

## AIRPLANE AND SYSTEMS DESCRIPTION

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## INTRODUCTION

This section provides description and operation of the airplane and its systems. Some equipment described herein is optional and may not be installed in the airplane. Refer to Section 9, Supplements, for details of other optional systems and equipment.

## AIRFRAME

The airplane is an all metal, four-place, high wing, single-engine airplane equipped with tricycle landing gear and is designed for general utility purposes.

The construction of the fuselage is a conventional formed sheet metal bulkhead, stringer, and skin design referred to as semimonocoque. Major items of structure are the front and rear carry through spars to which the wings are attached, a bulkhead and forgings for main landing gear attachment at the base of the rear door posts, and a bulkhead with attach fittings at the base of the forward door posts for the lower attachment of the wing struts. Four engine mount stringers are also attached to the forward door posts and extend forward to the firewall.

The externally braced wings, containing integral fuel tanks, are constructed of a front and rear spar with formed sheet metal ribs, doublers, and stringers. The entire structure is covered with aluminum skin. The front spars are equipped with wing-to-fuselage and wing-to-strut attach fittings. The aft spars are equipped with wing-to-fuselage attach fittings, and are partial span spars. Conventional hinged ailerons and single slot type flaps are attached to the trailing edge of the wings. The ailerons are constructed of a forward spar containing balance weights, formed sheet metal ribs and V type corrugated aluminum skin joined together at the trailing edge. The flaps are constructed basically the same as the ailerons, with the exception of the balance weights and the addition of a formed sheet metal leading edge section.

The empennage (tail assembly) consists of a conventional vertical stabilizer, rudder, horizontal stabilizer, and elevator. The vertical stabilizer consists of a forward and aft spar, formed sheet metal ribs and reinforcements, four skin panels, formed leading edge skins and a dorsal fin.

(Continued Next Page)

## **AIRFRAME** (Continued)

The rudder is constructed of a forward and aft spar, formed sheet metal ribs and reinforcements, and a wrap-around skin panel. The top of the rudder incorporates a leading edge extension which contains a balance weight.

The horizontal stabilizer is constructed of a forward and aft spar, ribs and stiffeners, center upper and lower skin panels and two, left and two right wrap-around skin panels which also form the leading edges. The horizontal stabilizer also contains the elevator trim tab actuator.

Construction of the elevator consists of formed leading edge skins, a forward spar, ribs, torque tube and bellcrank, left upper and lower V type corrugated skins, and right upper and lower V type corrugated skins incorporating a trailing edge cut-out for the trim tab. Both elevator tip leading edge extensions incorporate balance weights. The elevator trim tab consists of a spar, rib, and upper and lower V type corrugated skins.

## **FLIGHT CONTROLS**

The airplane's flight control system (Refer to Figure 7-1) consists of conventional aileron, rudder, and elevator control surfaces. The control surfaces are manually operated through mechanical linkage using a control wheel for the ailerons and elevator, and rudder/brake pedals for the rudder. The elevator control system is equipped with downsprings which provide improved stability in flight.

(Continued Next Page)

## **FLIGHT CONTROLS** (Continued)

### **TRIM SYSTEMS**

A manually operated rudder and elevator trim is provided (Refer to Figure 7-1). The rudder is trimmed through a bungee connected to the rudder control system and a trim control wheel mounted on the control pedestal. This is accomplished by rotating the horizontally mounted trim control wheel either left or right to the desired trim position. Rotating the trim wheel to the right will trim nose-right; conversely, rotating it to the left will trim nose-left. The elevator is trimmed through the elevator trim tab by utilizing the vertically mounted trim control wheel. Forward rotation of the trim wheel will trim nose-down, conversely, aft rotation will trim nose-up.

### **MANUAL ELECTRIC TRIM SYSTEM**

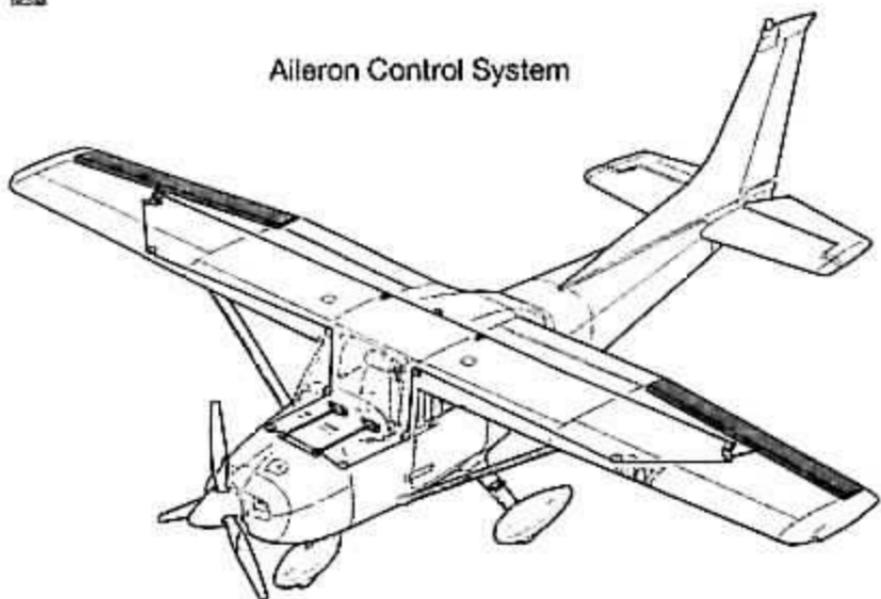
Refer to the Garmin G1000 Cockpit Reference Guide (CRG) for more information on system operation.

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## FLIGHT CONTROLS AND TRIM SYSTEM

1823M

Aileron Control System



Rudder and Rudder Trim  
Control Systems

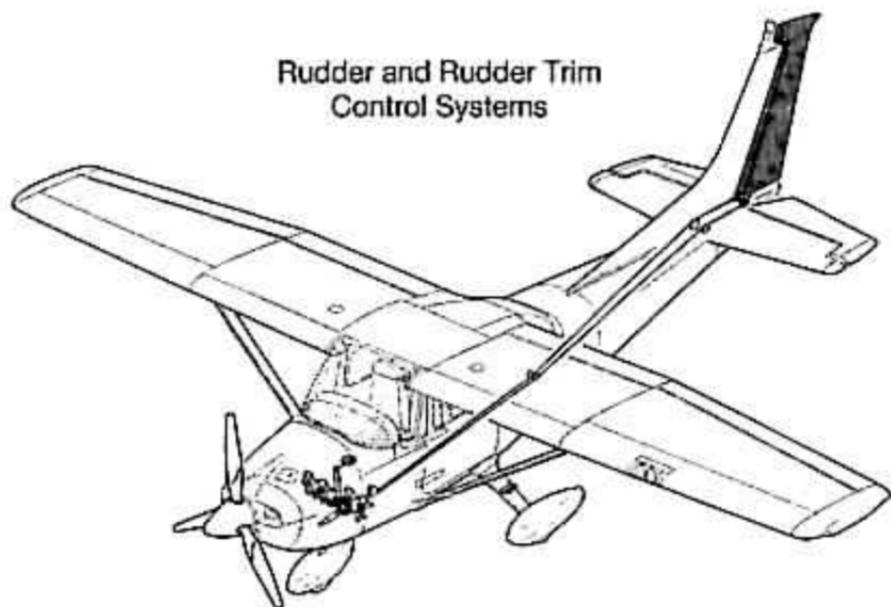
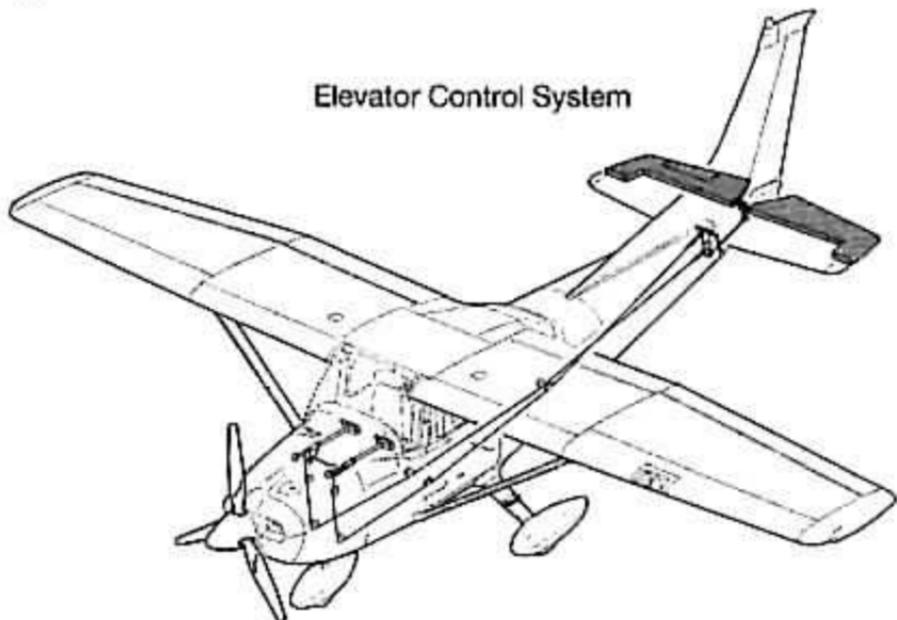


Figure 7-1 (Sheet 1 of 2)

## FLIGHT CONTROLS AND TRIM SYSTEMS

55399

Elevator Control System



Elevator Trim Control System

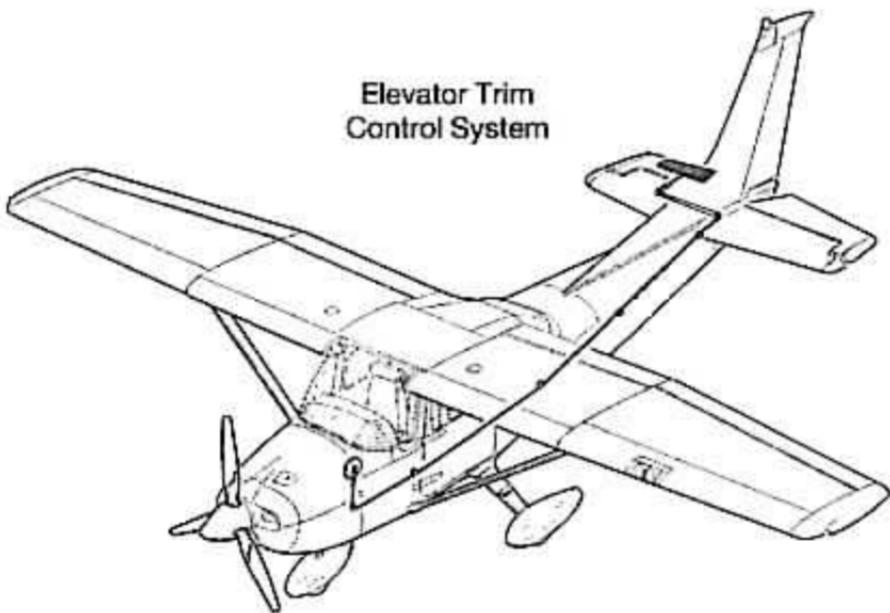


Figure 7-1 (Sheet 2)

## **INSTRUMENT PANEL**

The instrument panel (Refer to Figure 7-2) is of all metal construction and is installed in sections so equipment can be easily removed for maintenance. The glareshield, above and projecting aft from the instrument panel, limits undesirable reflections on the windshield from lighted equipment and displays mounted in the instrument panel.

The Nav III instrument panel contains the Garmin Display Unit (GDU) Primary Flight Display (PFD) and Multifunction Display (MFD) and the Garmin Audio Panel. For specific details regarding the instruments, switches, circuit breakers and controls on the instrument panel, refer to the related topics in this section.

## **PILOT PANEL LAYOUT**

The PFD, centered on the instrument panel in front of the pilot, shows the primary flight instruments during normal operation. During engine start, reversionary operation (MFD failure) or when the DISPLAY BACKUP switch is selected, the Engine Indication System (EIS) is shown on the PFD. Refer to the Garmin G1000 CRG for specific operating information.

The Standby Battery (STBY BATT) switch is found at the upper left corner of the pilot instrument panel on an internally lighted subpanel. The switch positions (ARM/OFF/TEST) select the standby battery operating modes. The rocker-type MASTER and AVIONICS switches are found immediately below the standby battery switch.

The controls for adjusting instrument panel, equipment, and pedestal lighting are found together on the subpanel below the MASTER and AVIONICS switches. See the INTERNAL LIGHTING paragraphs of this section for more information.

(Continued Next Page)

## **INSTRUMENT PANEL (Continued)**

### **PILOT PANEL LAYOUT (Continued)**

Switches for the airplane electrical systems and equipment are found on an internally lighted subpanel found below the lower left corner of the PFD. Each switch is labeled for function and is ON when the handle is in the up position. See the ELECTRICAL EQUIPMENT descriptions in this section for further information.

The circuit breaker panel is found along the lower edge of the pilot's instrument panel below the electrical equipment switch panel and pilot control wheel column. Each circuit breaker is identified for the equipment or function it controls and for the bus from which it receives power. Lighting for this subpanel is controlled using the SW/CB PANELS dimmer control. See the ELECTRICAL EQUIPMENT descriptions in this section for further information.

### **CENTER PANEL LAYOUT**

The Garmin audio panel is found on the upper half of the center instrument panel, immediately to the right of the PFD. A pushbutton switch labeled DISPLAY BACKUP to manually select display reversion mode is found on the lower face of the audio panel. Refer to the Garmin G1000 CRG for operating information.

The MFD is found on the upper center panel to the right of the audio panel. The MFD depicts EIS information along the left side of the display and shows navigation, terrain, lightning and traffic data on the moving map. Flight management or display configuration information can be shown on the MFD in place of the moving map pages. Refer to the Garmin G1000 CRG for operating information.

(Continued Next Page)

## **INSTRUMENT PANEL** (Continued)

### **CENTER PANEL LAYOUT** (Continued)

The standby instrument cluster is in the center instrument panel below the audio panel. A conventional (mechanical) airspeed indicator and a sensitive aneroid altimeter are on each side of the vacuum-powered attitude indicator. The pitot-static instruments share the airplane pitot head and static ports with the air data computer. The attitude indicator features a low vacuum flag to provide immediate warning of vacuum system failure.

The engine controls are found on the lower center instrument panel below the standby instrument cluster. The controls are conventional push-pull-type controls and include throttle, prop RPM and mixture. See **ENGINE** description in this section for operating information.

The alternate static air valve is found adjacent to the throttle control. Refer to the **PITOT-STATIC SYSTEM AND INSTRUMENTS** description in this section for operating information.

The wing flap control lever and indicator are found at the lower right side of the center panel. Refer to the **WING FLAP SYSTEM** description in this section for operating information.

(Continued Next Page)

## **INSTRUMENT PANEL** (Continued)

### **RIGHT PANEL LAYOUT**

The Emergency Locator Transmitter (ELT) remote switch (ON/ARM/TEST RESET) is positioned at the upper inboard corner of the right panel adjacent to the MFD. Refer to Section 9, Supplements, for ELT operating information.

