# SECTION 7

## AIRPLANE AND SYSTEMS DESCRIPTION

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### XL-2 Airplane

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INTRODUCTION

This section describes the construction of the airplane and the construction, layout, and operation of its systems. Some of the equipment described in this section is optional, and may not be installed in all airplanes.

Refer to Section 9 (Supplements) for information on the description and operation of other optional systems or equipment.

AIRFRAME

The Liberty XL-2 is a two-place airplane of typical low-wing configuration. Seating is side by side.

The wings are of conventional semi-monocoque aluminum construction, and consist of main and rear spars, ribs, stringers, and skins. Single-slotted flaps of aluminum construction, consisting of a spar, ribs, and skins, are attached to the inboard rear of each wing by three hinges each. Aluminum ailerons, consisting of a spar, ribs, skins, and inboard and outboard mass balance weight assemblies, are attached to the outboard rear of each wing by two lower surface hinges.

The fuselage is of hybrid construction. A center section or "chassis" of welded 4130 chrome-moly steel tubing structure supports the cockpit and provides attachment points for the wings, landing gear, and engine mounts. The aircraft fuel tank and empennage are also secured to this structure.
The fuselage, including the vertical stabilizer, is constructed from molded composite material. The aluminum skin rudder is hinged to one side on the rudder skin. Pushrods activate the stabilator and rudder.

Carbon fiber material is used in both the skin and internal structure of the vertical fin. Metallic material is bonded into certain areas of the fuselage to meet lightning strike resistance requirements.

**WING FLAPS**

Large single-slotted flaps are installed on each wing. Three hinges offset below the wing’s lower surface support each flap.

A single electric actuator in the center fuselage operates a cross-tube, which in turn, operates both flaps. It is powered by the airplane primary electrical system via the “FLAP” circuit breaker.

![Figure 7-2 Wing Flaps](image-url)
**FLAP POSITION SWITCH**

To operate the flaps, a spring loaded, center flap control switch that is located on the avionics panel right side must be held in the "retract" or "extend" position. As flaps move, illuminating one of three indicators above the flap control switch provides position information. Position switches that are located in the flap actuation mechanism activate each indicator. As flaps reach an angle of deflection to be reported, the corresponding position switch is depressed, which in turn activates the appropriate position indicator. As flaps continue to move however, position switch pressure is released extinguishing the indicator. Flap motor operation continues as long as flap control switch pressure is held. At flap full extension or full retraction, the actuator disengages permitting actuator motor free spool operation for as long as flap control switch pressure is applied. At the end of travel, depending on direction selected, a full extension indicator or full retraction position indicator will be illuminated. For the standard aircraft, flap positions are marked at 0, 10, and 30 degrees. It is possible to release the flap control switch at each of these positions or any point between. Flaps will hold the position achieved when the flap control switch is released.

![Figure 7-3 Wing Flap Switch and Indicator](image)

Figure 7-3 Wing Flap Switch and Indicator
PRIMARY FLIGHT CONTROLS

**Pitch Control System**

The left (pilot) and right (passenger) control sticks are attached to a common control column assembly. Forward and aft movement of the control sticks move about a center bearing and operate a pushrod which transmits this motion aft (via a "tunnel" through the fuselage-mounted fuel tank) to an idler bell-crank. The bell-crank is installed aft of the fuel tank below the floor of the baggage compartment. From there, a second pushrod transmits pitch control movement to a control arm attached to the center of the stabilator torque tube, thus rotating the torque tube and stabilator up and down.

To prevent aerodynamic flutter, the stabilator is mass-balanced by a weight inside the fuselage attached to an arm protruding forward from the stabilator torque tube.

![Figure 7-4 Pitch Control System](image)

**Roll Control System**

Side to side movement of the control sticks rotate a torque tube extending aft to approximately mid-chord of the wings. A bell-crank arm at the aft end of the torque tube changes its rotation to a linear motion of two push rods extending upward and outboard to operate "rocker" bell-cranks installed in the steel tube fuselage center section. The rockers bear against identical rockers in the wing roots, which transmit the motion, via push rods and bell-cranks, to the left and right ailerons.
**YAW CONTROL SYSTEM**

Rudder pedal assemblies are provided for the pilot and copilot. Each pedal assembly includes an adjustment crank, located at the bottom of the instrument panel on each side of the cockpit, which allows the pedals to be adjusted forward or aft.

Forward and aft movement of either rudder pedal rotates a torque tube extending across the airplane below the instrument panel. A series of links, bell-cranks, and pushrods changes this rotation to linear motion and transmits it aft through the fuselage to the rudder drive arm, moving the rudder left and right in response to pedal inputs.
TRIM SYSTEM

A trim tab is hinged to the rear spar of each horizontal stabilator. These tabs are geared to move in the same direction as the stabilator itself, thus providing an "anti-servo" effect and generating consistent pitch control forces.

![Figure 7-7 Pitch Trim System](image)

The trim tabs are connected, via link rods and a linkage which moves about the stabilator torque tube at the stabilator root rib, to an electric screw-jack actuator in the lower aft fuselage. This allows the pilot to set the neutral or faired position of the trim tabs to correspond with any desired angle of the stabilators, thus providing an aircraft pitch trim function.

![Figure 7-8 Trim Tab](image)
**PITCH TRIM SWITCH AND INDICATOR**

The actuator is controlled by a switch on the cockpit center console. The switch is labeled “NOSE DOWN” and “NOSE UP”. An indicator on the center console displays trim position to the pilot. With the stabilator held neutral, pressing NOSE DOWN/NOSE UP on the switch causes the trailing edge of the anti-servo tab to move and the position light on the indicator to change.

In the event of trim malfunction, out-of-trim forces are such that the airplane can be safely landed, even if trimmed to the full nose-up or nose-down position.

Refer to Section 3 – *Emergency Procedures*

![Figure 7-9 Pitch Trim Switch and Indicator](image)
AIRPLANE CABIN / FLIGHT DECK ARRANGEMENT

**Entrance Doors / Windows**

A single large combination entrance door and window on each side provides access to the cockpit. The doors are top-hinged, swinging upward to open. Integral gas springs aid in opening the doors and holds them in the open position.

Forward and aft door latch pins in the doors engage pin bushings in the fuselage. An interior and exterior door handle on either side operates the door latches.

**WARNING**

Prior to engine run-up and take-off, ensure the aircraft’s doors are properly closed, latched, and locked, by verifying each door pin (2 in each door) is seated in its respective receptacle by completing the following: For each door, with the door closed and handle in the fully closed position, apply pressure to the lower door frame (not the window glass), near the forward pin and then near the aft pin. If the applicable pin is seated in its receptacle, the door should not move and there should be no gap between the lower edge of the door and the fuselage. If movement or a gap is present, open and shut the door until each pin is properly seated in its receptacle.

If a door is opened or opens in-flight, it could result in the possible departure of the door from the aircraft, with potential damage to the aircraft that could result in a loss of aircraft performance. If a door is opened or opens in-flight, refer to door open in flight procedure in section 3 – emergency procedures and land as soon as practical.
The Liberty XL-2 instrument panel is subdivided into the upper left area, which contains the flight instruments (pitot-static and gyro instruments); the lower left area, containing essential switches and the integrated engine instrument display system, described later in this section; center area, containing the avionics “stack,” and the far right area, containing the circuit breaker panel.

The six primary flight instruments are installed in the “standard T” arrangement directly in-front of the pilot. The arrangement comprises six primary instruments: airspeed indicator, attitude indicator, altimeter, turn coordinator, directional gyro (heading indicator), and vertical speed indicator. The picture below is for reference only and is not intended to represent actual locations or availability of equipment shown.
**Center Console**

The center console panel, a small angled panel, directly below the avionics stack, accommodates the fuel pump mode switch, engine alternate induction air control, cabin heat, and trim indicator.

