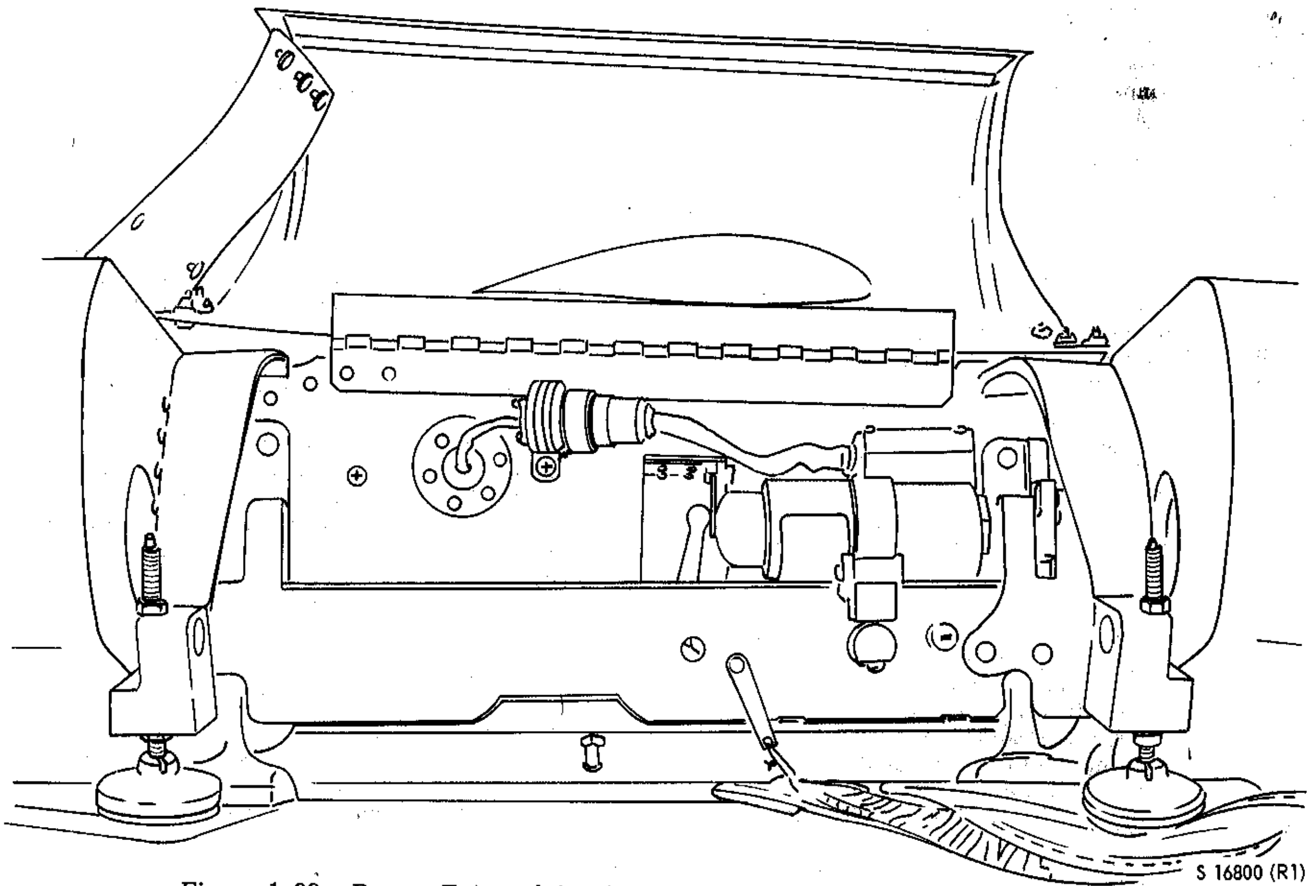


Figure 1-29. Proper External Auxiliary Fuel Tank To Bomb Shackle Installation

**Rapid Fuel Dumping System.**

CH-3E 16▸ and all HH-3E helicopters are equipped with a rapid fuel dumping system. The system consists of a fuel dump valve for each main tank and a fuel dump pump that pumps the fuel overboard through an outlet, located on the right rear fuselage. Fuel may be dumped from both main tanks at a rate of approximately 880 pounds per minute in level flight from 70 knots to V max knots and during descents through full autorotation at speeds from 70 to 100 knots. The system will pump all fuel overboard from the main tanks, except for 500 pounds in each main tank.

**Fuel Dump Valves.**

The fuel dump valves, one for each main tank, are electrically operated by switches on the pressure refueling control panel. The valves may also be manually opened and closed by use of a manual override handle located on each valve. The fuel dump valves are controlled by switches, marked FUEL DUMP, located on the pressure refueling panel. The switches have marked positions FWD and OFF and AFT and OFF to designate the tank valve they control. The FWD and AFT positions are the ON position for the switches. The fuel dump valves operate from power supplied by the dc essential bus and are protected by circuit breakers, marked FWD TANK and AFT TANK, under the general headings FUEL DUMP and INFLT REFUEL, located on the copilot's circuit breaker panel.

**Fuel Dump Pump.**

The fuel dump pump is automatically turned on and off when either or both of the fuel dump switches are actuated. The pump also contains a thermal protective circuit that will shut the pump off if fuel is no longer flowing through the pump to preclude damage from the pump overheating. The fuel dump pump operates from power supplied by the ac essential bus and is protected by circuit breakers, marked INFLT REFUEL PUMP DUMP, located on the pilot's circuit breaker panel.

## GROUND PRESSURE AND AIR REFUELING SYSTEMS.

HH-3E helicopters are equipped with ground pressure and air refueling systems (figure 1-32). Ch-3 E helicopters 16▸ are equipped with provisions for the ground pressure and air refueling systems. Both systems utilize the same plumbing and system components, except the ground pressure refueling system is refueled through a pressure refueling adapter located on the lower fuselage below the en-

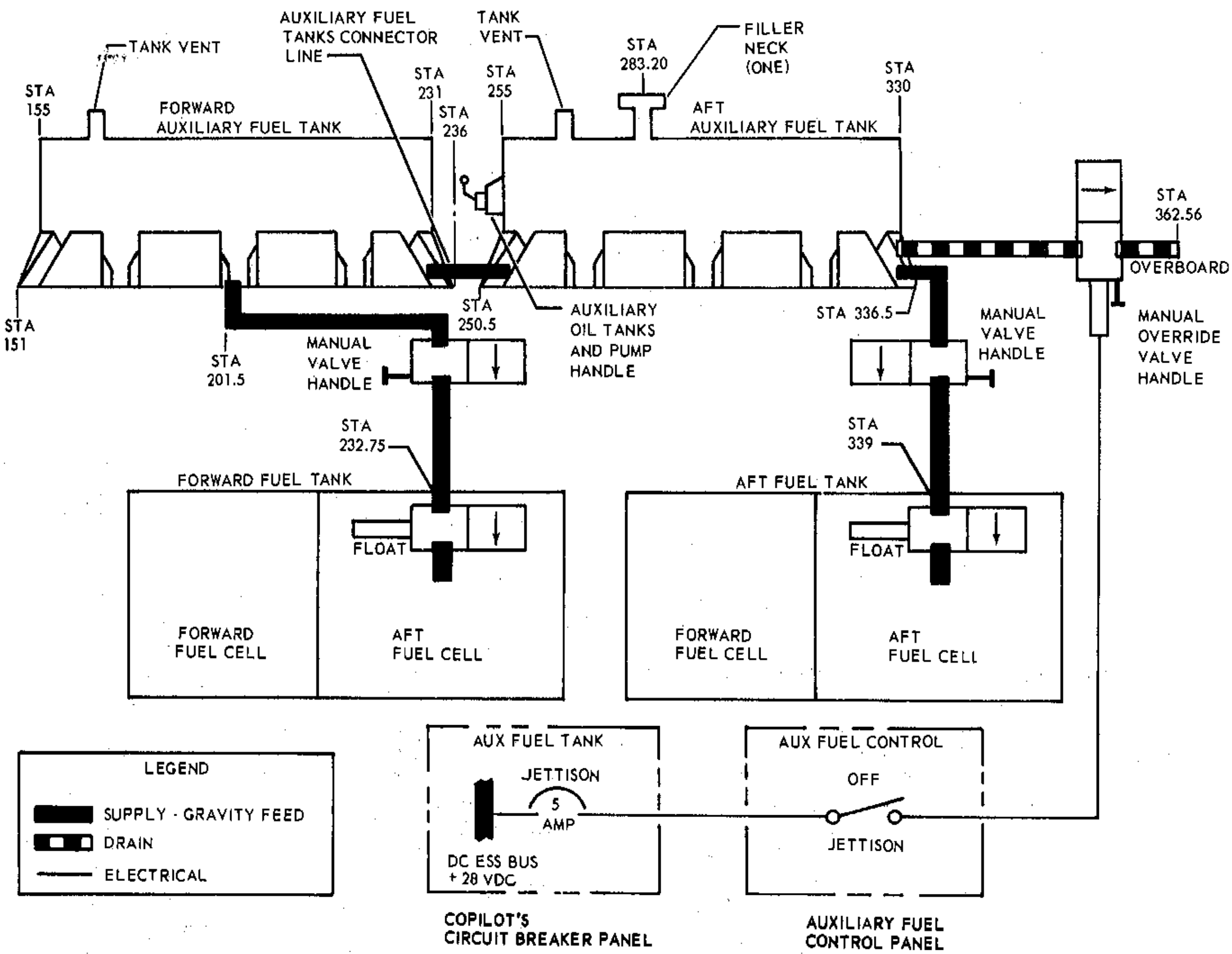


**Figure 1-30. Dual Internal Tank Auxiliary Fuel System Diagram**

trance to the cargo compartment. The ground pressure refueling system does not use bleed air. The air refueling system is refueled through a probe, located on the right side of the forward fuselage, which is extended and retracted by compressor bleed air. Both systems are controlled from the pressure refuel control panel marked PRESSURE REFUEL, located on the top and center of the instrument panel.

**Ground Pressure Refueling.**

Since the control switches, indicator lights, and system components operate in the same manner for both systems, they will be discussed under the heading Air Refueling System in this section.

**Pressure Refueling Adapter.**

The pressure refueling adapter (20, figure 1-3), located on the lower fuselage below the entrance to the cargo compartment, is the single-point hose attachment that is used to simultaneously pressure refuel all fuel tanks.

**Air Refueling System.**

The air refueling system consists of a refueling probe, bleed air selector and shut-off valves, flow sensors, vent valves, surge valves, pressure refueling shut-off and high-level sensor valves, and a control panel.

**Refueling Probe.**

The retractable refueling probe (16, figure 1-3), located on the forward right-hand side of the helicopter, is extended and retracted by compressor bleed air. The system can either operate on bleed air from either engine, or from both engines simultaneously. Check valves are installed to prevent bleed air from flowing to an inoperative engine during single engine operation. The probe contains lock actuators that lock the probe in the extended or retracted position and cause condition lights to illuminate to indicate the probe condition.

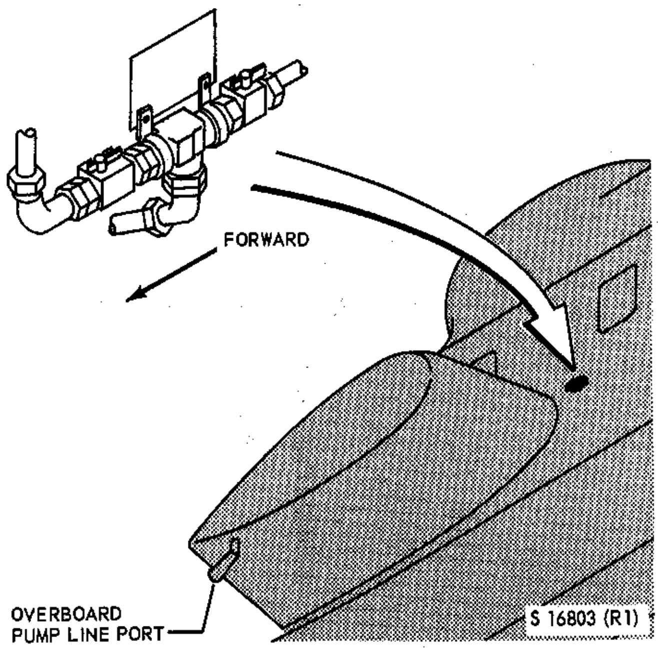


Figure 1-31. Manual Fuel Close Line and Dump Valves

**Bleed Air Selector and Shut-off Valves.**

The bleed air selector and shut-off valves control the flow of bleed air to the refueling probe. Both valves are controlled from the pressure refueling panel. The shut-off valve controls the flow of bleed air to the bleed air selector valve and contains a relief valve that opens when the shut-off valve is closed, if the pressure exceeds 220 psi, and closes when the pressure drops to 160 psi. The bleed air selector valve selects bleed air to extend or retract the refueling probe. The selector valve also contains a relief valve that opens if inlet port pressure exceeds 220 psi and closes when the pressure drops to 160 psi.

**Flow Sensors.**

The flow sensors, one located in the inlet fuel line to each tank, will sense a fuel flow above $2 \pm 1$ gpm and cause the appropriate fuel flow indicator light to illuminate.

**Pressure Refueling Shut-off and High Level Sensor Valves.**

The pressure refueling shut-off and high level sensor valves, located in each main tank, control the flow and fuel level during ground pressure and air refueling operations. The pressure refueling shut-off valves open when fuel pressure is applied and permit fuel to flow to the tanks. The high level sensors close when the fuel tanks have reached their full capacity and cause the pressure refueling shut-off valve to close and shut off the flow of fuel. The high level sensor valves are also the test component for the PRE-SHUT-OFF TEST function.

**Pressure Refueling Control Panel.**

The pressure refueling control panel (figure 1-33), located on the top and center of the instrument panel, marked PRESSURE REFUEL, contains the control switches and condition lights for the ground pressure and air refueling systems. The fuel tank selector switches and condition lights are used in the same manner for both ground pressure and air refueling operations. The control panel contains the fuel dump switches, the controllable spotlight rheostat, main and external auxiliary fuel tank selector switches and condition lights, and fuel level shut-off test switch, a control panel light test switch, the air refueling probe control switch and condition lights, and the master power switch. The fuel dump switches, under the heading FUEL DUMP, have marked positions FWD and OFF and AFT and OFF respectively. Placing the switches in the FWD and AFT positions will open respective dump valves to the forward and aft main fuel tanks, energize the fuel dump pump and cause all but 500 pounds of fuel in each main tank to be dumped overboard. The main fuel tanks may be simultaneously or individually selected for fuel dumping. The switches should be placed in the OFF position when 500 pounds of remaining fuel is noted on the fuel quantity indicators. The controllable spotlight rheostat, marked PROBE LIGHT, with marked positions OFF and MAX, is rotated out of the OFF position to vary the intensity of the controllable spotlight (14, figure 1-3) during night air refueling operations. The external auxiliary fuel tank selector switches and condition lights are under the heading EXT TANKS. The auxiliary fuel tank selector switches, marked SELECT, have marked postions LEFT and RIGHT and OFF. Placing the switches in the LEFT and RIGHT positions will open both auxiliary fuel tank fuel valves and allow fuel to flow to the tanks. The fuel flow condition lights, marked LEFT FLOW and RIGHT FLOW, respectively, will illuminate to indicate that fuel is flowing to the tanks through the flow sensors. The flow lights may flicker on and off any time the probe is in an intermediate position. The fuel level lights, will illuminate to show the tanks are full. When the fuel level lights illuminate, the fuel flow condition lights will go out as the thermistors, one located in each auxiliary tank, also close the fuel valves when they sense that the tanks are full. If the LEFT FULL or RIGHT FULL light remains on after fuel is transferred to the main tanks, reaccomplish the auxiliary tank transfer operation. The external auxiliary tanks may be simultaneously or individually selected for refueling. The main fuel tank selector switches and condition lights are under the heading MAIN TANKS. The main fuel tank selector switches, marked FWD and AFT, have marked positions SELECT and OFF. Placing the FWD and AFT switches in the SELECT position, opens the pressure refueling shut-off valves to allow fuel to flow to the tanks. The fuel flow condition lights, marked FWD FLOW and AFT FLOW, respectively, will illuminate to indicate that fuel is flowing to the tanks through the flow sensors. The

FWD and AFT selector switches should be left in the SELECT position at all times unless it is not desired to service a particular tank during a fuel transfer or air refueling operation. When the main tanks are full, the fuel flow condition lights will go out as the high level sensor valves, one located in each main tank, close the pressure refueling shut-off valves when they sense that the tanks are full. All four fuel tanks, two main and two external auxiliary, may be simultaneously or individually selected for refueling. The fuel level shut-off test switch, marked PRE-SHUT-OFF TEST, is depressed to test the integrity of the high level sensor and pressure refueling shut-off valves.

NOTE

> Master power switch must be ON to check PRE-SHUT-OFF TEST system.