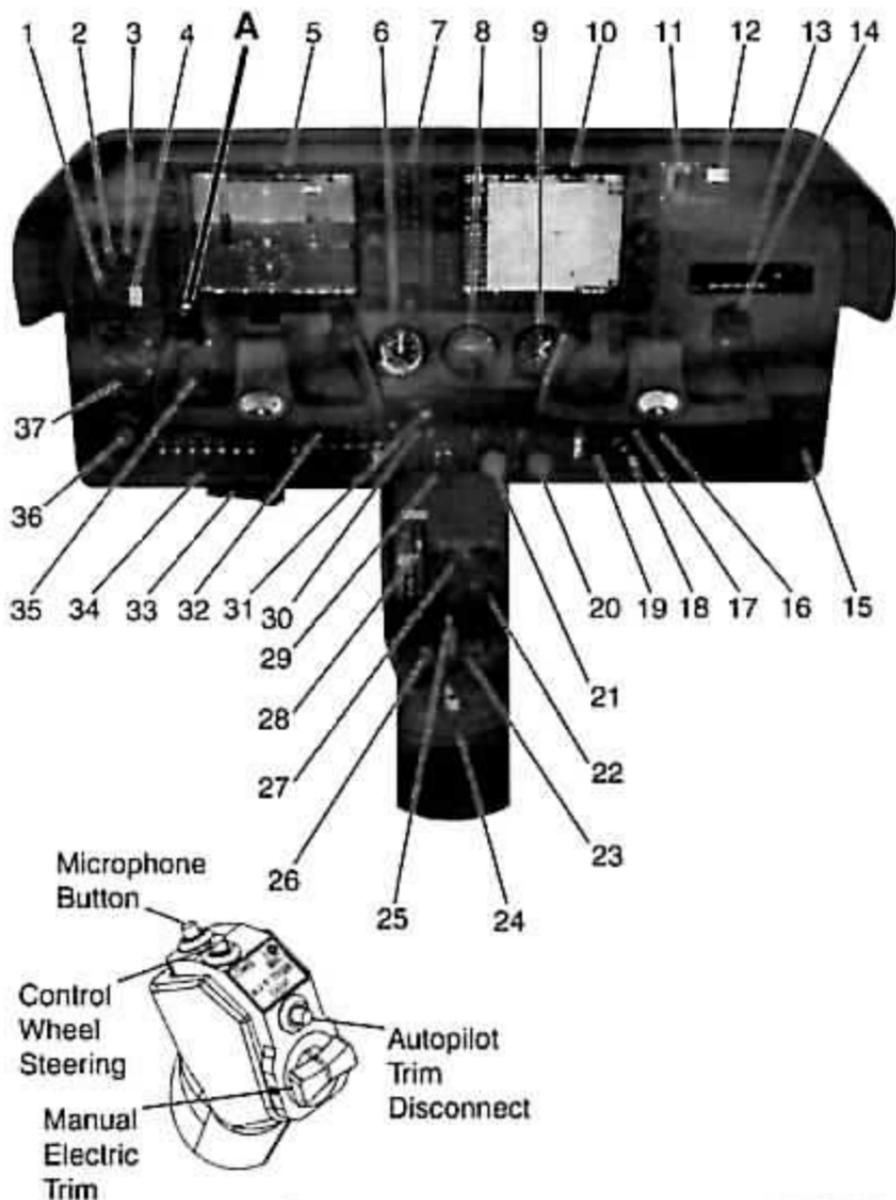
The Hour (Hobbs) meter is found to the right of the ELT switch and records engine operating time, when oil pressure is greater than 20 PSI, for maintenance purposes. Refer to the ENGINE INSTRUMENTS description in this section for further information.

### **CENTER PEDESTAL LAYOUT**

The center pedestal, located below the center panel, contains the elevator and rudder trim control wheels, trim position indicators, cowl flap control lever, 12V power outlet, aux audio input jack and a bracket for the microphone. The fuel selector valve handle is located at the base of the pedestal.

## INSTRUMENT PANEL

E6397



DETAIL A

0719P105B

Figure 7-2

## INSTRUMENT PANEL

1. MASTER Switch (ALT and BAT)
2. STBY BATT Switch
3. STBY BATT Test Annunciator
4. AVIONICS Switch (BUS 1 and BUS 2)
5. Primary Flight Display
6. Standby Airspeed Indicator
7. Audio Control Panel
8. Standby Attitude Indicator
9. Standby Altimeter
10. Multifunction Display
11. ELT Remote Switch/Annunciator
12. Flight Hour Recorder
13. Bendix/King KR87 Automatic Direction Finder (if installed)
14. Microphone Button
15. Glove Box
16. Defroster Control
17. Cabin Heat Control
18. Cabin Air Control
19. Wing Flap Control Lever And Position Indicator
20. Mixture Control Knob
21. Propeller Control Knob
22. Cowl Flap Control Lever
23. Aux Audio Input Jack
24. Fuel Selector Valve
25. Handheld Microphone
26. 12V/10A Power Outlet
27. Rudder Trim Control Wheel And Position Indicator
28. Elevator Trim Control Wheel And Position Indicator
29. Throttle Control Knob (With Friction Lock)
30. Go-Around Button
31. ALT Static Air Valve Control
32. Yoke Mounted Map Light
33. Parking Brake Handle
34. Circuit Breaker Panel
35. Electrical Switch Panel
36. MAGNETOS/START Switch
37. DIMMING Panel

## FLIGHT INSTRUMENTS

The G1000 Integrated Cockpit System primary flight instrument indications are shown on the PFD. The primary flight instruments are arranged on the PFD in the basic T configuration. The Attitude Indicator (AI) and Horizontal Situation Indicator (HSI) are centered vertically on the PFD and are conventional in appearance and operation. Vertical tape-style (scrolling scale) indicators with fixed pointers and digital displays, show airspeed, altitude, and vertical speed. The vertical indicators take the place of analog indicators with a fixed circular scale and rotating pointer.

Knobs, knob sets (two knobs on a common shaft) and membrane type push button switches, found on the bezel surrounding each GDU display, control COM, NAV, XPDR, AUTOPILOT, and GPS avionics, set BARO (barometric pressure), CRS (course), and HDG (heading), and work various flight management functions. Some push button switches are dedicated to certain functions (keys) while other switches have functions defined by software (softkeys). A softkey may perform various operations or functions at various times based on software definition. Softkeys are found along the lower bezel of the GDU displays.

(Continued Next Page)

## FLIGHT INSTRUMENTS (Continued)

### ATTITUDE INDICATOR

The G1000 attitude indicator is shown on the upper center of the PFD. The attitude indication data is provided by the Attitude and Heading Reference System (AHRS). The G1000 attitude indicator provides a horizon line that is the full width of the GDU display.

The roll index scale is conventional with 10° graduations to 30° and then 15° graduations to 60° of roll. The roll pointer is slaved to the airplane symbol. The pitch index scale is graduated in 5° increments with every 10° of pitch labeled. If pitch limits are exceeded in either the nose-up or nose-down direction, red warning chevrons will appear on the indicator to point the way back to level flight. A small white trapezoid located below the roll pointer moves laterally left and right to provide the slip-skid information previously supplied by the skid indicator ball. The trapezoid should be centered below the roll pointer for coordinated turns. The standby (vacuum) attitude indicator is found on the lower center instrument panel.

(Continued Next Page)

## **FLIGHT INSTRUMENTS** (Continued)

### **AIRSPEED INDICATOR**

The G1000 vertical tape airspeed indicator is shown along the upper left side of the PFD. The airspeed indication data is provided by the air data computer unit. Colored bands are provided to indicate the maximum speed, high cruise speed caution range, normal operating range, full wing flap operating range and low airspeed awareness band. Calculated true airspeed is displayed in a window at the bottom edge of the airspeed tape.

The standby (pneumatic) airspeed indicator is found on the lower center instrument panel. Colored arcs are provided to indicate the maximum speed, high cruise speed caution range, normal operating range, full wing flap operating range and low airspeed awareness band.

### **ALTIMETER**

The primary altitude indicator (altimeter) is found along the right side of the altitude indicator on the PFD. The altitude indication data is provided by the air data computer unit. The local barometric pressure is set using the BARO knob on the GDU displays.

A cyan selectable altitude reference pointer, bug, is displayed on the altimeter tape and is set using the ALT SEL knob on the GDU displays. The altitude bug set-point is shown in a window at the top edge of the altimeter.

The standby (aneroid) sensitive altimeter is found on the lower center instrument panel.

(Continued Next Page)

## FLIGHT INSTRUMENTS (Continued)

### HORIZONTAL SITUATION INDICATOR

The Horizontal Situation Indicator (HSI) is found along the lower center area of the PFD. The heading indication data is provided by the AHRS and magnetometer units. The HSI combines a stabilized magnetic direction indicator (compass card) with selectable navigation deviation indicators for GPS or VHF navigation. The HSI is conventional in appearance and operation.

Magnetic heading is shown numerically in a window centered above the heading index (lubber line) at the top of the HSI. Reference index marks are provided at 45° intervals around the compass card. A circular segment scale below the heading window at the top of the HSI shows half and standard rates of turn based on the length of the magenta turn vector.

The cyan HSI heading reference pointer, bug, is set using the HDG knob on the GDU display. The selected heading is shown digitally in a window above the upper left 45° index mark. The selected heading will provide control input to the autopilot when engaged in HDG mode.

The CDI navigation source shown on the HSI is set using the CDI softkey to select from GPS, NAV 1 or NAV 2 inputs. The course reference pointer is set using the CRS knob on the GDU display. The selected course is shown digitally in a window above the upper right 45° index mark. The selected navigation source will provide control input to the autopilot when engaged in NAV, APR or BC mode and it is receiving a navigation signal from the selected GPS or VHF NAV radios.

(Continued Next Page)

## **FLIGHT INSTRUMENTS** (Continued)

### **HORIZONTAL SITUATION INDICATOR** (Continued)

#### **WARNING**

WHEN THE AUTOPILOT IS ENGAGED IN NAV, APR OR BC OPERATING MODES, IF THE HSI NAVIGATION SOURCE IS CHANGED MANUALLY, USING THE CDI SOFTKEY, THE CHANGE WILL INTERRUPT THE NAVIGATION SIGNAL TO THE AUTOPILOT AND WILL CAUSE THE AUTOPILOT TO REVERT TO ROL MODE OPERATION. NO AURAL ALERT WILL BE PROVIDED. IN ROL MODE, THE AUTOPILOT WILL ONLY KEEP THE WINGS LEVEL AND WILL NOT CORRECT THE AIRPLANE HEADING OR COURSE. SET THE HDG BUG TO THE CORRECT HEADING AND SELECT THE CORRECT NAVIGATION SOURCE ON THE HSI, USING THE CDI SOFTKEY, BEFORE ENGAGING THE AUTOPILOT IN ANY OTHER OPERATING MODE.

### **VERTICAL SPEED INDICATOR**

The Vertical Speed Indicator (VSI) tape is found on the right side of the altimeter display along the upper right side of the PFD. The vertical speed pointer moves up and down the fixed VSI scale and shows the rate of climb or descent in digits inside the pointer. The VSI tape has a notch on the right edge at the 0 feet/min index for reference. Rate of descent is shown with a negative sign in front of the digits. Vertical speed must exceed 100 feet/min in climb or descent before digits will appear in the VSI pointer.

## GROUND CONTROL

Effective ground control while taxiing is accomplished through nosewheel steering by using the rudder pedals; left rudder pedal to steer left and right rudder pedal to steer right. When a rudder pedal is depressed, a spring loaded steering bungee, which is connected to the nose gear and to the rudder bars, will turn the nosewheel through an arc of approximately  $11^{\circ}$  each side of center. By applying either left or right brake, the degree of turn may be increased up to  $29^{\circ}$  each side of center.

Moving the airplane by hand is most easily accomplished by attaching a towbar to the nose gear strut. If a towbar is not available, or pushing is required, use the wing struts as push points. Do not use the vertical or horizontal surfaces to move the airplane. If the airplane is to be towed by vehicle, never turn the nosewheel more than  $29^{\circ}$  either side of center or structural damage to the nose gear could result.

The minimum turning radius of the airplane, using differential braking and nosewheel steering during taxi, is approximately 27 feet. To obtain a minimum radius turn during ground handling, the airplane may be rotated around either main landing gear by pressing down on a tailcone bulkhead just forward of the horizontal stabilizer to raise the nosewheel off the ground. Care should be exercised to ensure that pressure is exerted only on the bulkhead area and not on skin between the bulkheads. Pressing down on the horizontal stabilizer to raise the nosewheel off the ground is not recommended.

## WING FLAP SYSTEM

The single slot type wing flaps (Refer to Figure 7-3), are extended or retracted by positioning the wing flap control lever on the instrument panel to the desired flap deflection position. The wing flap control lever is moved up or down in a slotted panel that provides mechanical stops at the 10° and 20° positions. To change flap setting, the wing flap control lever is moved to the right to clear mechanical stops at the 10° and 20° positions. A scale and pointer to the left of the wing flap control lever indicates flap travel in degrees. The wing flap system circuit is protected by a 10-ampere circuit breaker, labeled FLAP, on the left side of the circuit breaker panel.

06A03

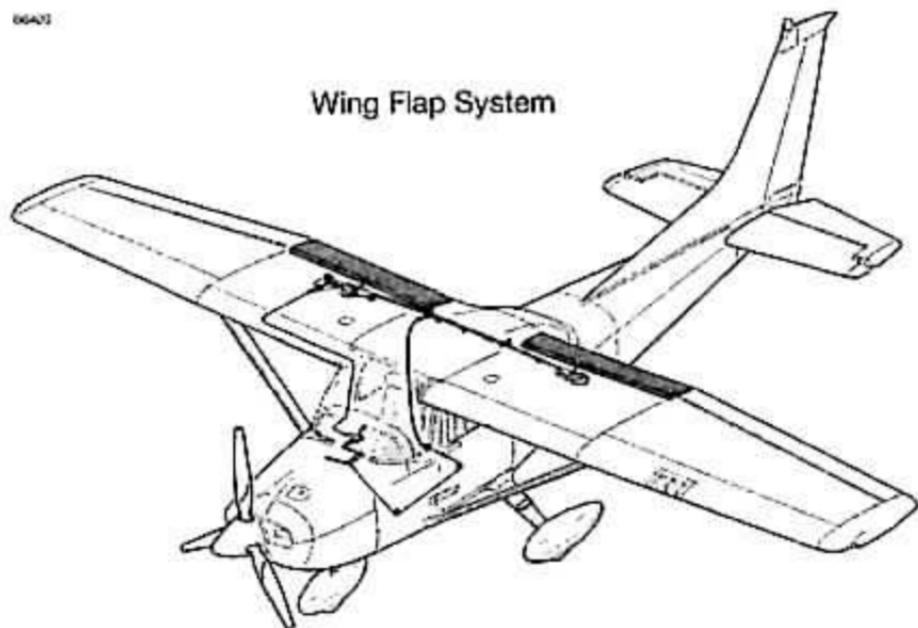


Figure 7-3

## **LANDING GEAR SYSTEM**

The landing gear is of the tricycle type, with a steerable nosewheel and two main wheels. Wheel fairings are standard equipment for both the main wheels and nosewheel. Shock absorption is provided by the tubular spring steel main landing gear struts and the air/oil nose gear shock strut. Each main gear wheel is equipped with a hydraulically-actuated disc type brake on the inboard side of each wheel.

## **BAGGAGE COMPARTMENT**

The baggage compartment consists of the area from the back of the rear passenger seats to the aft cabin bulkhead. Access to the baggage compartment is gained through a lockable baggage door on the left side of the airplane, or from within the airplane cabin. A baggage net with tiedown straps is provided for securing baggage and is attached by tying the straps to tiedown rings provided in the airplane. For baggage area and door dimensions, refer to Section 6.

## SEATS

The seating arrangement consists of two vertically adjusting crew seats for the pilot and front seat passenger, and an infinitely adjustable split aft bench seat for rear seat passengers.

Seats used for the pilot and front seat passenger are adjustable forward and aft, and up and down. Additionally, the angle of the seat back is infinitely adjustable.

Forward and aft adjustment is made using the handle located below the center of the seat frame. To position the seat, lift the handle, slide the seat into position, release the handle and check that the seat is locked in place. To adjust the height of the seat, rotate the large crank under the right corner of the seat until a comfortable height is obtained. To adjust the seat back angle, pull up on the release button, located in center front of seat, just under the seat bottom, position the seat back to the desired angle, and release the button. When the seat is not occupied, the seat back will automatically fold forward whenever the release button is pulled up.

The rear passenger seat consists of a fixed, one piece seat bottom and an infinitely adjustable split back. Seat back controls are located beneath each seat bottom and provide adjustment for each seat back. To adjust the seat back, raise the lever, position the seat back to the desired angle, release the lever, and check that the seat back is securely locked in place.

Headrests are installed on both the front and rear seats. To adjust the headrest, apply enough pressure to it to raise or lower it to the desired level.