The center console between the pilot and passenger seat contains the throttle lever, parking brake lever and/or fingerbrakes with parking brake lever, the electric pitch trim switch, and the emergency fuel shutoff valve.
CABIN HEAT / DEMIST

Cabin heating is accomplished by means of a passive system (no thermostatic control) supplying heated air for both heating the cabin and windshield demist. The cabin heat/demist system consists of a heater muff (heat exchanger) around the engine exhaust muffler, a heater control valve, air ducting for distribution, windshield demist vents, adjustable leg vents, and a cable control for turning the entire system “on” and “off.”
Heating is accomplished by directing outside fresh air through the heat exchanger, then through the heater control valve, and then distributing the ‘heated’ air through ducting to the occupants and windshield demist vents. The windshield demist system is automatically activated when the cabin heat control knob is pulled. Partial air flow to the windshield demist vents will occur anytime one or both of the adjustable leg vents are open. For full air flow to the windshield demist vents, both adjustable leg vents should be closed. The adjustable leg vents are directionally controllable and are located on the lower sides of center console near the inboard leg of each occupant. Very little air flow is generated during idle ground operations; therefore, cabin temperature (flow rates) will vary with aircraft movement, altitude, and airspeed.

**SEATS**

The Liberty XL-2’s seats are integral to the fuselage structure, and are not adjustable. The pilot and copilot rudder pedals are adjustable, in-flight or on the ground. The cushions are secured to the seats by hook-and-loop (“Velcro”) fasteners. Ensure cushions do not obstruct flight controls.

**NOTE**

Rudder pedals should only be adjusted on the ground to ensure proper positioning for takeoffs and landings.

**BAGGAGE COMPARTMENT**

The baggage compartment is contiguous with the passenger cabin, and extends from aft of the seatbacks to the rear baggage compartment bulkhead. The seat backs are sculpted to allow ease of access to the 4’ x 3’ x 2’ baggage area. Maximum allowable baggage weight is 100 pounds. The loads should be distributed over the baggage bay area at a maximum 29 lbs/ft². Reduced weights may be required to conform to aircraft weight and balance limitations.

**NOTE**

It is the pilot’s responsibility to ensure the baggage bay is loaded at and around the center of gravity (refer to Section 6 - Weight and Balance).
Figure 7-13 Seats / Baggage Compartment
CABIN SAFETY EQUIPMENT

This section contains information about the cabin safety systems

**SEATBELTS / SHOULDER HARNESS**

Seat belts and dual shoulder harnesses are provided for each seat. The seat belts are secured to the aircraft structure on either side of the seat. A single fitting in the seat back joins and secures each of the shoulder harness straps to the seatback structure directly behind each seat.

Each seat belt has a standard “lift-to-release” metal-to-metal buckle. Adjusters on each belt allow it to be tightened, and to place the buckle at an approximate central location.

The left and right shoulder harness straps for each seat have buckles to adjust for pilot comfort and asymmetric end fittings. To fasten the shoulder harness, slide both fittings over the “tongue” of the seat belt buckle end, and then fasten the seat belt. When the shoulder harness fittings are properly aligned, the harness straps should not cross above the seat belt buckle.

**EMERGENCY EGRESS HAMMER / FIRE EXTINGUISHER**

If the doors cannot be opened during an emergency egress, utilize the safety hammer, which is located in the passenger’s seat back storage compartment and is within easy reach of the pilot. The safety hammer can be used to smash out enough of the canopy door to exit the aircraft. Another standard feature in the Liberty XL-2 is the fire extinguisher, mounted behind the passenger’s seat within easy reach of the pilot.

**LANDING GEAR**

**MAIN GEAR**

The Liberty XL-2 has fixed tricycle landing gear. The main gear legs are fabricated from heat-treated aluminum alloy and they are fastened to the fuselage center section by saddle fittings, bolts, and bushings. Either main landing gear assembly can be removed from the airplane without affecting the opposite main gear.

The 5.00 x 5 main gear wheels are installed aluminum axles. Each incorporates a hydraulic disk brake. Aerodynamic wheel fairings (optional) can be installed one each main landing gear assembly. Normal tire pressure is 50 psi.
NOSE GEAR
The nose landing gear leg is fabricated from heat-treated steel. It is secured to the fuselage center section by a bushing and a bolt. It may be removed from the airplane without affecting the main landing gear.

The nose wheel is 5.00 x 5; normal tire pressure is 50 psi. The nose wheel assembly is secured to the nose gear leg by way of a ball bearing (the “caster” bearing) to allow rotation up to 80 degrees left or right for nose wheel steering. A stack of six spring-steel washers provides a constant force that applies to the friction dampener located on the nose wheel. An aerodynamic wheel fairing of composite material may be installed on the nose landing gear.

Ground steering of the Liberty XL-2 is achieved by differential application of the left and right main landing gear brakes.

BRAKE SYSTEM (FINGER BRAKES)
Individual disk brakes are installed on each main landing gear wheel. Individual master cylinders installed beneath the cockpit center console operate the brakes. Brake lines extend from each master cylinder to the brake calipers. Two brake levers are installed in the center console. To operate the brakes, pull either or both levers aft.

A ratchet type parking brake lever is installed in the center console between the brake levers. To apply the parking brake, pull both brake levers firmly aft, then pull the center parking brake lever aft and release the left and right brake levers. To release the parking brake, pull both brake levers firmly aft and allow the parking brake lever to spring forward.

Brake fluid is stored in a common reservoir for both master cylinders, located on starboard side of the engine bay aft of the firewall. Remove upper cowling for access to service the reservoir with MIL-PRF-5606 type hydraulic fluid.
**BRAKE SYSTEM (TOE BRAKES)**

Individual disk brakes are installed on each main landing gear wheel. Two pairs of master cylinders located on the vertical rudder bars and attached to toe brake pads above the rudder pedals for the pilot and co-pilot operate the brakes. Brake lines extend from the pilots rudder pedals through the parking brake valve located in the center console to the brake calipers. Under pressure, brake fluid is sent via this line to the bake calipers. To operate the right or left brake depress the corresponding toe brake pad on either the pilot or copilot’s rudder pedal.

A parking brake lever is installed in the center console that is attached to the parking brake valve. To apply the parking brake, depress both right and left toe brake pads and move the parking brake lever to the ON position, then release the pressure on the toe brakes. To release the parking brake, move the parking brake lever to the OFF position. To ensure the parking brake is released properly, depress and release toe brake pedals.
To ensure that parking brake is set, depress brake pedals and then move parking brake lever into the ON position. If brake pedals are not pressed down, brake fluid is not being held in the caliper.

Do not move the parking brake lever to the ON position in flight and particularly during landing with brake pedals depressed because you will lock the pressure in the brake lines. This could cause locked brakes and blown tires on landing.

Brake fluid is stored in a common reservoir for all four master cylinders. This reservoir is located on the starboard side of the engine bay aft of the firewall. Fluid from this reservoir flows first to the co-pilot’s brake cylinders and then to the pilot’s brake cylinders before going to the parking brake. To access the brake fluid reservoir, remove the upper cowling and service the reservoir with MIL-PRF-5606 type hydraulic fluid.
When depressing the brake pedals use extreme caution that your feet are on the pedals only and not on the brake cylinders or other support structures.

**ENGINE**

The Liberty XL-2 is powered by a Teledyne Continental IOF-240-B, Full Authority Digital Engine Control (FADEC), four-cylinder, horizontally-opposed, air-cooled, naturally-aspirated, fuel-injected engine rated at 125 HP at 2800 RPM.

