some of the items to inspect for are scratches, nicks, corrosion, and signs of overstress. Visually inspect the gear legs for straightness. The axles and other components should also be inspected for corrosion, cracks, signs of overstress, and wear. The main gear assembly should fit snugly together with very little, if any, free play.

4. Main Landing Gear Leg Abrasion Repair

A. The main landing gear legs may develop abrasions as a result of rubbing with the landing gear fairing upper rib. Typically, there are two abrasions opposite each other. The abrasions are located 16.0 in. from the upper tip of the gear leg; they are approximately 0.7 inch long and 0.020 inch deep.

(1) Clean the surface area surrounding the abrasions.

(2) Determine the depth and length of the abrasions. If the abrasion depth and length are greater than 0.020 inch and 1.0 inch respectively, contact Cessna for instructions.

(3) Use Scotch-Brite™ or other equivalent material to blend out nicks, scratches, or cuts at the abrasion area. The repair area must be blended over a distance at least 10 times the depth of the abrasion. Refer to Figure 209.

(4) After polishing or blending the abrasion, treat all bare metal with zinc chromate or zinc oxide.

(5) Before installing the repaired main landing gear on the aircraft, contact Cessna for instructions on modifying the main landing gear fairing to prevent future rubbing.

5. Gearbox

A. The main landing gear is attached to the fuselage structure via a welded steel gearbox. The gearbox is bolted directly to the fuselage structure.

(1) **Removal and Installation of the Gearbox** - The gearbox is not easily removable from the fuselage due to its location. Should the gearbox need to be removed or replaced, please contact the manufacturer for further instructions.

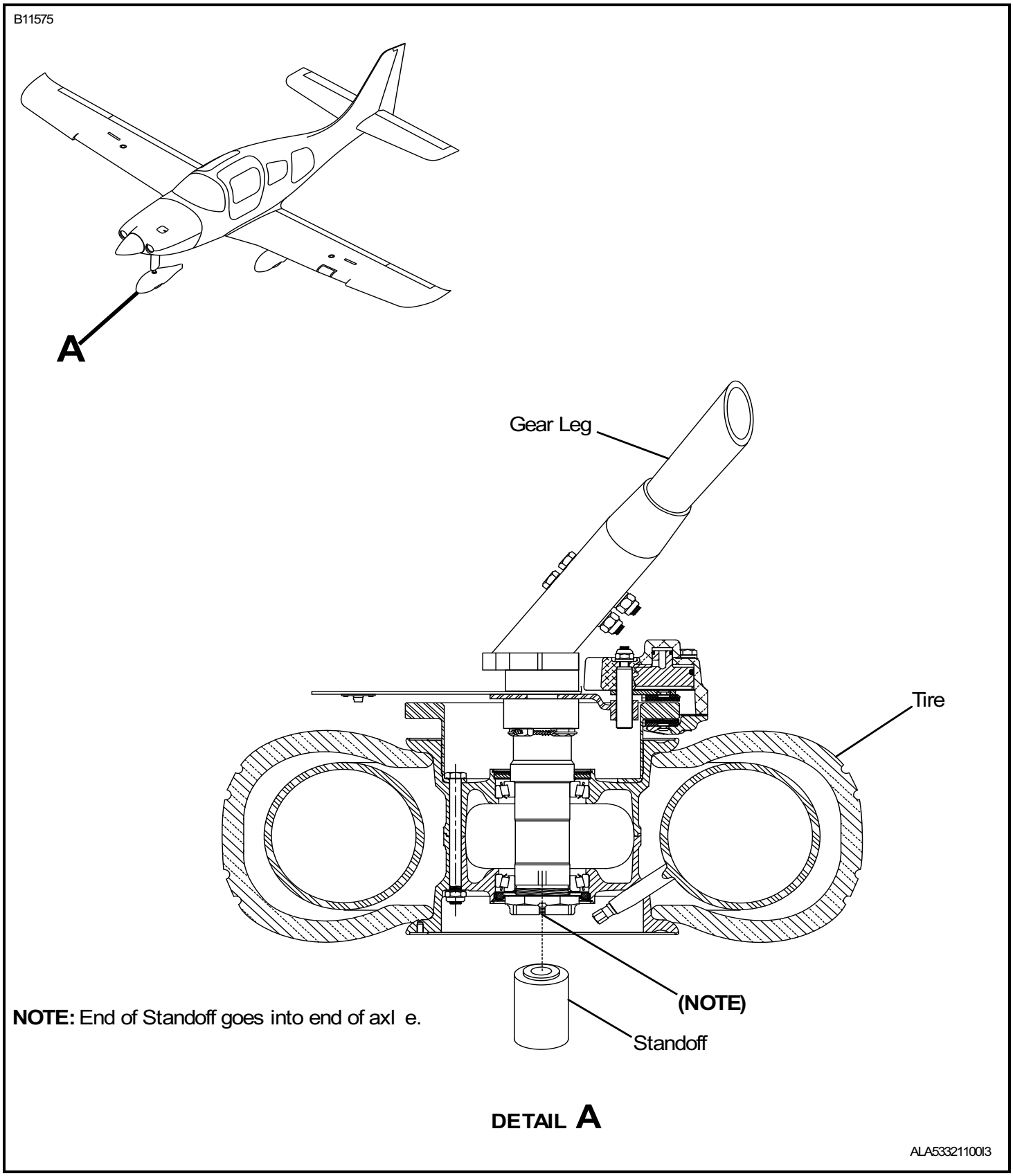
(2) **Servicing and Maintenance of the Gearbox** - The gearbox is a major structural component and should be inspected every 100 hours or annually for wear and corrosion. Remove the wheel pants and fairings per the Main Gear Wheel Pants and Fairings Removal and Installation procedures and remove the gear leg per the Main Gear

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Assembly Removal and Installation procedures. Inspect the gear leg sockets of the gearbox for wear and corrosion. If surface corrosion is present, lightly hone the interior diameter of the socket using a cylinder bore hone and light machining oil. A Scotch-Brite™ pad may also be used. After the corrosion has been removed, apply a thin film of machine oil or other surface protectant to help prevent further corrosion. The bushing at the lower end of the socket should also be inspected for excessive wear and replaced if required. When replacing the bushing, P/N 32FS16, thoroughly clean the surface of the socket and install with Loctite 660 retaining compound. Allow the Loctite to fully cure prior to reinstalling the gear leg. Every 3000 hours, the gearbox bolts must be checked for evidence of corrosion. Remove one attachment bolt from each side of the gearbox and check for corrosion. If corrosion is noted, replace all attachment bolts in the gearbox. If no corrosion is noted, replace bolts that were removed and check the torque on all bolts. Torque bolts 40 to 50 in.-lbs.

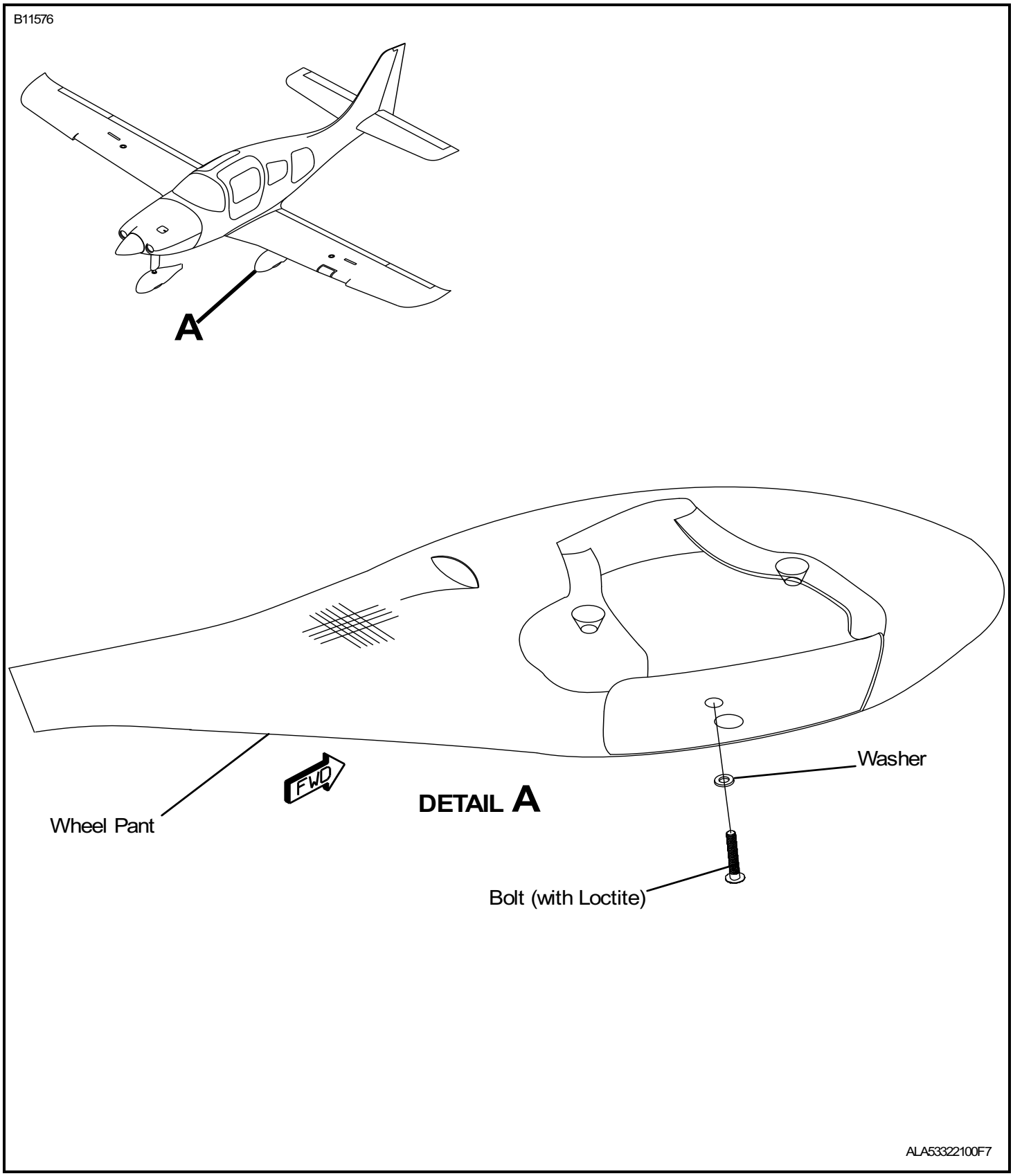
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Figure 201 : Sheet 1 : Wheel Pant Standoff Location



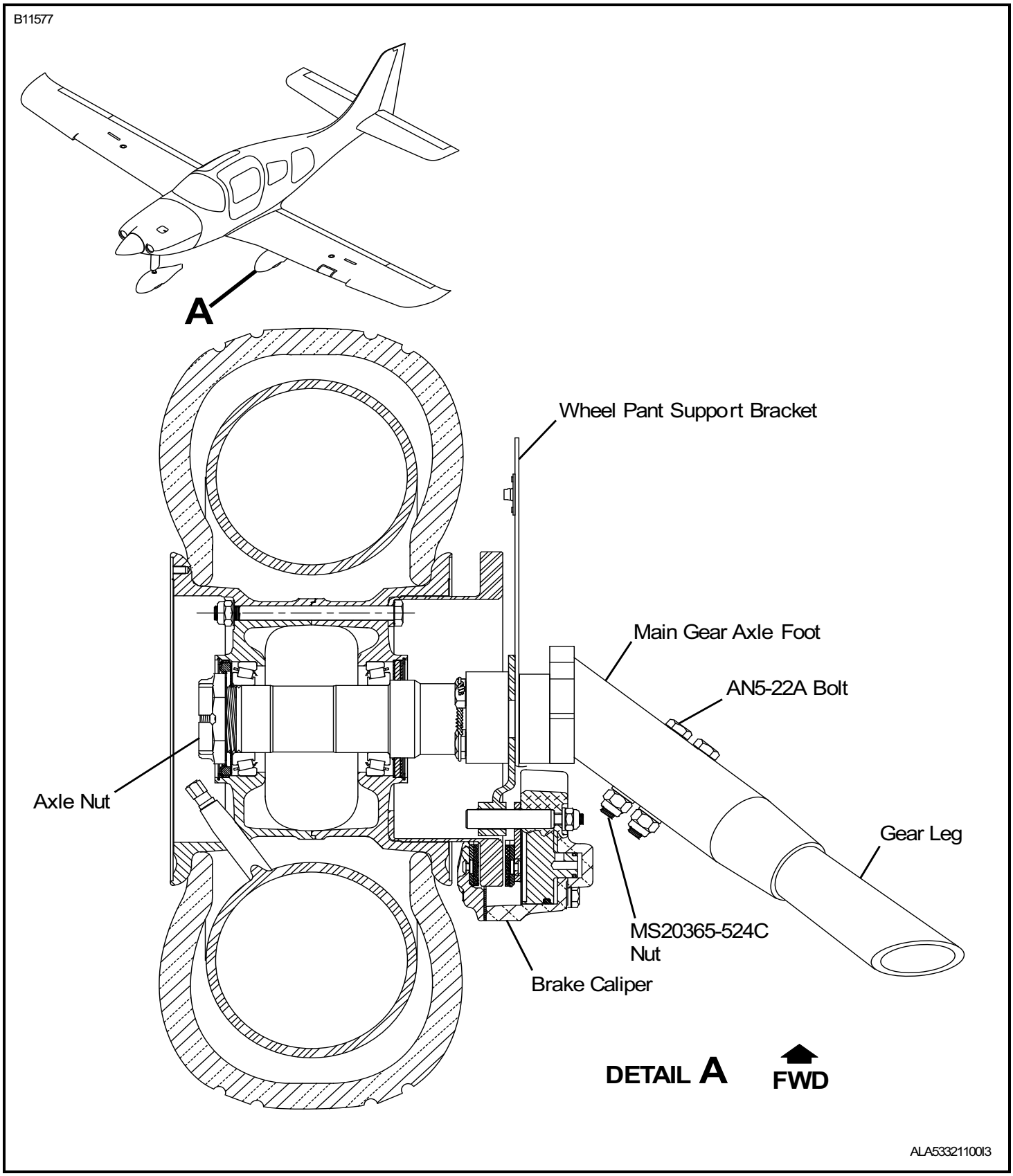
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Figure 202 : Sheet 1 : Hardware Attaching Wheel Pant to Standoff