## **INTEGRATED SEAT BELT/SHOULDER HARNESS**

All seat positions are equipped with integrated seat belts/shoulder harness assemblies (Refer to Figure 7-4). The design incorporates an overhead inertia reel for the shoulder portion, and a retractor assembly for the lap portion of the belt. This design allows for complete freedom of movement of the upper torso area while providing restraint in the lap belt area. In the event of a sudden deceleration, reels lock up to provide positive restraint for the user.

In the front seats, the inertia reels are located on the centerline of the upper cabin area. In the rear seats, the inertia reels are located outboard of each passenger in the upper cabin.

To use the integrated seat belt/shoulder harness, grasp the link with one hand, and, in a single motion, extend the assembly and insert into the buckle. Positive locking has occurred when a distinctive "snap" sound is heard.

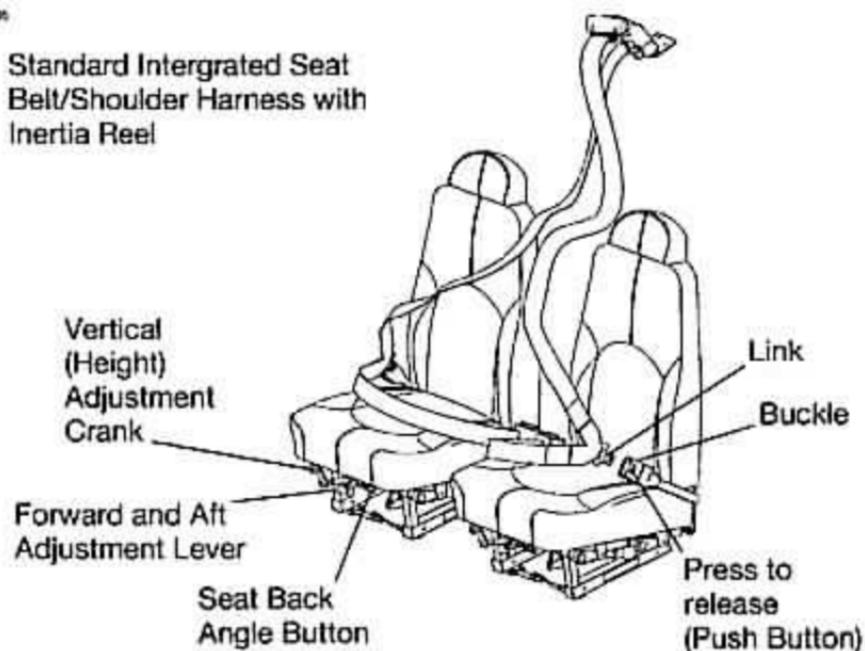
Proper locking of the lap belt can be verified by ensuring that the belts are allowed to retract into the retractors and the lap belt is snug and low on the waist as worn normally during flight. No more than one additional inch of belt should be able to be pulled out of the retractor once the lap belt is in place on the occupant. If more than one additional inch of belt can be pulled out of the retractor, the occupant is too small for the installed restraint system and the seat should not be occupied until the occupant is properly restrained.

Removal is accomplished by pressing the release button on the buckle and pulling out and up on the harness. Spring tension on the inertia reel will automatically stow the harness.

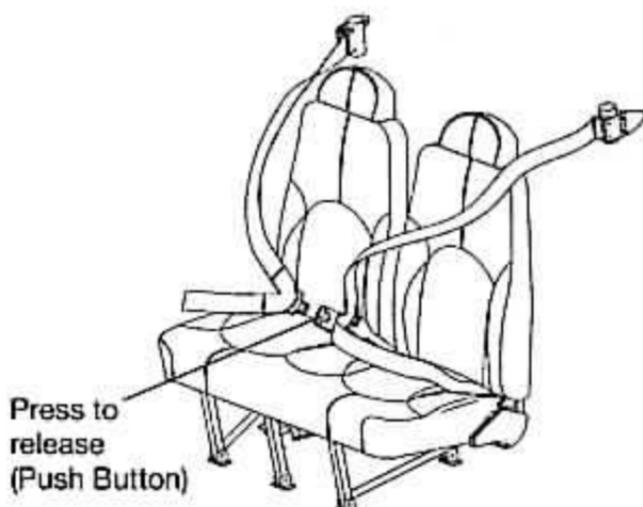
## INTEGRATED SEAT BELT/SHOULDER HARNESS

81326

Standard Intergrated Seat  
Belt/Shoulder Harness with  
Inertia Reel



Pilot and Front Passenger Seat



Aft Seat

Figure 7-4\*

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## ENTRANCE DOORS AND CABIN WINDOWS

Entry to, and exit from, the airplane is accomplished through either of two entry doors, one on each side of the cabin, at the front seat positions (Refer to Section 6 for cabin and cabin door dimensions). The doors incorporate a recessed exterior door handle, a conventional interior door handle, a key operated door lock (left door only), a door stop mechanism, and openable windows in both the left and right doors.

### NOTE

The door latch design on this model requires that the outside door handle on the pilot and front passenger doors be extended out whenever the doors are open. When closing the door, do not attempt to push the door handle in until the door is fully shut.

To open the doors from outside the airplane, utilize the recessed door handle near the aft edge of either door by grasping the forward edge of the handle and pulling outboard. To close or open the doors from inside the airplane, use the combination door handle and arm rest. The inside door handle has three positions and a placard at its base which reads OPEN, CLOSE, and LOCK. The handle is spring loaded to the CLOSE (up) position. When the door has been pulled shut and latched, lock it by rotating the door handle forward to the LOCK position (flush with the arm rest). When the handle is rotated to the LOCK position, an over center action will hold it in that position. Both cabin doors should be locked prior to flight, and should not be opened intentionally during flight.

### NOTE

Accidental opening of a cabin door in flight due to improper closing does not constitute a need to land the airplane. The best procedure is to set up the airplane in a trimmed condition at approximately 80 KIAS, momentarily shove the door outward slightly, and forcefully close and lock the door.

(Continued Next Page)

## **ENTRANCE DOORS AND CABIN WINDOWS**

(Continued)

Exit from the airplane is accomplished by rotating the door handle from the LOCK position, past the CLOSE position, aft to the OPEN position and pushing the door open. To lock the airplane, lock the right cabin door with the inside handle, close the left cabin door, and using the ignition key, lock the door.

The left and right cabin doors are equipped with openable windows which are held in the closed position by a detent equipped latch on the lower edge of the window frame. To open the windows, rotate the latch upward. Each window is equipped with a spring-loaded retaining arm which will help rotate the window outward, and hold it there. If required, either window may be opened at any speed up to 175 KIAS. The rear side windows and rear windows are of the fixed type and cannot be opened.

## **CONTROL LOCKS**

A control lock is provided to lock the aileron and elevator control surfaces to prevent damage to these systems by wind buffeting while the airplane is parked. The lock consists of a shaped steel rod and flag. The flag identifies the control lock and cautions about its removal before starting the engine. To install the control lock, align the hole in the side of the pilot's control wheel shaft with the hole in the side of the shaft collar on the instrument panel and insert the rod into the aligned holes. Installation of the lock will secure the ailerons in a neutral position and the elevators in a slightly trailing edge down position. Proper installation of the lock will place the flag over the ignition switch. In areas where high or gusty winds occur, a control surface lock should be installed over the vertical stabilizer and rudder. The control lock and any other type of locking device should be removed prior to starting the engine.

## ENGINE

The airplane is powered by a direct drive, horizontally opposed, six cylinder, overhead valve, air cooled, fuel injected engine with a wet sump lubrication system. The engine is a Lycoming Model IO-540-AB1A5 rated at 230 horsepower at 2400 RPM. Major accessories include a propeller governor, starter and belt driven alternator mounted on the front of the engine, dual magnetos, vacuum pump, engine driven fuel pump, and a full flow oil filter mounted on the rear of the engine accessory case.

## ENGINE CONTROLS

Engine manifold pressure is set using the throttle control. The throttle control is a smooth black knob located at the center of the instrument panel below the standby instruments. The throttle control is configured so that the throttle is open in the forward position and closed in the full aft position. A friction lock, located at the base of the throttle, is operated by rotating the lock clockwise to increase friction or counterclockwise to decrease friction.

Engine speed is controlled by the propeller control. The propeller control is a fluted blue knob located immediately to the right of the throttle control. This system is described under Propeller in this section.

Engine fuel mixture is controlled by the mixture control. The mixture control is a red knob, with raised points around the circumference, located immediately to the right of the propeller control and is equipped with a lock button in the end of the knob. The rich position is full forward, and full aft is the idle cutoff position. For small adjustments, the control may be moved forward by rotating the knob clockwise, and aft by rotating the knob counterclockwise. For rapid or large adjustments, the knob may be moved forward or aft by depressing the lock button in the end of the control, and then positioning the control as desired.

(Continued Next Page)

## ENGINE (Continued)

### ENGINE INSTRUMENTS

The G1000 Engine Indication System provides graphical indicators and numeric values for engine, fuel, and electrical system parameters to the pilot. The EIS is shown in a vertical strip on the left side of the PFD during engine starts and on the MFD during normal operation. If either the MFD or PFD fails during flight, the EIS is shown on the remaining display.

The EIS consists of three pages that are selected using the ENGINE softkey. The ENGINE page provides indicators for Manifold Pressure (MAN IN), Tachometer (RPM), Fuel Flow (FFLOW GPH), Oil Pressure (OIL PRES), Oil Temperature (OIL TEMP), Cylinder Head Temperature (CHT), Exhaust Gas Temperature (EGT), Fuel Quantity (FUEL QTY GAL), Electrical Bus Voltages (VOLTS), and Battery Currents (AMPS). When the ENGINE softkey is pressed, the LEAN and SYSTEM softkeys appear adjacent to the ENGINE softkey. The LEAN page provides simultaneous indicators for Exhaust Gas Temperature (EGT °F) and Cylinder Head Temperature (CHT °F) on all cylinders to be used for adjusting, or leaning, the fuel/air mixture along with an indicator for FUEL QTY GAL. The SYSTEM page provides numerical values for parameters on the ENGINE page that are shown as indicators only. The SYSTEM page also provides an indicator for vacuum (VAC) and a digital value for Engine Hours (ENG HRS), Fuel Used (GAL USED) and Fuel Remaining (GAL REM).

The engine and airframe unit, located forward of the instrument panel, receives signals from the engine/system sensors for the parameters that are being monitored. The engine and airframe unit provides data to the EIS, which displays the data for the ENGINE page described on the following pages.

(Continued Next Page)

## **ENGINE** (Continued)

### **ENGINE INSTRUMENTS** (Continued)

#### **MANIFOLD PRESSURE**

Manifold pressure is shown by the MAN IN indicator at the top of all EIS pages. The manifold pressure indicator uses a circular scale and moving pointer with a digital value. The pointer moves through a range of 10 to 35 in.hg. The digital manifold pressure value is shown in white numerals below the pointer.

Colored arcs on the manifold pressure indicator show different operating ranges. The normal operating range is from 15 to 23 in.hg. and indicated by the green arc. The acceptable operating range is indicated with a white arc from 10 to 15 in.hg. and 23 to 35 in.hg.

An absolute pressure transducer, located between the firewall and the instrument panel, provides a signal to the engine and airframe unit which processes and outputs the data to the EIS. A red X through the manifold pressure display shows that the indicating system is inoperative.

(Continued Next Page)

## **ENGINE** (Continued)

### **ENGINE INSTRUMENTS** (Continued)

#### **RPM (TACHOMETER)**

Engine speed is shown by the RPM indicator, found on all EIS pages below the manifold pressure indicator. The tachometer indicator uses a circular scale with moving pointer and a digital value. The pointer moves through a range from 0 to 2700 RPM. The numerical RPM value is displayed in increments of 10 RPM in white numerals below the pointer.

The normal engine speed operating limit (red line) is 2400 RPM. When engine speed is 2472 RPM or more, the pointer, digital value, and label (RPM) turn red to show engine speed is more than the limit. The digital value and label (RPM) will flash. The engine speed (tachometer) is displayed in the same configuration and location on the LEAN and SYSTEM pages. If engine speed becomes 2472 RPM or more, while on the LEAN or SYSTEM page, the display will return to the ENGINE page.

A speed sensor, mounted on the engine tachometer drive accessory pad, provides a digital signal to the engine and airframe unit which processes and outputs the RPM data to the EIS. A red X through the RPM indicator shows the indicating system is inoperative.

#### **FUEL FLOW**

Fuel flow is shown on the ENGINE page by the FFLOW GPH horizontal indicator. The indicator range is from 0 to 22 gallons per hour (GPH) with 2 GPH graduations, with a green band from 0 to 15 GPH. A white pointer shows the measured fuel flow.

A digital value for FFLOW GPH is included on both the EIS LEAN and SYSTEM pages.

The fuel flow transducer is located in the engine fuel injection system between the fuel/air control unit (servo) and the fuel distribution manifold (flow divider). The transducer provides a signal to the engine display that is processed and shown as fuel flow (FFLOW) on the EIS pages. A red X through the indicator means the indicating system is inoperative.

(Continued Next Page)

## **ENGINE** (Continued)

### **ENGINE INSTRUMENTS** (Continued)

#### **OIL PRESSURE**

Engine oil pressure is shown on the ENGINE page by the OIL PRES horizontal indicator. The indicator range is 0 to 120 PSI with a red band from 0 to 20 PSI, a green band from 50 to 90 PSI (normal operating range) and a red band from 115 to 120 PSI. A white pointer indicates actual oil pressure. Oil pressure is shown numerically on the SYSTEM page.

When oil pressure is 0 to 20 PSI or 115 to 120 PSI, the pointer, digital value, and label (OIL PRES) will change to red to show that oil pressure is outside normal limits. If oil pressure exceeds either the upper or lower limit while on the LEAN or SYSTEM page, the EIS will return to the ENGINE page.

When the engine speed (RPM) is in the green arc and the oil temperature is in the green band, the oil pressure should be in the green band. If oil pressure is below the green band or above the green band, adjust the engine speed to maintain adequate oil pressure. When engine speed is at idle or near idle, the oil pressure indication must be above the lower red band. With the engine at normal operating oil temperature, and engine speed at or close to idle, oil pressure below the green band, but above the lower red band, is acceptable.

In cold weather, the oil pressure will initially be high (close to the upper red band when the engine is started). As the engine and oil warm up, the oil pressure will come down into the green band range.

The oil pressure transducer, connected to the engine forward oil pressure port, provides a signal to the engine display that is processed and shown as oil pressure. A separate low oil pressure switch causes an OIL PRESSURE annunciation on the PFD when oil pressure is 0 to 20 PSI. A red X through the oil pressure indicator means that the indicating system is inoperative.

(Continued Next Page)

## **ENGINE** (Continued)

### **ENGINE INSTRUMENTS** (Continued)

#### **OIL TEMPERATURE**

Engine oil temperature is shown on the ENGINE page by the OIL TEMP horizontal indicator. The indicator range is from 75°F to 250°F with a green band (normal operating range) from 100°F to 245°F and a red band from 245°F to 250°F. A white pointer indicates actual oil temperature. Oil temperature is displayed numerically on the SYSTEM page.

When oil temperature is in the red band, 245°F to 250°F, the pointer and OIL TEMP turn red and flash to show oil temperature is higher than the limit. If oil temperature becomes hotter than 245°F while on the LEAN or SYSTEM page, the display will default to the ENGINE page.

The oil temperature sensor is installed in the engine oil filter adapter and provides a signal to the engine display that is processed and shown as oil temperature. A red X through the indicator shows that the indicating system is inoperative.

(Continued Next Page)

## **ENGINE** (Continued)

### **ENGINE INSTRUMENTS** (Continued)

#### **CYLINDER HEAD TEMPERATURE**

Cylinder head temperature is shown on the ENGINE page on the CHT horizontal indicator. The gage range is from 100°F to 500°F, with a green band from 200°F to 500°F and red line at 500°F. The white pointer indicates relative CHT while the number inside of the pointer identifies the hottest cylinder head. If a CHT probe or wire failure occurs for the hottest CHT, the next hottest CHT will be displayed.