Depressing the fuel level shut-off test switch, after ground pressure or air refueling operations have started, will cause the fuel level sensor valves to simulate a tank full condition and close the pressure refueling shut-off valve. This will stop the flow of fuel and cause the fuel flow condition lights to go out. The fuel leve shut-off test circuit is powered from the dc essential bus and is protected by circuit breakers, marked FWD TANK and AFT TANK and under the headings PRE-CHECK and INFLT REFUEL, located on the copilot's circuit breaker panel. Releasing the test switch will restore the system to a normal condition. The control panel light test switch, marked PNL LAMP TEST, is depressed to check the integrity of the bulbs in the various condition lights. The air refueling probe selector switch and condition lights are under the heading PROBE. The probe selector switch, with marked positions REFUEL and STOW, is actuated to extend and retract the refueling probe. When the switch is placed in the REFUEL position, the bleed air shut-off and selector valves are actuated to permit bleed air to extend the probe. When the probe is fully extended and locked, the READY condition light will illuminate. The probe will first bounce away from the fully extended position then reseat itself before the latching mechanism engages to hold it in the fully extended position and cause the READY light to illuminate. Placing the switch in the STOW position will cause the bleed air shut-off and selector valves to actuate to permit bleed air to retract the probe. If the probe should not fully retract and/or lock, the NOT STOWED condition light will illuminate. If the refueling probe fails to extend or retract completely, stop operation and recycle by first turning off the master power switch, recycling the probe selector switch, then turning the master power switch back on. The control circuits for the refueling probe are powered from the dc essential bus and protected by a circuit breaker, marked BOOM CONT and under the heading INFLT REFUEL, located on the copilot's circuit breaker panel. The master power switch, marked MASTER POWER, with marked positions ON and OFF, must be in the ON position to energize the ground pressure and air refueling systems. The master switch must be in the OFF position when fuel is being transferred from the external auxiliary fuel tanks to the main fuel tanks, and when it is desired to bypass the dimming circuit and have the controllable spotlight operate as a bright light only.

NOTE

> The standby compass is not reliable when the pressure refueling panel is energized and operating.

NOTE

> If the probe has just been stowed, wait approximately one minute to allow the system to bleed before extending the probe again.

## ELECTRICAL POWER SUPPLY SYSTEM.

The primary source of electrical power is supplied by a 115/200 volt ac system. Alternating current is rectified to provide a 28-volt dc system.

### ALTERNATING CURRENT SUPPLY SYSTEM.

Two alternating current generators are the primary source of power for the ac electrical supply system (figure 1-34). Other sources of alternating current are the dc powered ground inverter and the ac external power receptacle. An auxiliary power unit is also provided that drives the generators through the main gear box accessory section when the rotor rpm is below 100% $N_r$.

#### Generators.

The two 20KVA, 115/200-volt, three-phase, self-cooled, brushless, self-exciting generators are mounted on, and driven by, the accessory section of the main gear box. Each generator has a prorated capacity of 25KVA at 15,000 feet altitude and a OAT of 10°C. The auxiliary power unit powers the main gear box accessory section to drive the generators when the rotor rpm is below 100% $N_r$. When rotor speed reaches 100 percent rpm, the accessory section is driven through the main gear box. Each generator output is directed through a respective supervisory panel that provides control and protection of the electrical system from underfrequency, overvoltage, and open-phase protection. The underfrequency protection is not available when the weight of the helicopter is removed from the landing gear wheels. The No. 1 generator normally furnishes power to the ac essential bus. Each generator is controlled by a respective generator switch located on the overhead switch panel in the pilot's compartment. Generator power is supplied to the supervisory panel whenever the generators are operating above the cut-in speed (between 92 and 97 percent rotor speed). The generators are connected directly to the helicopter's system whenever the generator switches are placed in the ON position, provided the supervisory panel is satisfied that voltage output as

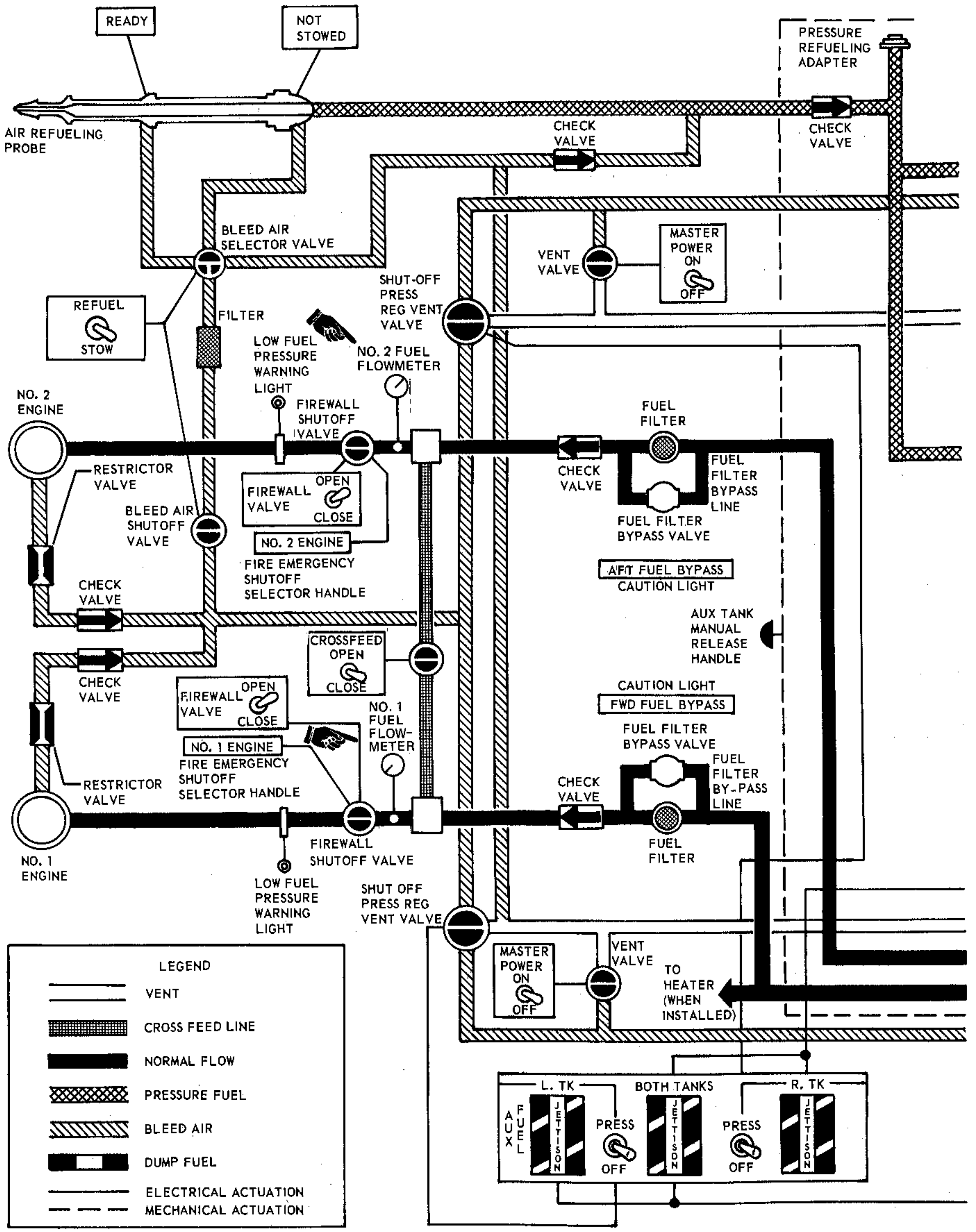


Figure 1-32. Ground Pressure and Air Refueling System (Sheet 1 of 2)

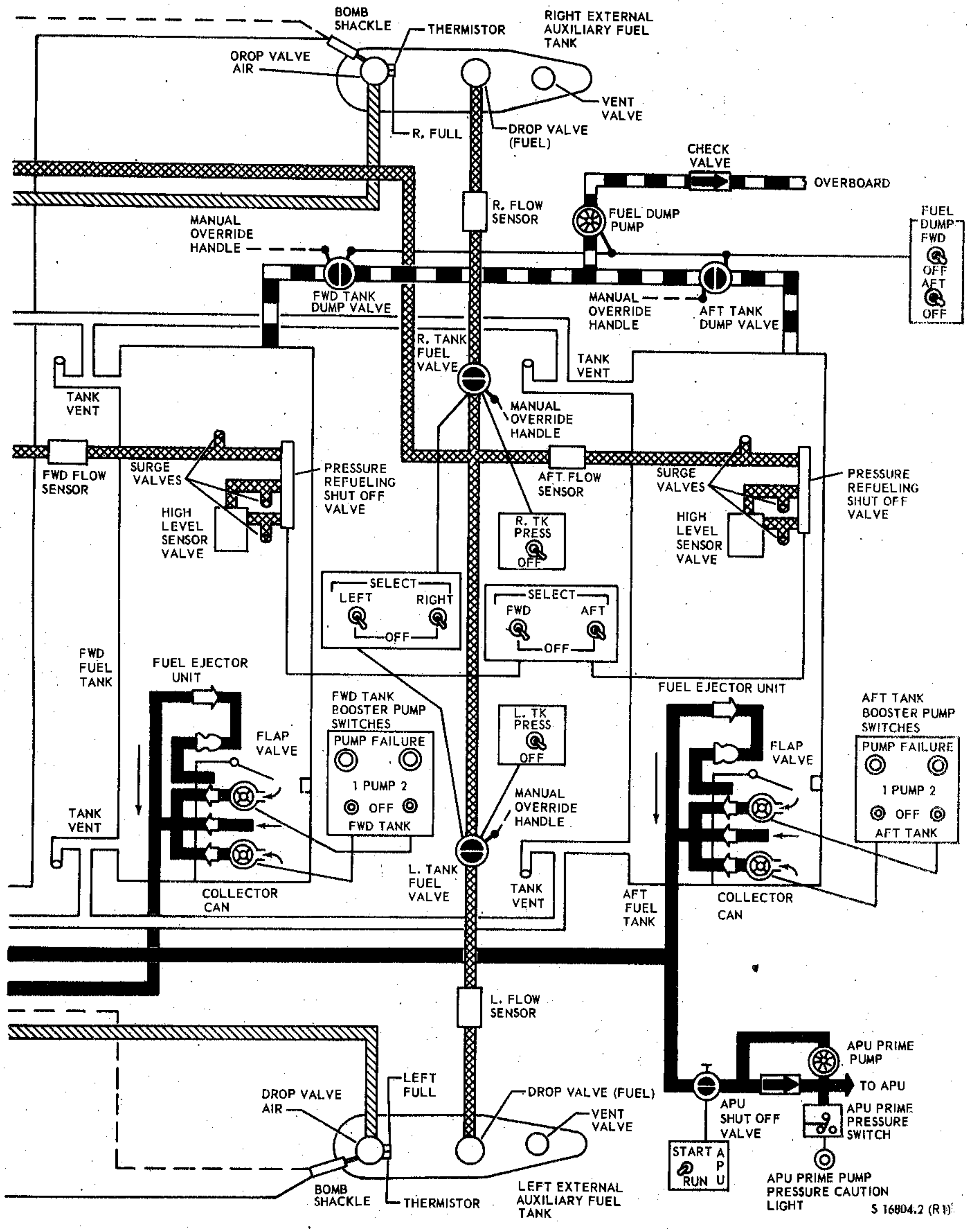


Figure 1-32. Ground Pressure and Air Refueling System (Sheet 2 of 2)

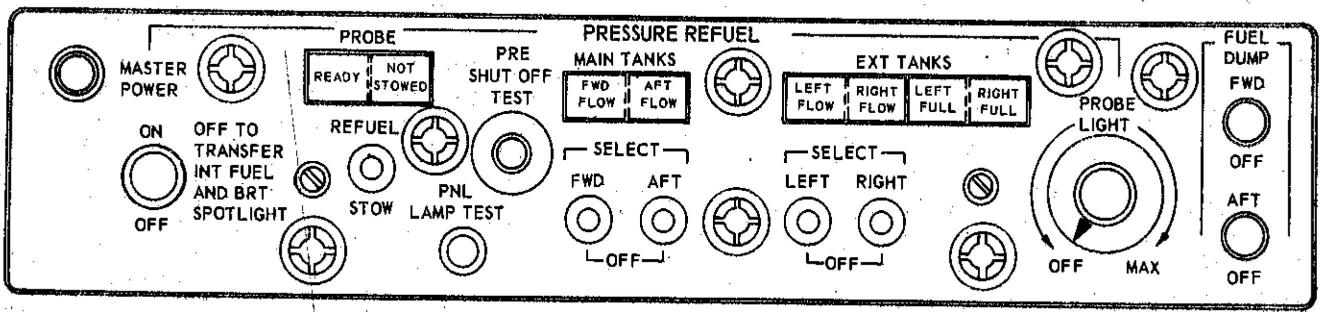


Figure 1-33. Pressure Refueling Control Panel

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well as frequency output is within the prescribed limits. Failure of either generator is indicated by failure caution lights, marked No. 1 GENERATOR and No. 2 GENERATOR, located on the caution panel. Both generators will continue to generate power during autorotation.

**Generator Switches.**

The generator switches, located on the overhead switch panel (figure 1-13) under the general heading 1 GEN 2, have marked positions ON, OFF/RESET, and TEST. When a generator switch is placed in the ON position, generator power is connected through the respective main line contactor relay to the appropriate bus. When the No. 1 generator switch is placed in the OFF/RESET position, the No. 1 generator is turned OFF and generator cycling is reset. Placing the switch in the OFF/RESET position, then ON, will bring the generator back on the line, provided a temporary over-voltage condition occurred and no longer exists. The TEST position of the switches is to be used by ground personnel when performing maintenance checks.

**Inverter.**

The 100 VA inverter, located in the electronics compartment, is powered by the 28-volt dc essential bus and supplies 115-volt ac to two 26-volt ac transformers, engine fire detectors, fuel quantity indicators, and turbine inlet temperature indicators. The transformers step the inverter voltage down to 26 volts ac for operation of instruments connected to the inverter bus. The inverter is energized and automatically connected to the inverter bus when the ac essential bus is not energized from the generators or ac external power, and the inverter switch is in the ON position. Whenever the ac essential bus is energized, a relay is actuated that disconnects the inverter from the dc power source, regardless of the inverter switch position, and connects the transformers, engine fire detectors, fuel quantity indicators, and turbine inlet temperature indicators, to the ac essential bus. The inverter is protected by a circuit breaker, marked INV, located on the overhead dc circuit breaker panel.

**Inverter Switch.**

The inverter switch, marked INVERTER, ON and OFF, is located on the overhead switch panel (figure 1-13). The ON position connects the inverter to the dc essential bus. The OFF position disconnects the inverter circuit from the dc essential bus.

**AC External Power Receptacle, Switch, and Advisory Light.**

The 115/200-volt ac external power receptacle (19, figure 1-3) is located on the right side of the helicopter below the pilot's window. External power is controlled by a switch, marked EXT PWR, with marked positions ON and OFF, located on the overhead switch panel, through a circuit breaker, marked EXT PWR, located on the overhead dc circuit breaker panel. The external power switch must be in the ON position to utilize an external power source connected to the external power receptacle. When the external power relay is energized by placing the external power switch in the ON position, and the external power source is connected, a light, marked EXT PWR ON, located on the advisory panel, will illuminate. The external power relay also serves to keep the external power receptacle from being energized, when not in use. External power is supplied, through the No. 2 line contactor relay to the No. 1 line contactor relay to energize the ac essential bus. The ac nonessential bus is energized directly from the external power relay. Whenever ac external power energizes the essential and nonessential buses, the respective transformer-rectifiers are energized to provide direct current power.

**NOTE**

All ac APU's are not equipped to provide a 28-volt dc input. When using this type of APU, it is necessary to activate the battery switch momentarily to provide dc current for energizing the ac external power relay.

**Alternating Current Distribution.**

Power for the operation of alternating current electrical equipment is distributed through supervisory panels that provide control and protection of the system, then through the appropriate line contactor relay to the essential and nonessential buses.

### AC Essential Bus.

The ac essential bus is normally powered by the No. 1 generator, or by ac external power, when connected, and the external power switch is in the ON position. The essential bus is energized through the No. 1 generator contactor relay and distributes power to all of the essential ac operated equipment, the No. 1 transformer-rectifier, and the two transformers, which step down voltage to 26 volts for the operation of certain radio facilities and pressure instruments. Failure of the No. 1 generator causes the No. 1 main line contactor to automatically transfer the essential bus to the No. 2 generator and drop the nonessential bus from the system.

### Inverter Bus.

The inverter bus is energized by alternating current through two transformers that step 115-volt ac power down to 26 volts for the operation of certain pressure instruments that are essential for safe engine operation. The transformers and inverter bus are energized by the ac essential bus when energized from any ac power source. The transformers and inverter bus are energized by the inverter whenever the ac essential bus is de-energized and the inverter switch is in the ON position.

NOTE

> The inverter bus is not separately identified on the circuit breaker panels.

### AC Nonessential Bus.

The ac nonessential bus is normally powered by the No. 2 generator, or by ac external power, when connected, and the external power switch is in the ON position. The nonessential bus is normally energized through the No. 2 generator contactor relay and the nonessential bus contactor relay. The nonessential bus distributes power to the nonessential equipment and the No. 2 transformer-rectifier. Failure of either generator will cause the nonessential bus to be dropped from the system. If the No. 2 generator should fail, the pilot's attitude indicator, which is normally powered by the No. 2 generator, is automatically transferred to the ac essential bus. The nonessential bus contactor relay, energized by the No. 1 generator, normally connects the No. 2 generator output to the nonessential bus. When the nonessential bus contactor relay is de-energized by loss of No. 1 generator output, the nonessential bus is dropped from the system and the No. 2 generator output is directed to the essential bus. When the nonessential bus contactor relay is energized by No. 1 generator output, and the No. 2 generator is inoperative, the No. 1 generator contactor relay insures that No. 1 generator output is directed only to the essential bus and the nonessential bus is dropped from the system.