**FADEC SYSTEM**

The engine is equipped with a Full Authority Digital Engine Control (FADEC) System for continuously monitoring and controlling ignition timing, fuel injection timing, and fuel mixture. The microprocessor-based FADEC system monitors engine operating conditions and then automatically sets the fuel mixture and ignition timing accordingly for any given power setting. Consequently, the FADEC equipped engine does not require magnetos and eliminates the need for a manual fuel/air mixture control.
The FADEC System provides control in both specified operating conditions and fault conditions. The system is designed to prevent adverse changes in power or thrust. In the event of loss of primary aircraft-supplied power, the engine controls continue to operate using a Secondary Power Source (SPS). As a control device, the system performs self-diagnostics to determine overall system status and conveys this information to the pilot by various indicators on the Health Status Annunciator (HSA) panel.

The FADEC System is able to withstand storage temperature extremes and operate at the same capacity as a non-FADEC equipped engine in extreme heat, cold, and high humidity environments.

The basic components of the FADEC System includes Two Electronic Control Units (ECUs), Health Status Annunciator (HSA) (panel installed in the cockpit), and FADEC Sensor Set.

**Electronic Control Units (ECU)**

An Electronic Control Unit (ECU) is assigned to a pair of engine cylinders. Since the engine has four cylinders, there are two ECUs, one unit for every pair of cylinders. The ECUs control the fuel mixture and spark timing for respective engine cylinders; ECU 1 controls opposing Cylinders 1 and 2; ECU 2 controls Cylinders 3 and 4.

Each ECU is divided into upper and lower portions. The lower portion contains an electronic circuit board; the upper portion houses the ignition coils. The electronic circuit board contains two, independent microprocessor controllers which serve as control channels. During engine operation, one control channel is assigned to operate a single engine cylinder. Therefore, one ECU can control two engine cylinders, one control channel per cylinder.

The control channels are independent and there are no shared electronic components between the control channel pair within one ECU. However, if a control channel fails, the other control channel in the pair within the same ECU is capable of operating both its assigned cylinder and the other opposing engine cylinder as backup control for fuel injection and ignition timing.
Each channel controls its assigned cylinder in a manner that will yield optimum performance for the current operating conditions to prevent exceeding normal operating parameters. The fuel mixture may be enriched or leaned and ignition timing may be retarded to minimize the extent of limit excursion for the given parameter. In this respect, a FADEC-controlled engine is different from a non-FADEC engine in that an individual cylinder can be leaned or enriched by its control channel without affecting the other cylinders.

![Figure 7-17 Electronic Control Unit (ECU)](image)

**HEALTH STATUS ANNUNCIATOR (HSA)**

The Health Status Annunciator (HSA) is located on the primary instrument panel towards the right side, above the electrical switches and aircraft annunciator lights. The HSA provides indications of primary or secondary (emergency) power malfunctions, erroneous sensor indications, fuel pump malfunctions, and possible misfiring cylinders.

Each HSA annunciator light is associated with a specific system condition. Illumination of a given light indicates that the associated condition has been detected and some response by the pilot may be necessary. In the event annunciator illumination occurs, follow the steps in Chapter 3 - Emergency Procedures for the specific indication.

Illumination of the FADEC CAUTION light indicates pressure or temperature sensor failures, abnormal pressure, or temperature above limits, misfire/cylinder not firing, or cylinder not enabled (ignition switch OFF or not in BOTH position).

Illumination of the FADEC WARN light indicates that more than one cylinder is faulted. The FADEC WARN light is always preceded and/or accompanied by illumination of the FADEC CAUTION light.
The EBAT FAIL light illuminates when there is a fault condition with the backup (secondary) power supply. The charging current into the backup battery is too high, indicating: low charge, bad battery, the wire charging the backup battery is not connected, or the primary power source is off or has failed.

Illumination of the PPWR FAIL light indicates that the FADEC system is drawing power from the backup power supply because the primary electrical power supply has been interrupted, the backup power supply potential is higher than the primary buss or the primary power source is off or has failed.

Refer to Section 3 - Emergency Procedures and Section 4 - Normal Procedures for explanation of FUEL PUMP indications.

![Figure 7-18 Health Status Annunciators (HSA)](image)

**FADEC SENSOR SET**

All essential components of the FADEC system are connected using a low voltage harness. This harness acts as a signal transfer buss interconnecting the two Electronic Control Units (ECUs) with aircraft power sources, the Ignition Switch, Speed Sensor Assembly (SSA), Health Status Annunciator (HSA), temperature and pressure sensors. The fuel injector coils and all sensors, except the speed sensor assembly, fuel pressure, and manifold pressure sensors, are hardwired to the low voltage harness.
FADEC IGNITION SYSTEM

The ignition system of the IOF-240-B engine is part of the FADEC. Unlike conventional magneto ignition systems, its timing is electronically determined by the FADEC computers. The ignition subsystem of the FADEC is not self-powered, but requires an adequate supply of DC power from the airplane electrical system.

To provide necessary system redundancy, the FADEC has two power sources. In the airplane, the primary power source, labeled FADEC A, is the airplane’s main power distribution bus, which is powered by the alternator and/or the airplane’s primary battery.
The secondary power source, labeled FADEC B, is powered by a separate battery. When the airplane’s main power distribution bus is energized, a charging circuit constantly recharges the FADEC B battery, thus indirectly powering the FADEC B bus.

In the event of failure of the airplane primary DC system, including discharge of the primary battery, the secondary battery will power the FADEC, via the FADEC B connection, for a period sufficient to locate and land at a suitable airport.

**WARNING**

**ENGINE MAY CONTINUE TO OPERATE NORMALLY FROM THE EMERGENCY BATTERY FOR UP TO 60 MINUTES IF THE BATTERY IS PROPERLY MAINTAINED AND FULLY CHARGED. PLAN TO LAND WELL WITHIN 60 MINUTES FROM ILLUMINATION OF EBAT FAIL AND PPWR FAIL ANNUNCIATORS.**

The FADEC is fully operational when powered by the FADEC A bus, the FADEC B bus, or both. Switches are provided to check operation on both systems before flight. PPWR FAIL and EBAT FAIL captions on the FADEC Health Status Annunciator illuminate to confirm operation of the FADEC from either or both power sources.

**FADEC FUEL INJECTION SYSTEM**

Fuel from the airframe fuel system is routed to the intake of the engine-driven fuel pump, which is mounted on the front left side of the engine and gear-driven from the camshaft.

When fuel entering the engine driven pump, it passes through a centrifugal separator. The separator is where fuel vapor is removed and returned to the airplane fuel tank. Next, it passes into the pump element, where its pressure is increased. Since pump effectiveness varies with engine speed, the pump is designed to produce more pressure and flow than is required by the engine fuel injectors.

An adjustable relief valve regulates pump output at lower RPM, while an adjustable internal orifice regulates pressure at high RPM. These adjustments are critical to proper function of the FADEC system.
Fuel pressure is displayed on the VM-1000FX Integrated Engine Instrument System, or can be checked by connecting a portable (“laptop”) computer to the data output of the FADEC system which is located underneath the circuit breaker panel on the co-pilots side of the airplane. Diagnostic software and the required connecting cable are available from Teledyne Continental Motors (TCM).

![Figure 7-20 FADEC Fuel Injection System](image)

**FUEL SYSTEM**

The Liberty XL-2 airframe fuel system incorporates a fuselage-mounted fuel tank, fuel strainer assembly (“gascolator”), electric fuel boost pump, cockpit fuel shutoff valve, and associated plumbing. There are no fuel tanks installed in the wings.

Additional fuel system components installed on the engine include an engine-driven fuel pump, fuel distribution manifold, inline fuel filter, and fuel injection nozzles.
**FUEL BOOST PUMP AND SWITCH**

Operation of this pump can be selected automatically by the engine FADEC system, or manually by the pilot using the Boost Pump Mode Switch (BPMS) on the center console. Automatic (FADEC) operation is controlled via a fuel pump relay. Manual operation (BPMS “ON” position) bypasses the relay and powers the electric fuel pump directly. The boost pump mode switch is a three position switch and is located in the middle of the lower center console.