When the CHT is 500°F or hotter, the pointer will change to red and the CHT label will change to red and flash to show that the CHT is greater than the limit. If CHT becomes hotter than 500°F while on the LEAN or SYSTEM page, the display will default to the ENGINE page.

The CHT for all six cylinders are shown on the LEAN page. The cylinder with the hottest CHT is indicated by a cyan bar graph. The CHT for cylinder number three has been demonstrated to be the most critical, and operation with CHT 3 inoperative is not allowed. The LEAN page will show a red X over any cylinder that has a probe or wiring failure.

A thermocouple is installed in each cylinder head and provides a signal to the engine display that is processed and shown as CHT on the EIS LEAN page.

(Continued Next Page)

## **ENGINE** (Continued)

### **ENGINE INSTRUMENTS** (Continued)

#### **EXHAUST GAS TEMPERATURE**

Exhaust gas temperature is shown by the EGT horizontal indicator on the EIS. The indicator range is from 1100°F to 1500°F with graduations every 50°F. The white pointer indicates relative EGT with the number of the hottest cylinder displayed inside the pointer. If a cylinder EGT probe or wiring failure occurs for the hottest EGT, the next hottest EGT will be displayed.

The EGT for all six cylinders is shown on the LEAN page of the EIS. The hottest cylinder is indicated by the cyan bar graph. The EGT for a particular cylinder may be shown by using the CYL SLCT softkey to select the desired cylinder. Automatic indication of the hottest cylinder will resume a short time after the CYL SLCT is last selected. The LEAN page will show a red X over a cylinder that has a probe or wiring failure.

A thermocouple is installed in the exhaust pipe of each cylinder which measures EGT and provides a signal to the engine display that is processed and shown as EGT on the EIS LEAN page.

(Continued Next Page)

## **ENGINE** (Continued)

### **NEW ENGINE BREAK-IN AND OPERATION**

The engine run-in was accomplished at the factory and is ready for the full range of use. It is suggested that cruising be accomplished at 75% power as much as practicable until a total of 50 hours has accumulated or oil consumption has stabilized. This will ensure proper sealing of the piston rings.

### **ENGINE LUBRICATION SYSTEM**

The engine utilizes a full pressure, wet sump type lubrication system with aviation grade oil as the lubricant. The capacity of the engine sump, located on the bottom of the engine, is nine quarts with one additional quart contained in the engine oil filter. Oil is drawn from the sump through a filter screen on the end of a pickup tube to the engine driven oil pump. Oil from the pump passes through a full-flow oil filter, a pressure relief valve at the rear of the right oil gallery, and a thermostatically controlled remote oil cooler. Oil from the remote cooler is then circulated to the left oil gallery and propeller governor. The engine parts are then lubricated by oil from the galleries. After lubricating the engine, the oil returns to the sump by gravity. The filter adapter in the full-flow filter is equipped with a bypass valve which will cause lubricating oil to bypass the filter in the event the filter becomes plugged, or the oil temperature is extremely cold.

An oil dipstick/filler tube is located on the upper left side of the engine case. The dipstick and oil filler tube are accessed through a door located on the left center portion of the upper engine cowling. The engine should not be operated on less than four quarts of oil. To minimize loss of oil through the breather, fill to eight quarts for normal flights of less than three hours. For extended flight, fill to nine quarts (dipstick indication only). For engine oil grade and specifications, refer to Section 8 of this POH.

(Continued Next Page)

## **ENGINE** (Continued)

### **IGNITION AND STARTER SYSTEM**

Engine ignition is provided by two engine driven magnetos, and two spark plugs in each cylinder. The left magneto fires the upper left and lower right spark plugs, and the right magneto fires the lower left and upper right spark plugs. Normal operation is conducted with both magnetos due to the more complete burning of the fuel/air mixture with dual ignition.

Ignition and starter operation is controlled by a rotary-type switch located on the left switch and control panel. The switch is labeled clockwise, OFF, R, L, BOTH, and START. The engine should be operated on both magnetos (BOTH position) except for magneto checks. The R and L positions are for checking purposes and emergency use only. When the switch is rotated to the START position, (with the MASTER switch in the ON position), the starter contactor is closed and the starter, now energized, will crank the engine. When the switch is released, it will automatically return to the BOTH position.

### **AIR INDUCTION SYSTEM**

The engine air induction system receives ram air through an intake on the lower front portion of the engine cowling. The intake is covered by an air filter which removes dust and other foreign matter from the induction air. Airflow passing through the filter enters an air box, which is equipped with a spring-loaded alternate air door. If the air induction filter should become blocked, suction created by the engine will open the door and draw unfiltered air from inside the lower cowl area. An open alternate air door will result in an approximate 10% power loss at full throttle. After passing through the air box, induction air enters a fuel/air control unit under the engine, and is then ducted to the engine cylinders through intake manifold tubes.

(Continued Next Page)

## **ENGINE** (Continued)

### **EXHAUST SYSTEM**

Exhaust gas from each cylinder passes through a riser or exhaust collector assembly (on the left or right side below the engine) to a muffler and overboard through a single tailpipe. Outside air is supplied to a shroud constructed around the outside of each muffler to form a heating chamber. The air heated by the shroud is then supplied to the cabin.

### **FUEL INJECTION SYSTEM**

The engine is equipped with a fuel injection system. The system is comprised of an engine driven fuel pump, fuel/air control unit, fuel manifold, fuel flow indicator, and air-bleed type injector nozzles.

Fuel is delivered by the engine driven fuel pump to the fuel/air control unit. The fuel/air control unit correctly proportions the fuel flow to the induction air flow. After passing through the control unit, induction air is delivered to the cylinders through the intake manifold tubes and metered fuel is delivered to a fuel manifold (flow divider). The fuel manifold, through spring tension on a diaphragm and valve, evenly distributes the fuel to an air-bleed type injector nozzle in the intake valve chamber of each cylinder. A turbine-type fuel flow transducer mounted between the fuel/air control unit and the fuel distribution unit produces a digital signal that displays fuel flow on the EIS pages.

(Continued Next Page)

## **ENGINE** (Continued)

### **COOLING SYSTEM**

Ram air for engine cooling enters through two intake openings in the front of the engine cowling. The cooling air is directed from above the engine, around the cylinders and other areas of the engine by baffling, and then exits through cowl flaps on the lower aft edge of the cowling. The cowl flaps are mechanically operated from the cabin by means of the cowl flap control lever located on the right side of the control pedestal and is labeled OPEN, COWL FLAPS, CLOSED. Any time the control lever is repositioned, it must first be moved to the right to clear the detent.

Before starting the engine, before takeoff and during high power operation, the cowl flap control lever should be placed in the OPEN position for maximum cooling. This is accomplished by moving the control lever to the right to clear a detent, then moving the control lever up to the OPEN position.

While in cruise flight, cowl flaps should be closed unless hot day conditions require them to be adjusted to keep the CHTCHT at approximately two-thirds of the normal operating range (green band).

During extended descents, it may be necessary to completely close the cowl flaps by pushing the cowl flap control lever down to the CLOSED position.

A winterization kit is available for the airplane. Refer to Section 9, Supplement 5 for description and operating information.

## PROPELLER

The airplane has an all metal, three-bladed, constant speed, governor regulated propeller. A setting introduced into the governor with the propeller control establishes the propeller speed, and thus the engine speed to be maintained. The governor then controls flow of engine oil, boosted to high pressure by the governing pump, to or from a piston in the propeller hub. Oil pressure acting on the piston twists the blades toward high pitch (low RPM). When oil pressure to the piston in the propeller hub is relieved, centrifugal force, assisted by an internal spring, twists the blades toward low pitch (high RPM).

A propeller control knob, located on the lower center instrument panel, is used to set the propeller and control engine RPM as desired for various flight conditions. The control knob is labeled PROPELLER, PUSH INCR RPM. When the control knob is pushed in, blade pitch will decrease, giving a higher RPM. When the control knob is pulled out, the blade pitch increases, thereby decreasing RPM. The propeller control knob is equipped with a vernier feature which allows slow or fine RPM adjustments by rotating the knob clockwise to increase RPM, and counterclockwise to decrease it. To make rapid or large adjustments, depress the button on the end of the control knob and reposition the control as desired.

## FUEL SYSTEM

The airplane fuel system (Refer to Figure 7-6) consists of two vented integral fuel tanks (one tank in each wing), two fuel manifolds (one in each aft doorpost), a dual stack, four-position selector valve, an electrically-driven auxiliary fuel pump, and a fuel strainer. The engine-mounted portion of the system consists of the engine driven fuel pump, a fuel/air control unit, fuel flow transducer, a fuel distribution valve (flow divider) and fuel injection nozzles.

The fuel system also incorporates a fuel return system that returns fuel from the top of the fuel servo back to each integral wing tank. The system includes a flexible fuel hose assembly between the servo and the firewall. Aluminum fuel lines return the fuel to the top portion of the selector valve and then to the airplane's integral tanks. One drain is added to properly drain the fuel return system.

### WARNING

**UNUSABLE FUEL LEVELS FOR THIS AIRPLANE WERE DETERMINED IN ACCORDANCE WITH FEDERAL AVIATION REGULATIONS. FAILURE TO OPERATE THE AIRPLANE IN COMPLIANCE WITH FUEL LIMITATIONS SPECIFIED IN SECTION 2 MAY FURTHER REDUCE THE AMOUNT OF FUEL AVAILABLE IN FLIGHT.**

(Continued Next Page)

**FUEL SYSTEM** (Continued)

**FUEL QUANTITY DATA IN U.S. GALLONS**

FUEL TANKS	FUEL LEVEL (QUANTITY EACH TANK)	TOTAL FUEL	TOTAL UNUSABLE	TOTAL USABLE ALL FLIGHT CONDITIONS
Two	Full (46.0)	92.0	5.0	87.0
Two	Reduced (39.5)	79.0	5.0	74.0
Two	Reduced (34.5)	69.0	5.0	64.0

Figure 7-5\*

**FUEL DISTRIBUTION**

Fuel flows by gravity from the two wing tanks through the fuel manifold (aft pickup only), and to a four position selector valve. From the selector valve, fuel flows through the auxiliary fuel pump, the fuel strainer, and to the engine driven fuel pump. A portion of the fuel (approximately 7 GPH) is returned to the wing tank currently selected through the use of the fuel return system. From the engine driven fuel pump, fuel is delivered to the fuel/air control unit on the bottom of the engine. The fuel/air control unit (fuel servo) meters fuel flow in proportion to induction air flow. After passing through the control unit, metered fuel goes to a fuel distribution valve (flow divider) located on the bottom of the engine. From the fuel distribution valve, individual fuel lines are routed to air bleed type injector nozzles located in the intake chamber of each cylinder.

**FUEL INDICATING SYSTEM**

Fuel quantity is measured by two fuel quantity sensors, one in each fuel tank, and is displayed on the EIS pages. The indicators are marked in gallons of fuel (GAL). An empty tank is displayed on the fuel quantity indicator (FUEL QTY GAL) as a red line on the far left of the indicator scale, and the number 0. When an indicator shows an empty tank, approximately 2.5 gallons of unusable fuel remain in the tank. The indicators should not be relied upon for accurate readings during skids, slips or unusual attitudes.

(Continued Next Page)

## FUEL SYSTEM (Continued)

### FUEL INDICATING SYSTEM (Continued)

The fuel quantity indicator shows the fuel available in the tank up to the limit of the sensor measurement range. At this level, additional fuel may be added to completely fill the tank, but no additional movement of the indicator will result. The limit for sensor measurement range is approximately 35 gallons and is indicated by the maximum limit of the green band. When the fuel level decreases below the maximum limit of the fuel sensor, the fuel quantity indicator will display fuel quantity measured in each tank. A visual check of each wing tank fuel level must be performed prior to each flight. Compare the visual fuel level and indicated fuel quantity to accurately estimate usable fuel.

The fuel quantity indicators detect low fuel conditions and incorrect sensor outputs. When fuel quantity is less than 8 gallons indicated (and remains less than this level for more than 60 seconds), LOW FUEL L (left) and/or LOW FUEL R (right) will be displayed in amber on the PFD and a tone will sound. The fuel quantity indicator pointer(s) and indicator label will change from white to steady amber. When fuel quantity reaches the calibrated usable fuel empty level, the LOW FUEL L and/or LOW FUEL R remain amber and the indicator pointer(s) and label change to flashing red.

#### NOTE

Takeoff is not recommended if both fuel quantity indicator pointers are in the yellow band range and/or amber LOW FUEL L or LOW FUEL R annunciator is displayed on the PFD.

In addition to low fuel annunciation, the warning logic is designed to report failures with each sensor. If the system detects a failure, the affected fuel indicator will display a red X. A red X through the top part of the indicator indicates a failure associated with the left fuel tank. A red X through the bottom part of the indicator indicates a failure associated with the right fuel tank.

(Continued Next Page)

## FUEL SYSTEM (Continued)

### FUEL INDICATING SYSTEM (Continued)

Fuel flow is measured by use of a turbine type transducer mounted on the centerline of the engine between the fuel/air control unit and the fuel distribution unit. This flow meter produces a signal that is displayed as the rate of fuel flow on the FFLOW GPH indicator on the EIS pages. FFLOW GPH is shown as either a horizontal analog indicator or a digital value, depending on the active EIS page.

### FUEL CALCULATIONS

#### NOTE

Fuel calculations do not use the airplane's fuel quantity indicators and are calculated from the last time the fuel was reset.

For fuel consumption information, a fuel used totalizer function is provided on the EIS SYSTEM page as GAL USED. This digital indicator shows total fuel used since last reset of the totalizer. To reset the GAL USED, the EIS SYSTEM page must be active and the RST USED softkey must be selected. GAL USED is calculated after reset using information from the fuel flow transducer signal.

For fuel remaining information, a count down fuel totalizer function is provided on the EIS System page as GAL REM. This digital indicator shows calculated fuel remaining since last GAL REM pilot adjustment. To adjust GAL REM, the EIS SYSTEM page must be active and the GAL REM softkey must be selected followed by the appropriate quantity adjustment softkeys. Refer to the Garmin G1000 CRG for details for resetting and adjusting fuel calculations. GAL REM is calculated after pilot adjustment using information from the fuel flow transducer signal.

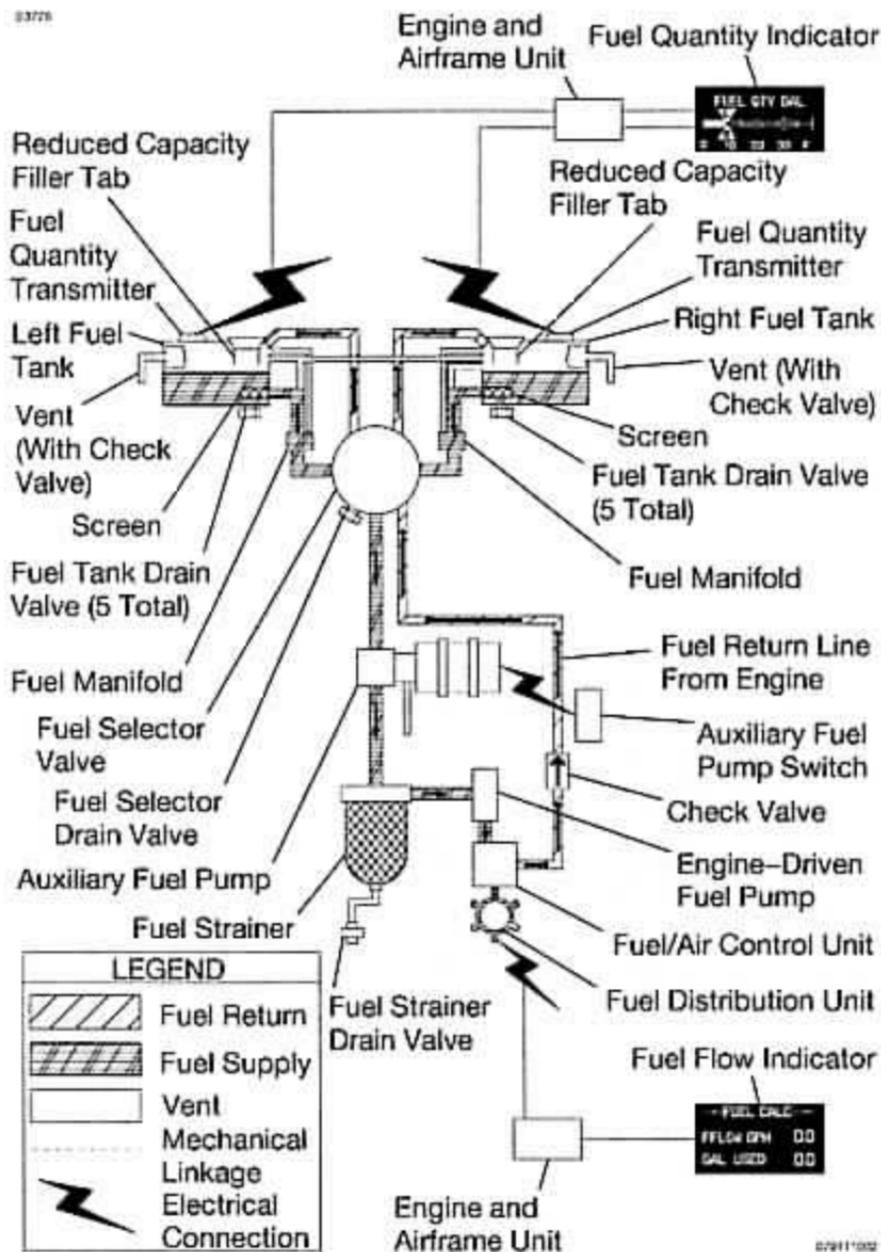
#### NOTE

GAL USED and GAL REM provide no indication of the actual amount of fuel remaining in each tank and should only be used in conjunction with other fuel management procedures to estimate total fuel remaining.