### Autotransformer.

An autotransformer on the pilot's circuit breaker panel support provides a source of 26-volt ac for the operation of radio equipment by reducing 115 volt ac power from the No. 1 generator or ac external power receptacle.

### Isolation and Autotransformers.

The 115/26-volt ac isolation and autotransformers on the pilot's circuit breaker panel support serve two functions. The isolation transformer isolates the operating circuit connected across its secondary winding and places a load across the phase C and phase A output of the inverter to provide a better balanced load. It also provides 26-volt ac for the utility hydraulic pressure, auxiliary hydraulic pressure, and No. 2 engine oil pressure indicators and No. 2 engine torquemeter. The autotransformer provides 26-volt ac for the primary hydraulic pressure, transmission oil pressure, No. 1 engine oil pressure indicators, and the No. 1 engine torquemeter.

### Alternating Current Circuit Breakers.

Alternating current circuit breakers, protecting the various ac circuits, are located on the overhead circuit breaker panels (figure 1-36) above the pilot's and copilot's heads in the pilot's compartment. The circuit breaker panel located above the pilot's head contains circuit breakers that protect the ac essential bus loads, and the circuit breaker panel located above the copilot's head contains circuit breakers that protect the ac nonessential bus loads. All circuit breakers are marked as to the circuit they protect and are of the push-pull type that may be reset. Any malfunctioning circuit may be isolated from the ac power supply system by pulling out its circuit breaker.

## DIRECT CURRENT POWER SUPPLY SYSTEM.

Two transformer-rectifiers are the primary sources of direct current power for the dc electrical supply system (figure 1-35). Other sources of direct current power are the battery and the dc external power receptacle.

### Transformer-Rectifiers.

Two 200-ampere transformer-rectifiers are located in the electronics compartment. Each transformer-rectifier is powered by a separate generator. The No. 1 generator powers the No. 1 transformer-rectifier through the ac essential bus, and the No. 2 generator powers the No. 2 transformer-rectifier through the ac nonessential bus. Each transformer-rectifier is controlled by a respective transformer-rectifier switch, located on the overhead switch panel in the pilot's compartment. The No. 1 transformer-rectifier is protected by a circuit breaker,

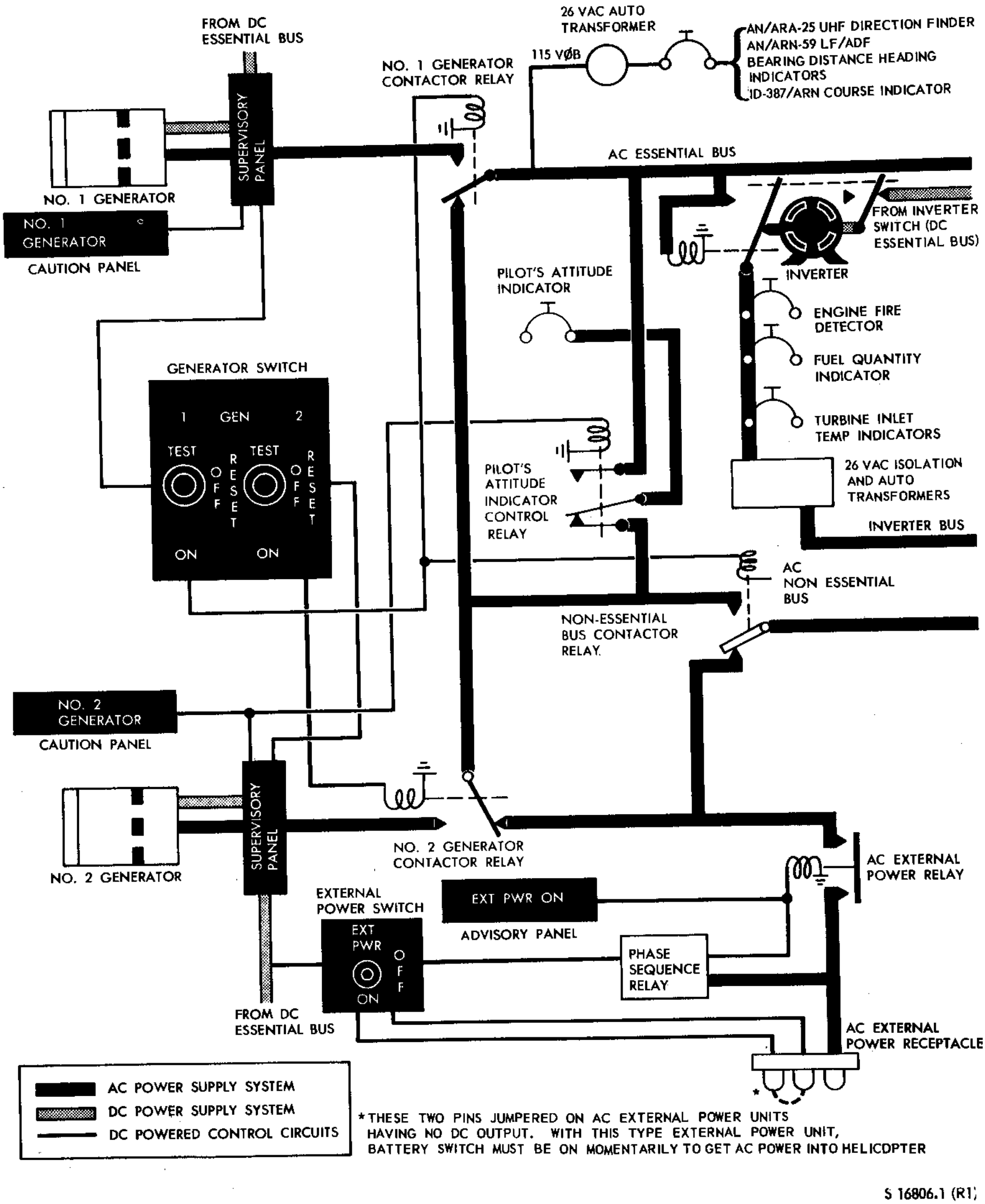


Figure 1-34. AC Electrical Power Supply System (Typical) (Sheet 1 of 2)


Figure 1-34. AC Electrical Power Supply System (Typical) (Sheet 2 of 2)

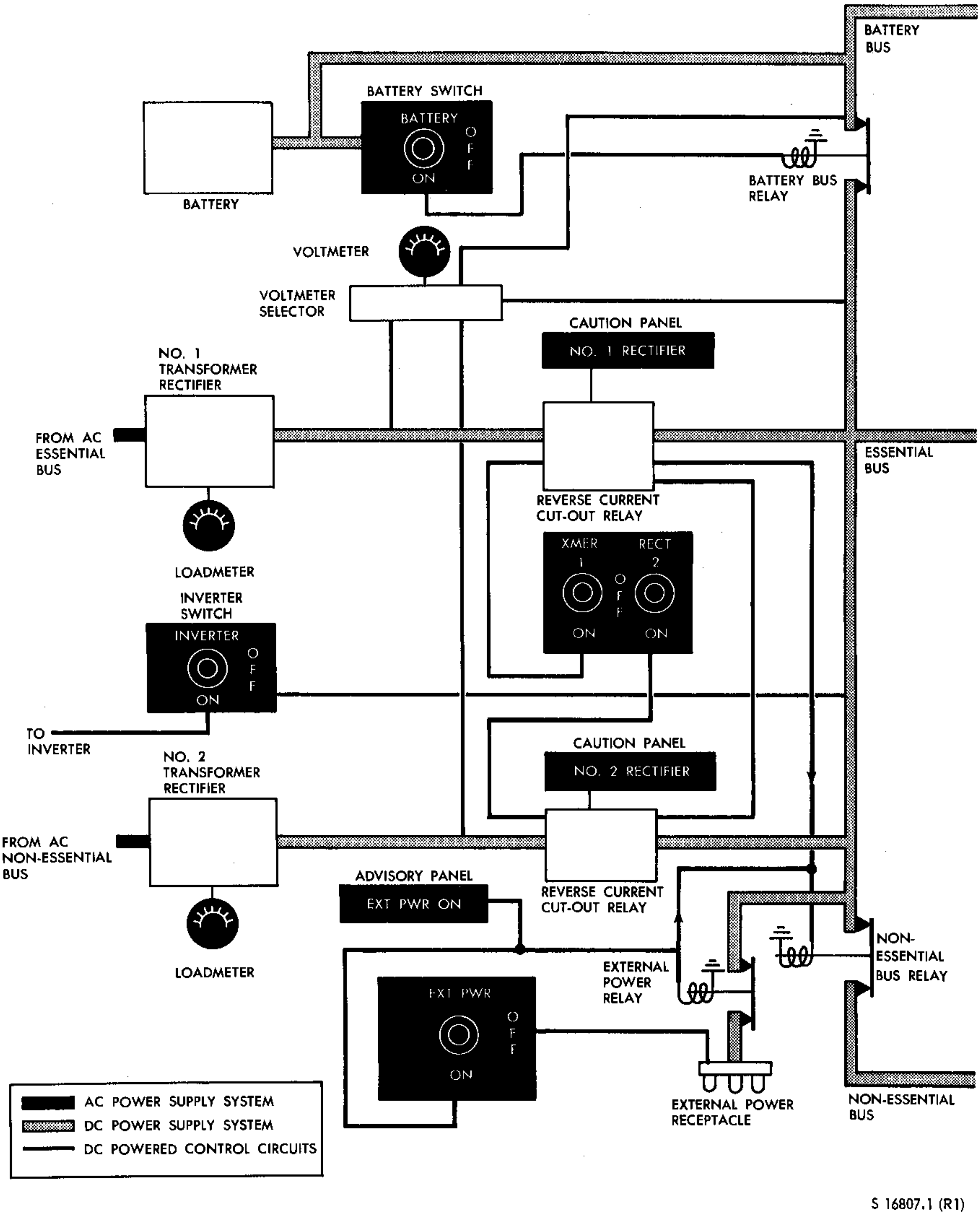


Figure 1-35. DC Electrical Power Supply System (Typical) (Sheet 1 of 2)

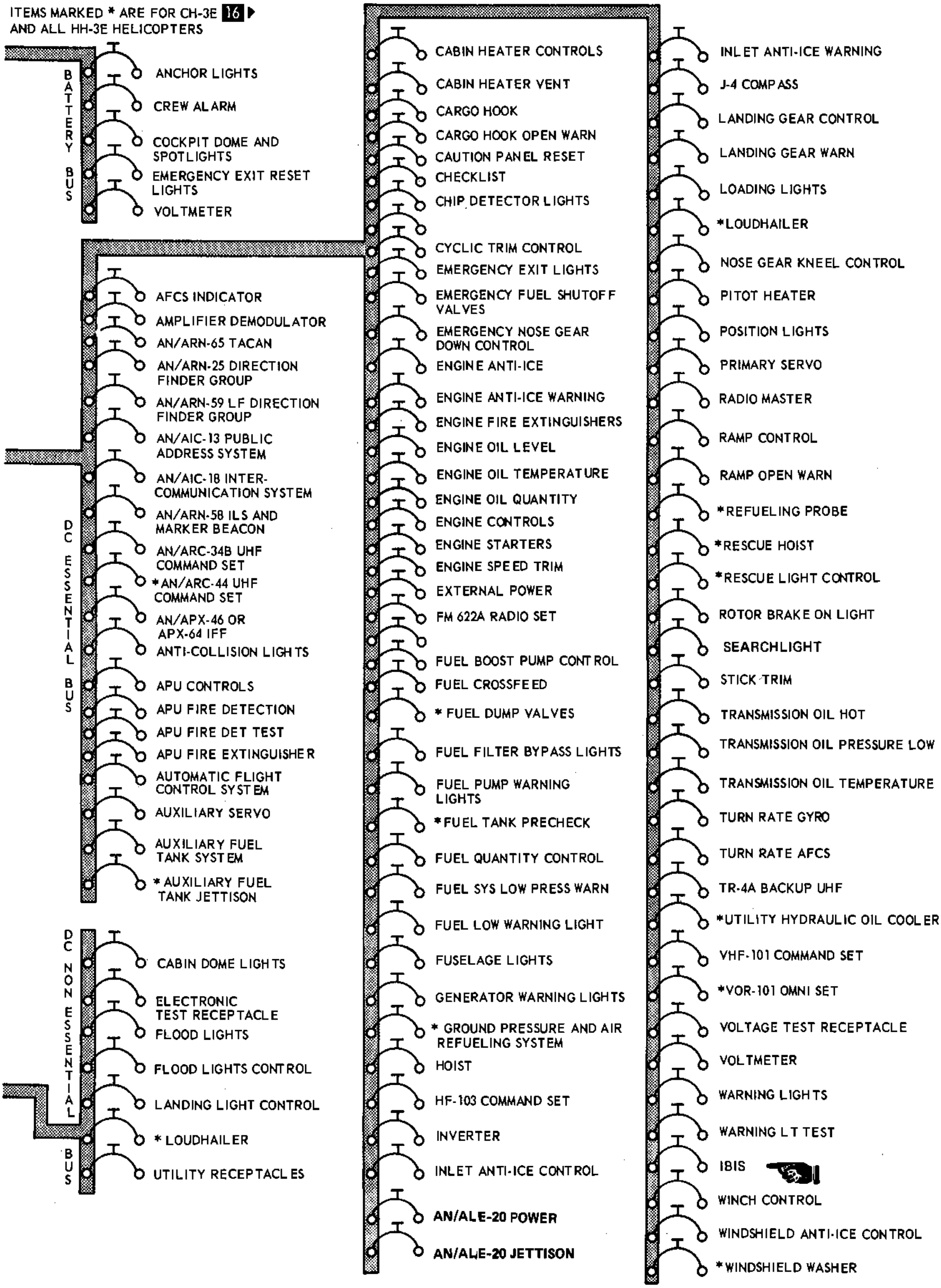


Figure 1-35. DC Electrical Power Supply System (Typical) (Sheet 2 of 2)

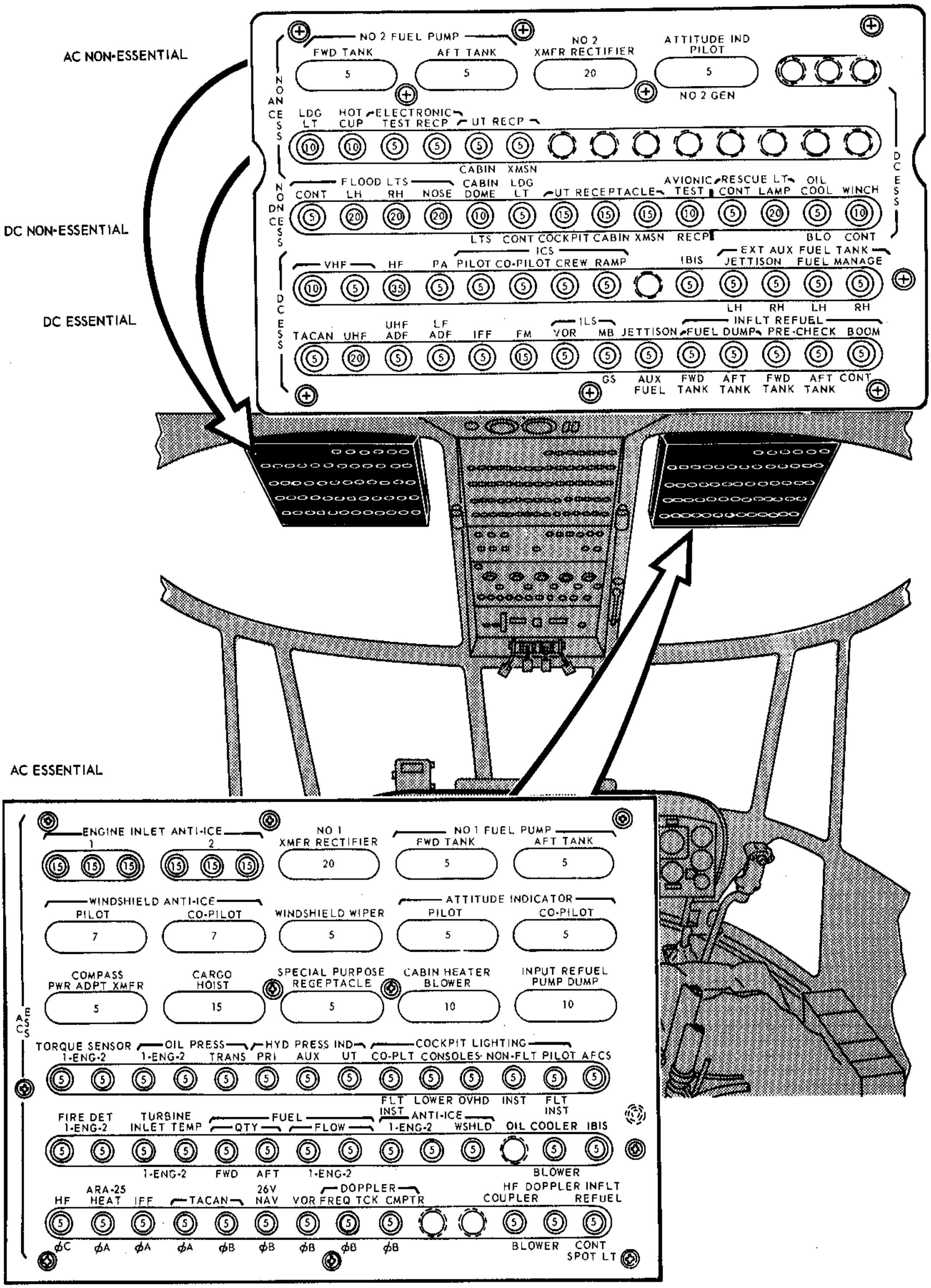


Figure 1-36. Circuit Breaker Panels (Typical) (Sheet 1 of 2)