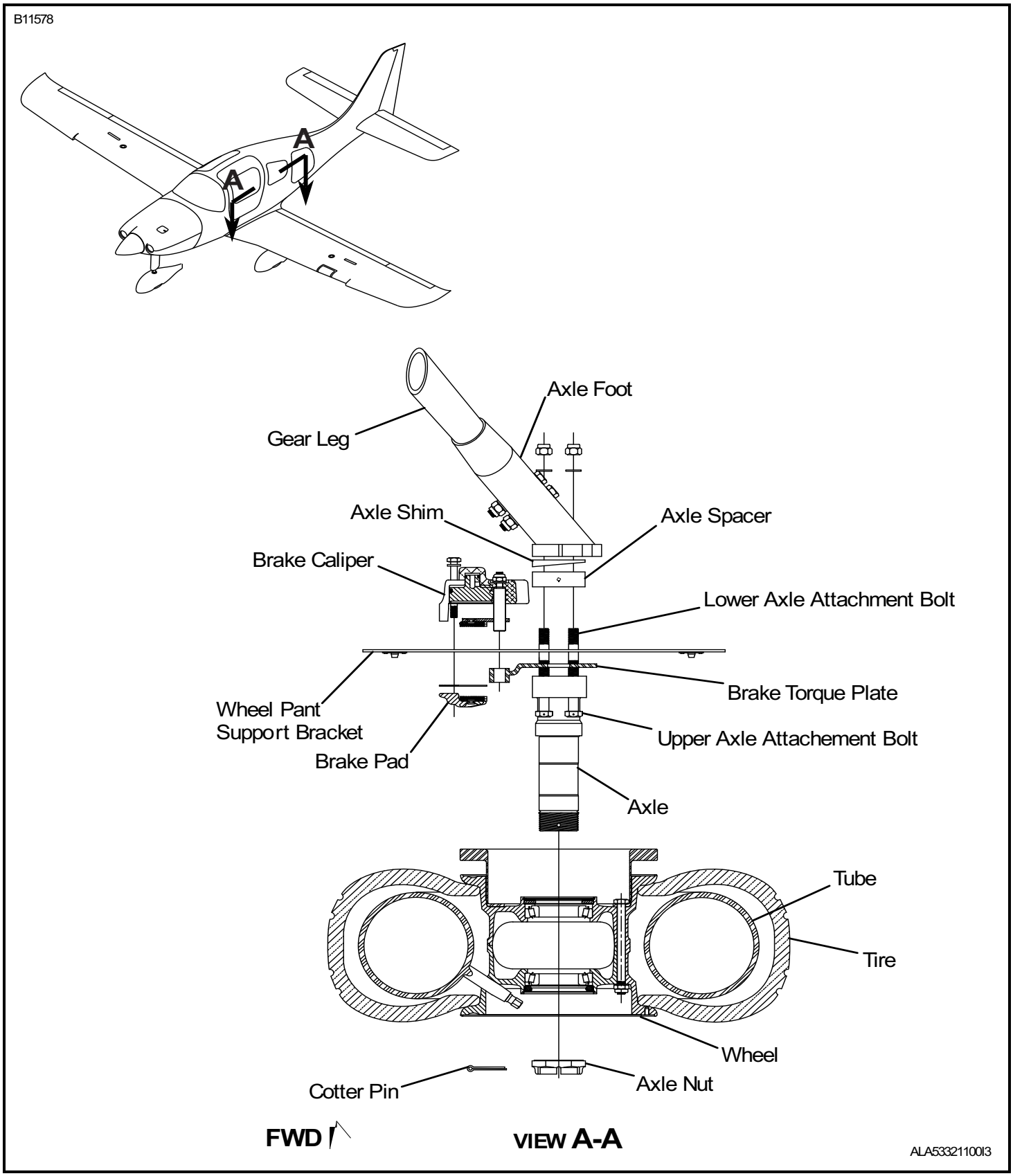
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Figure 203 : Sheet 1 : Main Gear Wheel Assembly



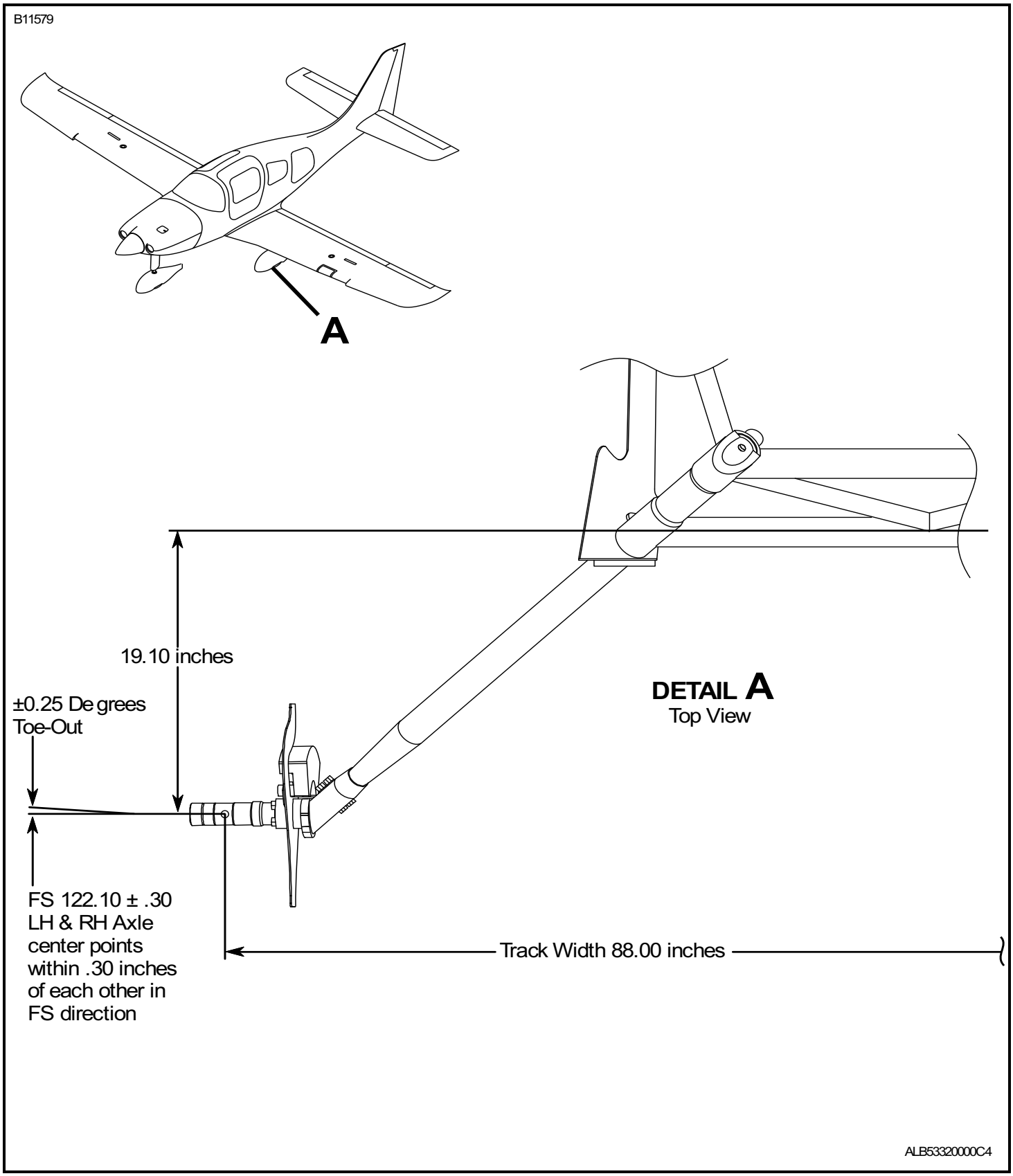
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Figure 204 : Sheet 1 : Main Gear Assembly, Exploded View



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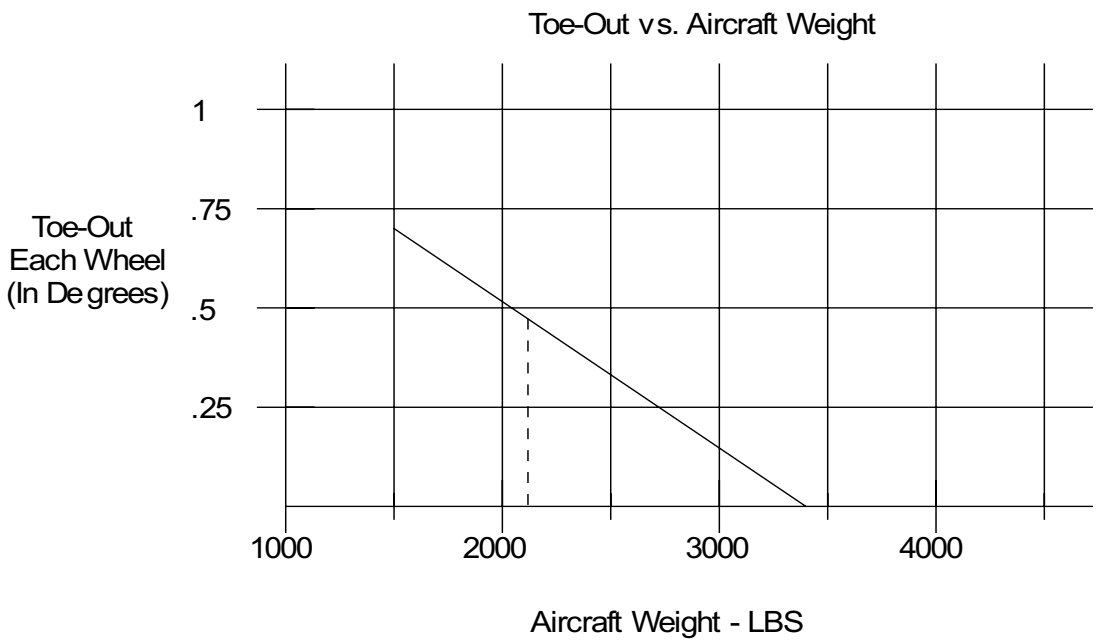
Figure 205 : Sheet 1 : Toe-in/Toe-out



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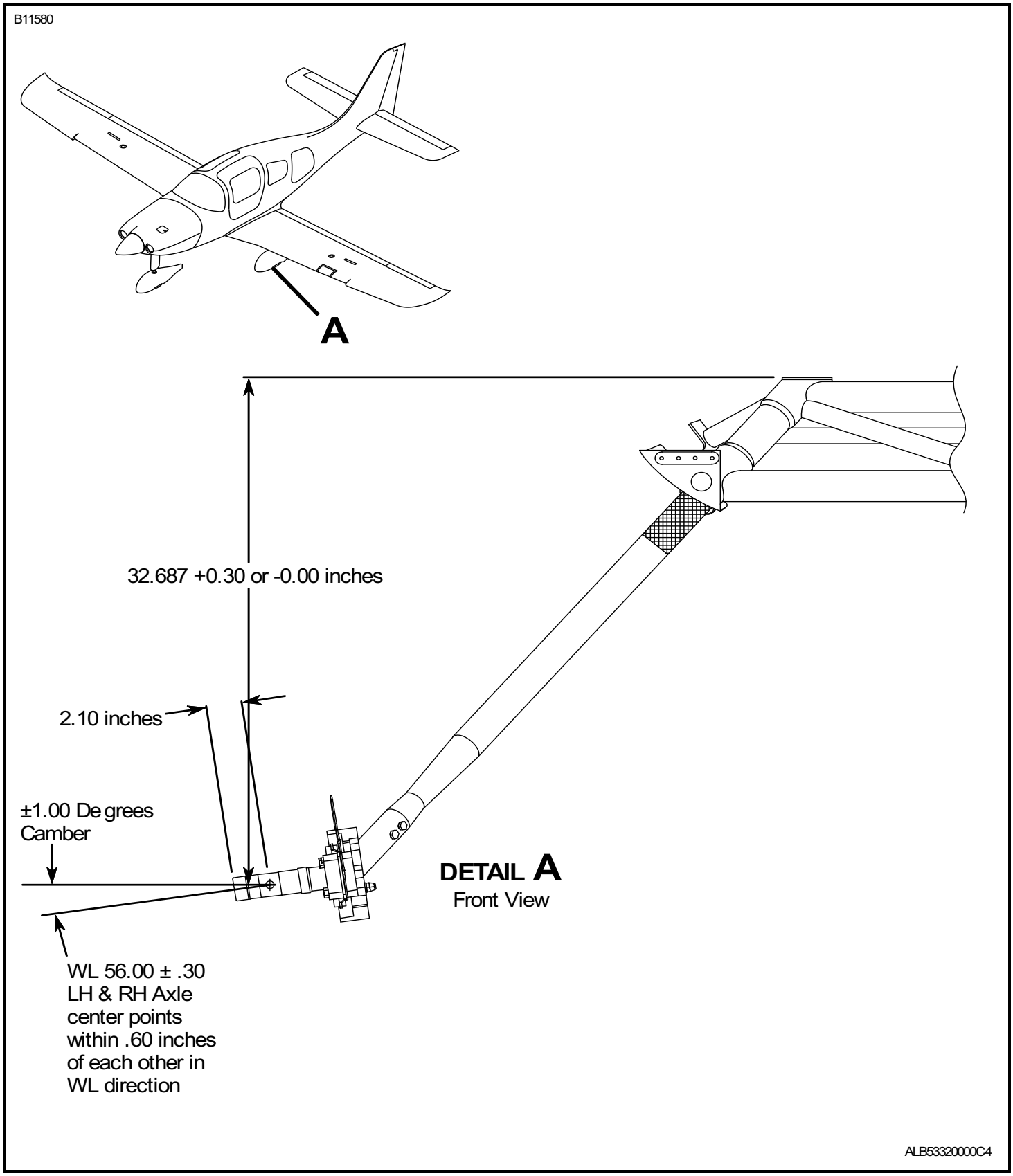
Figure 206 : Sheet 1 : Toe-out versus Aircraft (A/C) Weight