**FUEL QUANTITY INDICATOR**

The fuel indicating system includes a capacitance-type probe in the fuel tank and an indicator on the left instrument panel. The capacitance probe has no moving parts. The fuel indicating system is powered by the airplane electrical system via the Fuel “Gauge” circuit breaker.

**FUEL SHUTOFF VALVE**

The fuel shutoff valve is installed at the rear of the cockpit center console and has OFF and ON positions. The button in the center of the valve handle must be lifted to allow the valve to be moved to the OFF position.
The fuel tank sump drain valve is the press-to-open type. It is located at the bottom of the fuel tank, and is accessible via an opening in the fuselage belly fairing. If the valve is rotated after opening, it will remain open to allow aircraft de-fueling.

An additional valve of the same type, accessible through a similar opening slightly farther forward in the fuselage belly fairing, allows fuel and any accumulated water or sediment to be drained from the fuel system strainer ("gascolator").

Both fuel drain valves are installed in the system between the fuel tank and the fuel shutoff valve. Thus, all tank contents may be drained from either valve regardless of shutoff valve position.

**FUEL VENTING**

The fuel system is vented to the atmosphere via a vent line extending from the fuel vent/return fitting at the top of the tank to a labeled opening in the bottom of the fuselage. This vent is considered non-icing.

![Figure 7-22 Fuel Drain and Vent Locations](image_url)
ENGINE SYSTEMS

ENGINE OIL SYSTEM

Engine oil is stored in a pressed-steel tank (sump) of 6-quart capacity, attached to the bottom of the engine crankcase. An oil filler tube, with a filler cap incorporating an oil dipstick, is installed on the oil tank. The gear-type engine oil pressure pump is located at the rear of the engine, with its pickup tube extending into the oil tank. It is driven by the engine camshaft drive gear.

Oil leaves the oil pump under pressure. A regulating valve allows some oil to return to the oil tank to maintain oil pressure within limits at varying engine speeds. Oil from the pump is routed to an oil filter and oil cooler adapter on the left side of the engine accessory case.

The oil cooler routes oil to the externally mounted oil cooler, which incorporates a “Vernatherm” thermostatically controlled valve to control oil temperature, and to the oil filter element. An integral bypass valve will open in the event of blockage of the oil cooler or oil filter. The oil temperature sensor for the engine instruments is also located on the oil cooler adapter. The oil pressure sensor is mounted on the firewall and plumbed to the engine.

Oil galleries and drilled passages inside the engine route oil to the crankshaft and camshaft main bearings and to the hydraulic valve lifters.

Drillings inside the crankshaft route oil to the connecting rod lower bearings. Oil escaping from the main and connecting rod bearings creates an oil mist inside the crankcase that lubricates the connecting rod upper bearings and cylinder walls, as well as the cam lobes and lower faces of the lifters. In addition, oil nozzles on the main bearings direct a jet of oil at the undersides of the pistons to cool them.

Oil in the hydraulic lifters is routed via the hollow valve pushrods to the cylinder rocker boxes, where it lubricates the rockers, valve stems, and valve guides before returning, via the pushrod housings, to the crankcase. Any excess oil in the crankcase returns, via the large-diameter opening between the crankcase and the sump, to the oil sump.
ENGINE COOLING

Air enters the engine cowling through inlets on either side of the propeller. Baffles between the cylinders, forward of the front cylinders, and vertically behind the rear cylinders, force this high-pressure, low temperature air through the cylinder and cylinder head cooling fins. A duct in the rear baffle directs high-pressure, low temperature air to the firewall-mounted oil cooler.

Low-pressure warm air leaves the engine compartment via an opening at the rear of the lower cowling, forward of the firewall. Oil cooler exhaust air is also discharged at this location.

ENGINE AIR INDUCTION SYSTEM / ALTERNATE AIR CONTROL

Air at ambient temperature is admitted through an air filter assembly incorporating an oil-saturated textile filter element to trap and remove dust and other foreign material, located above the port-side cylinders. In case this filter becomes blocked or iced, pilot selection of alternate air admits warm unfiltered air from the tube forward of the number four cylinder exhaust stack. The alternate air control is a pull-push knob located in the lower left center console.

ENGINE EXHAUST

The exhaust system includes individual exhaust pipes from each cylinder and a single muffler located below the engine. A single overboard discharge pipe extends through the right side of the lower cowling.

The exhaust pipes from each side of the engine are connected to the left and right ends of the muffler, which is cylindrical in shape and which is installed below the engine with its axis running across the airplane. Slip joints in the exhaust pipe accommodate the dimensional changes that result from temperature changes in the exhaust system. Clamps secure the exhaust pipes to the muffler.

A single discharge pipe extends downward and to the (airplane’s) right from the muffler for overboard discharge of exhaust gases.

Reference Section 5 – Performance of this manual.

ENGINE CONTROLS

Engine operation is controlled by a single throttle lever in the cockpit center console. It operates in the conventional sense: moving the lever forward increases engine power.

Additional engine controls include a conventional key-type ignition switch with OFF, R, L, BOTH, and a spring-loaded START position, and two lever-lock type switches to control primary and secondary power to the FADEC system.
The Teledyne Continental Motors IOF-240B engine installed in the Liberty XL-2 is equipped with a “PowerLink™” Full Authority Digital Engine Control system (FADEC) manufactured by the Aerosance Corporation.

This system controls both ignition and fuel delivery functions for the engine, thus replacing conventional components such as magnetos (ignition) and carburetor or typical fuel injection system (fuel delivery). Advantages of the FADEC system include digitally controlled and optimized spark timing and duration and digitally controlled fuel delivery to each cylinder. FADEC incorporates closed-loop feedback for each individual cylinder to maximize both power output and fuel economy. During all operating modes (including startup, idle, takeoff, climb, cruise, descent, and landing), the FADEC system provides single-lever power control and optimized engine parameters.

**ENGINE DATA MANAGEMENT - VM1000FX**

All engine instruments in the Liberty XL-2 are consolidated onto the VM1000FX, a single electronic display on the lower left instrument panel. They include RPM, percent power, manifold pressure, oil pressure and oil temperature, fuel pressure, electrical system voltage and alternator output amperage, and individual graphic display of exhaust gas temperature (EGT) and cylinder head temperature (CHT) for each cylinder. The engine instrument display is powered by the airplane primary electrical system via the “VM1000FX” circuit breaker.

Five push buttons (buttons 1 through 5, from left to right) below the display panel control various functions. The VM1000FX will provide the pilot a “flashing signal” signifying an out-of-limits condition for any engine parameter that it monitors.

**TACHOMETER**

The tachometer system provides both a full sweep graphic analog display and a four place digital display. Color range marks provide a quick reference to monitor normal, caution and red line engine RPM. The digits in the center of the tachometer read engine speed with a resolution of 1 RPM. Allow about 3 seconds for RPM indications to stabilize after RPM changes.

A warning alert activates whenever the engines redline is reached. The RPM display will flash until this condition is corrected. When the engine is not running, the tachometer digital display reads the total accumulated engine hours to a maximum of 5999.9 hours. Engine hours are accumulated any time RPM is greater than 1500.
Figure 7-23 Engine Indicating - VM1000FX

**PERCENT POWER**

Percent power is displayed by both a full sweep graphic display and a digital display. The digital display also gives a numeric reading of percent power. The color range marks provide a quick reference to monitor changes in power. Percent power information is useful during all stages of flight and can be used as a guide when setting cruise power.

The percent power gauge shows the current power output of the engine in percentage of brake horsepower. The range of the indicator is 0 - 100. The maximum percent power is 100 and a warning will activate whenever this value is reached or exceeded.

**MANIFOLD PRESSURE GAUGE**

The manifold pressure system provides both a full sweep graphic analog display and a three place digital display with resolution to 0.1 inches of Mercury. The full sweep graphic display resolution is 1-inch of Mercury. The color range marks provide a quick reference to manifold pressure when making rapid power changes.

**OIL PRESSURE AND TEMPERATURE**

Both oil pressure and oil temperatures are displayed continuously in two separate full sweep graphic and digital areas.