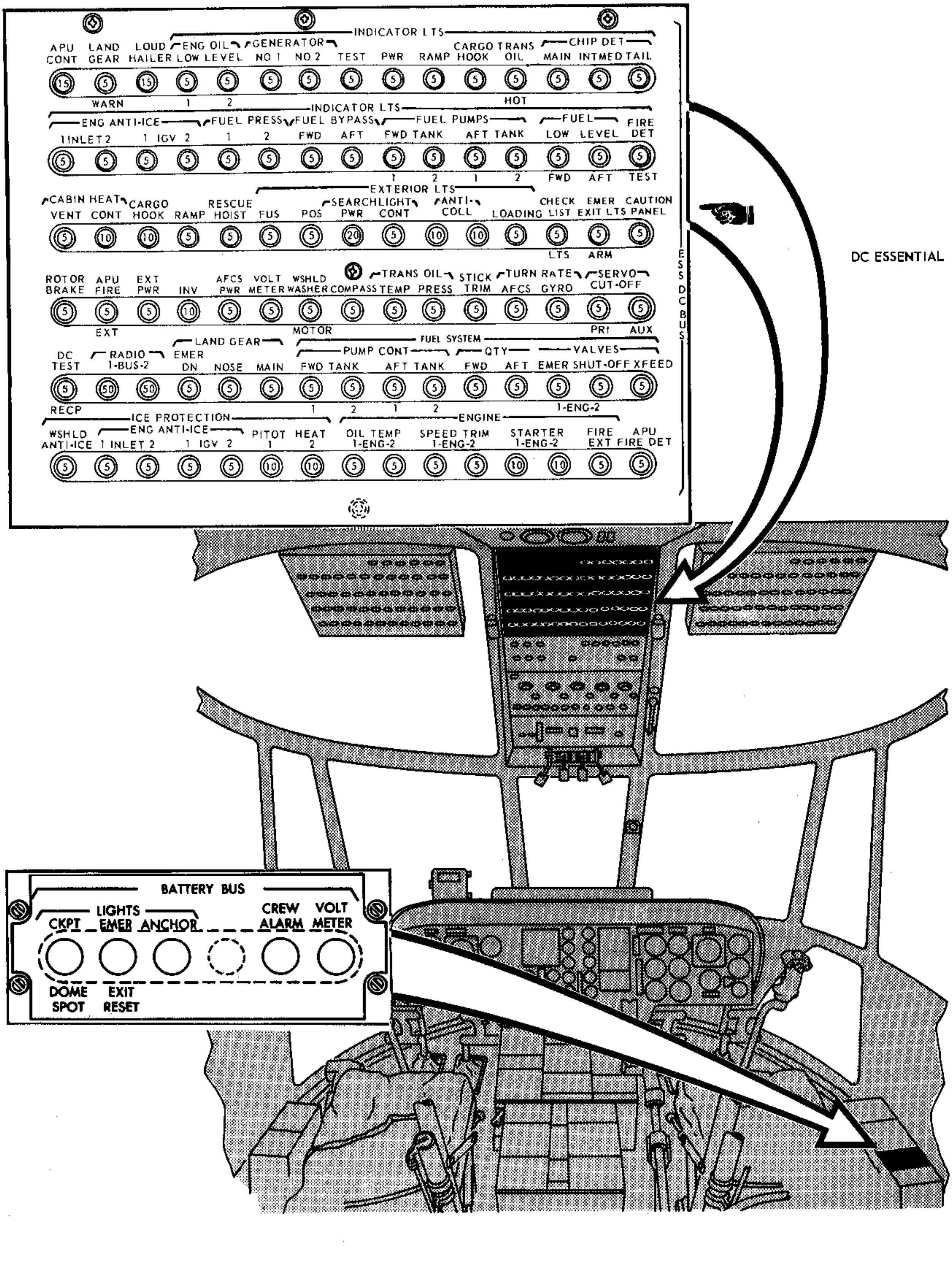


Figure 1-36. Circuit Breaker Panels (Typical) (Sheet 2 of 2)

marked No. 1 XMFR RECTIFIER, located on the ac essential bus circuit breaker panel, and the No. 2 transformer-rectifier is protected by a circuit breaker, marked No. 2 XMFR RECTIFIER, located on the ac nonessential bus circuit breaker panel. The transformer-rectifiers are connected directly to the helicopter's dc system whenever they are energized by ac power and the transformer-rectifier switches are in the ON position. Failure of the No. 1 transformer-rectifier will be indicated by a light on the caution panel, marked #1 XMFR RECT, and failure of the No. 2 transformer will be indicated by a light on the caution panel, marked #2 XMFR RECT.

**Transformer-Rectifier Switches.**

The transformer-rectifier switches, located on the overhead switch panel (figure 1-13) under the general heading XMFR RECT 1 and 2, have marked positions, ON and OFF. When the transformer-rectifier switches are placed in the ON position, dc power is connected through the respective reverse current cutout relay to the essential and nonessential buses. When both transformer-rectifier switches are placed in the OFF position, only battery power will be available to the dc essential bus. When either transformer-rectifier switch is placed in the OFF position, and the other transformer-rectifier, switch is in the ON position, power will only be supplied to the dc essential bus and the dc nonessential bus will be dropped from the system.

**Battery.**

The 24-volt, 22-ampere hour nickel cadmium battery (1, figure 1-44), located in the nose section forward of the pilot's compartment, is accessible from outside the helicopter for maintenance. Battery power is used for limited ground operations, when no external power is available, and as emergency source of power to the essential bus, if both generators and/or transformer-rectifiers should fail. The battery also powers the battery bus.

**Battery Switch.**

The battery switch, located on the overhead switch panel (figure 1-13), marked BATTERY, has marked positions ON and OFF. When the battery switch is placed in the ON position, battery power is supplied to the dc essential bus. Battery power is disconnected from the essential bus when the battery switch is placed in the OFF position.

**NOTE**

If battery power is excessively low, the battery switch will not actuate the battery relay to the dc essential bus.

**DC External Power, Receptacle, Switch, and Advisory Light.**

The 28-volt dc external power receptacle (18, figure 1-3) is located on the right side of the helicopter below the pilot's window. External power is controlled by a switch, marked EXT PWR, with marked positions ON and OFF, located on the overhead switch panel. The circuit is protected by a circuit breaker, marked EXT PWR, located on the overhead dc circuit breaker panel. The external power switch must be in the ON position to utilize an external power source connected to the dc external power receptacle. When the external power relay is energized, a light marked EXT POWER ON, located on the advisory panel, will illuminate. External power will also be provided to energize the nonessential bus relay, thereby allowing both the dc essential and dc nonessential buses to be energized. The external power relay also serves to keep the external power receptacle from being energized when not in use. DC external power, if used, should be used for all ground operations until after the generators are operating.

**NOTE**

DC power will be automatically provided when the ac system is energized and the transformer-rectifier switches are in the ON position.

**Direct Current Distribution.**

Power for the operation of direct current electrical equipment is distributed through the essential, nonessential, and battery buses.

**DC Essential Bus.**

The dc essential bus supplies power for the operation of all equipment necessary for safety of flight and limited mission accomplishment. The dc essential bus is powered by either or both transformer-rectifiers, when the transformer-rectifier switches are in the ON position, or by either transformer-rectifier if one should fail. Other dc power sources are external power, when connected and on, or the battery when the battery switch is on.

**DC Nonessential Bus.**

The dc nonessential bus supplies power for the operation of equipment not essential for safety of flight or limited mission accomplishment. The dc nonessential bus is powered by the transformer-rectifiers, when both are operating and the transformer-rectifier switches are in the ON position, or by external power, when connected and the external power switch is in the ON position. Loss of power from either transformer-rectifier will cause the nonessential bus relay to be de-energized and drop the nonessential bus from the system. Battery power is not distributed to the dc nonessential bus.

**Battery Bus.**

The battery bus is continuously energized by the battery and supplies power to anchor lights, cockpit dome and spotlights, emergency lights reset, crew alarm bell, and voltmeter. The equipment may be operated regardless of the position of the battery switch.

**Direct Current Circuit Breakers.**

Circuit breakers that protect the various dc circuits are located on the overhead dc circuit breaker panel and on a portion of the ac nonessential bus circuit breaker panel (figure 1-36) in the pilot's compartment. All circuit breakers are marked as to the circuit they protect and are of the push-pull type that may be reset. Any malfunctioning circuit may be isolated from the dc power supply system by pulling out its circuit breaker.

**Loadmeters.**

A loadmeter (46 and 51, figure 1-14) for each transformer-rectifier, located on the instrument panel, indicates the direct current being drawn from the respective transformer-rectifier. When either engine starter is engaged, relays disconnect the loadmeters from the dc system.

**Voltmeter and Voltmeter Selector.**

A voltmeter (53, figure 1-14), located on the instrument panel, indicates the voltage available from the power source selected by the voltage selector. The voltmeter selector panel (52, figure 1-14), located directly below the voltmeter, marked VOLT SELECTOR, has marked positions ESS DC BUS, BATT BUS, NO 1 TR, and NO 2 TR. When the selector knob is rotated to any marked positon, the voltage available from the power source selected will be indicated on the voltmeter.

## UTILITY HYDRAULIC SUPPLY SYSTEM.

The utility hydraulic system (figure 1-37) operates the main landing gear, nose landing gear, APU start system, and ramp actuating system. On helicopters CH-3E 16 ▶, HH-3E, or those helicopters modified by T.O. 1H-3(C)C-561, the utility hydraulic system also powers the rescue hoist. The utility hydraulic system reservoir (figure 1-44), located aft of the main gear box, has a capacity of 3.05 gallons of hydraulic fluid. The utility hydraulic pump is located on the accessory drive section of the main gear box and provides 3000 psi hydraulic pressure. An oil cooler is provided in the hydraulic line to maintain utility hydraulic oil temperature within limits on helicopters CH-3E 16 ▶, HH-3E, or those helicopters modified by T.O. 1H-3(C)-561. The oil cooler blower operates on power from the AC essential bus and is protected by a circuit breaker marked OIL COOLER BLOWER located on the AC essential bus circuit breaker panel. The blower is actuated by power from the DC essential bus through a circuit breaker marked OIL COOL BLD located on the DC essential bus circuit breaker panel. The blower operates continuously when the buses are energized. Helicopters modifed by T.O. 1H-3-581 have a screen-type fan guard over the blower inlet.

**WARNING**

On helicopters not modified by T.O. 1H-3-581, there is no protective cover over the utility hydraulic system oil cooler blower fan. When the APU is operating, the noise is such and the fan is turning so rapidly that personnel working in this area may not be aware that the fan is turning.

NOTE

Utility hydraulic system degradation may be experienced when operating combinations of rescue hoist, landing gear, etc.

### UTILITY HYDRAULIC PRESSURE INDICATOR.

The utility hydraulic pressure indicator (49, figure 1-14), located on the instrument panel, operates on 26-volt ac power from an autotransformer. The gage, marked UTI, indicates pressure in the utility hydraulic system in psi. The indicator receives electrical power from the ac essential bus through a circuit breaker, under the heading HYD PRESS IND and marked UT, located on the ac essential circuit breaker panel.

## FLIGHT CONTROL SYSTEM.

The flight control system is divided into three systems: the main rotor flight control system, the tail rotor flight control system, and the flight control hydraulic power supply systems. When the automatic flight control system is engaged, it provides corrections of limited authority to the flight control system, causing the helicopter to respond in a stable manner to the maneuver called for by the position of the cyclic stick. This equipment also functions to provide a constant altitude. The description and operation of the automatic flight control system are included in the paragraph AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) in this section. A cyclic stick trim system is installed to provide cyclic stick "feel" and to facilitate hands-off control with the AFCS in operation.

### MAIN ROTOR FLIGHT CONTROL SYSTEM.

The main rotor flight control system provides vertical, lateral, and longitudinal control. Control motions from the collective pitch lever for vertical control and from the cyclic stick for lateral and longitudinal control are combined in a mixing unit, located in the AFCS control compartment aft of the pilot's seat, and are transmitted to the main rotor assembly by mechanical linkage. Control action is assisted by two hydraulically operated flight control

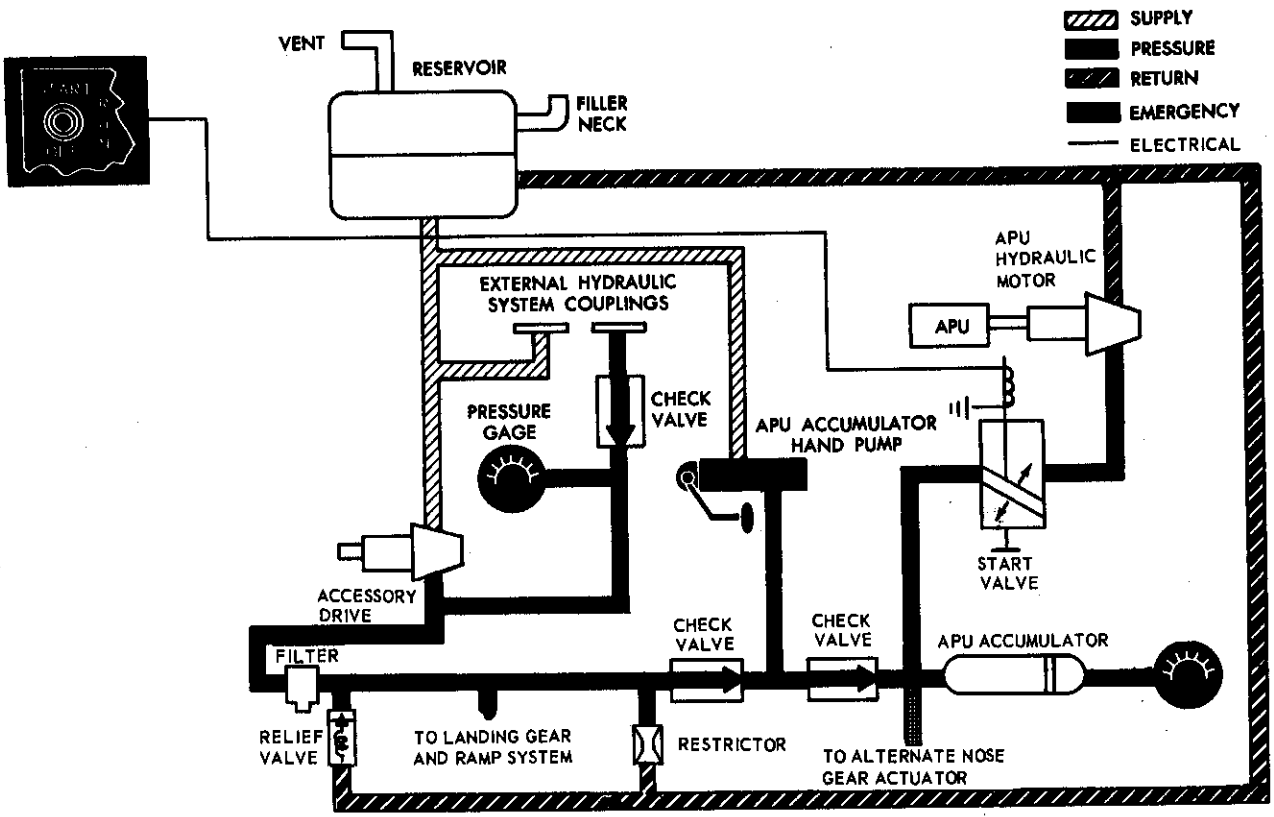


Figure 1-37. Utility Hydraulic System (Typical)

servo systems. The main rotor controls terminate at the stationary swashplate of the main rotor head. Control action is transmitted through the rotating swashplate and linkage on the main rotor hub to the blades.

**Collective to Yaw Coupling.**

A collective to yaw coupling, located in the mixing unit, provides automatic tail rotor blade angle changes to compensate for changes in collective pitch settings. The coupling is irreversible with the auxiliary servo system in operation and collective pitch motion will act to displace the tail rotor. Tail rotor pedal motion will not affect main rotor collective pitch blade angle. Tail rotor blade angle changes result from both collective pitch lever and tail rotor pedal inputs. A combination of collective pitch lever position and pedal position that would exceed the system limits cannot be obtained during flight. The collective pitch lever is always free to move within its full travel. During ground checks, if a collective pitch lever position is reached that adds to the pedal position and creates a tail rotor blade angle equal to the system limits, any further increase in collective pitch lever will cause a loss of pedal position. With auxiliary servo on, collective pitch lever low, and pedal full left, raising the collective pitch lever will be accompanied by pedal motion to the right. With the collective pitch lever high and pedal full right, reducing collective pitch will be accompanied by pedal motion to the left. With the auxiliary servo switch OFF, the irreversibility is not effective. When the combination of collective and yaw positions reaches the system limits, additional pedal motion is possible by a reduction in collective pitch. The trading of motion between collective and yaw will never occur in flight but may be encountered during ground checks. During rapid pedal motions on the ground, a noticeable noise can be heard behind the pilot's seat when the pedals reach their left or right limits. The sound is created by the system stops and indicates that collective pitch and the pedals have reached the limits of tail rotor control. Additional pedal motion is possible by reducing collective pitch.

**Collective to Cyclic Pitch Coupling.**

A bias in the collective cyclic pitch (lateral) coupling is incorporated in the mixing unit to apply a slight left roll correction when the collective pitch is raised.