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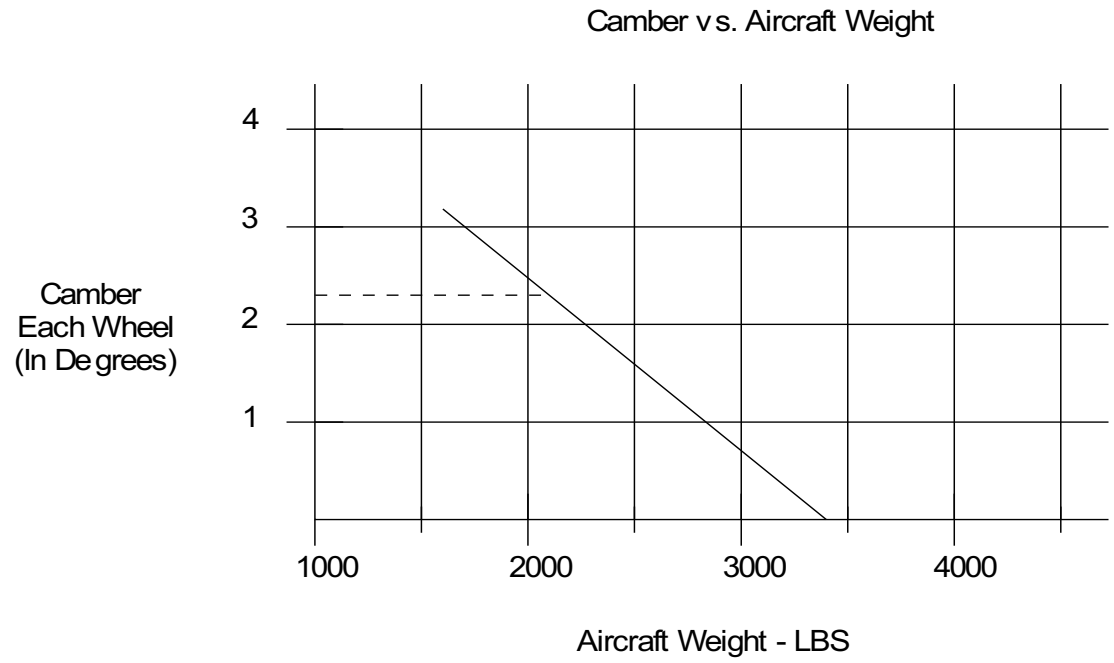
Figure 207 : Sheet 1 : Camber



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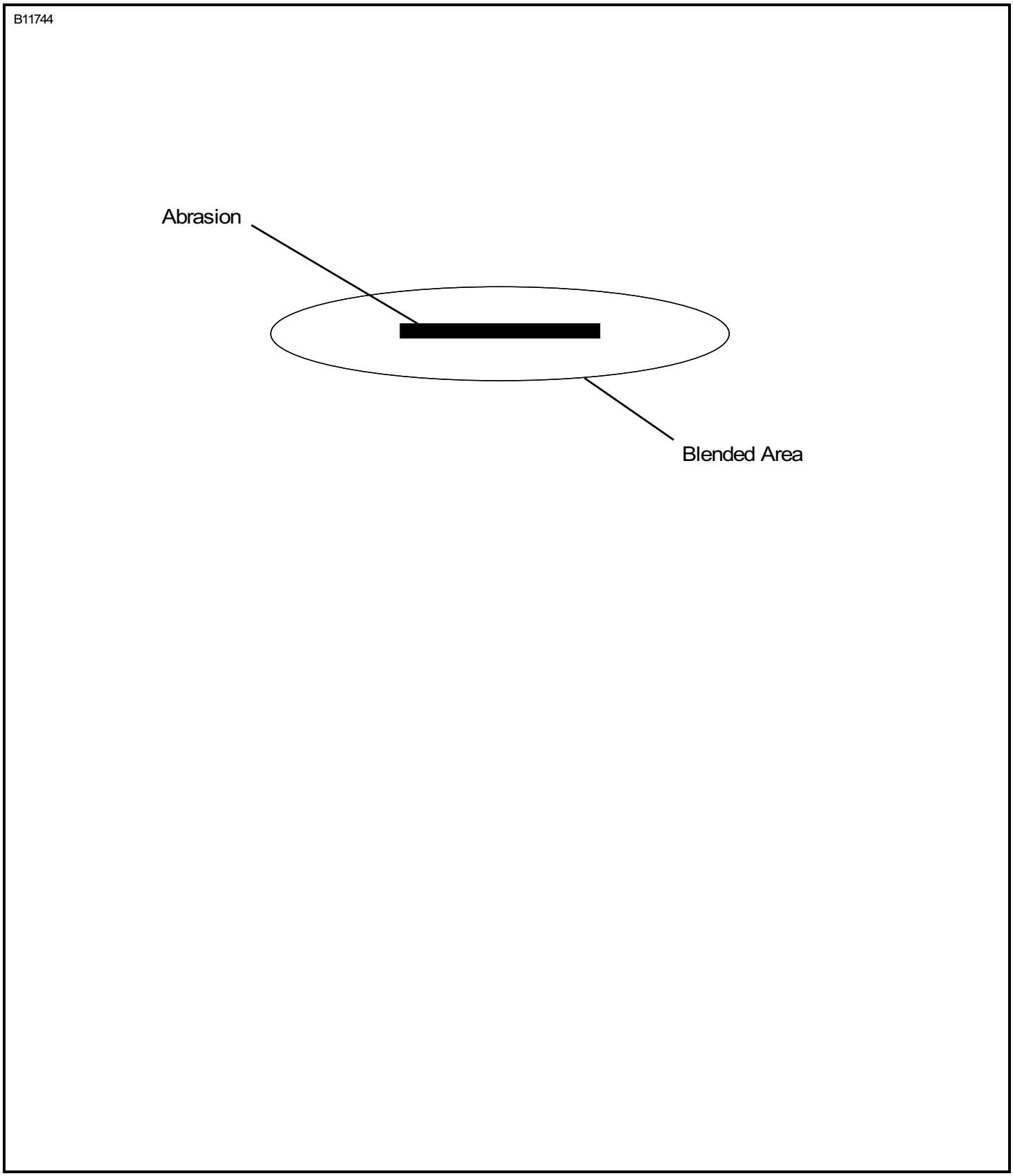
Figure 208 : Sheet 1 : Camber versus Aircraft (A/C) Weight

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Figure 209 : Sheet 1 : Blended Area for Abrasion Repair



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NOSE LANDING GEAR - MAINTENANCE PRACTICES

1. Nose Gear - General

A. The nose gear is a fixed hydraulic unit that casters freely during taxiing. Turning is accomplished via differential braking and optional rudder assist. Landing loads and shimmy dampening are controlled using hydraulic oil within the strut. Nitrogen gas pressure controls the ride height of the strut and supplies the necessary pressure to lock in the centering cam mechanism. On the ground with the weight of the aircraft bearing down on the nose strut, an internal keyway within the strut slides out from a locking cam making rotational movement possible. The moment the aircraft takes flight and all weight is off the nose gear, gas pressure moves the keyway into the locking cam eliminating any ability to caster during flight.During landing the weight of the aircraft overpowers the gas pressure and slides the keyway out from the locking cam, allowing full rotational movement of the nose wheel.

2. Nose Gear Wheel Pant and Fairing

A. **Nose Gear Wheel Pant and Fairing Removal and Installation** - Use the following procedures for removal and installation of the nose gear wheel pant and fairing.

(1) Remove the upper and lower engine cowls per instructions in Chapter 71-11-00.

(2) Remove the aft screw attaching the nose strut fairing to the fuselage.

(3) Raise the front of the airplane such that the nose wheel is completely clear of the ground.

(4) Remove the four AN5-12A bolts and washers attaching the nose wheel assembly to the base of the nose strut. Refer to Figure 201.

(5) For S/N 42001 to 42089, remove the four MS21207-6-8 self-tapping screws from the base of the strut fairing, and slide the internal rib straight down and off of the nose strut.

(6) For S/N 42090 and On, remove the two AN526C1032 or MS51958 truss head screws and NAS1149FN832P washers from the rear of the base of the strut fairing, and remove the heel plate.

(7) Slide the nose strut fairing straight down and off the nose strut.

(8) Remove the two AN526C1032 or MS51958 truss head screws from the top of the wheel pant attaching the wheel pant to the attach bracket. Refer to Figure 202.

(9) Remove the two AN526C1032 or MS51958 truss head screws from either side of the wheel pant. Refer to Figure 202.

(10) Remove the nose gear wheel pant from the wheel and fork assembly.

(11) Installation is in the reverse order. Torque all AN526C1032 or MS51958 screws 30 to 36 in.-lbs. Hand tighten the MS21207 screws holding the internal rib, or the AN526C1032 or MS51958 screws holding the heel plate, to the nose strut fairing snug. Torque the AN5-12A bolts 140 to 180 in.-lbs.

B. **Servicing and Maintenance** - The nose gear wheel pant and fairing are a glass fiber composite construction and do not require any servicing or maintenance. The wheel pants and fairings should be inspected periodically for cracking, chips, etc. and repaired and replaced as necessary. Contact the manufacturer for replacement parts. The interior of the wheel pant should also be cleaned of all foreign debris. A few owners or operators have reported hairline paint cracking in the area adjacent to the nose wheel tow bar hole. While the paint cracking does not present a structural problem, the visual presentation of the wheel pant may degrade somewhat over several years. The onset of the cracking depends, in part, on the method used to tow the airplane and the frequency of towing. Enlarging the wheel pant hole may mitigate the onset of paint cracking since the tow bar will be less likely to apply a lateral force directly to the wheel pant. To enlarge the wheel pant hole, complete the following instructions.

(1) Remove the nose wheel pant per instructions in this section.

(2) Using a suitable tool (i.e. die grinder or drill with sanding barrel), carefully enlarge the size of the lugholes to 1.25 to 1.30 in. diameter. If necessary, the hole may be enlarged eccentrically to the current hole in the wheel pant to maintain a consistent gap between the tow lug and wheel pant.

(3) Re-install the nose wheel pant per instructions in this section.

3. Nose Gear Wheel And Fork Assembly

A. **Disassembly - Wheel and Fork Assembly**

(1) Raise the front of the airplane such that the nose wheel is completely clear of the ground. Remove the four AN5-12A bolts and washers attaching the nose wheel assembly to the base of the nose strut. Refer to Figure 201.

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(2) Remove the wheel pant per the instructions in the Nose Gear Wheel Pant and Fairing section.

(3) Remove the four MS24693-C279 countersunk screws and remove the towbar plates and blocks from the fork. Refer to Figure 203.

(4) Secure the nose wheel assembly in a soft jaw vise or equivalent fixture. Remove the nose wheel axle nut, bolt, and two washers. Using a plastic mallet, gently tap inward on the fork area around the axle bushing areas until the bushings become dislodged from the fork. Remove the bushings, wheel assembly, and axle spacer from the fork. Refer to Figure 203.

B. **Reassembly - Wheel and Fork Assembly**

(1) Secure the nose wheel fork in a soft jaw vise or equivalent fixture with fork legs open and unobstructed.

(2) Insert the axle spacer into the wheel. Install the tire and wheel assembly into the nose wheel fork. Insert the axle bushings into the fork with the axle spacer inserted into the bushings. Refer to Figure 203.

(3) Install the axle bolt, nut, and two washers in reverse order as disassembled, but do not tighten the axle nut. Refer to Figure 203.

(4) With the axle bushings partially installed, check the clearance between the outside diameter of the tire and the edge of the nutplate on the fork. Refer to Figure 204. Clearance must not be less than 0.25 inches If clearance is less than 0.25 inches, replace the 5.00-5, 10 ply tire with one that has an inflated outside diameter of 14.20 inches or less per TSO-C62C.

(5) Rotate the axle bushings so that the flat edges of the bushings are parallel with the edge of the tow bar blocks (blocks prevent axle bushings from rotating within the fork). If necessary, temporarily install the tow bar plates to the fork to assist alignment of bushings during installation.

(6) Tighten the axle nut until the axle-bushing flanges are fully seated against the outside surface of the fork. If necessary, lightly tap the fork legs outward with a plastic or rubber mallet while tightening the axle bolt. Do not allow bushings to become cocked or jammed. If bushings become hard to install, stop and remove the bushings to check for any possible damage to the bushing or fork bushing holes. Replace all damaged parts. Do not file, grind, polish or sand bushings or fork bushing holes. Do not install a warm bushing into a cold fork. If necessary, place the axle bushings in a freezer for one hour and submerge fork in hot water for ten minutes prior to assembly. Anti-seize compound may be used to help install axle bushings into the fork.

(7) With axle bushings fully installed, tighten nut until all side play is eliminated from wheel bearings, then continue tightening an additional quarter turn. The wheel should have a slight drag when the bearings are properly preloaded. Verify that the wheel bearings are fully seated by attempting to rock the wheel from left to right. No movement or "play" should be felt.

(8) With towbar plates and blocks in place, torque countersunk screw 30 to 36 in.-lbs. Refer to Figure 203.