(Continued Next Page)

**FUEL SYSTEM (Continued)**

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Figure 7-6\*

## FUEL SYSTEM (Continued)

### AUXILIARY FUEL PUMP OPERATION

The auxiliary fuel pump is used primarily for priming the engine before starting. Priming is accomplished through the fuel injection system. The engine may be flooded if the auxiliary FUEL PUMP switch is accidentally placed in the ON position for prolonged periods, with MASTER Switch ON and mixture rich, with the engine stopped.

The auxiliary fuel pump is also used for vapor suppression in hot weather. Normally, momentary use will be sufficient for vapor suppression; however, continuous operation is permissible if required. Turning on the auxiliary fuel pump with a normally operating engine driven fuel pump will result in only a very minor enrichment of the mixture.

It is not necessary to operate the auxiliary fuel pump during normal takeoff and landing, since gravity and the engine driven fuel pump will supply adequate fuel flow. In the event of failure of the engine driven fuel pump, use of the auxiliary fuel pump will provide sufficient fuel to maintain flight at maximum continuous power.

Under hot day, high altitude conditions, or conditions during a climb that are conducive to fuel vapor formation, it may be necessary to utilize the auxiliary fuel pump to attain or stabilize the fuel flow required for the type of climb being performed. In this case, turn the auxiliary fuel pump on, and adjust the mixture to the desired fuel flow. If fluctuating fuel flow (greater than 1 GPH) is observed during climb or cruise at high altitudes on hot days, place the auxiliary fuel pump switch in the ON position to clear the fuel system of vapor. The auxiliary fuel pump may be operated continuously in cruise.

(Continued Next Page)

## FUEL SYSTEM (Continued)

### FUEL RETURN SYSTEM

A fuel return system was incorporated to improve engine operation during extended idle operation in hot weather environments. The major components of the system include an orifice fitting located in the top of the fuel servo, a dual stack fuel selector and a drain valve assembly. The system is designed to return fuel/vapor back to the main fuel tanks at approximately 7 GPH. The dual stack fuel selector ensures that fuel/vapor returns only to the fuel tank that is selected as the feed tank. For example, if the fuel selector is positioned to use fuel from the left fuel tank, the fuel return system is returning fuel/vapor to the left fuel tank only.

### FUEL VENTING

Fuel system venting is essential to system operation. Complete blockage of the fuel venting system will result in decreasing fuel flow and eventual engine stoppage. The fuel venting system consists of an interconnecting vent line between the fuel tanks and check valve equipped overboard vents in each fuel tank. The overboard vents protrude from the bottom surface of the wings behind the wing struts, slightly below the upper attach points of the struts. The fuel filler caps are vacuum vented; the fuel filler cap vents will open and allow air to enter the fuel tanks in case the overboard vents become blocked.

### REDUCED TANK CAPACITY

The airplane may be serviced to a reduced capacity to permit heavier cabin loadings. This is accomplished by filling each tank to the bottom edge of the fuel filler indicator tab, thus giving a reduced fuel load of 32.0 gallons usable in each tank or to the line of holes located inside the filler indicator tab, thus giving a reduced fuel load of 37.0 gallons usable in each tank.

(Continued Next Page)

## FUEL SYSTEM (Continued)

### FUEL SELECTOR VALVE

The fuel selector is a four-position selector valve, labeled BOTH, RIGHT, LEFT and OFF. The selector handle must be pushed down before it can be rotated from RIGHT or LEFT to OFF. The top portion of the valve is the return portion of the valve, while the bottom portion of the valve is the supply portion. Each side is isolated from the other.

The fuel selector valve should be in the BOTH position for takeoff, climb, landing, and maneuvers that involve prolonged slips or skids of more than 30 seconds. Operation on either LEFT or RIGHT fuel tank is reserved for level cruising flight only.

#### NOTE

- When the fuel selector valve is placed in the BOTH position, while in cruise flight, unequal fuel flow from each tank may occur if the wings are not maintained exactly level. Unequal fuel flow can be detected by one fuel tank indicating more fuel than the other on the L FUEL and R FUEL indicators. The resulting fuel imbalance can be corrected by turning the fuel selector valve to the fuel tank indicating the highest fuel quantity. Once the L FUEL and R FUEL indicators have equalized, position the fuel selector valve to the BOTH position.
- It is not practical to measure the time required to consume all of the fuel in one tank, and, after switching to the opposite tank, expect an equal duration from the remaining fuel. The airspace in both fuel tanks is interconnected by a vent line and, therefore, some sloshing of fuel between tanks can be expected when the tanks are nearly full and the wings are not level.
- When the fuel tanks are 1/4 tank or less, prolonged uncoordinated flight, such as slips or skids, can uncover the fuel tank outlets causing fuel starvation and engine stoppage. Therefore, if operating with one fuel tank dry or operating on either LEFT or RIGHT tank with 1/4 tank or less, do not allow the airplane to remain in uncoordinated flight for periods in excess of 30 seconds.

(Continued Next Page)

## FUEL SYSTEM (Continued)

### FUEL DRAIN VALVES

The fuel system is equipped with drain valves to provide a means for the examination of fuel in the system for contamination and grade. The system should be examined before each flight and after each refueling, by using the sampler cup provided to drain fuel from each wing tank sump, the fuel return side sump, the fuel selector drain and the fuel strainer sump. If any evidence of fuel contamination is found, it must be eliminated in accordance with the preflight inspection checklist and the discussion in Section 8. If takeoff weight limitations for the next flight permit, the fuel tanks should be filled after each flight to prevent condensation.

### BRAKE SYSTEM

The airplane has a single-disc, hydraulically-actuated brake on each main landing gear wheel. Each brake is connected, by a hydraulic line, to a master cylinder attached to each of the pilot's rudder pedals. The brakes are operated by applying pressure to the top of either the left (pilot's) or right (copilot's) set of rudder pedals, which are interconnected. When the airplane is parked, both main wheel brakes may be set by utilizing the parking brake which is operated by a handle under the left side of the instrument panel. To apply the parking brake, set the brakes with the rudder pedals, pull the handle aft, and rotate it 90° down.

For maximum brake life, keep the brake system properly maintained, and minimize brake usage during taxi operations and landings.

Some of the symptoms of impending brake failure are: gradual decrease in braking action after brake application, noisy or dragging brakes, soft or spongy pedals, and excessive travel and weak braking action. If any of these symptoms appear, the brake system is in need of immediate attention. If, during taxi or landing roll, braking action decreases, let up on the pedals and then reapply the brakes with heavy pressure. If the brakes become spongy or pedal travel increases, pumping the pedals should build braking pressure. If one brake becomes weak or fails, use the other brake sparingly while using opposite rudder, as required, to offset the good brake

## ELECTRICAL SYSTEM

The airplane is equipped with a 28-volt direct current (DC) electrical system (Refer to Figure 7-7). A belt-driven 60 ampere or optional 95 ampere alternator powers the system. A 24-volt main storage battery is located in the tailcone of the airplane. The alternator and main battery are controlled through the MASTER switch found near the top of the pilot's switch panel.

Power is supplied to most electrical circuits through two primary buses (ELECTRICAL BUS 1 and ELECTRICAL BUS 2), with an essential bus and a crossfeed bus connected between the two primary buses to support essential equipment.

The system is equipped with a secondary or standby battery located between the firewall and the instrument panel. The STBY BATT switch controls power to or from the standby battery. The standby battery is available to supply power to the essential bus in the event that alternator and main battery power sources have both failed.

The primary buses are supplied with power whenever the MASTER switch is turned on, and are not affected by starter or external power usage. Each primary bus is also connected to an avionics bus through a circuit breaker and the AVIONICS BUS 1 and BUS 2 switches. Each avionics bus is powered when the MASTER switch and the corresponding AVIONICS switch are in the ON position.

### CAUTION

**BOTH BUS 1 AND BUS 2 AVIONICS SWITCHES SHOULD BE TURNED OFF TO PREVENT ANY HARMFUL TRANSIENT VOLTAGE FROM DAMAGING THE AVIONICS EQUIPMENT PRIOR TO TURNING THE MASTER SWITCH ON OR OFF, STARTING THE ENGINE OR APPLYING AN EXTERNAL POWER SOURCE.**

The airplane includes a power distribution module, located on the left forward side of the firewall, to house all the relays used in the airplane electrical system. The Alternator Control Unit (ACU), main battery current sensor, and the external power connector are also housed within the module.

(Continued Next Page)



**ELECTRICAL SYSTEM (Continued)**

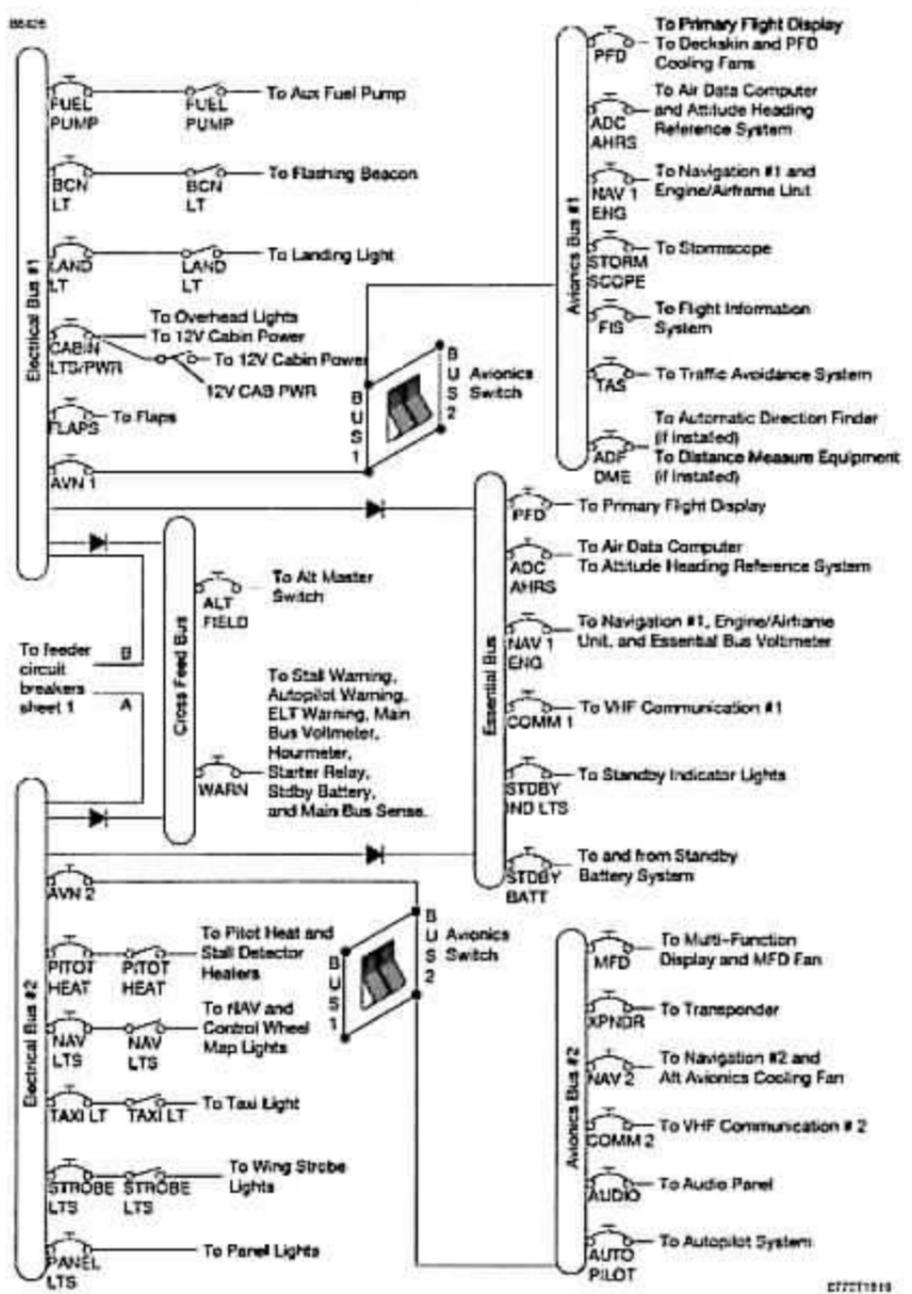
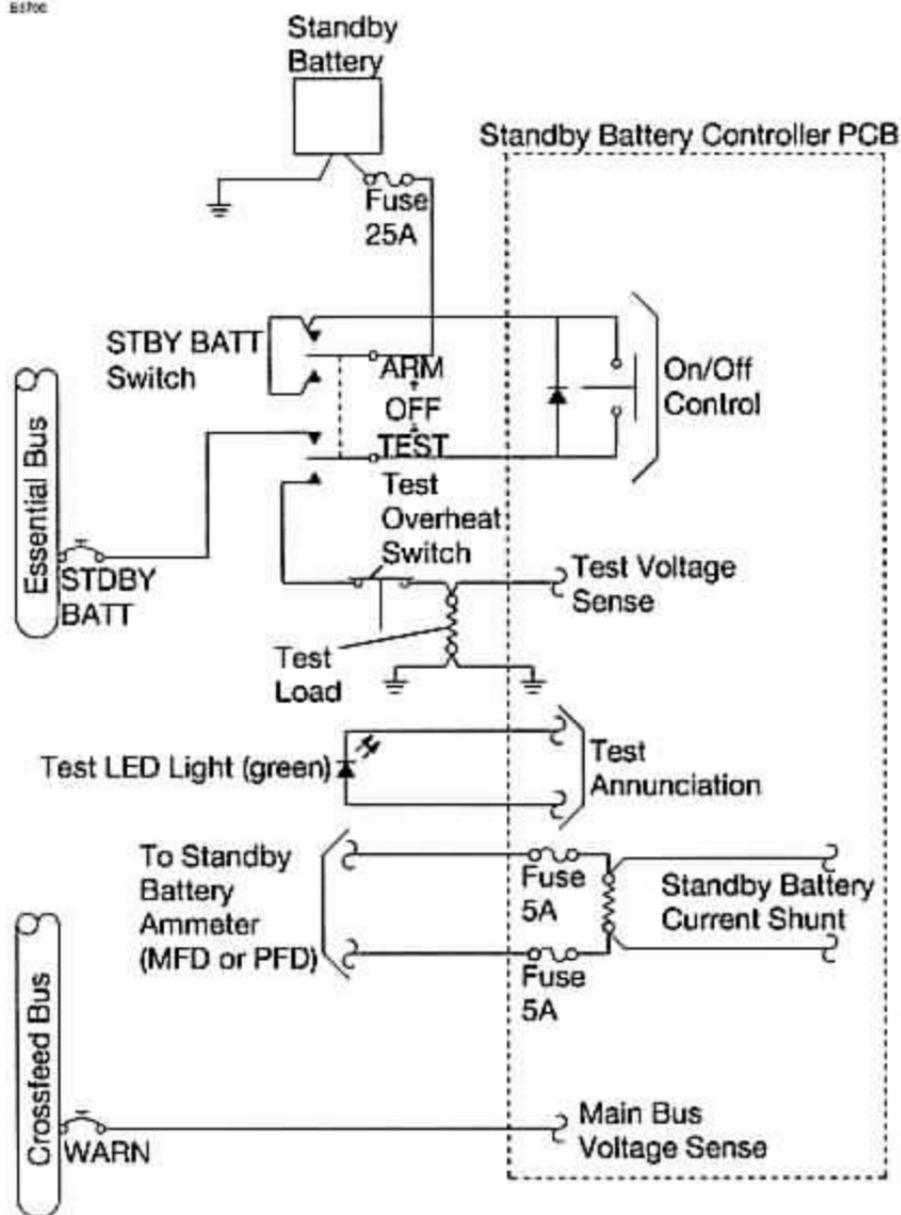


Figure 7-7 (Sheet 2)

**ELECTRICAL SYSTEM** (Continued)

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Figure 7-7 (Sheet 3)

## ELECTRICAL SYSTEM (Continued)

### G1000 ANNUNCIATOR PANEL

All system alerts, cautions and warnings are shown on the right side of the PFD screen adjacent to the vertical speed indicator. The following annunciations are supported:

OIL PRESSURE	LOW VACUUM
LOW FUEL L	LOW FUEL R
LOW VOLTS	HIGH VOLTS
STBY BATT	CO LVL HIGH

Refer to the Garmin G1000 CRG Appendix A for more information on system annunciations.