**Collective Pitch Levers.**

Two collective pitch levers (13 and 26, figure 1-5) are located in the pilot's compartment. Both levers operate simultaneously to change the collective pitch of the main rotor blades. A friction lock on the pilot's collective pitch lever, marked COLLECTIVE PITCH LOCK with an arrow pointing left, marked INCREASE FRICTION, can be rotated to apply friction to prevent the collective pitch lever from creeping while in flight.

**Cyclic Sticks.**

The cyclic sticks (10 and 31, figure 1-5) provide lateral and longitudinal control of the helicopter. Moving the cyclic stick in any direction tilts the tip path plane of rotation of the main rotor blades in that direction and moves the helicopter in the same direction. The stick grip (figure 1-38) contains pushbutton and thumb-operated switches for controlling various equipment installed in the helicopter.

**Cyclic Stick Trim System.**

The cyclic stick trim system permits a fine degree of adjustment of the cyclic stick and provides cyclic stick feel. When used with the AFCS engaged, the system permits hands-off flight by holding the stick in a selected trim position. However, hands should be kept close to flight controls in case of an AFCS or cyclic stick trim system malfunction.

NOTE

With loss of DC electrical power, the cyclic stick will remain in the last trimmed position.

Two actuators are hydraulically powered by the auxiliary servo system and energized electrically from the dc essential bus. One actuator positions the cyclic stick laterally and the other actuator positions the cyclic stick fore-and-aft. The actuators are operated by a four-position cyclic trim switch mounted on both the pilot's and copilot's cyclic stick grips. To trim the cyclic stick, the cyclic trim switch is pushed in the direction of desired cyclic stick movement and the actuators move the stick until the cyclic trim switch is released. The cyclic stick may be manually displaced from the trimmed position, but a resistance force caused by the spring tension increases progressively. The spring tension provides cyclic stick "feel" and amounts to approximately 1-1/2 pounds initial force plus 1/2 pound for each one inch of cyclic stick movement. When the pressure on the cyclic stick is released, spring tension returns the stick to the original trim position. The cyclic stick trim system will operate as long as there is both DC power to the essential bus and auxiliary hydraulic pressure to the actuators.

**Cyclic Stick Trim Master Switch.**

A cyclic stick trim master switch, marked STICK TRIM MASTER, ON and OFF, is located on the overhead switch panel (figure 1-13). The switch is the master control for the cyclic stick trim system. When the switch is placed in the ON position, hydraulic pressure holds the cyclic stick in position. If the cyclic stick is moved from this position, the spring action of the force gradient system will resist any movement and attempt to return the cyclic stick to the initial position. When the switch is placed in the OFF position, the force gradient system is inoperative and the cyclic trim system will not position the cyclic stick.

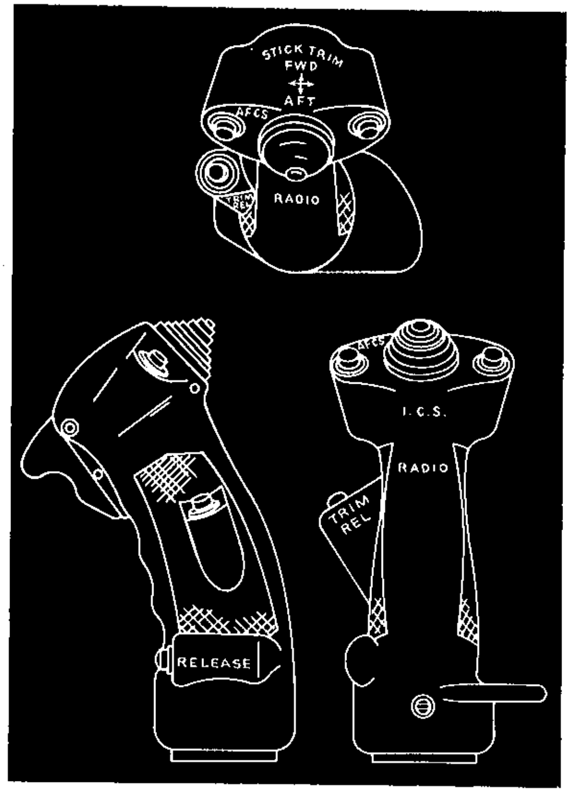


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Figure 1-38. Cyclic Stick Grip

**Cyclic Trim Switches.**

The cyclic trim switches, located on the pilot's and copilot's cyclic stick grips (figure 1-38), have marked positions FWD, AFT, L, and R. The four-way thumb switch is spring-loaded to the center (off) position. When the switch is placed in any of the four positions, hydraulic pressure will drive the cyclic stick in the same direction. When the desired cyclic stick position is obtained, the switch is released. The action of the cyclic stick trim system will then function about this location of the cyclic stick. The cyclic trim switches receive electrical power from the dc essential bus through a circuit breaker, marked STICK TRIM, located on the overhead dc circuit breaker panel.

**Cyclic Trim Release Button.**

The spring-loaded, pushbutton switches, located on the pilot's and copilot's cyclic stick grips (figure 1-38), marked TRIM REL, are used to change trim position (without using the cyclic trim switch). Cyclic trim position is changed by depressing the cyclic trim release button, moving the cyclic stick to the new position, and then releasing the cyclic trim release button. The cyclic trim system will then hold the selected position of the cyclic stick. The cyclic trim release button controls dc essential bus power to the trim actuators.

## TAIL ROTOR FLIGHT CONTROL SYSTEM.

The functions of the tail rotor flight control system are to compensate for main rotor torque and to provide a means for changing the heading of the helicopter. The torque developed by the main rotor blades turning counterclockwise tends to rotate the fuselage in a clockwise direction. Any change in power setting will vary the amount of main rotor torque. To compensate for torque variations, the pitch and resulting thrust of the tail rotor blades can be increased or decreased. Turns are accomplished by increasing tail rotor thrust, which overcompensates for main rotor torque and changes the heading of the fuselage to the left, or by decreasing the tail rotor thrust, which undercompensates for the main rotor torque and changes the heading of the fuselage to the right. Tail rotor control pedal movements are transmitted to the tail rotor assembly by mechanical linkage and cables. Control action is assisted by the auxiliary servo system only. A hydraulic damping device incorporated in the auxiliary servo prevents abrupt movements of the pedals, which would cause sudden changes in thrust developed by the tail rotor with resulting rapid yaw acceleration and possible damage to the helicopter. The pedal damper is inoperative when the auxiliary servo system is inoperative or shut off. Yaw compensation is accomplished by mechanical linkage in the mixing unit which automatically changes tail rotor blade angles for changes in collective pitch. If both collective pitch and tail rotor blade angle are at their maximum limits, the pedal will be forced back with collective pitch change. A tail rotor negative force gradient system is installed to relieve the pilot of tail rotor forces created by aerodynamic loads when the auxiliary servo system is inoperative. Because of this, when the system is checked on the ground with tail rotor stationary and the auxiliary servo off, a negative spring centering effect is created. The normal tendency of the pedals is then to go to either extreme. Under these conditions, considerable force is required to push the pedals from the extreme positions; however, the force will decrease as the neutral pedal position is approached. The initial force to move the pedals toward the right from a full left position is approximately 10 to 15 pounds.

### Tail Rotor Pedals.

The tail rotor pedals (17 and 27, figure 1-5) change the pitch and thrust of the tail rotor and consequently the heading of the helicopter. Electrical switches, mounted on the force link assembly, cancel the directional signals of the automatic flight control system when approximately 8 pounds of pressure is exerted on either pedal. Toe brake pedals for the main landing gear wheel brakes are mounted on both the pilot's and copilot's pedals.

### Tail Rotor Pedal Adjustment Knobs.

Pedal adjustment knobs (14 and 29, figure 1-5) are located on each side of the fuselage, just forward of the ash trays in the pilot's compartment. The adjustment knobs are connected to mechanical linkage that provide for fore-and-aft adjustment of the pedals. The knobs are rotated to the right, as indicated by the arrow marked FWD, for forward adjustment and to the left, as indicated by the arrow marked AFT, for aft adjustments. The pilot's pedals are adjusted with the knob on the right side of the fuselage, and the copilot's pedals are adjusted by the knob on the left side of the fuselage.

NOTE

Adjust tail rotor pedals with feet off to avoid damage or breakage to pedal adjustment cables.

## FLIGHT CONTROL HYDRAULIC POWER SUPPLY SYSTEM.

The flight control hydraulic power supply system (figure 1-39) consists of a primary and an auxiliary flight control servo system. The servo systems are required by the pilot for a power boost to operate the controls. The servos also prevent feedback of main rotor vibratory loads to the control sticks. Both servo systems operate from independent hydraulic systems and utilize similar servo hydraulic units to vary the main and tail rotor blade pitch through the mechanical linkage of the regular flight control system. The servo unit output is connected to the flight control linkage to provide the power boost. The continuity of the direct control linkage is maintained from the controls in the pilot's compartment through the auxiliary and the primary servos to the main rotor blades, except for a slight amount of end play at each servo unit to permit the pilot valves to move before the direct control linkage. Normally, both servo systems are in operation at all times.

### Primary Flight Control Servo System.

The primary flight control servo system consists of three hydraulic servo units which connect the flight control linkage to the stationary swashplate of the main rotor assembly. The servos provide the power necessary for the pilot in the operation of the main rotor flight control system only. The primary servo hydraulic pump is driven by the accessory section of the main gear box. The primary hydraulic system reservoir, mounted aft of the main gear box, has a capacity of approximately 0.45 gallon of hydraulic oil. When primary servo pressure fails, the PRI SERVO PRESS light on the caution panel illuminates.

### Auxiliary Flight Control Servo System.

The auxiliary flight control servo system, consisting of a bank of four hydraulic servo packages, located below the main rotor flight control system mixing unit, provides the means of introducing AFCS corrective signals into the flight control systems and, in the event of primary servo failure, reacts to flight loads and tail rotor loads. The auxiliary servo hydraulic system reservoir, has a capacity of

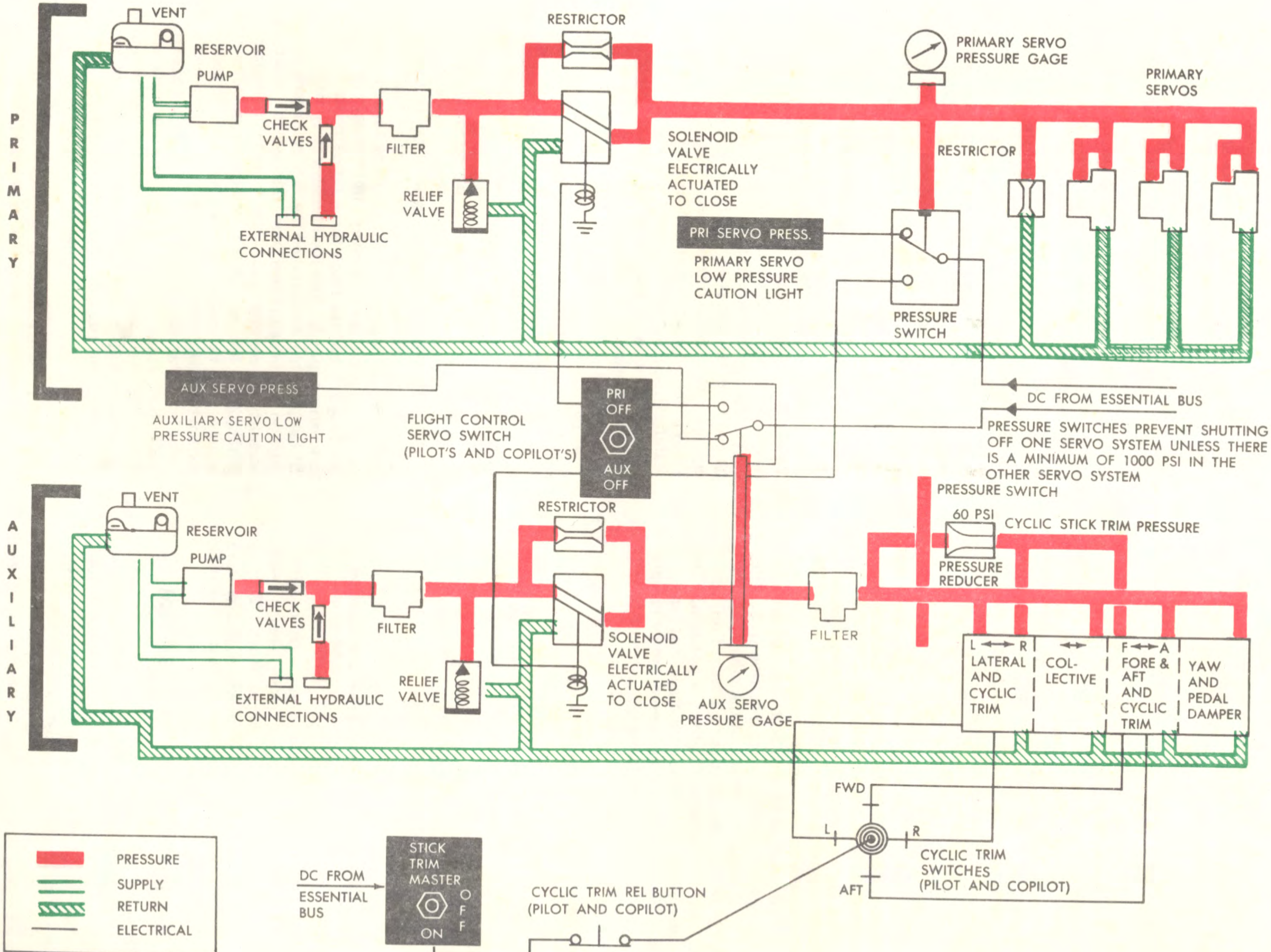


Figure 1-39. Flight Control Servo Hydraulic Systems

approximately 0.45 gallon of hydraulic oil. When auxiliary servo pressure fails, the AUX SERVO PRESS light on the caution panel illuminates.

**Flight Control Servo Switches (Servo Switches).**

The primary and the auxiliary flight control servo systems are controlled by the servo switches, marked SERVO, located on the pilot's and copilot's collective pitch lever grips (figure 1-11). The marked switch positions are PRI OFF and AUX OFF. When functioning properly, both servo systems are in operation when both the pilot's and copilot's switches are in the unmarked center (ON) position. To turn off the primary servos, either the pilot's or copilot's switch is placed in the forward PRI OFF position, and to turn off the auxiliary servos, the switch is placed in the aft (AUX OFF) position. The systems are interconnected electrically in such a way that, regardless of the switch position, it is impossible to turn either system off, or for it to remain off, unless there is a minimum of 1000 psi in the other system for proper operation. Therefore, it is impossible to turn both systems off by placing the pilot's switch in one position (i.e. PRI OFF) and the copilot's switch in the other position (i.e. AUX OFF). The first switch turned off has control until it is returned to the ON position. The servo shut-off valves operate on direct current from the essential bus and are protected by circuit breakers on the overhead dc circuit breaker panel, marked SERVO CUT-OFF, PRI and AUX.

**Servo Hydraulic Pressure Indicators.**

The primary and auxiliary servo hydraulic pressure indicators (figure 1-14) are located on the instrument panel. The indicators operate on 26 volts ac from the inverter bus and are protected by circuit breakers under the heading HYD PRESS IND marked PRI and AUX located on the ac essential circuit breaker panel.

**Primary and Auxiliary Servo Pressure Caution Lights.**

The primary and auxiliary servo pressure caution lights are located on the caution panel (figure 1-20) on the pilot's side of the instrument panel. The lights will illuminate when pressure in the respective servo system drops below 1000 psi or the system is turned off.

## AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS).

The automatic flight control system (AFCS) maintains the stability of the helicopter in its reference pitch and roll attitudes, about the reference directional heading, and at the engaged altitude, to permit automatic hands-off flight and controlled hovering operations. The AFCS used in this helicopter differs from the auto-pilot used in fixed-wing aircraft in that it may be engaged at all times, has less control authority than the primary flight control system, and may be easily overridden through normal use of the flight controls. The pilot has direct control of the system at all times and can engage or disengage the entire AFCS or any channel, as desired, by means of switches on the AFCS control panel, channel monitor control panel, cyclic sticks, and collective pitch levers. The AFCS indicator and attitude indicators provide the pilot and copilot with visual indication of all AFCS signals. AFCS has two modes of operation: (1) attitude and directional stabilization, and (2) barometric altitude stabilization. Attitude and direction stabilization is controlled through the pitch, roll, and yaw channels; and barometric altitude stabilization is controlled through the collective channel. AFCS is capable of maintaining the barometric altitude of the helicopter within $\pm$ 40 feet or $\pm$ 5% of the altitude, whichever is less, during straight unaccelerated flight, or when hovering out of ground effect by utilizing barometric altitude reference. In the pitch and roll channels, the fuselage attitude is held constant by comparing the acutal attitude signal received from the vertical gyro with the reference attitude signal received from the stick position sensor (senses positon of cyclic stick). Automatic pitch and roll attitude stability correction occurs any time the helicopter is displaced from the reference attitude. On helicopters equipped with the navigation set, radar (AN/APN-175(V), the doppler antenna receives pitch and roll stabilization information from the gyro selected on the channel monitor panel. In the yaw channel, the helicopter heading is held constant by comparing actual heading signals received from the J-4 compass system with reference heading signals received from the YAW TRIM knob and the tail rotor pedals. While the pilot establishes a reference heading by use of the pedals, the yaw channel is placed in a synchronizing mode (no heading correction signal is developed) until his feet are removed from the pedals. During the synchronizing mode, the yaw rate gyro develops a signal proportional to the manual heading displacement rate of the helicopter. This signal initiates an open-loop spring condition that produces a proportional feedback force at the pedals. As the pilot presses either pedal, he feels the proportional feedback force opposing the pedal pressure applied. The feedback force remains until the pilot has established the new reference heading and removes his feet from the pedals. Heading stability correction occurs any time the helicopter is displaced left or right from the desired reference heading. In the collective channel, the engaged barometric altitude of the helicopter is held constant by signals developed from the altitude controller, which senses changes in barometric pressure. Automatic barometric altitude stability correction occurs any time the helicopter is displaced up or down from the engaged reference altitude.