(9) Attach the nose wheel assembly to the base of the nose strut with the four AN5-12A bolts and washers. Refer to Figure 201. Torque the bolts 140 to 180 in.-lbs.

(10) Install the nose wheel pant over the fork and wheel assembly. Install four AN526C1032 wheel pant to fork attachment screws and washers in reverse order as disassembled. Torque screws 30 to 36 in.-lbs.

C. **Servicing and Maintenance** The nose wheel assembly does not require any service or maintenance. The nose wheel assembly should be inspected for corrosion and other damage annually or every 100 hours. Damaged parts should be replaced as required. The nose wheel is to be serviced per the procedures outlined in Chapter 12.

4. Nose Strut

A. The nose gear may be equipped with either the Cessna nose strut or the ESCO nose strut. The Cessna strut nitrogen gas inlet Schrader valve is located at the bottom rear of the nose strut. The ESCO strut has "ESCO" clearly labeled and the nitrogen gas inlet Schrader valve is located at the upper front of the strut.

(1) **Cessna Nose Strut**

(a) **Removal of the Nose Strut**

1 Remove the nose strut fairing per the instructions in the Nose Gear Wheel Pant and Fairing Removal and Installation procedures.

2 Relieve all load from the nose gear by pushing down on the tail of the aircraft.

3 Remove the valve cap from the strut Schrader valve. Using a small flat bladed screwdriver covered by a rag, gently press in on the Schrader valve core stem to slowly bleed off nitrogen gas pressure in the strut.

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CAUTION: Always wear safety goggles when releasing gas pressure.

4 Using a felt tip marker, apply an aligning index mark between the engine mount and the strut housing.

5 Remove the bolts, nuts (either castellated or self-locking) and washers connecting the nose strut to the engine mount. Refer to Figure 205.

6 Using a twisting motion, pull the nose strut straight down and out of the engine mount.

(b) **Installation of the Nose Strut**

1 Using a twisting motion, slide the nose strut housing into the engine mount until the housing collar is seated flush against the engine mount tube and index marks are aligned.

2 Install the bolts, nuts, and washers connecting the nose strut to the engine mount in reverse order as disassembled. Refer to Figure 205. Torque nuts 72 to 84 in.-lbs. Install new cotter pins in the castellated nuts.

3 Service the nose strut per the instructions in Chapter 12-30-00.

4 Reinstall the nose strut fairing per the instructions in the Nose Gear Wheel Pant and Fairing Removal and Installation procedures.

5 Reinstall the nose wheel assembly per the instructions in the Nose Gear Wheel Pant and Fairing Removal and Installation procedures.

(c) **Inspecting and Servicing the Nose Strut**

1 Inspection (every 100 hours or annually)

a The nose gear strut should be inspected for cracks, corrosion, scratches on the strut shaft, fluid leakage, and signs of overstress. The strut attachment to the engine mount should be checked for security. If oil leakage is present at any one of these locations, replacement of the strut is required.

b With the full weight of the aircraft on the nose and main gear inspect the ride height of the nose gear strut. There should be 3 to 4 inches of the chrome strut rod visible between the lower flange and the wiper seal. If ride height is low and no oil leakage is present, recharge the strut per Chapter 12-30-00. Recheck the filler valve and the filler valve gasket for signs of leakage.

c Remove weight off the nose gear to allow rotation of the nose wheel assembly left and right to its full range of rotation. A smooth resistance from the internal hydraulic shimmy damper device should be felt throughout its rotation. If binding is felt during rotation service the strut per Chapter 12-30-00. If the strut has been properly serviced and continues to bind, replacement of the strut is required.

d Rotate the nose gear wheel assembly to one side and remove just enough weight off the nose gear so the strut is fully extended and the wheel assembly is no longer in contact with the floor. If the wheel assembly does not center, in line with the direction of flight, service the strut per Chapter 12-30-00. If the strut has been properly serviced and does not center, replacement of the strut is required.

2 Servicing (every 200 hours or annually) - The strut should be serviced per the instructions in Chapter 12-30-00.

(2) **ESCO Nose Strut**

(a) **Removal of the Nose Strut**

1 Remove the nose strut fairing per the instructions in the Nose Gear Wheel Pant and Fairing Removal and Installation procedures.

2 Relieve all load from the nose gear by pushing down on the tail of the aircraft.

3 Remove the valve cap from the strut schrader valve. Using a small flat bladed screwdriver covered by a rag, gently press in on the Schrader valve core stem to slowly bleed off nitrogen gas pressure in the strut.

CAUTION: Always wear safety goggles when releasing gas pressure. Apply rags around the Schrader valve and a drip pan under the strut to catch any oil that may discharge with the nitrogen gas.

4 Remove the Schrader valve from the strut housing.

5 Using a felt tip marker, apply an aligning index mark between the engine mount and the strut housing.

6 Remove the bolts, nuts (either castellated of self-locking), and washers connecting the nose strut to the engine mount. Refer to Figure 206.

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7 Using a twisting motion, pull the nose strut straight down and out of the engine mount.

(b) **Installation of Nose Strut**

1 Using a twisting motion, slide the nose strut housing into the engine mount until the housing collar is seated flush against the engine mount tube and index marks are aligned.

2 Install bolts, nuts, and washers connecting the nose strut to the engine mount in reverse order as disassembled. Refer to Figure 206. Torque nuts 72 to 84 in.-lbs. Install new cotter pins in the castellated nuts.

3 Screw the Schrader valve into the nose strut and torque the valve 150 to 200 in.- lbs.

4 Service the nose strut per the instructions in Chapter 12-30-00.

5 Reinstall the nose strut fairing per the instructions in the Nose Gear Wheel Pant and Fairing Removal and Installation procedures.

6 Reinstall the nose wheel and wheel pant assembly per the instructions in the Nose Gear Wheel Pant and Fairing Removal and Installation procedures.

(c) **Servicing and Maintenance**

1 Maintenance (every 100 hours or annually)

a The nose gear strut should be inspected for cracks, corrosion, scratches on the strut shaft, fluid leakage, and signs of overstress. The strut attachment to the engine mount should be checked for security. If oil is present at any one of these locations, replacement of the strut is required.

b With the full weight of the aircraft on its nose and main gear inspect the ride height of the nose gear strut. There should be 3 to 4 inches of the chrome strut rod visible between the lower flange and the wiper seal. If ride height is low and no oil is present, recharge the strut per Chapter 12-30-00. Recheck the filler valve and the filler valve gasket for signs of leakage.

c Remove weight off the nose gear to allow rotation of the nose wheel assembly left and right to its full range of rotation. A smooth resistance from the internal hydraulic shimmy damper device should be felt throughout its rotation. If binding is felt during rotation service the strut per Chapter 12-30-00. If the strut has been properly serviced and continues to bind, replacement of the strut is required.

d Remove just enough weight off the nose gear so the strut is fully extended and the wheel assembly is no longer in contact with the floor. If the nose wheel assembly does not center, in line with the direction of flight, service the strut per Chapter 12-30-00. If the strut has been properly serviced and does not center, replacement of the strut is required.

2 Servicing (every 200 hours or annually) - The strut should be serviced per the instructions in Chapter 12-30-00.

5. Hard Landing Inspection

A. After a hard landing or a landing over maximum landing weight, the aircraft must be inspected for damage in accordance with this section. If any damage is found, it must be repaired prior to the next flight.

(1) Nose Gear - The nose gear strut should be inspected for cracks, scratches on the strut shaft, fluid leakage, and signs of overstress. Check for proper nitrogen pressure in the strut. The strut attachment to the engine mount should be checked for security. If oil leakage is present at any one of these locations, replacement of the strut is required.Inspect the ride height of the nose gear strut. There should be 3 to 4 inches of the chrome strut rod visible between the lower flange and the wiper seal. If ride height is low and no oil leakage is present, recharge the strut per Chapter 12. Recheck the filler valve and the filler valve gasket for signs of leakage.

(2) Main Gear - Inspect the wheel, axle, axle mounting bolts, wheel shims, gear leg, and gear leg attachment bolts to assure there are no broken, bent, or loose parts. The main gear assembly should fit snugly together with very little, if any, free play.

(3) Wheel Toe-in and Camber - Assure that wheel toe-in and camber meets the requirements of the Main Gear Assembly section.

(4) Wheel Pants and Fairings - Visually inspect the main and nose gear wheel pants and fairings for chips, cracks, or other damage.

(5) Gearbox - Determine if the gearbox is bent or broken. With equal pressure and wear on the tires, the aircraft evenly loaded (including fuel), and the aircraft on a level surface, the wings should be level. If a bent or broken gearbox is

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suspected, contact the manufacturer, and do not fly the aircraft.

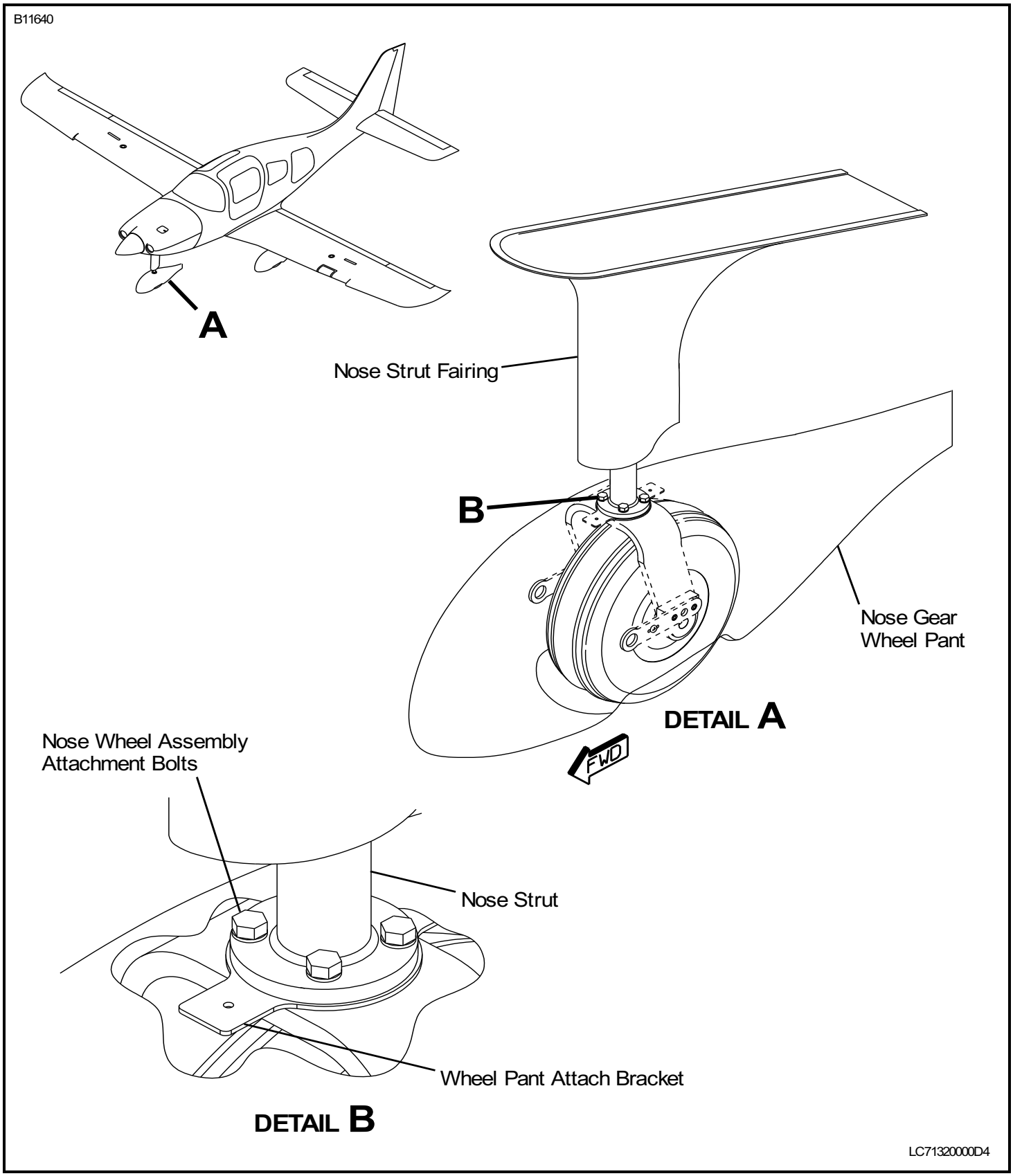
(6) Tires - Visually inspect the tires for cuts, wear, or broken belts.

(7) Engine Mount - Visually inspect the welds on the engine mount using a bright light and magnifying glass.

(8) Propeller - Visually inspect the propeller for nicks or chips.