### MASTER SWITCH

The MASTER switch is a two-pole, rocker-type switch. The BAT side of the switch controls the main battery electrical power to the airplane. The ALT side of the switch controls the alternator system.

In normal operation, both sides of the switch (ALT and BAT) are ON simultaneously; however, the BAT side of the switch may be selected separately as necessary. The ALT side of the switch can not be set to ON without the BAT side of the switch also being set to ON.

If the alternator system fails, the MASTER switch may be set in the OFF position to preserve main battery capacity for later in the flight. With the MASTER switch OFF and the STBY BATT switch in the ARM position, the standby battery will power the essential bus for a limited time. Time remaining may be estimated by monitoring essential bus voltage. At 20 Volts, the standby battery has little or no capacity remaining.

(Continued Next Page)

## **ELECTRICAL SYSTEM** (Continued)

### **STANDBY BATTERY SWITCH**

The STBY BATT master switch is a three position (ARM-OFF-TEST) switch that tests and controls the standby battery system. The energy level of the battery shall be checked before starting the engine (Refer to Section 4) by placing the switch in the momentary TEST position and observing the correct illumination of the TEST lamp found to the right of the switch. Energy level tests after starting engine are not recommended.

Placing the switch in the ARM position during the engine start cycle allows the standby battery to help regulate and filter essential bus voltage during the start cycle. The switch is set to the ARM position during normal flight operation to allow the standby battery to charge and to be ready to power the essential bus in the event of alternator and main battery failure. Placing the switch in the OFF position disconnects the standby battery from the essential bus. Operation with the STBY BATT switch in the OFF position prevents the standby battery from charging and from automatically providing power should an electrical system failure occur.

### **AVIONICS SWITCH**

The AVIONICS switch is a two-pole rocker-type switch that controls electrical power to AVIONICS BUS 1 and BUS 2. Placing either side of the rocker switch in the ON position supplies power to the corresponding avionics bus. Both sides of the AVIONICS switch should be placed in the OFF position before turning the MASTER switch ON or OFF, starting the engine, or applying an external power source.

(Continued Next Page)

## **ELECTRICAL SYSTEM** (Continued)

### **ELECTRICAL SYSTEM MONITORING AND ANNUNCIATIONS**

#### **BUS VOLTAGE (VOLTMETERS)**

Voltage indication (VOLTS) for the main and essential buses is provided at the bottom of the EIS bar (along the left margin of the MFD or PFD), labeled M BUS E. Main bus voltage is shown numerically below the M. Essential bus voltage is displayed numerically below the E. The main bus voltage is measured at the WARN circuit breaker on the crossfeed bus. The essential bus voltage is measured at the NAV1 ENG circuit breaker on the essential bus.

Normal bus voltages with the alternator operating shall be about 28.0 volts. When the voltage for either main or essential buses is above 32.0 volts, the numerical value and VOLTS text turns red. This warning indication, along with the HIGH VOLTS annunciation, is an indication that the alternator is supplying too high of a voltage. The ALT MASTER Switch should immediately be positioned to OFF (Refer to Section 3, Emergency Procedures, HIGH VOLTS ANNUNCIATOR COMES ON).

When the voltage for either main or essential buses is below 24.5 volts, the numeric value and VOLTS text turns red. This warning indication, along with the LOW VOLTS annunciation, is an indication that the alternator is not supplying all the power that is required by the airplane. Indicated voltages between 24.5 and 28.0 volts may occur during low engine RPM conditions (Refer to note under LOW VOLTAGE ANNUNCIATION).

(Continued Next Page)

## **ELECTRICAL SYSTEM** (Continued)

### **ELECTRICAL SYSTEM MONITORING AND ANNUNCIATIONS**

#### **AMMETERS**

Current indication (AMPS) for both the main and standby batteries is provided at the bottom of the EIS bar (along the left margin of the MFD or PFD), labeled M BATT S. Main battery current is numerically displayed below the M. Main battery current greater than -1.5 amps is shown in white. Standby battery current is displayed numerically below the S. A positive current value (shown in white) indicates that the battery is charging. A negative current value (shown in amber) indicates that the battery is discharging. In the event the alternator is not functioning or the electrical load exceeds the output of the alternator, the main battery ammeter indicates the main battery discharge rate.

In the event that standby battery discharge is required, normal steady state discharge should be less than 4 amps. The STBY BATT annunciator will come on when discharge rates are greater than 0.5 amps for more than 10 seconds. After engine start, with the STBY BATT switch in the ARM position, the standby battery ammeter should indicate a charge showing correct charging of the standby battery system.

#### **STANDBY BATTERY ANNUNCIATION**

The STBY BATT annunciator will come on when discharge rates are greater than 0.5 amps for more than 10 seconds. This caution annunciation is an indication that the alternator and the main battery are not supplying the power that is required by the essential bus. If the condition causing the caution can not be resolved, flight should be terminated as soon as practicable.

(Continued Next Page)

## **ELECTRICAL SYSTEM (Continued)**

### **ELECTRICAL SYSTEM MONITORING AND ANNUNCIATIONS (Continued)**

#### **LOW VOLTAGE ANNUNCIATION**

A signal from the ACU, located inside the power distribution module, provides the trigger for a red LOW VOLTS annunciation shown on the PFD. LOW VOLTS is displayed when the main bus voltage measured in the power distribution module is below 24.5 volts. The LOW VOLTS warning annunciation is an indication that the alternator is not supplying the power that is required by the airplane. If the conditions causing the LOW VOLTS warning can not be resolved, nonessential electrical loads should be eliminated and the flight should be terminated as soon as practicable.

#### **NOTE**

During low RPM operation, with a high electrical load on the system, such as during a low RPM taxi, the LOW VOLTS annunciation may come on, the bus voltage values may turn red, and main battery ammeter discharge indications may occur. Under these conditions, increase RPM or decrease electrical loads to reduce demand on the battery.

In the event an overvoltage condition (or other alternator fault) occurs, the ACU will automatically open the ALT FIELD circuit breaker, removing alternator field current and stopping alternator output. The main battery will then supply current to the electrical system as shown by a discharge (negative number) on the M BATT ammeter. The LOW VOLTS annunciator will come on when the system voltage drops below 24.5 volts. Set the ALT FIELD circuit breaker to the ON position (push in) to energize the ACU. If the warning annunciation goes out and the main battery (M BATT) ammeter indicates positive current, normal alternator charging has resumed. If the annunciator comes on again, or the ALT FIELD circuit breaker opens again, an alternator malfunction has occurred. If the circuit breaker opens again, do not SET it to the ON position again. Have a qualified technician determine the cause and correct the malfunction. Turn off nonessential electrical loads and land as soon as practicable.

(Continued Next Page)

## **ELECTRICAL SYSTEM** (Continued)

### **ELECTRICAL SYSTEM MONITORING AND ANNUNCIATIONS** (Continued)

#### **LOW VOLTAGE ANNUNCIATION** (Continued)

The ALT FIELD circuit breaker may open on occasion during normal engine starts due to transient voltages. Provided that normal alternator output is resumed after the ALT FIELD circuit breaker is reset, these occurrences are considered nuisance events. If the ALT FIELD circuit breaker opens after reset, do not close again. Repeated occurrences indicate a problem with the electrical system that must be corrected by a qualified maintenance technician before flight.

#### **HIGH VOLTAGE ANNUNCIATION**

The HIGH VOLTS annunciator will come on when main or essential bus voltage is above 32.0 volts. This warning annunciation is an indication that the alternator is supplying too high of a voltage. The ALT MASTER switch should immediately be positioned to OFF (Refer to Section 3, Emergency Procedures, HIGH VOLTS ANNUNCIATOR COMES ON).

In the event a HIGH VOLTS condition occurs, the ACU will automatically open the ALT FIELD circuit breaker, removing alternator field current and stopping alternator output. The HIGH VOLTS annunciator is a warning that the ACU automatic alternator shutdown circuit is not operational and an action from the pilot is required to position the ALT MASTER to OFF.

(Continued Next Page)

## **ELECTRICAL SYSTEM** (Continued)

### **CIRCUIT BREAKERS AND FUSES**

Individual system circuit breakers are found on the circuit breaker panel below the pilot's control wheel. All circuit breakers on ESSENTIAL BUS, AVIONICS BUS 1 and AVIONICS BUS 2 are capable of being opened or disengaged from the electrical system by pulling straight out on the outer ring for emergency electrical load management. Using a circuit breaker as a switch is discouraged since the practice will decrease the life of the circuit breaker. All circuit breakers on ELECTRICAL BUS 1, ELECTRICAL BUS 2 and CROSSFEED BUS are not capable of being opened or disengaged.

The power distribution module uses three push-to-reset circuit breakers for the electrical bus feeders. A fast blow automotive type fuse is used at the standby battery. The standby battery current shunt circuit uses two field replaceable fuses located on the standby battery controller printed circuit board.

Most Garmin G1000 equipment has internal non-field replaceable fuses. Equipment must be returned to Garmin by an approved service station for replacement.

(Continued Next Page)

## **ELECTRICAL SYSTEM** (Continued)

### **EXTERNAL POWER RECEPTACLE**

A external power receptacle is integral to the power distribution module and allows the use of an external power source for cold weather starting or for lengthy maintenance work on electrical and avionics equipment. The receptacle is located on the left side of the cowl near the firewall. Access to the receptacle is gained by opening the receptacle door.

#### **NOTE**

Set the AVIONICS switches BUS 1 and BUS 2 to OFF if no avionics are required. If maintenance on the avionics equipment is required, a 28 VDC regulated and filtered external power source must be provided to prevent damage to the avionics equipment from transient voltages. Set AVIONICS switches BUS 1 and BUS 2 to OFF before starting the engine.

The following check should be made whenever the engine has been started using external power (after disconnecting the external power source).

1. MASTER Switch (ALT and BAT) - OFF
2. TAXI and LAND Light Switches - ON
3. Throttle Control - REDUCE TO IDLE
4. MASTER Switch (ALT and BAT) - ON (with taxi and landing lights turned on)
5. Throttle Control - INCREASE (to approximately 1500 RPM)
6. Main Battery (M BATT) Ammeter - CHECK (Battery charging, Amps Positive)
7. LOW VOLTS Annunciator - CHECK (Verify annunciator is not shown)

#### **WARNING**

**IF M BATT AMMETER DOES NOT SHOW POSITIVE CHARGE (+ AMPS), OR LOW VOLTS ANNUNCIATOR DOES NOT GO OFF, REMOVE THE BATTERY FROM THE AIRPLANE AND SERVICE OR REPLACE THE BATTERY BEFORE FLIGHT.**

## LIGHTING SYSTEMS

### EXTERIOR LIGHTING

Exterior lighting consists of navigation lights on the wing tips and the tip of the rudder, landing/taxi lights located on the left wing leading edge, a flashing beacon mounted on top of the vertical stabilizer, and a strobe light on each wing tip.

Two courtesy lights are recessed into the lower surfaces of each wing and provide illumination for each cabin door area. The switch for the courtesy lights is found on the pilot's overhead console. The rear dome light and under-wing courtesy lights share the same control switch. Pressing the rear dome light switch will make the lights come on and pressing it again will make the lights go out.

All other exterior lights are operated by switches found on the lighted switch panel to the left of the PFD. Exterior lights are grouped together in the LIGHTS section of the switch panel. To activate the BEACON, LAND (landing light), TAXI (taxi light), NAV, and STROBE light(s), place the switch in the up position. Circuit breakers for the lights are found on the lighted circuit breaker panel on the lower left instrument panel, below the PFD. Circuit breakers are grouped by electrical bus with BEACON and LAND on ELECTRICAL BUS 1 and TAXI, NAV and STROBE on ELECTRICAL BUS 2.

#### NOTE

The strobes and flashing beacon should not be used when flying through clouds or overcast; the flashing light reflected from water droplets or particles in the atmosphere, particularly at night, can produce vertigo and loss of orientation.

(Continued Next Page)

## LIGHTING SYSTEMS (Continued)

### INTERIOR LIGHTING

Interior lighting is controlled by a combination of dimmable crew area flood lighting, internally lit switch and circuit breaker panels, avionics panel lighting, standby instrument lighting, pedestal lighting, pilot control wheel map lighting and passenger area flood lighting.

Flood lighting is accomplished using two dimmable lights in the front crew area and one dome light in the rear passenger area. These lights are contained in the overhead console, and are controlled by dimmer controls for the front flood lights, and an on-off type push button switch for the rear dome light. The front flood lights can be rotated to provide directional lighting for the pilot and front passenger. The rear dome light provides for general illumination in the rear cabin area. The rear dome light and under-wing courtesy lights share the same switch.

Lighting of the switch panel, circuit breaker panel, engine controls and environmental control panel is accomplished by using internally lit panels. Rotating the SW/CB PANELS dimmer, found on the switch panel in the DIMMING group, controls the lighting level for both panels. Rotating the dimmer counterclockwise decreases light intensity from the highest level to off.

For airplane serials 18281869 and 18281876 thru 18282021 and 18282023 thru 18282045, pedestal lighting consists of three hooded lights found at various locations on the pedestal. Rotating the PEDESTAL light dimmer, found on the switch panel in the DIMMING group, controls the pedestal lights. Rotating the dimmer counterclockwise decreases light intensity from the highest level to off.

For airplane serials 18282022 and 18282046 and On, pedestal lighting is accomplished by using an internally lit panel for the cowl flap control lever, aux audio in jack and the 12 volt power outlet, along with a LED strip light located at the top of the pedestal. Lighting for the fuel selector valve is accomplished by a LED strip light incorporated into the bottom of the internally lit cowl flap control panel. Rotating the PEDESTAL light dimmer, found on the switch panel in the DIMMING group, controls the pedestal lights. Rotating the dimmer counterclockwise decreases light intensity from the highest level to off.

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## LIGHTING SYSTEMS (Continued)

### INTERIOR LIGHTING (Continued)

Avionics panel lighting consists of the PFD and MFD bezel and display lighting and Audio Panel lighting. Rotating the AVIONICS dimmer, found on the switch panel in the DIMMING group, controls the lighting level. Positioning the dimmer control in the off position (fully counterclockwise) causes the avionics displays to use internal photocells to automatically control the lighting levels. This is the recommended use of the avionics lighting for all day and lower lighting levels where lighting of the avionics bezels and keys is not required. In low to night lighting levels rotating the AVIONICS dimmer control clockwise from the off position places all avionics lighting level control to the AVIONICS dimmer control. This is the recommended use of avionics lighting for night and low lighting conditions to allow the pilot control of the avionics illumination levels as dark adaptation occurs.