As oil pressure rises, the graph size increases proportionately. The digital display reads out in 1 pound per square inch (psi) increments to a maximum of 99 psi. A warning alert activates whenever the engine’s oil pressure reaches the redline condition. The display will flash until there is correction to the redline condition.
Oil temperature is displayed both graphically and digitally. As oil temperature rises, the graph size increases proportionately. The digital display reads out in 1 degree Fahrenheit increments to a maximum of 300 degrees. If the oil temperature rises above redline, the system captures the event and the display is flashed until the problem is corrected.

**CYLINDER ANALYZER OPERATION - CHT AND EGT**

The engine analyzer system displays all cylinder information (such as cylinder head temperature and exhaust gas temperature). The display is both graphical and digital, and is referred to as the diamond graph display. Color reference marks provide for cylinder head green, yellow, and redline temperatures.

The default mode for the diamond graph display is the normal mode. In this mode the system displays CHT between the green, yellow and red range marks, left to right, cylinders 1 through 4. EGT graphics are displayed above the CHT redline marks. A defective CHT or EGT probe will leave the respective graph blank. A flashing CHT graph indicates a cylinder is too hot or is being shock cooled.

The high resolution mode is selected by pressing button 1 while in the normal mode. In this mode the entire diamond graph display is temporarily used for high resolution monitoring of EGT. The display can be returned to the normal mode by pressing button 1 again. Notice that left and right brackets appear on the sides of the graphs when in high resolution mode.

The digital display shows the temperatures for each EGT and CHT pair and periodically shows the cylinder number (i.e. E1 C1). A warning message is shown if a cylinder has reached red line temperature (i.e. H2 for hot cylinder 2), or is being shock cooled (i.e. C3 for cooled cylinder 3). The default for the digital display is peak display mode (i.e. P1 means cylinder 1 EGT is the hottest, H3 means cylinder 3 CHT is the hottest). Select any combination by pressing button 2 as described below.

<table>
<thead>
<tr>
<th>Display Mode</th>
<th>Cylinder Numbers</th>
<th>Probes Displayed</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cylinder 1 Pair</td>
<td>E1 C1</td>
<td>EGT 1 &amp; CHT 1</td>
</tr>
<tr>
<td>Cylinder 2 Pair</td>
<td>E2 C2</td>
<td>EGT 2 &amp; CHT 2</td>
</tr>
<tr>
<td>Cylinder 3 Pair</td>
<td>E3 C3</td>
<td>EGT 3 &amp; CHT 3</td>
</tr>
<tr>
<td>Cylinder 4 Pair</td>
<td>E4 C4</td>
<td>EGT 4 &amp; CHT 4</td>
</tr>
<tr>
<td>Peak Mode</td>
<td>P# C#</td>
<td>Max EGT/CHT</td>
</tr>
</tbody>
</table>

(Highest EGT and CHT may not be from the same cylinder)
ELECTRICAL SYSTEM MONITORING

The VM1000FX displays volts and amps both graphically and digitally. Color range marks provide a reference for levels. For a more detailed description of the operation of the voltmeter and ammeter, refer to Electrical System later in this section.

VM1000FX AUTOTRACK SYSTEM OPERATION

The 'Autotrack' system is designed to reduce pilot workload by monitoring engine parameters for deviations. Subtle changes may occur in engine parameters that can precede major problems. 'Autotrack' provides automatic alerts if such changes occur, allowing the pilot to analyze the situation and take appropriate action.

When to use 'Autotrack':

- **Climb** - Activate during climb to alert periodically as CHT and/or Oil Temperature increases.
- **Cruise** - Activate during cruise to alert if any parameter begins to drift from the selected starting point.
- **Descent** - Activate during descent to alert to increasing manifold pressure.

How to use 'Autotrack':

1. Stabilize the aircraft. Set up desired power (RPM and manifold pressure). Allow the engine time to stabilize (i.e., engine temps and pressures, etc.).
2. Press button 3. The 'Autotrack' indicator will activate in the display and the system will begin tracking the engine's performance from this point.

The 'Autotrack' system is now armed and monitoring for engine deviation from the values present when it was activated. To cancel, simply press button 3 again to extinguish the 'Autotrack' indicator. Re-arm again at any time.

Any important alert condition, (i.e., low oil pressure, high CHT, etc.) automatically cancels 'Autotrack' mode.

AUTOTRACK ALERT INDICATIONS

If any engine parameter deviates beyond the initial set point, the system will flash the corresponding graphic display and the 'AUTOTRACK' indicator.
If the deviation is large enough, a graphic pointer (circular sweep displays only) will show where the parameter was before the deviation occurred. This allows the pilot to evaluate the magnitude of the deviation and take appropriate action.

To shut off the alert condition, return the parameter to its previous value (example: adjusting manifold pressure due to a climb) or simply press button 3 to shut off the ‘AUTOTRACK’ system.

**VM1000FX FLIGHT DATA RECORDER SYSTEM OPERATION**

The integrated engine instrument display incorporates a Flight Data Recorder that stores certain engine operating parameters for each flight.

Data may be retrieved from the Flight Data Recorder in-flight or after shutdown. It will be retained, even when the airplane electrical power is removed, until overwritten by the next flight.

Minimum and maximum values are automatically recorded during the flight and can be reviewed at any time before the next flight. Actual time the engine is running is also recorded for all the time above 1500 RPM.

**How to use 'Flight Data Recorder':**

1. Press button 5. The first set of data displayed is flight minimums encountered (i.e., lowest oil pressure, lowest voltage, amperage, etc.). The RPM digital display shows the actual flight hours and tenths
2. Press button 5 again. The next set of data is flight maximums encountered (i.e., max CHT, max Oil Temp, max RPM, etc.).
3. Press button 5 again. The Flight Data Recorder display is shut off. This will also occur in approximately 20 seconds if no button is pressed.

**VM1000FX ADDITIONAL FUNCTIONS**

Pressing and holding button 1 while initially applying power to the engine instrument display system (turning airplane master switch ON) changes the graphic display from “sweep” mode, in which all pointer segments up to and including the currently displayed value are visible, to “pointer” mode, in which only the single pointer segment closest to the current value is displayed. The system will retain the chosen mode until it is changed by once again pressing and holding Button 1 during power-up.

Button 4 is used in conjunction with fuel flow indicating modes. This button, the fuel flow, and the totalizer feature are not implemented in this installation.
PROPELLER

The MT Propeller Installation consists of a two-blade, fixed pitch, wood composite propeller. The Spinner Assembly includes spacer and supporting hardware (P-1026). The model is MT175R127-2Ca and manufactured by MT-Propeller Entwicklung GmbH.

Torque values should be checked after initial 25 hours of operation, and 50 hours thereafter, or more frequently if the airplane has been moved from humid to dry conditions. Consult the Aircraft maintenance manual for detailed procedures to torque the propeller bolts.

Figure 7-24 MT-Propeller and Footsteps

FOOTSTEPS

The Footstep Assembly consists of two (port and starboard) steel weldments and two (port and starboard) carbon hardpoints along with supporting hardware. The footstep weldments are fastened to the exterior fuselage as a foothold for ease of accessibility into the cabin. The footstep hardpoints are bonded inside the fuselage footwells to reinforce the fuselage.
ELECTRICAL SYSTEM

**PRIMARY BATTERY**

The primary battery is of the recombinant-gas type and has a nominal capacity of 24 ampere-hours (Ah). It provides power for engine starting, is a backup source of power in case of alternator failure, and helps damp electrical system fluctuations. The primary battery is connected to the airplane electrical system (with the exception of the engine starting circuit) via a 70-ampere circuit breaker. It is secured to a battery and electrical equipment shelf in the aft fuselage which is accessible by removing the aft baggage compartment closeout. This type of battery is considered maintenance-free.