NOTE

If strong updrafts or downdrafts cause the helicopter to be displaced more than 200 feet from the engaged altitude, the barometric altitude channel should be disengaged to prevent possible damage to the barometric altitude controller.

AFCS utilizes both ac power from the ac essential bus and dc power from the dc essential bus. A thermal time delay relay is incorporated to allow approximately 3 minutes for the vertical gyros to reach a stabilized state before dc power is applied to the system. The AFCS ENG button may then be depressed to engage the pitch, roll, and yaw channels. The BAR ALT ENG button is then depressed to engage the collective channel. The AFCS ENG button must be depressed before the BAR ALT ENG button is depressed. AC power to the AFCS is protected by a circuit breaker, marked AFCS, located on the ac essential bus circuit breaker panel. DC power to the AFCS is protected by a circuit breaker, marked AFCS PWR, and two circuit breakers under the general heading TURN RATE and marked AFCS and GYRO, all of which are located on the overhead dc circuit breaker panel.

**AUTOMATIC FLIGHT CONTROL SYSTEM CONTROL PANEL.**

The AFCS control panel marked AFCS CONT, is located on the cockpit console (figures 1-17 and 1-18) between the pilot and copilot. Controls consist of two engage buttons; one marked AFCS ENG and the other BAR ALT ENG, an off button, marked BAR ALT OFF, a yaw trim knob marked, YAW TRIM; and a center-of-gravity trim knob, marked CG TRIM. The AFCS ENG button is depressed to engage the pitch, roll, and yaw channels, and the BAR ALT ENG button is depressed to engage the barometric altitude controller. Each engage button is equipped with a light that will illuminate to indicate engagement. Once engaged, the entire AFCS can be disengaged by depressing the button, marked AFCS REL, located on the pilot's and copilot's cyclic stick grips. The barometric altitude controller can be completely disengaged by depressing the BAR ALT OFF button, or released momentarily to make changes in altitude by depressing the BAR REL button located on the pilot's and copilot's collective pitch lever grip. The YAW TRIM and CG TRIM knobs, located on the bottom half of the panel, are designed with characteristic shapes to enable the pilot to easily distinguish between them. The YAW TRIM knob is triangular-shaped and the CG TRIM knob is clover leaf-shaped. The YAW TRIM knob enables the pilot to accurately trim the heading of the helicopter, provided his feet are off the tail rotor pedals. One rotation of the knob turns the helicopter 72 degrees. The CG trim control permits the pilot to recenter the pitch servo valve after a shift in the center of gravity. By a coordinated movement of the CG trim knob and the cyclic stick, the pilot may fly the helicopter at any pitch attitude, and the AFCS will retain its full ability to provide stabilization.

**CAUTION**

Do not adjust CG trim during a transitional phase of flight, such as takeoffs and landings or when operating near the ground and visual references are not available.

**CHANNEL MONITOR CONTROL PANEL.**

The channel monitor control panel, marked CHANNEL MONITOR, is located on the pilot's console (figure 1-40)to the right of the pilot's seat. The controls consist of four toggle switches across the top of the panel, marked PITCH, ROLL, COLL, and YAW, under the general heading CHANNEL DISENGAGE, with marked position ON and OFF. These toggle switches permit individual disengagement of the the pitch, roll, collective, and yaw channels of AFCS. They are usually left in the ON position, except when the pilot wishes to disengage a malfunctioning channel. Directly below the four toggle switches are four guarded three-position switches, under the general heading HARDOVER, that are normally used only to check system authority during a ground operational check-out. The PITCH switch has marked positions FWD and AFT. The ROLL and YAW switch have marked positions LEFT and RIGHT The COLL switch has marked positions UP and DOWN. Each switch may be placed in a position to check the corresponding PITCH, ROLL, COLL, or YAW channels of AFCS. When the switch guards are closed, the switches are held in the ON position. The switch guards must be lifted before override checks can be accomplished. The toggle switch on the bottom of the panel is marked GYRO SELECT with marked positions, LEFT and RIGHT. Pitch and roll references for AFCS are selected from either the left or right gyro; however, the normal position is LEFT. With the gyro select switch in the LEFT position, the left gyro provides signals for AFCS and the copilot's attitude indicator while the right gyro feeds the pilot's attitude indicator. If a gyro should fail with the switch in the LEFT position, the pilot will not lose the services of both his attitude indicator and AFCS pitch and roll channels simultaneously.

**WARNING**

Actuation of the hardover switches with the AFCS engaged or disengaged can result in a full AFCS command hardover condition for the channel actuated.

**AFCS (AUTOMATIC FLIGHT CONTROL SYSTEM) INDICATOR.**

Two indicators (34 and 59, figure 1-14) (referred to as AFCS indicator) are installed in front of the pilot and the copilot on the instrument panel. The indicators provide a visual indication of the output signals from the AFCS pitch, roll, yaw, and collective channels. On helicopters equipped with a doppler navigation system, the indicators provide a visual indication of doppler reliability during cruise, and displays ground speed, drift, and vertical velocity information. Each indicator contains scale increment marks located across the center vertical and horizontal axis and along the left and bottom sides of the dial face. Two movable bars coincide with the center vertical and horizontal axis scale marks of the dial and intersect perpendicularly at a small circle marked on the dial face. There are two arrowhead-type pointers provided, one located on the left-hand side of the indicator which moves vertically, up or down, coinciding with the vertical scale, while the other pointer at the bottom of the indicator moves horizontally, left or right, coinciding with the horizontal scale. On helicopters not equipped with a navigation set, radar, the AFCS indicator has provisions for three modes of operation

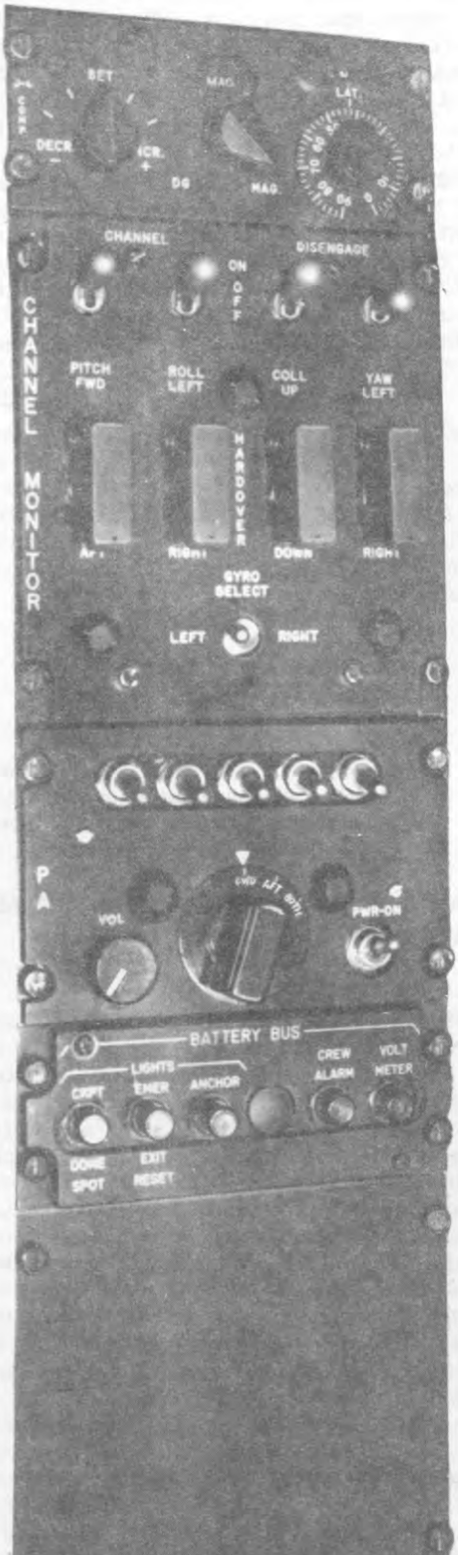


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Figure 1-40. Pilot's Console (Typical)

but utilizes only the A mode for monitoring AFCS output signals. On helicopters equipped with a navigation set, radar, the indicator uses the A mode to monitor AFCS output signals and the D mode to monitor doppler navigation reliability and indications. The C mode is inoperative. The mode selector knob, located on the lower left side of the indicator, is marked A, D, and C. An index mark on the case indicates the mode selector knob position. The mode indicator window in the upper right quadrant of the dial indicates which mode has been selected. The selector should remain in the A position for AFCS indications and the D position for doppler navigation indications. With the AFCS ENG button engaged and mode A selected on the indicator, the AFCS indicator will monitor the input to the AFCS servo valves. The horizontal bar is used to monitor the pitch channel, the vertical bar monitors the roll channel, the vertical pointer monitors the altitude channel, and the horizontal pointer monitors the yaw channel. The scale reference marks are spaced to equal zero, 25, 50, 75, and 100 percent of full scale deflection in either direction. All AFCS output signals are routed through the channel monitor control panel to the AFCS indicator. On helicopters equipped with a navigation set, radar, operation in the D mode provides a visual indication of doppler reliability, during cruise, and displays ground speed, drift, and vertical velocity during hover operation at speeds up to 22.5 knots ground speed. When operating in the D mode, the AFCS indicator is connected to the navigation set, radar. Below 22.5 knots ground speed, the horizontal bar indicates forward or rearward velocities and the vertical bar indicates left or right drift. Each increment of the AFCS indicator horizontal and vertical scales indicates 5 knots with a maximum indication of 20 knots. The vertical pointer indicates vertical velocity, with each increment equal to 500 feet per minute. Full scale deflection is equal to 2000 feet per minute up and 2000 feet per minute down. To indicate forward flight up to 22.5 knots, the horizontal bar will move downward and, to indicate a drift, the vertical bar will move in a direction opposite to the direction of the drift. Therefore, the pilot flies into the bar intersection for correction. In cruise flight above 22.5 knots ground speed, only the OFF flag and ground speed bar are operative. In the D mode the yaw pointer is disconnected and should not move. An OFF flag on the upper dial face of the AFCS indicator is used in both modes of operation. In the A mode, the flag disappears when the AFCS is engaged. In the D mode, the flag disappears when the doppler transmitter is turned on and doppler return signals are being received. Engage voltage for warning flag operation is supplied from the dc essential bus through a circuit breaker, marked AFCS PWR, located on the overhead dc circuit breaker panel.

### AFCS RELEASE BUTTON.

AFCS is disengaged by depressing the buttons marked AFCS REL which are located on both the pilot's and copilot's cyclic stick grips (figure 1-38).

**BAR ALT RELEASE SWITCHES.**

The barometric altitude controller is momentarily disengaged, when changing altitude, by holding down the buttons, marked BAR REL, which are located on both the pilot's and copilot's collective pitch lever grips (figure 1-11). After stabilizing attitude and airspeed at the new altitude, the BAR REL button is then released and the helicopter will be stabilized at the new altitude.

## INSTRUMENTS.

The instruments that operate on either alternating current, direct current, or both are protected by appropriately marked circuit breakers, located on the overhead circuit breaker panels in the pilot's compartment.

**MAGNETIC COMPASS.**

A magnetic compass (9, figures 1-5 and 1-6) is located at the top center of the instrument panel. A standby compass correction card is located on the pilot's side of the instrument panel.

**NOTE**

On helicopters equipped with a ground pressure and air refueling system, the compass will be unreliable when the pressure refueling panel is energized and operating.

**FREE AIR TEMPERATURE GAGE.**

The bimetallic free air temperature gage (38, figure 1-6), located in the windshield glass above the instrument panel, is a direct reading instrument that is calibrated in degrees Centigrade.

**CLOCKS.**

Two eight-day, 12-hour clocks (30 and 54, figure 1-14) are installed on the instrument panel. The control knob for the elapsed-time mechanism is located at the upper right corner of the clock face. The clock is stem-wound and stem-set with a knob located in the lower left corner of the clock face.

**PITOT-STATIC SYSTEM.**

There are two pitot-static systems. The pitot portions of the pilot's and copilot's systems are independent of each other, but the static portion of each system uses common tubing. Each pitot-static pressure system consists of a heated pitot-static tube, altimeter, airspeed and velocity instruments. The pitot and static lines both originate at the pitot-static tubes. The opening at the head of the tubes furnishes total pressure, and ports near the center of the tubes furnish static pressure. The static system vents the airspeed, altimeter, and vertical velocity instruments to atmospheric pressure. The pitot-static tube on the right side of the cockpit canopy furnishes ram air pressure to the pilot's airspeed indicator and static pressure to the common static system. On helicopters equipped with a doppler navigation system, the pitot tube on the right side of the cockpit canopy also furnishes pitot and static pressures to the true airspeed transmitter. The pitot-static tube on the left side of the cockpit canopy furnishes ram air pressure to the copilot's airspeed indicator and static pressure to the common static system. Capped tees in the lines in the electronics compartment and in the cargo compartment permit draining moisture from the lines. The AFCS barometric altitude sensing unit is connected into the pitot-static tube line. A resistance-type heater in the pitot-static tubes, controlled by the PITOT HEAT switch on the overhead control panel, prevents formation of ice at the openings. A drain hole near the forward edge of the pitot-static tubes permits water to escape. Power for the pitot-static tube heaters is supplied by the essential dc bus system, through the ice protection PITOT HEAT 1 and 2 circuit breakers on the overhead switch panel.

**ALTIMETER-ENCODER AAU-21/A.**

One altimeter-encoder (figure 1-41) is installed in the pilot's instrument panel. The altimeter-encoder combines a conventional barometric type altimeter, possessing a counter-drum-pointer display, with an altitude reporting encoder in one self-contained unit. The 10,000 and 1,000 foot counters and the 100 foot drum provide a direct digital output and readout of altitude in increments of 100 feet, from -1000 to 38,000 feet. The digital output is referenced to 29.92 in $H_g$. and is not affected by changes of barometric setting. The pointer repeats the indications of the 100 foot drum, and serves both as a vernier for the drum and as a quick indication of the rate and sense of altitude changes. Two methods may be used to read indicated altitude on the counter-drum-pointer altimeter: (1) read the counter-drum window, without reference to the pointer, as a direct digital readout in thousands and hundreds of feet, or (2) read the thousands of feet on the two counter indicators, without referring to the drum, and then add the 100 foot pointer indication. The self-contained servo driven encoder provides altitude encoded in 100 foot increments for automatic transmission when the AIMS/IFF transponder is interrogated on mode C. In case of power loss to the encoder servo system, a CODE OFF flag appears automatically in a window in the upper left portion of the display, indicating that altitude information is no longer being transmitted to the ground. In this condition, the instrument continues to function as a normal barometric altimeter. The barometric pressure is entered by use of a barometric set knob in the lower left front of the instrument case. The altimeter setting appears on counters in the window at the lower right of the display and has a range of settings from 28.1 to 31.0 in $H_g$. An internal vibrator operates continuously whenever aircraft DC power is turned on. The vibrator minimizes internal mechanical friction, enabling the instrument to provide a smoother display during changing altitude conditions. Should vibrator failure occur, the altimeter will continue to function pneumatically, but a less-smooth movement of the instrument display will be evident with changes in altitude.

**WARNING**

If the internal vibrators of the altimeter encoder or altimeter are inoperative due to either internal failure or dc power failure, the 100-foot pointers may momentarily hang up when passing through 0 (12 o'clock position). If the vibrators have failed, hangup of the 100-foot pointers can be minimized by tapping the case of the altimeters. Pilots should be especially watchful for this failure when the minimum approach altitude lies within the 800-1000 foot part of the scale (1800-2000 feet, etc.).

**NOTE**

When each 1,000 foot increment is nearly completed, the counter(s) abruptly index to the next digit. The counter-drum-pointer altimeter mechanism may also cause a noticeable pause or hesitation of the pointer due to the additional intermittent friction and inertia loads applied to the mechanism to turn over the thousand-foot counter. This affect may be more pronounced at ten-thousand foot intervals where both counters are turned over simultaneously. This momentary pause is followed by a noticeable acceleration as the altimeter mechanism overcomes the counter wheel load and rolls the dial over to the next digit. The pause occurs during the "9" to "1" portion of the scale. The pause-and-accelerate behavior is normally more pronounced at high altitudes and high rates of ascent and descent. During normal rates of descent or ascent and at low altitudes, the effect will be minimal.

## ALTIMETER AAU-27/A.

One altimeter (figure 1-41) is installed in the copilot's flight instrument panel. The instrument is similar to the altimeter-encoder except it does not have an altitude encoder nor the CODE OFF display mechanism. The indicated altitude on the altimeter is from -1000 to 50,000 feet. The altitude display, altimeter setting, and vibrator considerations described for the altimeter-encoder also apply to the altimeter.