Rotating the STBY IND dimmer, found on the switch panel in the DIMMING group, controls lighting of the standby airspeed indicator, attitude indicator, altimeter and non-stabilized magnetic compass. Rotating the dimmer counterclockwise decreases light intensity from the highest level to off.

Pilot's chart (map) lighting is accomplished by use of a rheostat and a light assembly, both found on the lower surface of the pilot's control wheel. The light provides downward illumination from the bottom of the control wheel to the pilot's lap area. To operate the light, first turn the NAV light switch ON, and then adjust the map light intensity using the knurled rheostat knob. Rotating the dimmer clockwise (when facing up) increases light intensity, and rotating the dimmer counterclockwise decreases light intensity.

Regardless of the light system in question, the most probable cause of a light failure is a burned out bulb. However, in the event any lighting systems fails to come on, check the appropriate circuit breaker. For the interior lights the PANEL LTS circuit breaker, and for the exterior lights the associated light function circuit breaker (i.e. landing light, LAND LT circuit breaker). If the circuit breaker has opened, and there is no obvious indication of a short circuit (smoke or odor), turn the affected lights OFF, reset the circuit breaker, and turn the lights ON again. If the circuit breaker opens again, do not reset until maintenance has been performed.

## **CABIN HEATING, VENTILATING AND DEFROSTING SYSTEM**

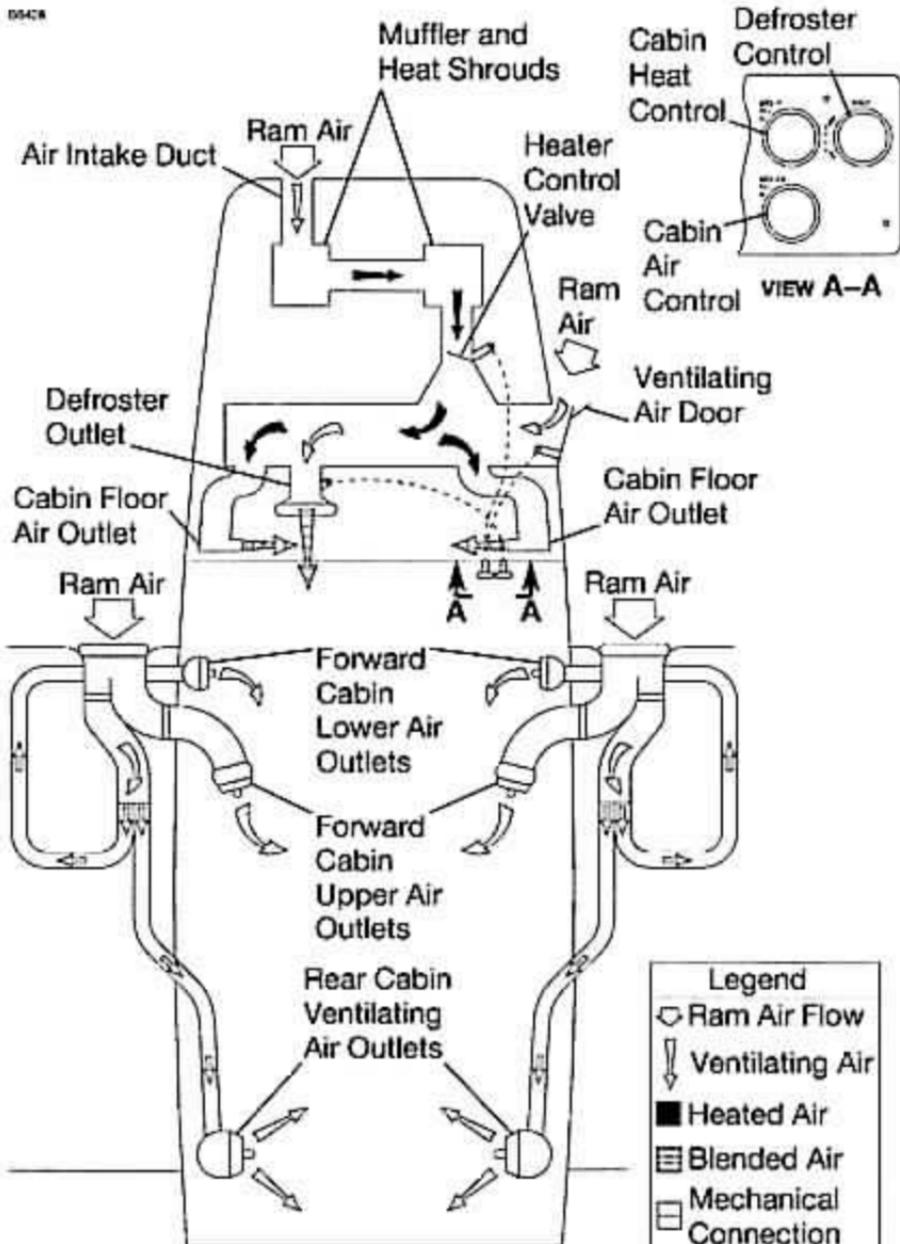
The temperature and volume of airflow into the cabin can be regulated by manipulation of the push-pull CABIN HT and CABIN AIR controls (Refer to Figure 7-8). Both controls are the double button locking-type and permit intermediate settings.

For cabin ventilation, pull the CABIN AIR knob out. To raise the air temperature, pull the CABIN HT knob out approximately 1/4 to 1/2 inch for a small amount of cabin heat. Additional heat is available by pulling the knob out farther; maximum heat is available with the CABIN HT knob pulled out and the CABIN AIR knob pushed full in. When no heat is desired in the cabin, the CABIN HT knob is pushed full in.

Front cabin heat and ventilating air is supplied by outlet holes spaced across a cabin manifold just forward of the pilot's and front passenger's feet. Rear cabin heat and air is supplied by two ducts from the manifold, one extending down each side of the cabin to an outlet just aft of the rudder pedals at floor level. Windshield defrost air is also supplied by two ducts from the cabin manifold outlet on top of the glareshield; therefore, the temperature of the defrosting air is the same as heated cabin air. A rotating control knob, labeled DEFROST, regulates the volume of air to the windshield. Turn the knob clockwise to ON and counterclockwise to OFF.

Separate adjustable ventilators supply additional air; one near each upper corner of the windshield supplies air for the pilot and front passenger, and two ventilators are available for the rear cabin area to supply air to the rear seat passengers. There are additional ventilators located in various positions in the cockpit.

**CABIN HEATING, VENTILATION AND DEFROSTING SYSTEM**



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Figure 7-8

## PITOT-STATIC SYSTEM AND INSTRUMENTS

The pitot-static system uses a heated total pressure (pitot) head mounted on the lower surface of the left wing, external static ports mounted on both sides of the forward fuselage and associated plumbing to connect the air data computer and the conventional pitot-static instruments to the sources.

The heated pitot system uses an electrical heating element built in the body of the pitot head. The PITOT HEAT control switch is found on the switch panel below the lower left corner of the PFD. The PITOT HEAT circuit breaker is found on the circuit breaker panel at the lower left side of the pilot panel.

A static pressure alternate source valve (ALT STATIC AIR) is located adjacent to the throttle control. The ALT STATIC AIR valve provides static pressure from inside the cabin if the external static pressure source becomes blocked.

If erroneous instrument readings are suspected due to water or ice in the pressure line going to the standard external static pressure source, the alternate static source valve should be pulled on.

Pressures within the cabin will vary with open heaters/vents and windows. Refer to Section 5, Figure 5-1 (Sheet 2), for the Airspeed Calibration, Alternate Static Source correction chart and Figure 5-2 for the Altimeter Correction, Alternate Static Source correction chart.

## VACUUM SYSTEM AND INSTRUMENTS

The vacuum system (Refer to Figure 7-9) provides the vacuum necessary to operate the standby attitude indicator. The system consists of one engine driven vacuum pump, a vacuum regulator, the standby attitude indicator, a vacuum system air filter, and a vacuum transducer. The vacuum transducer provides a signal to the engine display that is processed and displayed as vacuum on the EIS SYSTEM page. If available vacuum, from the engine-driven vacuum pump, drops below 3.5 in.hg., the LOW VACUUM annunciator will display in amber on the PFD.

### ATTITUDE INDICATOR

The standby attitude indicator is a vacuum-powered gyroscopic instrument, found on the center instrument panel below the MFD. The attitude indicator includes a low-vacuum warning flag (GYRO) that comes into view when the vacuum is below the level necessary for reliable gyroscope operation.

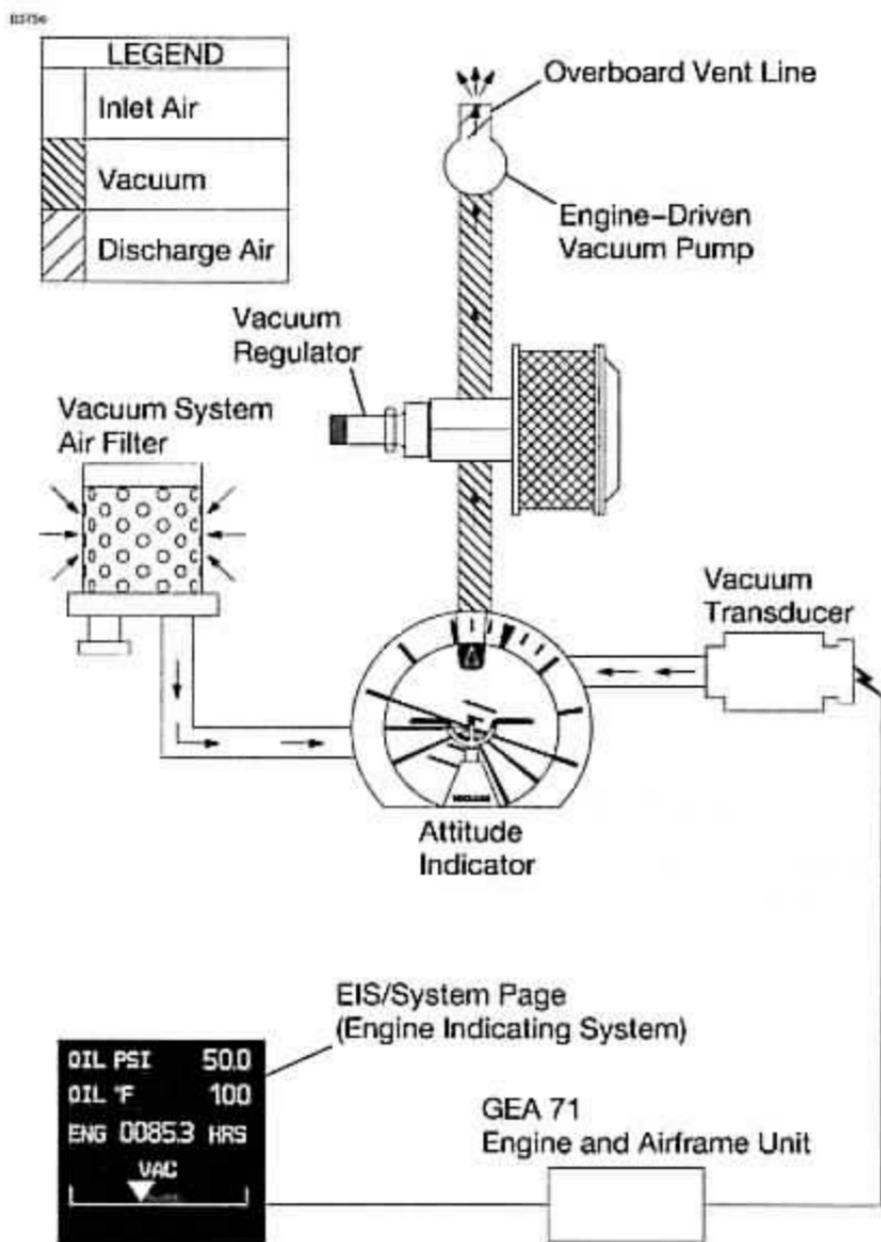
### VACUUM INDICATOR

The vacuum indicator is incorporated on the EIS SYSTEM page, found along the left side of the PFD during engine start or the left edge of the MFD during normal operation. During reversionary operation, the EIS bar appears along the left side of the operational display.

### LOW VACUUM ANNUNCIATION

A low vacuum condition is annunciated along the right side of the PFD by a amber LOW VACUUM annunciator.

## VACUUM SYSTEM



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Figure 7-9

## CLOCK/O.A.T. INDICATOR

A numerical time or clock window (based on GPS time) and an outside air temperature (O.A.T.) indicator window are provided along the lower edge of the PFD. The O.A.T. indicator uses an air temperature sensor located on top of the cabin.

## STALL WARNING SYSTEM

The airplane is equipped with a vane-type stall warning system consisting of an inlet in the leading edge of the left wing, which is electrically connected to a stall warning horn located in the headliner above the left cabin door. A 5-amp push-to-reset circuit breaker labeled WARN, on the left side of the circuit breaker panel, protects the stall warning system. The vane in the wing senses the change in airflow over the wing, and operates the warning horn at airspeeds between 5 and 10 knots above the stall in all configurations.

The airplane has a heated stall warning system, the vane and sensor unit in the wing leading edge is equipped with a heating element. The heated part of the system is operated by the PITOT HEAT switch, and is protected by the PITOT HEAT circuit breaker.

The stall warning system should be checked during the preflight inspection by momentarily turning on the MASTER switch and actuating the vane in the wing. The system is operational if the warning horn sounds as the vane is pushed upward.

## **STANDARD AVIONICS**

The Garmin G1000 Avionics System is an integrated flight control and navigation system. The system combines primary flight instruments, communications, airplane system information and navigational information all displayed on two color displays. The G1000 system consists of the following pieces of equipment:

### **GARMIN DISPLAY UNITS (GDU)**

Two identical units are mounted on the instrument panel. One, located in front of the pilot, is configured as a PFD. A second panel, located to the right, is configured as a MFD.

The PFD displays roll and pitch information, heading and course navigation information, plus altitude, airspeed and vertical speed information to the pilot. The PFD also controls and displays all communication and navigation frequencies as well as displaying warning/status annunciations of airplane systems.

The MFD displays a large scalable, moving map that corresponds to the airplane's current location. Data from other components of the system can be overlaid on this map. Location and direction of movement of nearby aircraft, lightning and weather information can all be displayed on the MFD. The MFD is also the principle display for all of the engine, fuel, and electrical system parameters.

The reversionary mode places the flight information and basic engine information on both the PFD and the MFD. This feature allows the pilot full access to all necessary information should either of the display screens malfunction.

(Continued Next Page)

## **STANDARD AVIONICS** (Continued)

### **AUDIO PANEL (GMA)**

The audio panel for the G1000 system integrates all of the communication and navigation digital audio signals, intercom system and marker beacon controls in one unit. It is installed on the instrument panel between the PFD and the MFD. The audio panel also controls the reversionary mode for the PFD and MFD.

#### **NOTE**

Use of the COM 1/2 function is not approved.

### **INTEGRATED AVIONICS UNIT (GIA)**

Two integrated avionics units are installed in the G1000 system. They are mounted in racks in the tailcone, behind the baggage curtain. These units act as the main communications hub linking all of the other peripheral parts to the GDU displays. Each unit contains a GPS receiver, a VHF navigation receiver, VHF communication transceiver and the main system microprocessors. The first GIA unit to acquire a GPS satellite 3-D navigation signal is the active GPS source.

### **ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS) AND MAGNETOMETER (GRS)**

The AHRS provides airplane attitude and flight characteristics information to the G1000 displays and to the integrated avionics units, which is located in the tailcone of the airplane. The AHRS unit contains accelerometers, tilt sensors and rate sensors that replace spinning mass gyros used in other airplanes. The magnetometer is located inside the left wing panel and interfaces with the AHRS to provide heading information.