**SECONDARY BATTERY**

The secondary battery is a maintenance-free recombinant-gas type, has a nominal capacity of 12-ampere hours (Ah). This battery is secured to the battery and electrical equipment shelf in the aft fuselage. Its purpose is to provide emergency backup power to the engine FADEC system, Attitude Indicator, and Turn Coordinator in the event of loss of primary power (alternator and primary battery). As such, its output is entirely dedicated to the FADEC B system and back up flight instruments (Attitude Indicator and Turn Coordinator). The secondary battery will operate the FADEC B system for up to 60 minutes after loss of all other aircraft power under worst-case conditions (high engine RPM and fuel flow requirement).

During normal electrical system operation the secondary battery is continually recharged by the airplane primary electrical system via the SPSC (Standby Power Source Circuit) circuit breaker.

Charging of the secondary battery is monitored and controlled by the FADEC Health Status Annunciator (HSA).

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**WARNING**

The secondary battery does not power the airplane electric fuel boost pump. Simultaneous loss of airplane primary electrical power and the engine driven fuel pump will result in engine stoppage.
ENGINE MAY CONTINUE TO OPERATE NORMALLY FROM THE EMERGENCY BATTERY FOR UP TO 60 MINUTES IF THE BATTERY IS PROPERLY MAINTAINED AND FULLY CHARGED. PLAN TO LAND WELL WITHIN 60 MINUTES FROM ILLUMINATION OF EBAT FL AND PPWR FL ANNUNCIATORS.

Figure 7-25 Electrical Block Diagram
**ALTERNATOR SYSTEM**

The primary source of power for the airplane electrical system is a 60-ampere alternator installed on the right forward side of the engine and driven by a V-belt installed around a pulley on the main engine shaft.

An alternator control unit (ACU) controls the output of the alternator to maintain voltage and current within its operating limits. The ACU provides over voltage protection. If output voltage from the alternator exceeds 15 VDC, the ACU will automatically disconnect the alternator from the airplane electrical system. If the over voltage was momentary, the ACU may be reset by cycling the ALT half of the airplane master switch OFF, then ON.

**MASTER SWITCH**

A two-part “split” master switch controls application of battery and alternator power to the airplane main electrical distribution bus. Moving the BATT half of the switch to the ON position completes a circuit to ground the coil of the primary battery contactor (relay) installed on the battery and electrical equipment shelf, closing the contactor and connecting the battery to the airplane electrical system.

Moving the ALT half of the switch to ON completes the circuit from the 5-ampere “ALT” circuit breaker on the main distribution bus to the alternator control unit (ACU), thus energizing the alternator. The BATT and ALT sections of the master switch are mechanically interconnected such that the BATT switch must be ON in order for the ALT switch to be ON, thus ensuring that the battery will be connected to the main distribution bus any time the alternator is in operation. However, the ALT section of the switch may be turned OFF while the BATT section remains ON, allowing the alternator to be turned off while the battery continues to provide power to the electrical system.

**AVIONICS POWER SWITCH**

All avionics units installed in the Liberty XL-2 receive power from individual circuit breakers on the avionics bus. This bus is connected to the main distribution bus via a 25-ampere circuit breaker and a master avionics relay.
The master avionics relay is controlled by the panel-mounted avionics master switch and is of the “fail operational” type, i.e., the relay is of the normally closed type. When power is applied to the airplane main distribution bus and the avionics master switch is in the OFF position, the relay is energized to the open position, thus removing power from the avionics bus. When the avionics master switch is moved to the ON position, the avionics master relay relaxes to its normally closed position, applying power to the avionics bus. Failure of the relay or the avionics master switch will also result in power being applied to the avionics bus; in this instance, individual avionics units can still be de-powered by operation of their individual ON-OFF switches.

**VOLTMETER / AMMETER**

Voltmeter and ammeter functions are provided as part of the Integrated Engine Instrument Display System, described briefly earlier in this section.

Both the voltmeter and ammeter display include an analog (pointer) and digital indication. Each has an alarm function, which will cause the display to flash in case of operation outside the normal range.

During and immediately after engine start-up, both displays may indicate current and/or voltage in the Yellow arc portion of their displays. This is normal and the indicators should move into the Green arc (within 30 seconds) as the batteries come up to a fully charged condition.

Voltage is displayed both graphically and digitally. Color range marks provide a reference for voltage levels. As voltage rises, the graph size increases proportionately. The system has a built-in warning system that flashes the graph when system voltage is out of nominal range (either too low or too high).

The ammeter is a load meter type device, i.e., it displays the amount of current delivered by the alternator to the main distribution bus. In case of alternator failure, the rate of battery discharge will not be displayed.

Amperage is displayed both graphically and digitally. Color range marks provide a reference for amperage levels. As amperage rises, the graph size increases proportionately. The digital readout displays amperage at one amp resolution.

The ammeter functions as an 'alternator load meter' displaying current flow FROM the alternator TO the aircraft electrical system. Selection of large electrical load items should cause an increase in ammeter indication.
The system has a built-in warning that flashes if alternator output exceeds preset limits. This situation may occur at very low engine idle speeds on the ground; this is considered normal under those circumstances.

**NOTE**

*There is an additional Voltmeter located in the Clock/OAT gauge.*

**ALT FAIL ANNUNCIATOR**

An amber ALT FAIL annunciator on the instrument panel illuminates to indicate that the alternator is not delivering reliable power to the airplane electrical system.

If the alternator disconnects automatically due to a momentary over voltage condition, it may be reset by cycling the ALT side of the airplane master switch OFF, then ON. If the alternator comes back online, the annunciator will extinguish. Continued illumination of the ALT FAIL annunciator indicates that the alternator has failed, and the airplane should be landed as soon as practicable to avoid loss of primary electrical power to the engine and FADEC system.

**CAUTION**

*Reduce electrical load and turn off all unnecessary electrical equipment after alternator failure. If primary power cannot be brought online, the FADEC system will draw power from the backup battery once the aircraft’s primary battery discharges to a level near or below the backup battery. The FADEC backup battery will power the FADEC for up to 1 hour. The time the aircraft is operating on backup battery should be counted from the time the PPWR FL lamp comes on the HSA panel. Land the airplane as soon as practical.*

A push-to-test switch is provided to test the ALT FAIL annunciator. This switch tests the annunciator lights only, not the warning circuitry in the ACU.

**CIRCUIT BREAKERS AND FUSES**

Most electrical circuits in the Liberty XL-2 are protected by circuit breakers on the main distribution bus and avionics bus. These circuit breakers will trip (“pop”) when the associated circuit has experienced an over-current condition. They can be manually tripped by pulling the circuit plunger.

To reset a tripped circuit breaker, wait approximately 15 seconds for the circuit breaker to cool, then push its plunger in. If it trips again do not make further attempts to reset it in-flight; notify maintenance.
Fuses installed at various locations in the airplane protect some non-critical circuits. Fuses are of the inline type. Refer to the following table for fuse locations, circuits, and ratings.

Some fuses are not accessible by the pilot.

<table>
<thead>
<tr>
<th>Fuse #</th>
<th>Device (Circuit)</th>
<th>Fuse Location</th>
<th>Fuse Rating (Amps)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Clock</td>
<td>At Battery Relay</td>
<td>1</td>
</tr>
<tr>
<td>2</td>
<td>FADEC B Power</td>
<td>At Backup Battery</td>
<td>10</td>
</tr>
<tr>
<td>3</td>
<td>Alternator Fail Annunciation</td>
<td>At Alternator</td>
<td>5</td>
</tr>
<tr>
<td>4</td>
<td>Starter Engaged Annunciation</td>
<td>At Starter Relay</td>
<td>1</td>
</tr>
<tr>
<td>5</td>
<td>Avionics Master Relay</td>
<td>Circuit Breaker Panel</td>
<td>2</td>
</tr>
<tr>
<td>6</td>
<td>VM1000FX Voltage Sense</td>
<td>Behind the Circuit Breaker Panel</td>
<td>1</td>
</tr>
<tr>
<td>7</td>
<td>Voltage Regulator</td>
<td>At Battery Relay</td>
<td>5</td>
</tr>
</tbody>
</table>

Table 7 - 1 Fuse Locations
**EXTERIOR LIGHTING**

Exterior lighting includes combination position, navigation, and strobe (anti-collision) lights on each wingtip and a landing/taxi light installed in the airplane nose.