## ATTITUDE INDICATORS.

Two (Lear 4005T) attitude indicators (5 and 26, figure 1-14), installed on the instrument panel in front of the pilot and copilot give a visual indication of the helicopter's flight attitude. The indicator face consists of a stationary miniature airplane representing the helicopter, a bank angle scale, bank index, and a moving two-colored sphere with a district white horizon line dividing the two colors, white above, black below, and a turn-and-slip indicator, located on the bottom of each attitude indicator that gives visual indication of the helicopter's rate-of-turn and balanced flight. The turn needle is a two

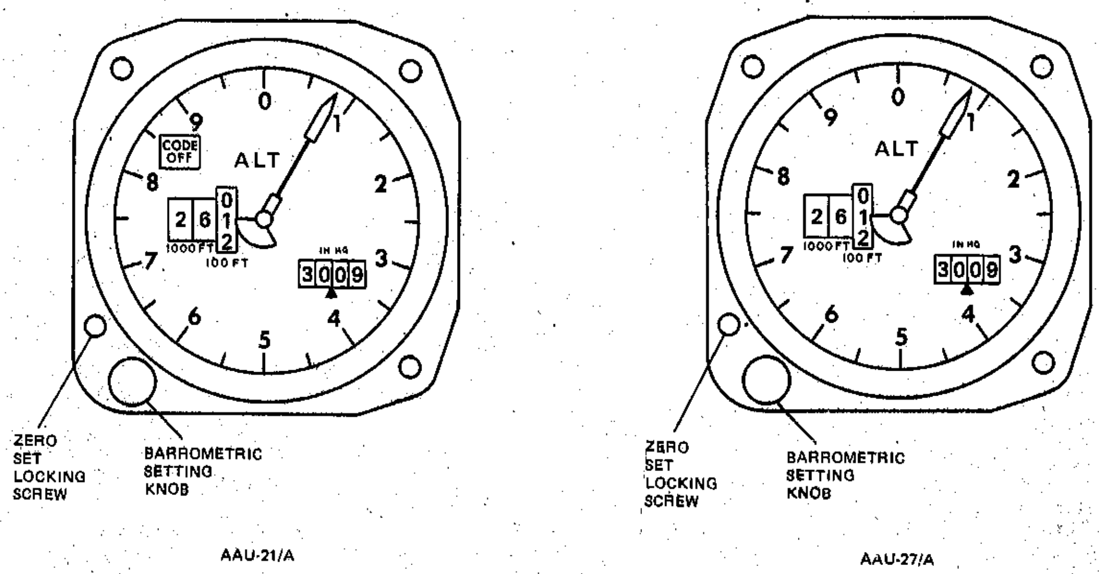



Figure 1-41. Altimeter-Encoder AAU-21/A and Altimeter AAU-27/A

minute needle; however, the calibration marks are the same as a four minute needle. For a standard rate turn of 3° per second, one needle width deflection is required. The pilot's attitude indicator receives pitch and roll information from the right vertical gyro and the copilot's attitude indicator receives pitch and roll information from the left vertical gyro. A power warning flag, marked OFF, will appear in the face of the indicator under the following circumstances: (1) when ac power has not been applied, (2) for approximately the first 68 seconds after ac power has been applied, (3) and when any unbalance or failure of the three phases of ac power occurs.

**WARNING**

A slight reduction in electrical power or failure of certain electrical components within the system will not cause the attitude warning flag to appear even though the system may not be functioning properly. Therefore, it is imperative that the attitude indicator is periodically cross-checked with other flight instruments.

Two trim adjustment knobs are located on the front of the attitude indicators, one at the lower left of the panel for adjusting roll, and the other at the lower right of the panel for adjusting pitch. The roll trim knob adjusts the bank index position from 8 to 20 degrees, right and left bank. The pitch trim knob rotates the sphere to deflect the horizon line upward, when the pitch trim knob is rotated clockwise from the zero pitch trim adjustment white dot to indicate between 4 and 10 degrees dive. The sphere may be rotated downward when the pitch trim knob is rotated counterclockwise from the zero pitch trim adjustment white dot to indicate between 8 and 20 degrees climb. The pilot's attitude indicator operates on current from the No. 2 generator and is protected by a circuit breaker, marked ATTITUDE IND PILOT, located on the ac nonessential circuit breaker panel. The copilot's attitude indicator operates on current from the ac essential bus and is protected by a circuit breaker, marked ATTITUDE INDICATOR CO-PILOT, located on the ac essential circuit breaker panel. Should the No. 2 generator fail, the pilot's attitude indicator will automatically be transferred to the ac essential bus.

## TURN RATE SWITCHES.

Two turn rate switches (32 and 56, figure 1-14), one each for the pilot and copilot, are located on the instrument panel. The switches, marked TURN RATE, have marked positions NORM and ALT. When the switches are placed in the NORM position, the copilot's turn and slip indicator receives rate-of-turn information from the AFCS yaw rate gyro, and the pilot's turn and slip indicator receives rate-of-turn information from a separate rate gyro. When the switches are placed in the ALT position, rate-of-turn information is received by the pilot's turn and slip indicator from the AFCS yaw rate gyro, and the copilot's turn and slip indicator receives rate-of-turn information from the separate rate gyro.

## VERTICAL VELOCITY INDICATOR.

Two vertical velocity indicators (6 and 29, figure 1-14), located on the instrument panel, indicate the rate of ascent or descent in feet-per-minute up to 6000 feet-per-minute. The vertical velocity indicators are connected to the pitot-static system in series with the altimeter and utilize the same static source as the altimeter.

## J-4 COMPASS SYSTEM.

The J-4 compass system provides precise heading reference information outputs for the azimuth coupler of the TACAN receiver-transmitter, the compass card of the pilot's and co-pilot's BDHI's, and to the heading pointer of the pilot's and copilot's course indicators. The compass system also provides separate heading reference outputs for the Automatic Flight Control System (AFCS) that are not affected by the selection of either the MAG mode or the DG mode of operation. On those helicopters equipped with a navigation set, radar, the compass system augments the doppler system with magnetic heading information. The system can be operated as a magnetic slaved compass or as a directional gyro, manually corrected for the particular latitude and hemisphere where the helicopter is operating. When the system is operated in the magnetic compass (MAG) mode, the gyro is slaved to the earth's magnetic field. The compass magnetic azimuth detector then acts as the direction detector device and the compass directional gyro provides stability influence. When the system is operated in the directional gyro (DG) mode, the compass directional gyro rotates freely in the horizontal plane without reference to the earth's magnetic field. Heading indications are then corrected for apparent gyro drift due to earth's rotation by manually setting in a latitude correction signal and a hemisphere correction signal from the compass control panel. Operation as a latitude corrected directional gyro meets the requirements of flights in high latitude or where poor magnetic references exist. The system consists of a magnetic flux valve, located in the tail cone; a directional gyro and signal synchro amplifier; an amplifier and a control panel, located in the pilot's compartment; and a power adapter transformer, located in the cargo compartment. The J-4 compass system receives dc electrical power from the essential bus through a circuit breaker, marked COMPASS, located on the overhead dc circuit breaker panel, and ac electrical power from the ac essentail bus through a circuit breaker, marked COMPASS PWR ADPT XMFR, located on the ac essential circuit breaker panel.

### J-4 Compass Control Panel.

The J-4 compass control panel, located on the pilot's console (figure 1-40), contains all the controls for operation of the compass system. The function selector switch has marked positions DG and MAG. When the function selector switch is placed in the DG position, the compass system will function as a free directional gyro, with either north or south latitude corrections for the drift effect of the rotation of the earth. When the switch is placed in the MAG position, the gyro is slaved to the earth's magnetic field. Directional reference signals are obtained from the remote compass flux valve. In this mode of operation, the gyro is used as the stabilizing element of the compass system. The rotating azimuth card on the BDHI's will indicate stabilized magnetic headings. The annunciator meter, marked MAG on the compass control panel, provides indication of the synchronization of the compass system. In magnetic compass (MAG) operation, the synchronization circuits automatically synchronize the gyro to the remote compass flux valve. Automatic synchronization is established in the first 15 seconds of MAG operation and is maintained during operation of the compass system in this mode. In the directional gyro (DG) mode of operation, synchronization is not established or maintained automatically. Manual synchronization is accomplished by operating the SET synchronizer switch on the compass control panel. When the SET synchronizer switch, with marked positions DECR (-) and INCR (+), is displaced to the right or left, a synchronized signal is provided through the system at the rate and direction required to set the compass card of the pilot's and copilot's BDHI's, as desired. Synchronization of the system is indicated on the annunciator meter when the pointer is in line with the white arrow on the control panel. Reference heading data for the AFCS system is not affected by the synchronization cycles. The BDHI's compass cards can also be repositioned by use of the SET synchroizer switch when the compass is in magnetic compass (MAG) operation. The switch marked LAT is used to set in the proper magnitude of latitude correction during the DG mode of operation. Maximum drift of the directional gyro occurs when the gyro spin axis is at right angles to the earth's spin axis (north and south poles), and zero drift occurs when the two axes are aligned at the equator. When the helicopter is being flown in a northerly or southerly direction, the latitude compensation control should be set periodically to the latitude at which the helicopter is flying. The hemisphere selector switch, marked N and S, provides selection of the hemissphere in which the helicopter is being flown.

## CAUTION AND ADVISORY PANELS.

### CAUTION PANEL.

The caution panel (figure 1-20), marked CAUTION, is located in the center of the instrument panel. The caution panel gives the pilots visual indication of failure or unsafe conditions of certain critical items in the helicopter. The caution lights, each having its own operating circuit, indicate a particular condition in the helicopter. If a failure or unsafe condition occurs in one of the systems, the caution light for that particular condition remains on until the failure or unsafe condition is corrected. The warning lights operate through two circuit breakers, marked PWR and TEST, located on the overhead dc circuit breaker panel. The circuit breaker marked PWR provides electrical power for the normal operation of the warning lights, and the circuit breaker marked TEST provides power for the test circuit only. For caution light panel indications, refer to figure 7-5 in section VII.

#### Master Caution Light.

The master caution light (23, figure 1-14), marked MASTER CAUTION, located on the instrument panel, illuminates whenever a caution light illuminates to focus the attention of the pilot on a condition or malfunction within the helicopter. However, the light is a press-to-reset type and, after the specific condition or malfunction has been noticed on the caution panel, the light can be reset to provide a similar indication if a second condition or malfunction should occur while the first is still present.

**NOTE**

The master caution light will go out if the malfunction corrects itself.

### ADVISORY PANEL.

The advisory panel (figure 1-20), marked ADVISORY PANEL, is located below the caution panel on the instrument panel. The advisory panel gives the pilots visual indication of certain operating conditions that exist in flight or while on the ground. The advisory panel contains placard-type green advisory lights, each having its own operating circuit, to indicate a particular system is in operation or a certain flight condition exists. When a system is in operation or a certain flight condition exists, the advisory light for that particular system or condition comes on and remains on until the system is turned off or the condition no longer exists. For advisory light panel indications, refer to figure 7-6 in section VII.

### CAUTION AND ADVISORY LIGHTS TEST SWITCH.

The caution and advisory lights test switch, marked LAMP TEST, located on the caution panel, provides a means of simultaneously checking all light filaments by a single pushbutton type switch. The switch receives power from the 28 volt dc essential bus through a circuit breaker, under the heading INDICATOR LTS, and marked TEST, located on the overhead dc circuit breaker panel.

### DIM-BRIGHT SWITCH.

The switch, marked DIM and BRT, located on the caution panel, enables selection of a dim or bright brilliance of the caution and advisory lights. The switch cannot be utilized until the rheostat, marked PILOT FLT INST, located on the overhead switch panel, has been turned on.

## LANDING GEAR SYSTEM.

The tricycle configured landing gear system consists of dual retractable main landing gear assemblies, a single retractable nose gear assembly, and a hydraulic system (figure 1-42). The landing gear hydraulic system operates on 3000 psi hydraulic pressure from the utility hydraulic system. A landing gear warning system is installed to alert the pilots that the landing gear is not down and locked when at airspeeds of approximately 60 knots. The necessary electrical power is provided from the dc essential bus, through circuit breakers, under the general heading LAND GEAR and marked EMER DN, NOSE, MAIN, located on the overhead dc circuit breaker panel. The landing gear warning system is powered by the dc essential bus through a circuit breaker, marked LAND GEAR WARN, located on the dc overhead circuit breaker panel. The main landing gear system is equipped with a one-shot pneumatic, alternate extension system, and the APU accumulator is utilized as an emergency extension power source for the nose gear assembly. The nose gear assembly retraction system can be further utilized on the ground to kneel the helicopter to facilitate cargo handling. The landing gear control panel, marked LANDING GEAR CONTROL, is located on the cockpit console (figures 1-17 and 1-18). The watertight hull, plus sponsons, provide the helicopter with amphibious capabilities. Helicopters CH-3E 16▸ and HH-3E 25▸ are equipped with provisions for a fixed landing gear.

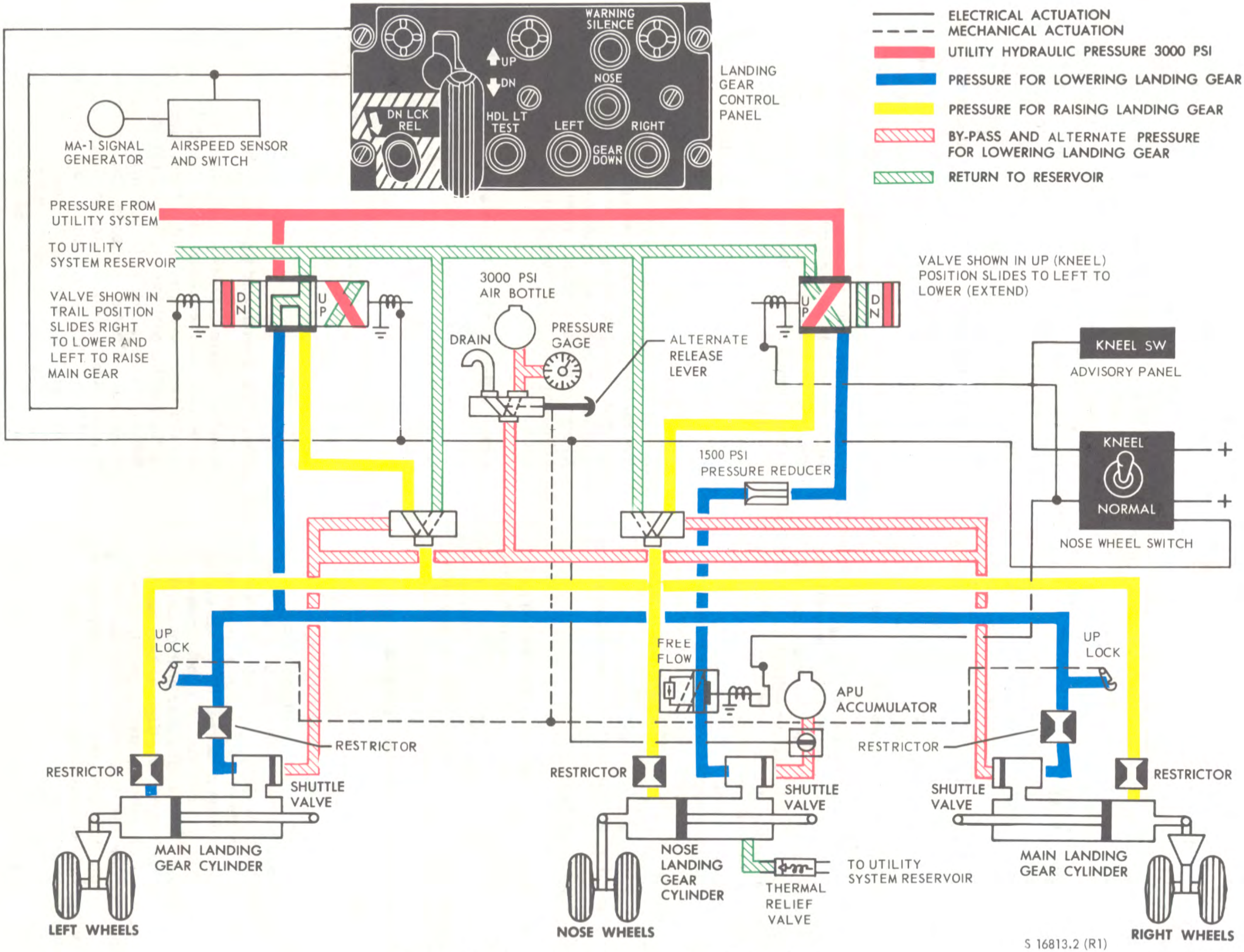



Figure 1-42. Landing Gear Hydraulic System

## MAIN LANDING GEAR.

The two main landing gear assemblies (27, figure 1-3) are located below the sponsons and retract forward and upward into the sponsons. Each main landing gear is equipped with dual wheels and hydraulic brakes, a retracting cylinder, a pneudraulic strut, attaching drag links and supports, and up and down lock mechanisms.

## NOSE LANDING GEAR.

The single nose landing gear (24, figure 1-44), mounted vertically at the centerline of the helicopter, is free to rotate 360 degrees about the strut centerline. All nose gear motion, including shock stroke, kneeling, jacking, and retraction, is vertical. The nose gear assembly is equipped with dual wheels, a retracting cylinder, a pneudraulic strut and shimmy damper, and attaching drag links and supports. The entire pneudraulic strut acts as a piston which is lowered or raised for retracting, jacking, and kneeling. The nose gear may be kneeled (retracted) 14.8 inches so as to alter the ground clearance of the tail section to facilitate cargo handling. The nose gear assembly is hydraulically locked in the extended, retracted, or kneeled position. The shimmy damper is utilized to minimize and offset nose gear vibration encountered during forward motion on the ground. A centering cam centers the nose gear assembly when the helicopter is airborne. A nose wheel lock is installed to improve ground stability of the helicopter on uneven terrain and for use during rotor shutdown and engagement.