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## **STANDARD AVIONICS (Continued)**

### **AIR DATA COMPUTER (GDC)**

The Air Data Computer (ADC) compiles information from the airplane's pitot/static system. The ADC unit is mounted behind the instrument panel, just forward of the MFD. An outside air temperature probe, mounted on top of the cabin, is connected to the ADC. The ADC calculates pressure altitude, airspeed, true airspeed, vertical speed and outside air temperature.

### **ENGINE MONITOR (GEA)**

The Engine Monitor is responsible for receiving and processing the signals from all of the engine and airframe sensors. It is connected to all of the CHT measuring sensors, EGT sensors, manifold pressure, RPM, fuel flow and to the fuel gauging system. This unit transmits this information to the engine display computers.

### **TRANSPONDER (GTX)**

The full-featured Mode S transponder provides Mode A, C and S functions. Control and operation of the transponder is accomplished using the PFD. The transponder unit is mounted in the tailcone avionics racks.

### **XM WEATHER AND RADIO DATA LINK (GDL)**

The XM weather and radio data link provides weather information and digital audio entertainment in the cockpit. The unit is mounted behind the instrument panel, just forward of the MFD. This unit communicates with the MFD on the high-speed data bus. XM weather and XM radio operate in the S-band frequency range to provide continuous uplink capabilities at any altitude throughout North America. A subscription to the XM satellite radio service is required for the XM weather and radio data link to be used.

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## **STANDARD AVIONICS** (Continued)

### **GFC 700 AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)**

Refer to the Garmin G1000 CRG for more information on system operation.

### **CONTROL WHEEL STEERING (CWS)**

The Control Wheel Steering (CWS) button, located on the pilot's control wheel, immediately disconnects the pitch and roll servos when activated. Large pitch changes while using CWS will cause the airplane to be out of trim. Retrim the airplane as necessary during CWS operation to reduce control forces or large pitch oscillations that may occur after releasing the CWS button.

#### **WARNING**

**WHEN THE AUTOPILOT IS ENGAGED IN NAV, APR OR BC OPERATING MODES, IF THE HSI NAVIGATION SOURCE IS CHANGED MANUALLY, USING THE CDI SOFTKEY, THE CHANGE WILL INTERRUPT THE NAVIGATION SIGNAL TO THE AUTOPILOT AND WILL CAUSE THE AUTOPILOT TO REVERT TO ROL MODE OPERATION. NO AURAL ALERT WILL BE PROVIDED. IN ROL MODE, THE AUTOPILOT WILL ONLY KEEP THE WINGS LEVEL AND WILL NOT CORRECT THE AIRPLANE HEADING OR COURSE. SET THE HDG BUG TO THE CORRECT HEADING AND SELECT THE CORRECT NAVIGATION SOURCE ON THE HSI, USING THE CDI SOFTKEY, BEFORE ENGAGING THE AUTOPILOT IN ANY OTHER OPERATING MODE.**

## GFC 700 SYSTEM SCHEMATIC

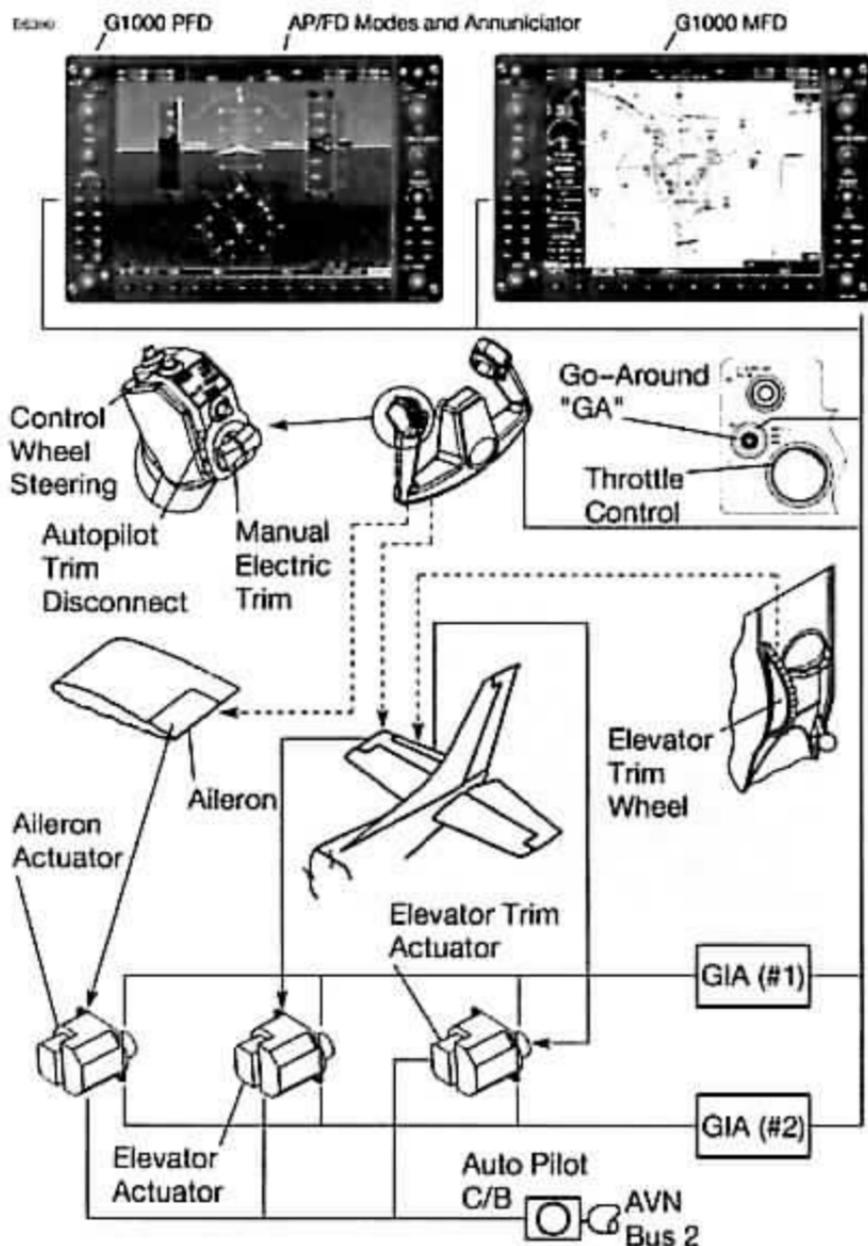


Figure 7-10

## **STANDARD AVIONICS** (Continued)

### **L3 COMMUNICATIONS WX-500 STORMSCOPE**

Refer to Section 9, Supplement 3, for operating information.

### **BENDIX/KING KTA 870 TRAFFIC ADVISORY/MULTI-HAZARD AWARENESS SYSTEM**

The Bendix/King KTA 870 Traffic Advisory/Multi-Hazard Awareness System provides the pilot with supplemental flight information through the G1000 system. This system includes a Traffic Advisory System (TAS) for air traffic and an Enhanced Ground Proximity Warning System (EGPWS) for terrain. The KTA 870 unit is mounted in the tailcone avionics racks. Refer to the Bendix/King KTA 870/KMH 880 Traffic Advisory System/Multi-Hazard Awareness System Pilot's Guide, Honeywell part number 006-18265-0000, Revision 3, dated January 2005, or later revision and the Garmin G1000 CRG for more information on system operation of the KTA 870 system.

#### **NOTE**

When TAS is enabled TIS will be unavailable.

## AVIONICS SUPPORT EQUIPMENT

Avionics cooling fans, antennas, microphone and headset provisions, power converter and static discharge wicks support the operation of the avionics equipment installations.

### AVIONICS COOLING FANS

Four DC electric fans provide forced air and ambient air circulation cooling for the G1000 avionics equipment. A single fan in the tailcone provides forced air cooling to the integrated avionics units and to the transponder. A fan located forward of the instrument panel removes air from between the firewall bulkhead and instrument panel, directing the warm air up at the inside of the windshield. Two additional fans blow air directly onto the heat sinks located on the forward sides of the PFD and MFD.

Power is provided to these fans when the MASTER (BAT) switch and the AVIONICS (BUS 1 and BUS 2) switch are all ON.

#### NOTE

None of the cooling fans will operate when the essential bus avionics equipment is being powered by the standby battery.

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## AVIONICS SUPPORT EQUIPMENT (Continued)

### ANTENNAS

Two combination VHF COM/GPS antennas are mounted on the top of the cabin. The dual-mode COM 1/GPS 1 antenna is mounted on the right side. Earlier production airplanes are configured with a tri-mode COM 2/GPS 2/XM antenna mounted on the left side, while later production airplanes are configured with a dual-mode COM 2/GPS 2 antenna mounted on the left side and a separate XM antenna mounted on top of the cabin. The combination VHF COM/GPS antennas are connected to the two VHF communication transceivers, the two GPS receivers in the integrated avionics units, and the GDL.

A blade-type navigation antenna is mounted on either side of the vertical stabilizer. This antenna provides VOR and glideslope signals to the VHF navigation receivers contained in the integrated avionics units.

The marker beacon antenna is mounted on the bottom of the tailcone. It provides the signal to the marker beacon receiver located in the audio panel.

The transponder antenna is mounted on the bottom of the cabin and is connected to the Mode S transponder by a coaxial transmission cable.

The L3 Communications WX-500 Stormscope antenna is mounted on the bottom of the tailcone and is connected to the L3 Communications WX-500 Stormscope receiver by an antenna cable assembly.

The Bendix/King Distance Measuring Equipment (DME) antenna (if installed) is mounted on the bottom of the tailcone and is connected to the Bendix/King DME receiver by a coaxial cable.

The Bendix/King Traffic Advisory System (TAS) (if installed) has two antennas. A directional antenna is mounted on top of the cabin forward of the COM antennas and an omni-directional antenna is mounted on the bottom of the tailcone. Both antennas are connected to the Bendix/King TAS receiver by coaxial cables.

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## AVIONICS SUPPORT EQUIPMENT (Continued)

### MICROPHONE AND HEADSET INSTALLATIONS

Standard equipment for the airplane includes a hand-held microphone, an overhead speaker, two remote-keyed microphone switches on the control wheels, and provisions for communications headsets at each pilot and passenger station.

The hand-held microphone includes an integral push-to-talk switch. This microphone is plugged in at the center pedestal and is accessible to both the pilot and front passenger. Pressing the push-to-talk switch allows voice transmission on the COM radios.

The overhead speaker is located in the center overhead console. Volume and output for this speaker are controlled through the audio panel.

Each control wheel contains a push-to-talk switch. This switch allows the pilot or front passenger to transmit on the COM radios using remote microphones.

Each seat position of the airplane has provisions for aviation-style headsets. Microphone and headphone jacks are located on each respective sidewall panel for communications between passengers and pilot. The system is designed so that microphones are voice activated. Only the pilot or front passenger can transmit through the COM radios.

#### NOTE

To ensure audibility and clarity when transmitting with the hand-held microphone, always hold it as closely as possible to the lips, then press the transmit switch and speak directly into it. Avoid covering the opening on back side of microphone for optimum noise canceling.

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## AVIONICS SUPPORT EQUIPMENT (Continued)

### AUXILIARY AUDIO INPUT JACK

An auxiliary audio input jack (AUX AUDIO IN) is located on the right side of the center pedestal (Refer to Figure 7-2). It allows entertainment audio devices such as cassette, compact disc, and MP3 players to play music over the airplane's headsets.

The signal from AUX AUDIO IN is automatically muted during radio communications or pilot selection of crew intercom isolation modes located on the audio panel. The AUX key on the audio panel does not control the AUX AUDIO IN signal. For a more complete description and operating instructions of the audio panel, refer to the Garmin G1000 CRG.

Since the entertainment audio input is not controlled by a switch, there is no way to deselect the entertainment source except to disconnect the source at the audio input connector. In the event of a high pilot workload and/or heavy traffic, it is wise to disable the entertainment audio to eliminate a source of distraction for the flight crew.

#### NOTE

- Passenger briefing should specify that AUX AUDIO IN (entertainment audio input) and Portable Electronic Device (PED) use is permitted only during the enroute phase of flight.
- Disconnect the cable from the AUX AUDIO IN jack when not in use.
- Use caution with audio cables in the cabin to avoid entangling occupants or cabin furnishings and to prevent damage to cables.

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## AVIONICS SUPPORT EQUIPMENT (Continued)

### 12V POWER OUTLET

A power converter, located on the cabin side of the firewall just forward of the right instrument panel, reduces the airplane's 28 VDC power to 12 VDC. This converter provides up to 10 amps of power to operate portable devices such as notebook computers and audio players. The power output connector (POWER OUTLET 12V -10A) is located on the center pedestal (Refer to Figure 7-2).

A second power outlet connector (POWER OUTLET 12V - 10A) is located in the right sidewall by the aft passenger seat. A switch located on the switch panel labeled CABIN PWR 12V controls the operation of the power outlets.

#### NOTE

- Charging of lithium batteries may cause the lithium batteries to explode.
- Take care to observe the manufacturer's power requirements prior to plugging any device into the 12 volt cabin power system connector. This system is limited to a maximum of 10 amps, or if equipped with two connectors 10 amps total.
- Use caution with power/adaptor cables in the cabin to avoid entangling occupants or cabin furnishings and to prevent damage to cables supplying live electric current.
- Disconnect power/adaptor cables when not in use.

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## AVIONICS SUPPORT EQUIPMENT (Continued)

### STATIC DISCHARGERS

Static dischargers are installed at various points throughout the airframe to reduce interference from precipitation static. Under some severe static conditions, loss of radio signals is possible even with static dischargers installed. Whenever possible, avoid known severe precipitation areas to prevent loss of dependable radio signals. If avoidance is impractical, minimize airspeed and anticipate temporary loss of radio signals while in these areas.

Static dischargers lose their effectiveness with age, and therefore, should be checked periodically (at least at every annual inspection) by a qualified technician.

## CABIN FEATURES

### EMERGENCY LOCATOR TRANSMITTER (ELT)

Refer to Section 9, Supplements 1 or 2, for ELT operating information.

### CABIN FIRE EXTINGUISHER

A portable Halon 1211 (Bromochlorodifluoromethane) fire extinguisher is installed in a holder on the floorboard between the front seats to be accessible in case of fire. The extinguisher is classified 5B:C by Underwriters Laboratories.

The extinguisher should be checked prior to each flight to ensure that the pressure of the contents, as indicated by the gage at the top of the extinguisher, is within the green arc (approximately 125 psi) and the operating lever lock pin is securely in place.

To operate the fire extinguisher:

1. Loosen retaining clamp(s) and remove extinguisher from bracket.
2. Hold extinguisher upright, pull operating ring pin, and press lever while directing the liquid at the base of the fire at the near edge. Progress toward the back of the fire by moving the nozzle rapidly with a side-to-side sweeping motion.

#### WARNING

**VENTILATE THE CABIN PROMPTLY AFTER SUCCESSFULLY EXTINGUISHING THE FIRE TO REDUCE THE GASES PRODUCED BY THERMAL DECOMPOSITION.**

3. The contents of the cabin fire extinguisher will empty in approximately eight seconds of continuous use.

Fire extinguishers should be recharged by a qualified fire extinguisher agency after each use. After recharging, secure the extinguisher to its mounting bracket.

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## **CABIN FEATURES** (Continued)

### **CARBON MONOXIDE DETECTION SYSTEM**

The carbon monoxide (CO) detection system consist of a single detector located behind the instrument panel, powered by the airplanes DC electrical system and integrated in the Garmin G1000 system with a warning annunciation and alert messages displayed on the PFD.

When the CO detection system senses a CO level of 50 parts-per-million (PPM) by volume or greater the alarm turns on a flashing warning annunciation, CO LVL HIGH, in the annunciation window on the PFD with a continuous tone until the PFD softkey below WARNING is pushed. It then remains on steady until the CO level drops below 50 PPM and automatically resets the alarm.

If the CO system detects a problem within the system that requires service, a CO DET SRVC message is displayed in the alerts window of the PFD. If there is an interface problem between the G1000 system and the CO system a CO DET FAIL message is displayed in the alerts window of the PFD.