Each of the two navigation/position and strobe light units include a red (port wing) or green (starboard wing) navigation light visible from the front and from the side, a white position light visible from the side and the rear, and a strobe light visible through over 180 degrees on each side of the airplane.

The navigation/position lights are controlled by an instrument panel switch. The airplane electrical system powers these lights via the “NAV/POS” circuit breaker.

The strobe lights are powered by a high voltage power supply installed on the aft baggage bay bulkhead. An instrument panel switch powered by the airplane electrical system via the “STROBE” circuit breaker controls primary DC power to the strobe light power supply.

**INTERIOR LIGHTING**

Interior lighting includes internal illumination of many primary flight instruments, post lighting of additional instruments, internal lighting of the integrated engine instrument display, and an overhead LED floodlight. In addition, internal and panel lighting is provided for optional avionics equipment and indicators.

All interior lighting except the overhead floodlight is powered by the airplane electrical system via the “INST” circuit breaker. The overhead light is powered via the “CABIN” circuit breaker.

Electro-luminescent devices operating on small amounts of 120 VAC current provide the internal lighting for some instrument lights and legends. A solid-state inverter provides this current. Other internal instrument lighting uses low-voltage DC. The “Upper” INSTRUMENT LIGHT dimmer on the instrument panel regulates power to the primary input of the inverter, low-voltage lights, and post lights. The “Lower” INSTRUMENT LIGHT dimmer controls the VM1000FX display, GMA 340, and CLOCK/OAT indicator.

An on-off switch located on the light assembly controls the overhead CABIN LED floodlight.
CABIN VENTILATION

Two NACA-type flush inlets on either side of the fuselage admit ambient air to adjustable cooling vents in the cabin sidewalls. The direction of these vents can be adjusted by the pilot and passenger, and their airflow can be regulated or turned off.

STALL WARNING SYSTEM

A stagnation probe (stall warning vane) on the port wing leading edge is wired to a warning device installed behind the instrument panel. The device will sound approximately 5 to 10 knots above airplane stalling speed to warn the pilot of an impending stall.

The stall warning system is powered by the airplane primary electrical system via a 2-amp circuit breaker.

PITOT-STATIC SYSTEM

The pitot-static system provides pitot and static air pressure to operate the airspeed indicator, altimeter, vertical speed indicator, and optional altitude encoder.

A combination electrically heated probe installed below the port wing provides pitot pressure to the airspeed indicator and static pressure to all the pneumatically operated instruments. Drain loops in the pneumatic lines to both the pitot and static connections on the probe serve to prevent any ambient moisture from blocking the system or damaging any instruments. If necessary to drain these loops, they are accessible when the fuselage belly fairing is removed.

AIRSPEED INDICATOR

The airspeed indicator is a pneumatic instrument operated by pitot and static pressure. It registers indicated airspeed (IAS) in knots, and requires no aircraft power for operation. It is located in the upper left position of the “standard T” layout. See Figure 7-27 Primary Instrument Panel.

ALTIMETER

The altimeter is a pneumatic instrument operated by static pressure. It indicates airplane altitude above mean sea level (MSL) by means of three hands. The largest hand indicates hundreds of feet; the smaller hand indicates thousands of feet; and the smallest hand (a narrow line from the center of the instrument to an inward-pointing triangle at the edge of the instrument) indicates tens of thousands of feet.
A setting knob at the 7 o'clock position and a setting window (“Kollsman window”) at the 3 o'clock position allow the altimeter to be set to the local barometric pressure. It is recommended that rotation of the barometric adjustment control knob results in a movement of both the pressure setting scale and the altimeter pointers. The altimeter reading should be compatible with the setting on the barometric adjustment scale.

The altimeter requires no aircraft power for operation and is located at the top right of the “standard T” layout. See Figure 7-27 Primary Instrument Panel.

If a second altimeter is needed it will be normally located down and to the left of the fuel quantity gauge. Other optional locations may be utilized. See Figure 7-27 Primary Instrument Panel.

**Vertical Speed Indicator**

The vertical speed indicator is a pneumatic instrument, operated by static pressure that indicates airplane vertical speed, in feet per minute, up to 2000 FPM for either a climb or descent. Vertical rates in excess of this value will “peg” the indicator, but will not cause damage. Normal indication will resume when airplane vertical speed is within the instrument’s normal operating range.

The Vertical Speed Indicator does not require any aircraft power for operation. It is installed in the lower right position of the “standard T” layout. See Figure 7-27 Primary Instrument Panel.
**PITOT HEAT SWITCH**

The pitot heat switch is located to the left of the Battery/Master switch. Activation of the switch applies power to the heater unit in the pitot blade. The light on the switch indicates power is not being supplied to the pitot heat relay, because of the switch being placed in the “off” position. Use of pitot heat is only recommended in suspected icing conditions.

**PITOT HEAT FAIL LIGHT**

Above and slightly to the left of the pitot heat switch is the pitot heat annunciator light. The LED illuminates when there is an undercurrent condition in the pitot heat circuit with the pitot heat switch placed in the “ON” position. The light can be tested and adjusted for bright or dim operation using the “BRIGHT/DIM/TEST” switch, directly to the right of the indicator.

**ALTERNATE STATIC SOURCE**

The alternate static source is a redundant system to the pitot blade static port. It is used when the blade static source has become inoperative for some reason. It is located to the left of the center console, above the pilots right knee, under the edge of the instrument panel. When the pilot selects the alternate static source, the instruments relying on the static pressure may operate slightly differently (Refer to Section 5 for calibration chart).

**AVIONICS AND NAVIGATION**

**MAGNETIC COMPASS**

A magnetic standby compass is installed near the top center of the windshield. No aircraft power is required for operation of the compass; however, internal compass lighting is controlled by the “Upper”instrument dimmer.

**ATTITUDE INDICATOR**

The attitude indicator (“artificial horizon”) is a gyroscopic instrument that indicates the pitch and bank attitude of the airplane. The airplane primary electrical system electrically powers the attitude indicator via the “ATT” circuit breaker.

An adjustment knob at the 6 o’clock position allows the aircraft symbol in the instrument display to be positioned vertically to compensate for changes in aircraft cruise pitch attitude and/or pilot eye position.
A red flag appears either to warn the pilot of a loss of electrical power or to indicate the gyro is providing unreliable information. When electrical power is first applied, up to three minutes may elapse before the flag is removed from view to allow for proper gyro operation. The flag will appear immediately when power is removed. During loss of primary aircraft power, the Attitude indicator will receive electrical power from the secondary power source. No pilot action is required.

The attitude indicator is located in the top center position of the “standard T” layout. See Figure 7-27 Primary Instrument Panel.

**DIRECTIONAL GYRO**

The directional gyro, or heading indicator, is a gyroscopic instrument that registers aircraft heading. It is electrically powered by the airplane primary electrical system via the “DG” circuit breaker.

A “push to turn” adjustment knob at the seven o-clock position allows the instrument to be set to correspond with the magnetic compass. This adjustment must be made by the pilot upon initial startup of the gyro, and at intervals thereafter to compensate for drift and precession.

A red flag appears either to warn the pilot of a loss of electrical power or to indicate the gyro is providing unreliable information. When electrical power is first applied, up to three minutes may elapse before the flag is removed from view to allow for proper gyro operation. The flag will appear immediately when power is removed.

The directional gyro is installed in the lower center position of the “standard T” layout. See Figure 7-27 Primary Instrument Panel.