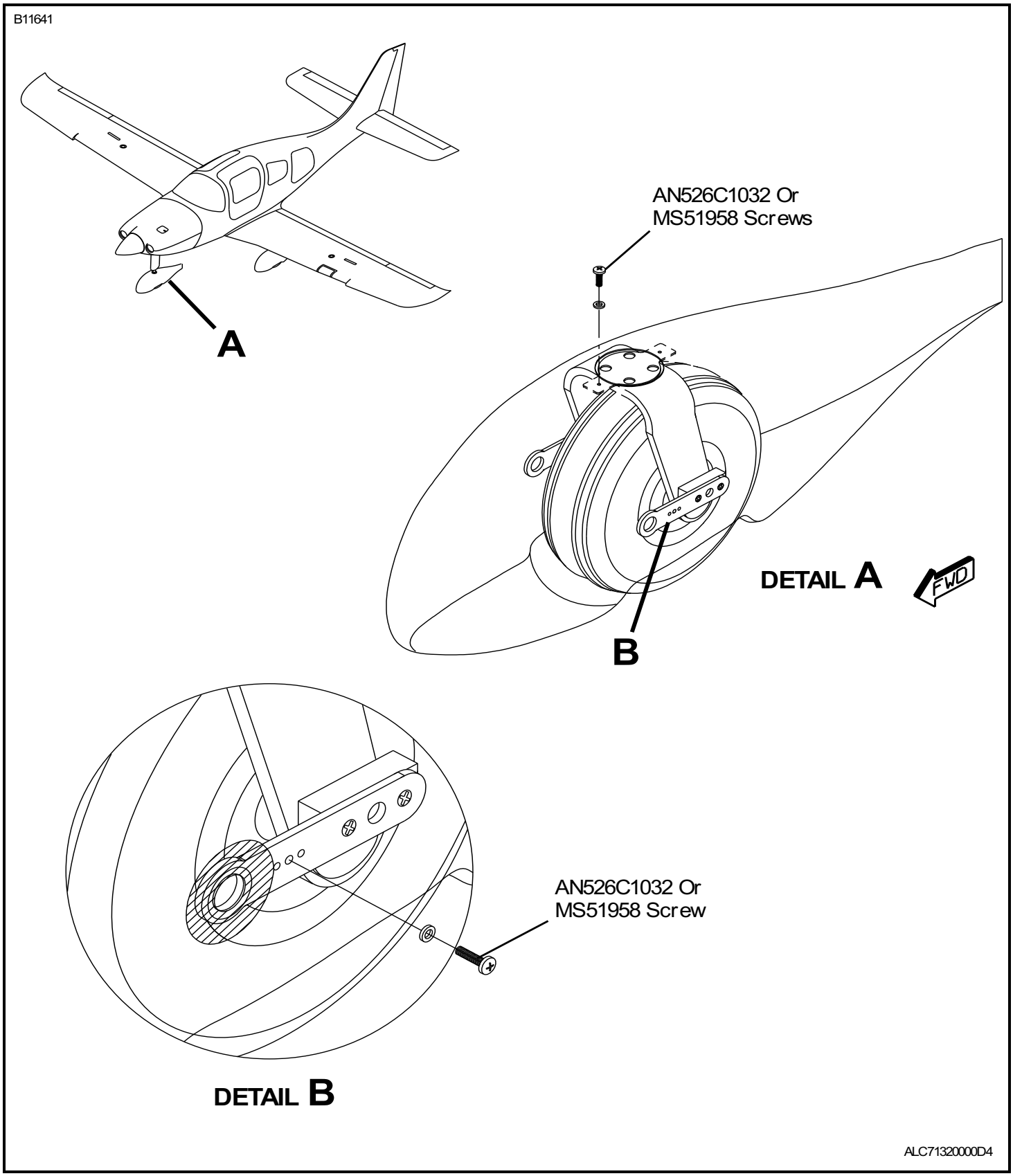
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Figure 201 : Sheet 1 : Nose Wheel Pant and Leg Fairing



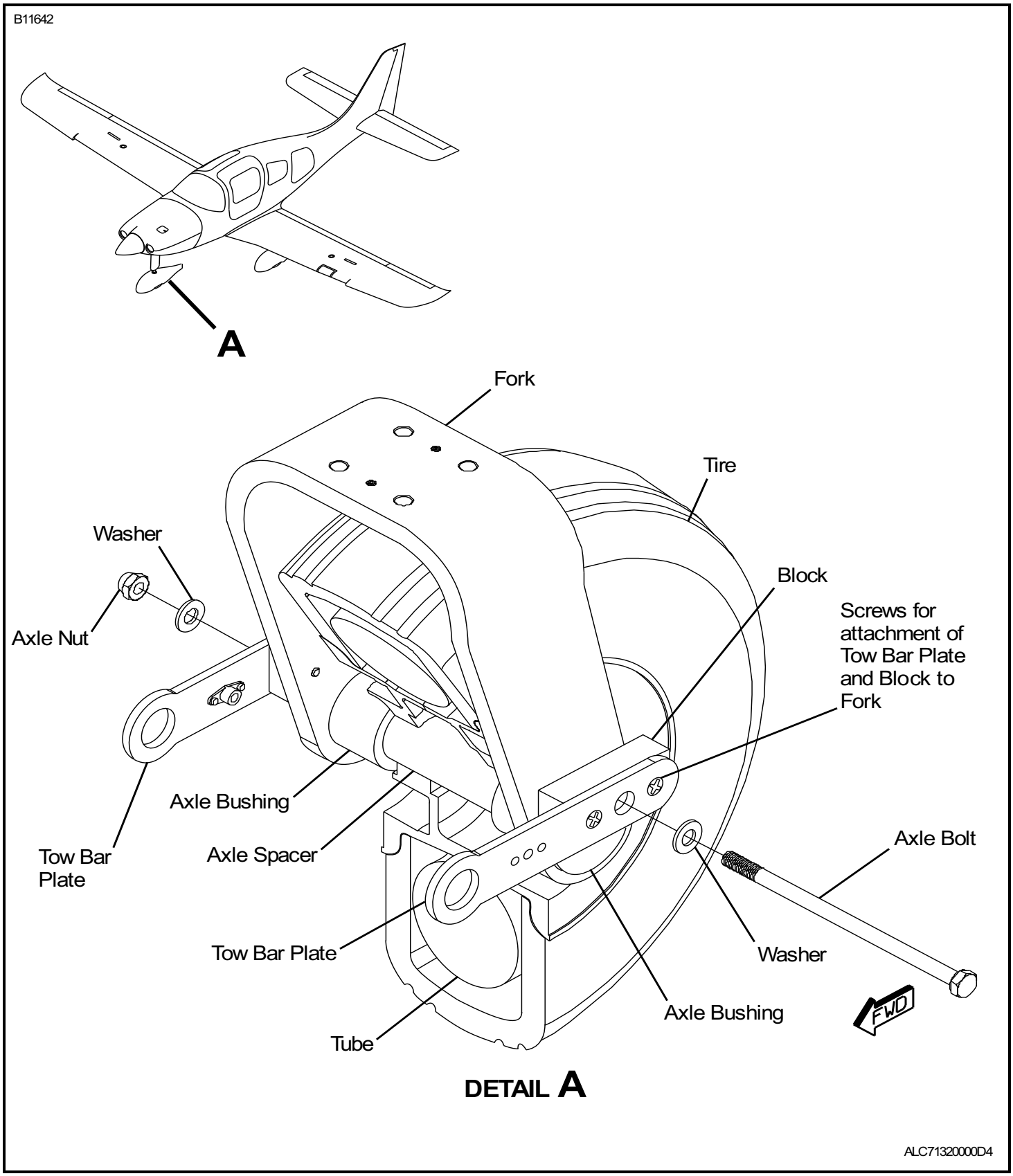
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Figure 202 : Sheet 1 : Wheel Pant Attachment Hardware



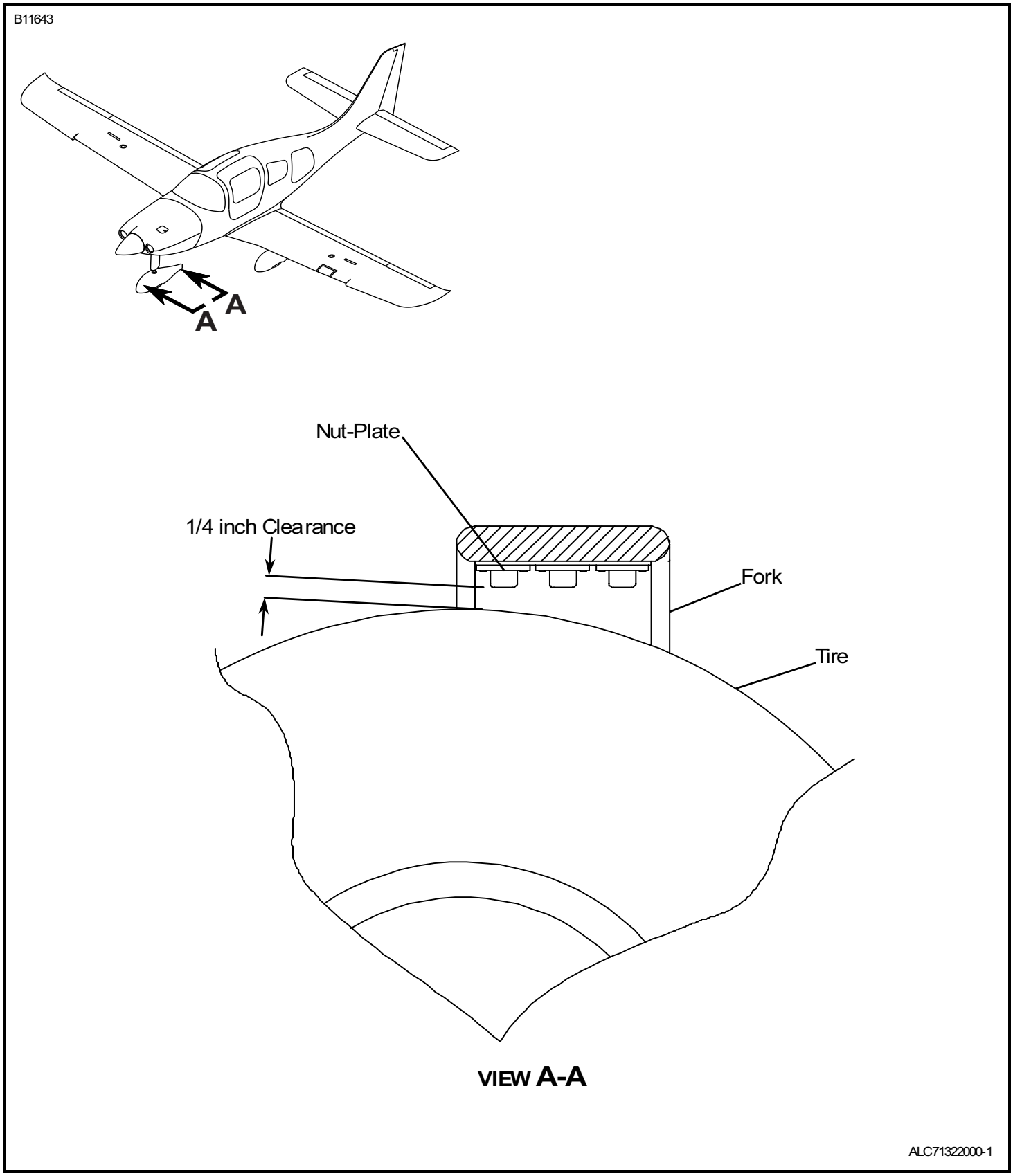
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Figure 203 : Sheet 1 : Nose Gear Wheel and Fork Assembly



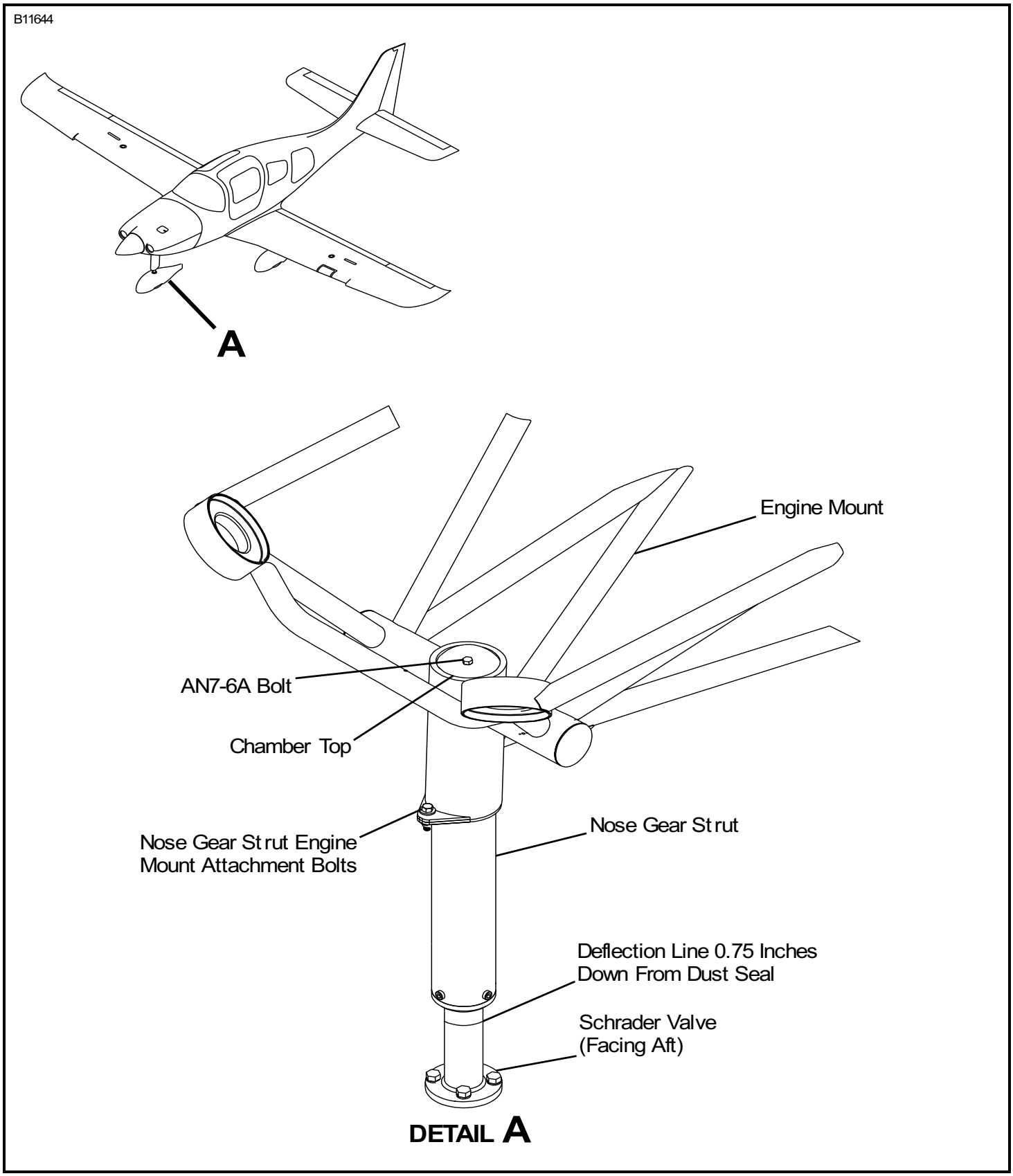
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Figure 204 : Sheet 1 : Minimum Tire Clearance



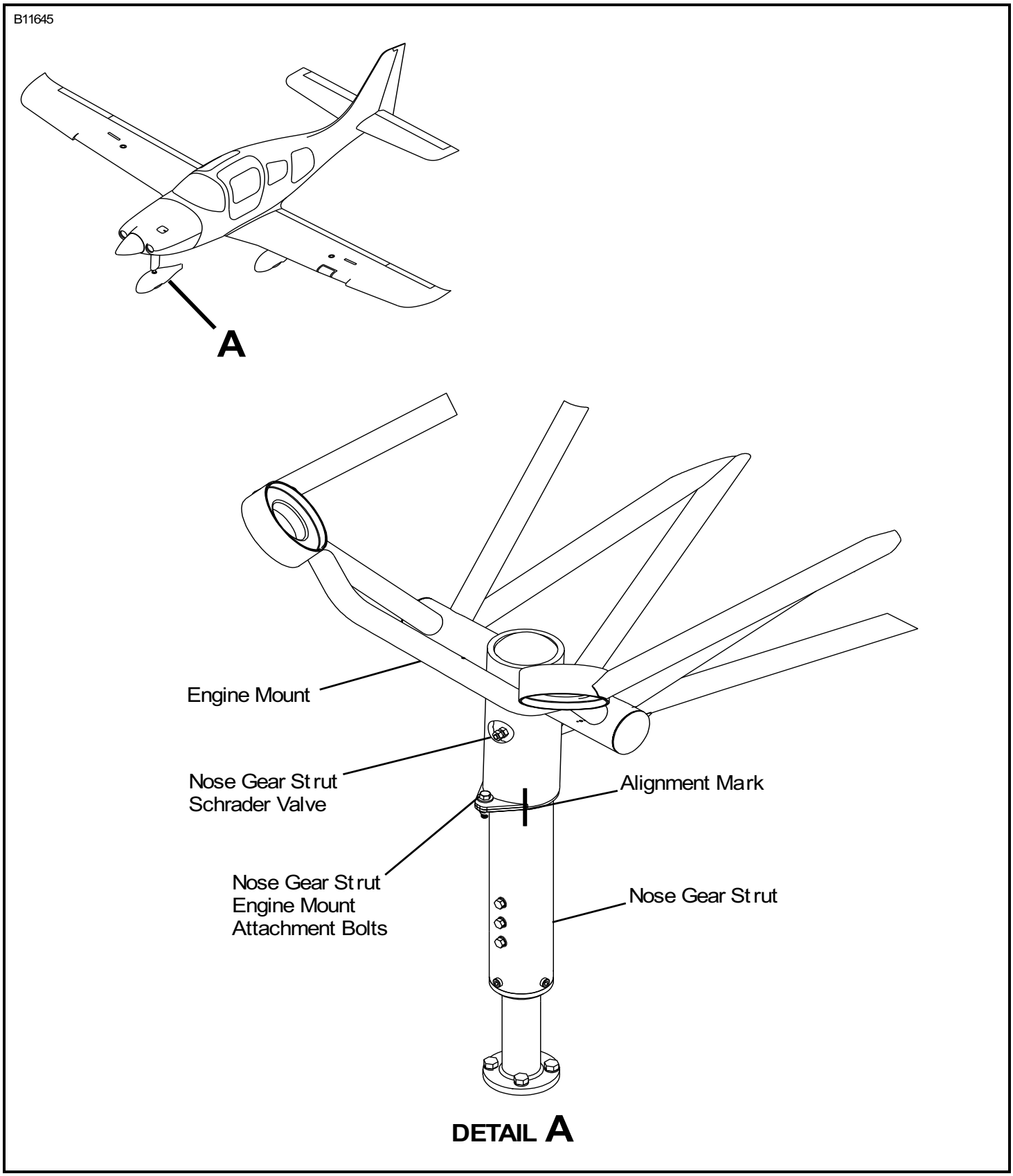
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Figure 205 : Sheet 1 : Cessna Nose Strut Assembly



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Figure 206 : Sheet 1 : ESCO Nose Strut Assembly



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WHEELS, TIRES AND BRAKE SYSTEM - MAINTENANCE PRACTICES

1. Wheels, Tires, and Brakes - General

A. The Cessna 350 (LC42-550FG) uses a 5.00-5 nose wheel and a 6.00-6 main wheel. The nose and main wheels are both tube type wheels. The nose wheel uses a 5.00-5 10-ply tire and the main wheel uses a 15x6.00-6 6-ply tire. Wheels and tires are certified under the requirements of TSOC26c, and all replacement parts should be in accordance with the TSO.

2. Wheels, Tires, and Brakes

A. All service and maintenance practices for the main wheels are in accordance with Cleveland Wheels and Brakes Maintenance Manual AWBCMM0001. The manual includes the disassembly and reassembly of the wheel and brake disk, the installation of tires and tubes, and the service and repair of wheel bearings and bearing races. Service and maintenance of the brake calipers and brake master cylinders is also covered in the Cleveland maintenance manual.

3. Brake System

A. The Cessna 350 has dual brakes located on the main gear. The brakes are hydraulically actuated and used for steering (differential braking) as well as stopping.

(1) **Removal and Installation of the Brake Calipers**

(a) Gain access to the brake calipers by removing the main gear wheel pant access panel.

(b) Remove the four bolts from the brake caliper that hold on the brake pads. Pads will drop away from behind the wheel disc.

(c) Slide the brake caliper straight out from the torque plate.

CAUTION: Depressing the brake pedals at this point will push the brake caliper pistons out of the calipers and release brake fluid.

(d) Disconnect the brake line from the brake caliper. Once the brake line is removed, use a plug or cap to seal the end of the brake line to minimize the amount of brake fluid that will drain from the line.

(e) Reinstall the brake calipers in the reverse order. Torque the brake pad attachment bolts per Cleveland Maintenance Manual AWBCMM0001.

(f) Service the brake system per the Servicing and Maintenance of the Brake System procedures section.

(2) **Removal and Installation of the Brake Master Cylinders** - The brake master cylinders are located in the cabin. There are four master cylinders, one at each rudder pedal. The brakes are actuated by depressing the rudder pedals, which in turn depresses the brake master cylinders. Use the following procedure for removing and installing the master cylinders.

(a) Remove the main gear wheel pant access panel to gain access to the brake caliper.

(b) Place a small pan or bucket under the brake caliper, and remove the lower bleeder valve.

(c) Depress the rudder pedals several times to facilitate draining of the brake system.

(d) Once the brake system is drained of hydraulic fluid, disconnect the brake lines from the brake master cylinder.

NOTE: Place some rags or a small pan underneath the master cylinder and rudder pedal to catch any excess hydraulic fluid that may still be present in the line.

(e) Remove the cap plug, cotter pins, nuts, washers, and bolts attaching the master cylinder to the rudder pedal and the rudder crossover tube (refer to Figure 201 and Figure 202).

(f) Install the brake master cylinder in the reverse order. Check all fittings and attachments for security. Fill a new MS-3 cap plug half full with Wacker SWS-951 sealant, and install over the nut with the cotter pin.

(g) After the master cylinder is reinstalled, refill the brake system with hydraulic fluid, MIL-H-5606, per the instructions in the Servicing and Maintenance of the Brake System section.

(3) **Servicing and Maintenance of the Brake System** – The brake system is a self-contained system and under normal operating conditions should not need any service or maintenance. The system will need to be serviced if the brake calipers or other components are replaced. If the brakes begin to feel "soft", the brake system may have air in the lines and will need to be serviced. The brake system will also need to be serviced if any of the brake lines are disconnected. Servicing the brake system will require the use of a small "garden sprayer" with a modified nozzle or other similar device. The spray nozzle should be replaced with a small section of tubing that can be fitted to the brake caliper bleeder valve. Use the following procedure for bleeding and servicing the brake system.

WARNING: Only service the brake system when the brake calipers are attached to the wheel and disc

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assembly. If the brake system is pressurized with the caliper unattached to the wheel and disc, the caliper pistons will be pushed out of the caliper.

(a) Place a small bucket or pan under the drain valve located at the rear of the lower engine cowl.

(b) Gain access to the brake caliper by removing the inner wheel pant access panel.

(c) Loosen the bleeder valve located on the bottom of the brake caliper.

(d) Fill the "garden sprayer" with MIL-H-5606 hydraulic fluid, and connect the end to the bleeder valve. Pressurize the sprayer.

(e) Depress the handle to release hydraulic fluid into the brake system.

(f) Depress the brake pedal several times until all air is forced out of the system. When all the air is forced out of the system, the brakes will feel “firm”.

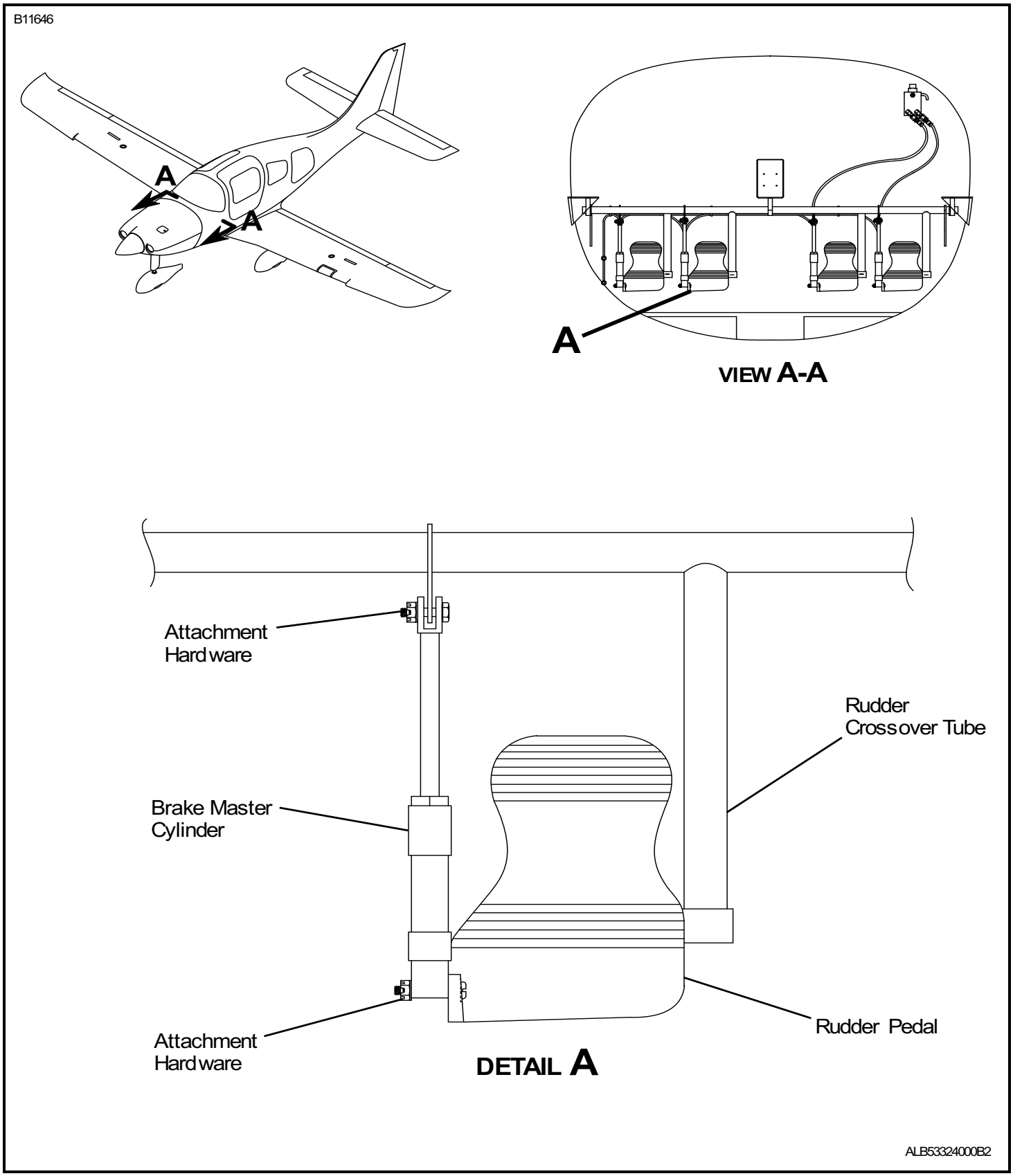
(g) Once the brake system has been serviced, tighten the bleeder valve prior to removing the "sprayer".

(h) Clean any excess brake fluid that may have spilled on or near the brake caliper, wheel, and brake disc.

(i) Reinstall the wheel pant access panel.