**TURN COORDINATOR**

The turn coordinator is a gyroscopic instrument that registers aircraft rate of turn. A tilted rate gyro is used so that the initial bank used to begin a turn will also be displayed on the turn coordinator. In a standard rate turn (3 degrees/sec) the wingtips of the miniature airplane in the instrument will be aligned with the white index marks.

The turn coordinator gyro is electrically powered by the airplane primary electrical system via the “TURN” circuit breaker. A red flag appears either to warn the pilot of a loss of electrical power or to indicate the gyro is providing unreliable information. During loss of primary aircraft power, the Turn Coordinator will receive electrical power from the secondary power source. No pilot action is required.
A weighted ball, moving in a curved tube filled with damping liquid, indicates aircraft coordination or displacements about the longitudinal axis. The ball requires no aircraft power.

The turn coordinator is located at the lower left of the “standard T” layout See Figure 7-27 Primary Instrument Panel.

**NAVIGATION/COMMUNICATION SYSTEMS**

See AFM Supplements for installed equipment.

**AUDIO SYSTEM**

The Garmin GMA 340 audio control panel, is located in the top position of the avionics stack and protected by a 5 amp circuit breaker labeled “AUDIO”. It provides audio amplification, audio selection, marker beacon control, and a voice activated intercom system for headsets and microphones. The system allows audio switching for up to three transceivers, COM 1, COM 2, and COM 3, and five receivers, NAV 1, NAV 2, ADF, DME, and MKR. Push buttons select the receiver audio source provided to the headphones. A fail-safe mode connects the pilot headphone and microphone to COM 1 if power is removed or if the MIC selector switch is turned off.

**HEADSET AND MICROPHONE INSTALLATION**

The airplane is equipped with provisions for two headsets with integrated microphones. The microphone headsets use a remote push to talk switch located on the top of the pilot and co-pilot/passenger control sticks. The headset and microphone (MIC) power jacks are located in the headliner above and behind the pilot, co-pilot/passenger seats. Individual audio selector switches on the audio control panel controls audio for the headsets. By using the selected receiver volume controls, the pilot or copilot controls the audio volume to their headsets. There is an auxiliary music input jack.

![Figure 7-28 Garmin GMA 340 Audio Panel](image-url)
TRANSPONDER

The airplane is equipped with a single Garmin GTX 327 ATC Mode A/C (identification and Altitude) transponder with squawk capability. The transponder system consists of the integrated receiver/transmitter control unit, an antenna, and an altitude encoder. The receiver/transmitter receives interrogations from a ground based secondary radar transmitter, and then transmits to the interrogating ATC center. Digitized altitude information is provided by an altitude digitizer (encoder) plumbed into the airplane static system. The transponder control provides active code display, code selection, IDENT button, and test functions. The display is daylight readable and protected by a 5-amp circuit breaker labeled “XPNDR”. The transponder antenna is located on the under side of the fuselage just aft of the belly panel.

Figure 7-29 Garmin GTX 327

HOUR METER

The airplane is equipped with an hour meter to record engine operating time. The hour meter is located in the lower right side of the circuit breaker panel, and is protected by a 1-amp fuse. It is wired in such a way that positive oil pressure will enable the hour meter.

DIGITAL CLOCK / OAT

A digital electric clock is installed in the instrument panel below and a little to the left of the turn coordinator. It is powered directly from the airplane battery, via a remote 2-amp fuse, so that it continues to keep correct time when the airplane electrical system is not energized. A digital Outside Air Temperature gauge is integrated in the digital clock. The clock also integrates a voltmeter. To switch the display between Volts and OAT, push the center top button. The clock functions are controlled using the bottom two buttons.
EMERGENCY LOCATOR TRANSMITTER

One of two types of Emergency Locator Transmitter (ELT) is available for the XL2 airplane. The ELT 121.5/243.0 MHz and the 121.5/406 MHz. The next two sections describe each of these transmitters.

**EMERGENCY LOCATOR TRANSMITTER 121.5/243.0 MHz**

A self-contained Emergency Locator Transmitter (ELT) is installed aft of the mid-fuselage bulkhead and is accessed through the baggage bay closeout. It will function automatically in the event of sudden impact or excessive deceleration forces and will broadcast an internationally recognized distress signal on frequencies of 121.5 MHz and 243.0 MHz for a minimum of 72 hours after activation. The ELT has a dedicated crash-resistant antenna (flexible whip) installed on the upper fuselage.

Forces generated by hard landings, taxiing over rough surfaces, inadvertent bumps during ground handling, minor “hangar rash” collisions, etc., may be sufficient to activate the ELT. It is good practice to monitor 121.5 MHz before engine shutdown to ensure that the ELT is not operating. An annunciator and switch on the instrument panel monitor and control operation of the ELT. A flickering annunciator light indicates that the ELT is operating.
ELT batteries must be replaced every 12 months, or earlier if prior to battery expiration date, the ELT fails an operational check. The battery must also be changed if 50% of the battery life has expired or it has been in use for more than 1 cumulative hour. The battery expiration date shall be marked externally on the ELT case. Only the Duracell MN1300, or PC 1300, “D” size, Alkaline Manganese Dioxide batteries (six required) are approved for the main unit. The Duracell DL 1/3 NB, Lithium Cell battery, is approved for the remote unit (one required).

**NOTE**

Avoid unnecessary ELT operational checks, as any operation of the ELT reduces its battery life.

![ELT Location Diagram](image)

**Emergency Locator Transmitter 121.5/406 MHz**

A self-contained Emergency Locator Transmitter (ELT) is installed aft of the mid-fuselage bulkhead and is accessed through the baggage bay closeout. It will function automatically in the event of sudden impact or excessive deceleration forces and will broadcast an internationally recognized distress signal on frequencies of 121.5 MHz and 406.0 MHz for a minimum of 72 hours after activation. The ELT has a dedicated crash-resistant antenna (flexible whip) installed on the upper fuselage.
Forces generated by hard landings, taxiing over rough surfaces, inadvertent bumps during ground handling, minor “hangar rash” collisions, etc., may be sufficient to activate the ELT. It is good practice to monitor 121.5 MHz before engine shutdown to ensure that the ELT is not operating. An annunciator and switch on the instrument panel monitor and control operation of the ELT. A flickering annunciator light indicates that the ELT is operating.

ELT batteries must be replaced every five years or earlier if prior to battery expiration date, the ELT fails an operational check. The battery must also be changed if 50% of the battery life has expired or it has been in use for more than one cumulative hour. The battery expiration date shall be marked externally on the ELT case. Use only an Artex part number 452-6499 battery. Replacement batteries are obtained in kits containing all parts and labels that must be replaced with the battery. See Artex battery replacement kit 455-0012.

Avoid unnecessary ELT operational checks, as any operation of the ELT reduces its battery life.
Figure 7-33 ELT Location
ELT Operational Check

(PER FAA ORDER 7310.3s, Chapter 3-3-7)

ELT operation on 121.5MHz and either 243MHz or 406MHz may be tested for no more than three sweeps of the distress tone and only during the first five minutes of each hour.

1. Place cushioned support under the fuselage next to the rear access panels.
2. Remove aft baggage compartment closeout to access ELT.
3. Disconnect ELT antenna coaxial cable from fitting on ELT unit.
4. Turn on BAT/MASTER and AVIONICS MASTER switches.
5. Tune COM receiver to 121.5 MHz and ensure receiver audio is selected to headphones.
6. During time period from 00 to 05 minutes past any hour, press ELT “ON” button.
7. Monitor headphones for no more than 3 sweeps of ELT tone.
8. Press ELT “RESET” button. Verify that tone ceases and light in ELT annunciator panel extinguishes.
9. Turn off BAT/MASTER and AVIONICS MASTER switches.
10. Reconnect ELT antenna coaxial cable
11. Replace aft baggage compartment closeout.
12. Make required entry in aircraft records.