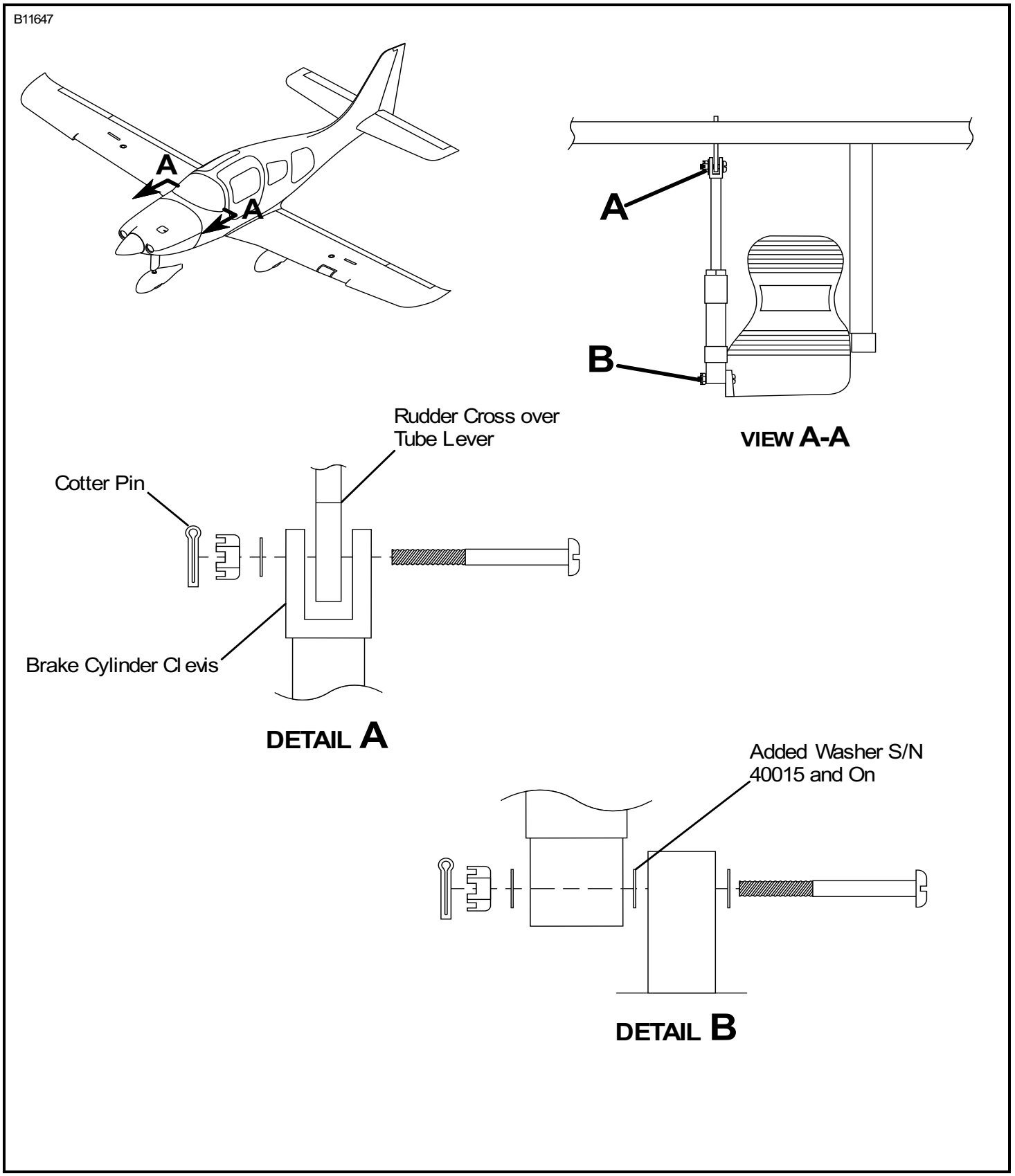
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Figure 201 : Sheet 1 : Rudder Pedal and Brake Master Cylinder Attachment



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Figure 202 : Sheet 1 : Rudder Pedal and Brake Master Cylinder Attachment Details



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PARKING BRAKE - MAINTENANCE PRACTICES

1. The Parking Brake

A. The Cessna 350 uses a parking brake to lock the brakes and prevent the aircraft from rolling. The parking brake is located on the left side of the cabin just forward of the circuit breaker panel.

(1) **Removal and Installation**

(a) Remove the parking brake interior panel per the instructions in Chapter 25-30-00.

(b) Drain the hydraulic system per the instructions in Section 32-40-00, Removal and Installation of the Brake Master Cylinders.

(c) Disconnect the brake lines from the parking brake. Place a rag under the brake lines to prevent any excess brake fluid that may still be in the lines from leaking onto the carpet.

(d) Remove the two AN526C1032R24 truss head screws and two NAS1149F0363P washers that attach the parking brake (refer to Figure 201), and slide the brake and spacer straight out.

(e) Reinstall the parking brake in the reverse order. Torque the attachment screws 30 to 36 in.-lbs.

(f) Service the brake system per the instructions in Section 32-40-00, Removal and Installation of the Brake Master Cylinders.

(2) **Service and Maintenance** - The parking brake is a sealed system and should not need any service or maintenance. The parking brake should be inspected periodically to ensure that it is still functioning properly. Use the following procedure to check if the parking brake is functioning properly.

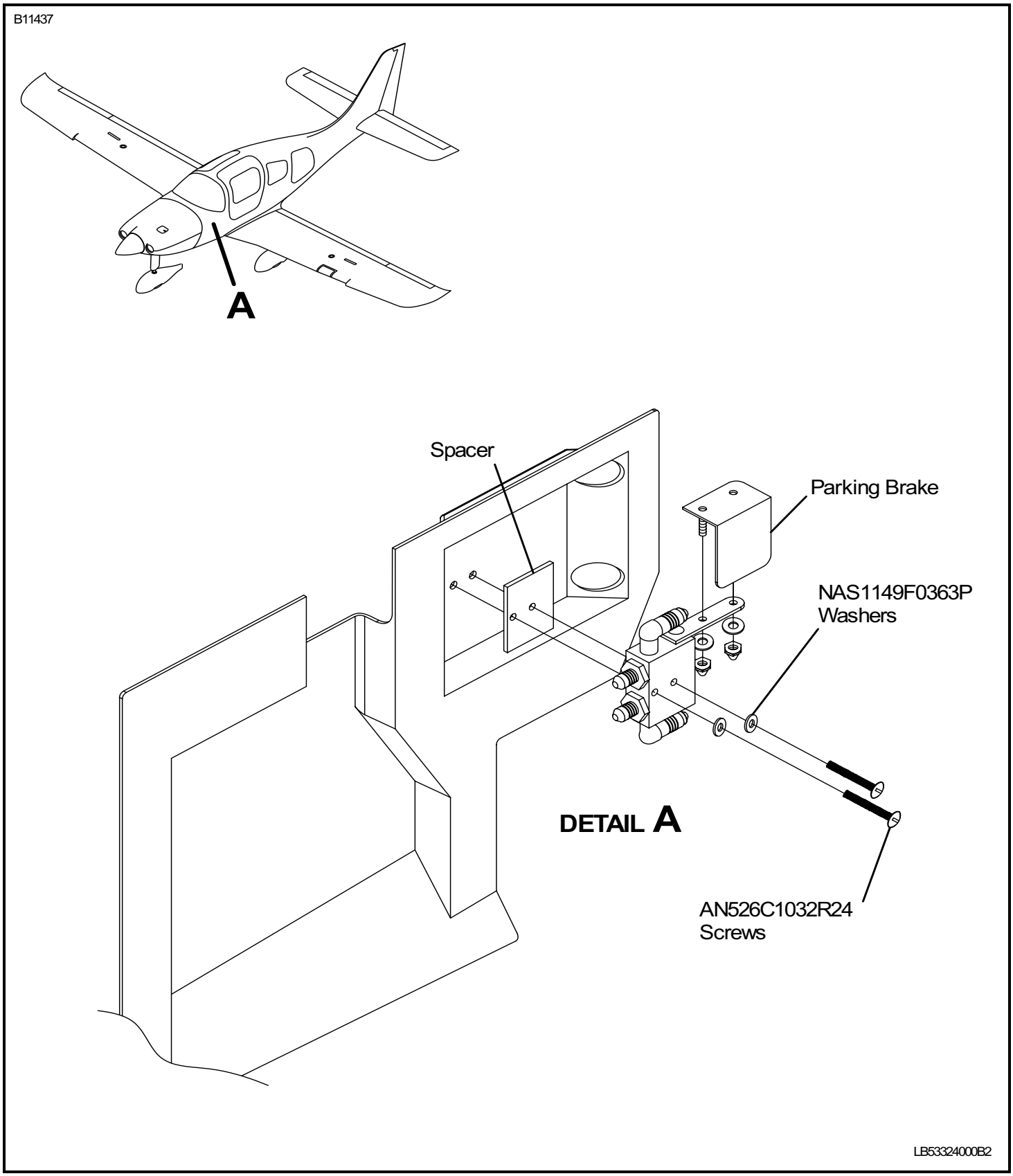
(a) Depress the rudder pedals to actuate the brakes.

(b) While pressing on the rudder pedals, pull the parking brake lever out and towards the pilot’s seat to engage the parking brake and lock the brakes.

(c) Using the tow bar, attempt to pull the airplane forward. If the airplane will not roll or the tires skid, the parking brake is functioning correctly. If the airplane does roll forward with little or no resistance the parking brake should be inspected for proper attachment of the brake lines (refer to Figure 202) and possible service of the parking brake itself. Service of the parking brake is in accordance with Cleveland Maintenance Manual AWMCMM0001.

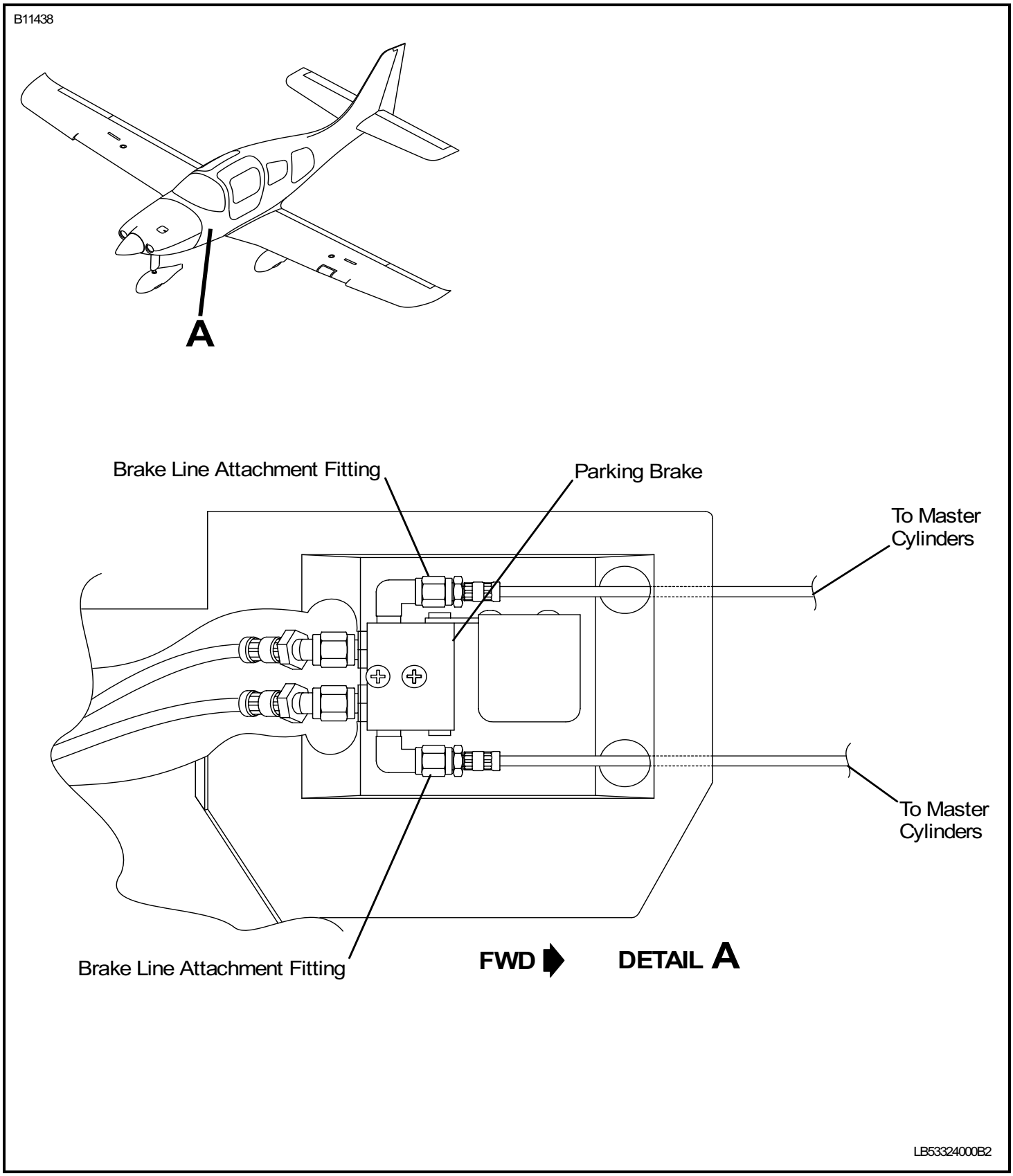
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Figure 201 : Sheet 1 : Parking Brake Installation



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Figure 202 : Sheet 1 : Parking Brake and Brake Line Attachment



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DIMENSIONS AND AREAS - GENERAL

1. Powerplant Limitations

A. Number of Engines: One (1)

B. Number of Cylinders: Six (6)

C. Engine Manufacturer: Teledyne Continental

D. Engine Model Number: IO-550-N

E. Maximum Power: 310 BHP at 2700 RPM

F. Maximum Manifold Pressure: Full power at sea level

G. Maximum Cylinder Head Temperature: 460° F

2. Powerplant Fuel And Oil Data

A. Oil Grades Recommended for Various Average Air Temperature Ranges

(1) Below 40° F: SAE 30, 10W30, 15W50, or 20W50

(2) Above 40° F: SAE 50, 15W50, 20W50, or 20W60

B. Oil Temperature

(1) Maximum Allowable: 240° F

(2) Recommended takeoff minimum: 75° F

(3) Recommended flight operations: 170° F to 200° F

C. Oil Pressure

(1) Normal Operations: 30-60 psi (pounds per square inch)

(2) Idle, minimum: 10 psi

(3) Maximum allowable (cold oil): 100 psi

D. Approved Fuel Grades

(1) 100LL Grade Aviation Fuel (Blue)

(2) 100 Grade Aviation Fuel (Green)

E. Fuel Flow and Fuel Pressure

(1) Normal Operations: 10 to 22 GPH (7 to 16 psi)

(2) Idle, minimum: 1 to 2 GPH (4 psi)

(3) Maximum allowable: 25 GPH (18 psi)

3. Propeller Data And Limitations

A. Number of Propellers: One

B. Propeller Manufacturer: Hartzell

C. Propeller Hub and Blade Model Numbers: PHC-J3YF-IRF and F7691D-14

D. Propeller Diameters

(1) Minimum: 76 inches

(2) Maximum: 77 inches

E. Propeller Blade Angle at 30-Inch Station

(1) Low: 14.1±0.2 degrees

(2) High: 34.7±1 degrees

4. Powerplant Instrument Markings

A. Table 1 shows applicable color-coded ranges for the various powerplant instruments within the aircraft.

Table 1. Powerplant Instrument Markings

INSTRUMENT	RED LINE Minimum Limit	GREEN ARC Normal Operating	RED LINEMaximum Limit

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Tachometer	Minimum for idle600 RPM (No Placard)	2000 - 2700 RPM	2700 RPM
Manifold Pressure	N/A	18-28 Inches Hg.(No Placard)	
Oil Temperature	Minimum for takeoff 75° F(No Placard)	160 - 210° F	240° F
Oil Pressure	Minimum for idle 10 psi(No Placard)	30 - 60 psi	100 psi (Cold Oil)(No Placard)
Fuel Quantity	"E" or "zero" reading in level flight indicates the remaining four gallons (S/N 42001 to 42567), or two gallons (S/N 42568 and on), in each tank cannot be used safely in flight.	N/A	N/A
Fuel Pressure/Fuel Flow	N/A	10 - 22 GPH4 - 14 psi(No Placard)	25 GPH (18 psi)
Cylinder Head Temperature	N/A	240 - 460° F	460° F

5. Power Setting Limitations

A. Do not exceed 20 in. of Hg manifold pressure below 2200 RPM. This requirement is not an engine limitation, but rather a harmonic condition inherent in the Cessna 350 (LC42-550FG).

6. Weight Limits (Utility Category)

A. Maximum Ramp Weight: 3400 lbs.

B. Maximum Zero Fuel Weight: FS (103) 2725 lbs. - FS (110) 3228 lbs.

C. Maximum Empty Weight: 2708 lbs.

D. Maximum Takeoff Weight: 3400 lbs.

E. Minimum Flying Weight: FS (103) 2240 lbs. - FS (110) 2500 lbs.

F. Maximum Landing Weight: 3230 lbs.

G. Maximum Baggage Weight: 120 lbs.

7. Center Of Gravity Limits

A. Table 2 specifies the center of gravity limits for utility category operations. The variation along the arm between the forward and aft datum points is linear (straight line). The straight-line variation means that at any given point along the arm, an increase in moments (expressed as inch-pounds) changes directly according to the variations in weight and distance from the datum.

Table 2. Center of Gravity Limits

CATEGORY	FORWARD DATUM POINT AND WEIGHT	AFT DATUM POINT AND WEIGHT	VARIATION
Utility Category	103 inches 2240 to 2500 lbs.	110 inches 2500 to 3400 lbs.	Straight Line

Reference Datum: The reference datum is located at the tip of the propeller spinner. This location causes all arm distances and moments (the product of arm and weight) to be positive values.

8. Cabin And Entry Dimensions

A. Maximum Cabin Width: 50 inches (Outside Dimension)/49 inches (Inside Dimension)

B. Maximum Cabin Length: (Firewall to aft limit of baggage compartment): 139.6 inches

C. Maximum Cabin Height: 47 inches

D. Minimum Entry Width: 33 inches

E. Minimum Entry Height: 28 inches/Clearance with Door Open: 46 inches

9. Space And Entry Dimensions Of Baggage Compartment

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A. Maximum Baggage Compartment Width: 38.5 inches
B. Maximum Baggage Compartment Length: 52 inches
C. Maximum Baggage Compartment Height: 34.5 inches
D. Maximum Baggage Entry Width: 27 inches

10. Specific Loadings

A. Wing Loading: 24.08 lbs/sq. ft.
B. Power Loading: 11.3 lbs/hp

11. Other Specifications

A. Wing Area: 141.2 sq. ft.
B. Wing Span: 35.8 ft
C. Angle of Incidence Wing: 0 degrees
D. Angle of Incidence Horizontal Stabilizer: -2.0 degrees
E. Dihedral: 4.5 degrees
F. Aspect Ratio: 9.18
G. Mean Aerodynamic Chord: 48.275 inches
H. Fuselage Length: 25.7 ft
I. Height of Vertical Stabilizer: 9 ft
J. Total Horizontal Stabilizer Area: 36.4 sq. ft.
K. Stabilizer Span: 14.1 ft
L. Vertical Stabilizer Area: 15.68 sq. ft.
M. Total Aileron Area: 5.75 sq. ft.
N. Rudder Area:4.46 sq. ft.
O. Total Flap Area: 20.36 sq. ft.
P. Total Elevator Area: 11.48 sq. ft.

12. Landing Gear

A. Nose Strut Pressure: 250 psi (Nitrogen)
B. Landing Gear Type: Fixed (Steel)
C. Landing Gear Track Width: 89.5 inches
D. Nose Tire Size: 5.00-5 (10 Ply)
E. Nose Tire Pressure: 88 psi
F. Main Gear Tire Size: 15x6.00-6 (6 Ply)
G. Main Gear Tire Pressure: 55 psi

13. Fuselage Stations

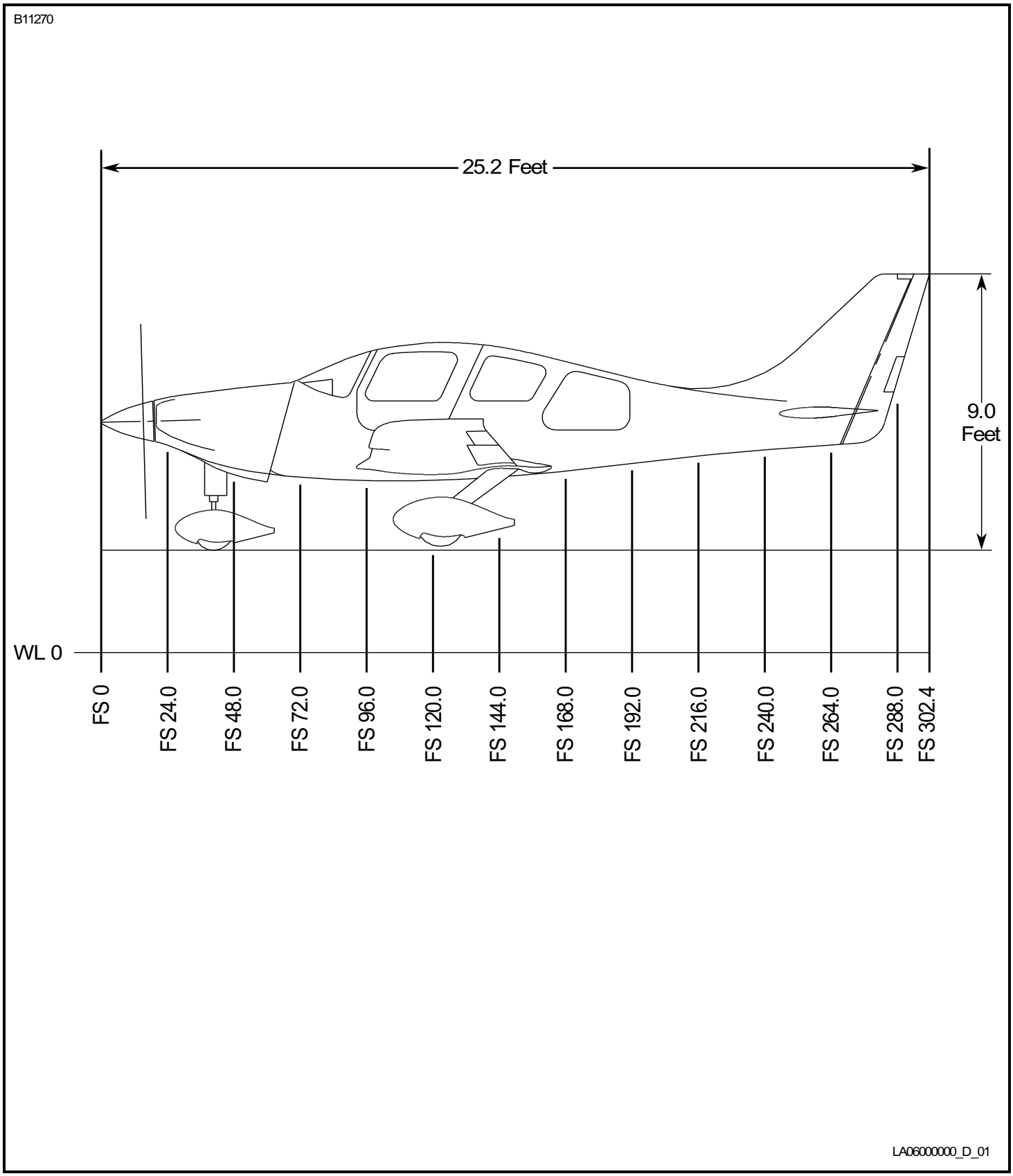
A. Refer to Figure 1 for Fuselage Stations.

14. Wing Stations

A. Refer to Figure 2 for Wing Stations.

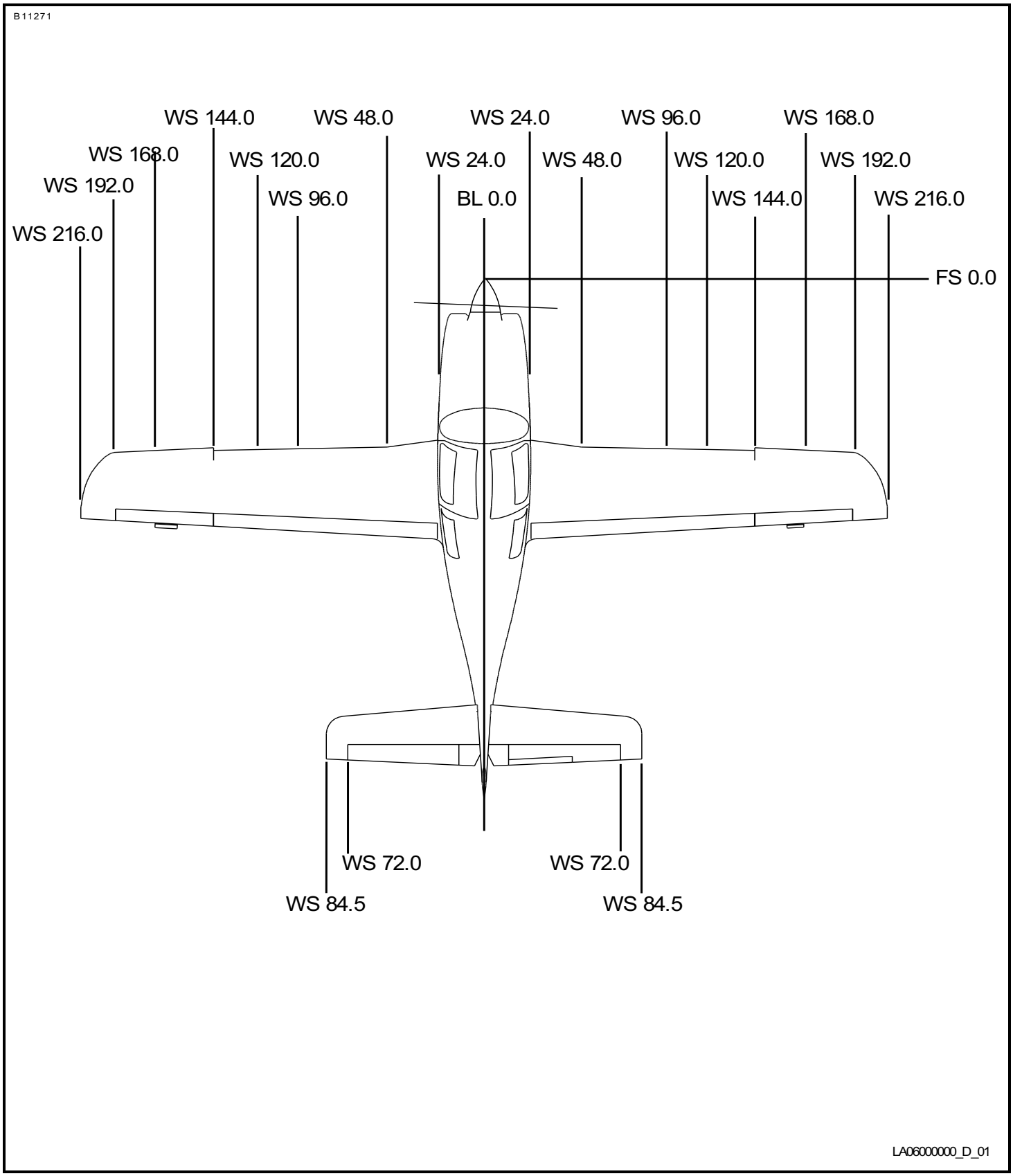
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Figure 1 : Sheet 1 : Fuselage Stations



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Figure 2 : Sheet 1 : Wing Stations



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ELECTRICAL PANELS - GENERAL

1. General

A. Because there are a limited number of electrical panels in the airplane, and the topic is interrelated with equipment and furnishings, this subject is covered in Section 25-11-00.

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VACUUM - GENERAL

1. General

A. The Cessna 350 does not have a vacuum system. All flight instruments are electrically powered.

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