### Nose Wheel Lock Handle.

The nose wheel lock handle (12, figures 1-5 and 1-6), marked PARK LOCK, is located below a plate marked NOSE GEAR PULL TO LOCK, on the pilot's side of the cockpit console. The nose wheel is locked by pulling the lock handle aft and up, and unlocked by pushing aft and down.

## LANDING GEAR ACTUATING SYSTEM.

The landing gear actuating system operates on 3000 psi hydraulic pressure, supplied by the utility hydraulic system to raise or lower the main and nose landing gear assemblies. Each main landing gear is equipped with down-lock release limit switches which prevent inadvertent retracting of the landing gear when the weight of the helicopter compresses the oleo struts. When airborne, the struts extend and close the contacts of the down-lock release limit switches. The landing gear control panel is located on the cockpit console (figures 1-17 and 1-18). Placing the landing gear control handle in the up position retracts the landing gear. As the landing gear retracts, limit switches are actuated that cause the landing gear control handle warning light to show an unsafe condition, the landing gear position lights to go out, and a circuit to be completed that assures electrical power to lower the gear. When the landing gear is fully retracted, limit switches are actuated causing the landing gear warning light to go out. The main gears have then engaged the up-lock mechanism. The nose landing gear retraction phase is then initiated and fluid is simultaneously directed to the retraction port of the nose gear cylinder. As the nose gear starts to retract, another limit switch is acutated and causes the landing gear warning light to show an unsafe condition. When the nose gear is fully retracted, the up limit switch is actuated and causes the nose landing gear indicating light to go out, and the portion of the circuit to the control handle warning light that pertains to the nose gear is de-energized. The retraction side of the nose gear cylinder remains energized to maintain the nose gear in the retracted position. The landing gear is extended by placing the landing gear lever in the DN position. This completes the electrical circuit to the solenoid valve that directs fluid to the uplock cylinders of the main landing gears, unlocks them from the up position, simultaneously directs fluid to the actuator, and causes the landing gear to extend. The landing gear control handle warning light is energized to indicate the system is in operation. When the main gears are fully extended, limit switches are actuated that energize the main landing gear green lights and show a safe (down) condition. The landing gear control handle warning light will flash whenever the helicopter is flown at airspeeds of approximately 60 knots or less and the landing gear is not down and locked. A mechanical spring-loaded lock is engaged to lock the gears in the down position. As the main landing gear extension phase is initiated, the retraction port of the actuator is vented to return pressure that had been holding the nose gear in the retracted position, and hydraulic pressure is directed to the extension port of the actuator to lower the nose gear. Hydraulic pressure that is retained in the actuating cylinder prevents the nose gear from inadvertently retracting.

### Landing Gear Control Handle Down-Lock Release.

A manually operated down-lock release, located on the landing gear control panel, marked DN LCK REL, provides a mechanical override of the landing gear control handle down-lock solenoid because of an interruption of electrical power to the solenoid. If the down-lock solenoid becomes inoperative, the down-lock release can be actuated to mechanically release the landing gear control handle from the DN position.

### Nose Gear Switch and Caution Light.

Kneeling (figure 1-43) is accomplished by placing the switch, marked NOSE GEAR, NORMAL, KNEEL, located on the overhead switch panel (figure 1-13), in the KNEEL position. After the helicopter is loaded, the nose gear is jacked (hydraulically extended) to the static position by placing the switch, marked NOSE GEAR, located on the overhead switch panel, in the NORMAL position. An advisory light, marked KNEEL SW ARMED, located on the pilot's advisory panel will illuminate when the kneel switch is in the KNEEL position. The green nose gear down light will be OUT and the red warning light in the landing gear handle will be on when the nose gear is kneeled.

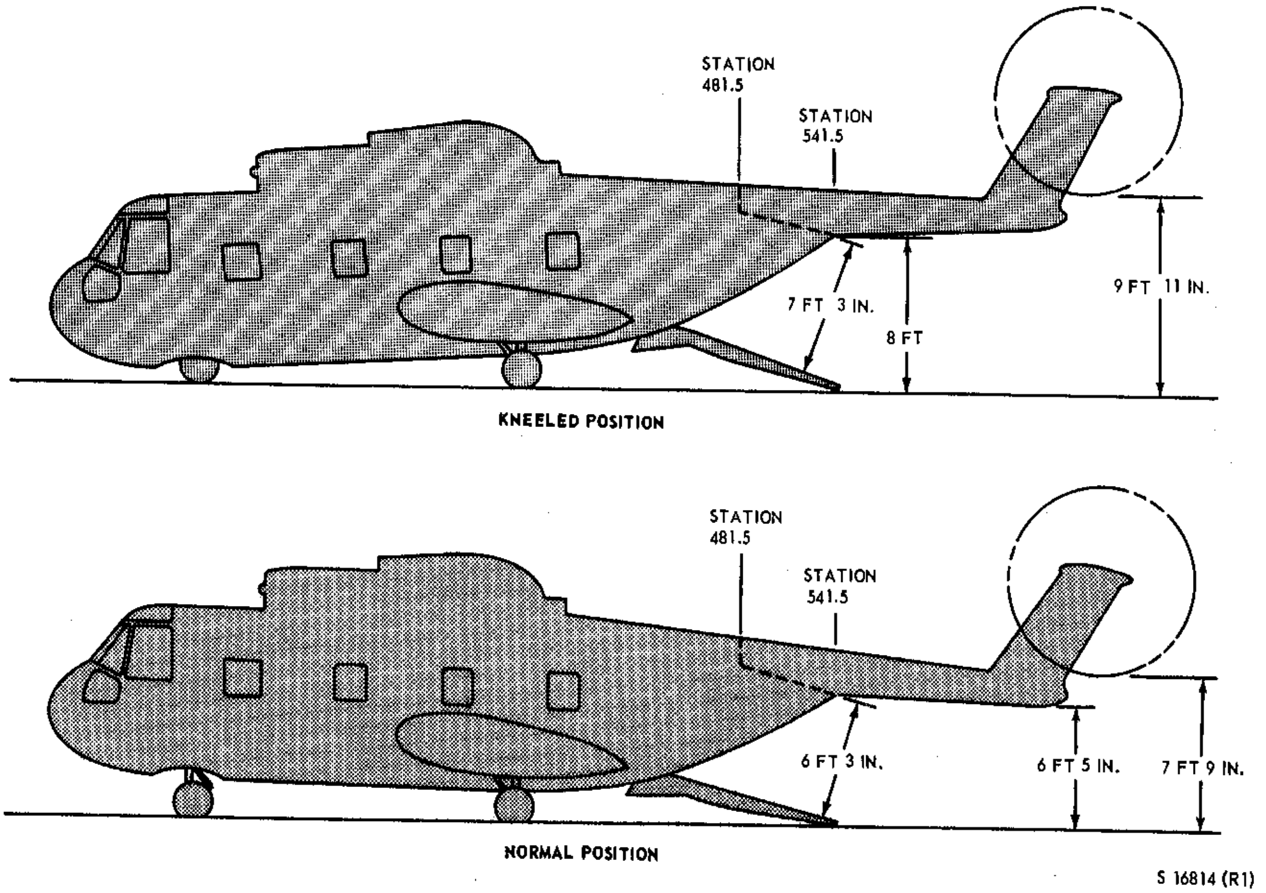



Figure 1-43. Landing Gear Kneeled

**Landing Gear Position Lights.**

The landing gear position indicators, located on the landing gear control panel under the general heading, GEAR DOWN and marked LEFT, RIGHT, and NOSE, are three press-to-test green lights. The lights operate on direct current from the dc essential bus through a circuit breaker, marked LAND GEAR WARN, located on the overhead dc circuit breaker panel. The lights will illuminate when the landing gear is down and locked and will be out when all three landing gears are up and locked. When any one of the three landing gears is neither fully up and locked nor fully down and locked, a red warning light in the landing gear control handle will illuminate.

**LANDING GEAR WARNING SYSTEM.**

The landing gear warning system consists of an airspeed sensor, an audible (beeper) signal, and a flasher light all located in the pilot's compartment. The landing gear warning system is designed to alert the pilot's whenever the landing gear is not down at airspeeds below an actuating range of approximately 60 knots or less. Within this range, the airspeed sensor will actuate a signal warning unit which injects an interrupted time (beeper) signal through both pilot's headsets and flashes the light in the landing gear control handle. The warning signal and flasher will continue to operate until any one of the following happens:

1. Airspeed is increased above the actuating range of approximately 60 knots.

2. The landing gear is extended and locked.

3. The warning silence button is depressed.

**NOTE**

The warning signal cannot be silenced by depressing the silence button at airspeeds within the actuating range of approximately 60 knots.

The landing gear warning system operates on current from the dc essential bus through the dc circuit breaker, marked LAND GEAR WARN, located on the overhead circuit breaker panel.

#### Landing Gear Warning Signal Light and Test Button.

The landing gear warning signal light, located in the handle of the landing gear control handle lever, provides the pilots with a visual warning that the landing gear is up or not locked down safely when at airspeeds below the actuating range. The signal light is the flasher type located in the landing gear control handle on the landing gear control panel. The signal light will flash whenever the audible signal is heard, or the gears are in transit between the up and down positions. When all wheels are fully up and locked or fully down and locked, the circuit is de-energized. The warning light operates on direct current from the dc essential bus through a circuit breaker, marked LAND GEAR WARN, located on the overhead dc circuit breaker panel. A landing gear warning light test button, marked HDL LT TEST, is located on the landing gear control panel. The test button is pressed to test the warning signal light located in the landing gear control handle. The warning light test circuit operates on direct current from the dc essential bus, through a circuit breaker, under the general heading INDICATOR LTS and marked TEST, located on the overhead dc circuit breaker panel.

#### Landing Gear Warning System Silence Button.

The landing gear warning system silence button, marked WARNING SILENCE, is located on the landing gear control panel above the landing gear position lights. Whenever the warning system has been silenced, the system will automatically reset when the airspeed of the helicopter exceeds the actuating range of approximately 60 knots. The landing gear warning system operates on current from the dc essential bus through the circuit breaker, marked LAND GEAR WARN, located on the overhead dc circuit breaker panel.

### LANDING GEAR ALTERNATE SYSTEM.

An alternate gear handle (25, figure 1-5), located on the left side of the cockpit console, is used to lower the landing gear should an electrical or hydraulic failure occur with the landing gears in the up and locked position. The handle mechanically unlocks the main landing gear up-locks, positions a directional valve, and discharges a 3000 psi preloaded air bottle. The compressed air charge actuates valves that vent the return side of the actuators to the reservoir and then drives the actuator to lower the landing gears. Fluid from the APU accumulator is used to lower the nose gear. The fluid is directed through an electrically actuated valve to the top side of the nose gear actuating cylinder to lower the nose gear. After actuation of the alternate gear system, the emergency release valves, located on the right side of the main transmission well, must be manually reset before the main gear emergency air cylinder can be recharged and/or the landing gear operated in a normal manner. If the air charge in the cylinder has been depleted, when the alternate landing gear handle is actuated, main landing gear hydraulic pressure is vented back to the utility reservoir, the up-locks are disengaged, and the main landing gears will lower by their own weight.

**NOTE**

> In case of complete electrical system failure, the nose gear will automatically extend.

## BRAKE SYSTEM.

The main landing gear wheels are each equipped with hydraulic brakes that are operated by toe brakes located on the pilot's and copilot's pedals. A parking brake system is also provided. The parking brake handle (11, figures 1-5 and 1-6), marked PARKING BRAKE, and a decal, marked ON-DEPRESS TOE BRAKE THEN PULL-OFF-DEPRESS TOE BRAKE, is located on the side of the cockpit console. The parking brakes are applied by depressing the brake pedals, manually pulling the parking brake handle to the PARK position, and then releasing the brake pedals. Depressing the left brake pedal will release the parking brakes, causing the parking brake handle to return to the OFF position.

## EMERGENCY EQUIPMENT.

### PORTABLE FIRE EXTINGUISHERS.

One hand-operated fire extinguisher (11, figure 3-6) is located on the bulkhead behind the pilot's seat. A second fire extinguisher (17, figure 3-6) is located on the left-hand side panel above the ramp. The fire extinguishers are filled with bromochloromethane (CB). The extinguishers are held in place by a bracket with a tight-fitting quick-release spring.

**WARNING**

> Prolonged exposure (5 minutes or more) to high concentrations of bromochloromethane (CB) or its decomposition products will cause pronounced irritation of eye and nose and should be avoided. CB is an anesthetic agent of moderate intensity. It is safer to use than previous fire extinguishing agents (carbon tetrachloride, methylbromide). However, especially in confined spaces, adequate respiratory and eye protection from excessive exposure, including the use of oxygen when available, should be sought as soon as the primary fire emergency will permit.

**NOTE**

> Bromochloromethane is highly corrosive to helicopter metal, paint, and plexiglass. If the fire extinguisher starts to leak, the extinguisher should be inverted and the control valve depressed (cover the nozzle with a cloth or aim into a can etc., to catch any liquid which may be discharged). This will re-

lease the stored charge of pressurizing gas with a minimum discharge of fluid and render the extinguisher harmless. The extinguisher should be returned to its bracket and be reported for replacement in Form 781.

## FIRST AID KITS.

One first aid kit (7, figure 3-6) is mounted on the bulkhead behind the pilot's or copilot's seat. Five additional first aid kits are installed in the cargo compartment on the left and right-hand cargo compartment side panels. Each kit is held in place by a metal cover and supporting clips.

## CREW ALARM BELL.

The crew alarm bell (1, figure 3-6), located overhead in the cargo compartment, is used to warn crew and passengers in an emergency. The crew alarm bell operates on direct current from the battery bus and is protected by a circuit breaker, marked CREW ALARM, located on the battery bus circuit breaker panel. The alarm bell can be sounded regardless of the position of the battery switch.

### Crew Alarm Bell Switch.

The crew alarm bell guarded switch, with marked positions ON and OFF, is located on the overhead switch panel (figure 1-13) in the pilot's compartment. To operate the alarm bell, lift the guard and place the switch in the ON position. To stop the alarm bell, return the switch to the OFF position.

## PYROTECHNIC KIT.

Provisions are made on the bulkhead behind the copilot's seat for mounting a pyrotechnic kit (10, figure 3-6) that contains a pistol and 12 cartridges. The pyrotechnic kit is a water-tight floatable container in which pistol and cartridges are stowed and is easily removed. On CH-3E helicopters Serial Nos. AF66-13285 add subsequent, and all HH-3E helicopters, provisions are made to stow the pyrotechnic kit on the upper right-hand side of the entrance to the pilot's compartment.

## CRASH AXE.

One crash axe (12, figure 3-6) is installed below the step at the pilot's compartment entrance. The axe is secured by a bracket and a strap.

## LIFE RAFTS.

There are provisions for three life rafts that can be secured by straps to fittings in the ceiling of the cargo compartment. Two 6-man life rafts (8, figure 3-6) can be installed above the cargo compartment escape hatch, and one 20-man life raft (18, figure 3-6) can be installed above the aft ramp on the right-hand side of the cargo compartment.

## FIRE DETECTION SYSTEM.

The engines and APU fire detector systems provide a warning in the event of fire in either of the engines or the APU compartment. The APU fire detector system uses probe type detector elements. The heat of a fire causes illumination of the warning light on the fire warning panel in the appropriate fire emergency T-handle on the overhead control panel for the engines, and on the APU control panel. As the temperature in the fire zone drops below the warning point, the warning lights go out. Power for the system is normally supplied by the No. 1 generator at 115 volts ac through the fire detector circuit breakers located on the ac circuit breaker panel. The engine sensing elements are mounted on the firewalls and are arranged in close proximity to the compressor section, the turbine section, and the exhaust tailpipe of each engine. The sensing element portion of the circuit terminates at the firewall, with the sensors interconnected to form a single circuit for each engine. The control units are located overhead in the forward portion of the cargo compartment. The function of the control unit is to continuously monitor the sensing elements and the probe type detector elements, and upon occurrence of fire, indicate its location via a cockpit warning light for each engine and an APU unit. An audible fire warning system supplements the existing fire warning system by providing an audible tone through the interphone system. The tone is the same as used for the landing gear warning system, but the system is designed to prevent one warning system from activating the other system. The tone will be heard if a fire or temperature condition in excess of 575°F is experienced in either or both engine compartments. The tone will continue until the temperature drops below 575°F or until the affected fire warning light is depressed. If a fire warning light is depressed to deactivate the audible warning, the light will stay illuminated until the fire is out or the temperature has dropped below the warning point. When normal conditions prevail, the system will automatically reset to the off position. The fire emergency T-handle for the affected engine will illuminate whenever a fire occurs. A main system test switch is located in the cockpit, adjacent to the warning lights, for testing circuit electrical continuity when power is applied to the system.

## ENGINE FIRE WARNING LIGHTS AND TEST SWITCH PANEL.

The red engine fire warning lights and test switch panel (20, figure 1-14) are located on a plate, marked FIRE WARN, on the instrument panel. The lights are marked No. 1 ENG and No. 2 ENG. The switch has two marked positions, FIRE TEST and OFF. The lights are pushbutton type that have a manual warning reset capability. The light(s) are depressed when it is desired to de-energize the audible fire warning system. In addition, four red engine fire warning lights, two for each engine, are installed in the engine fire emergency T-handles, located below the decal marked FIRE EMER SHUTOFF SELEC-

TOR on the overhead switch panel. The left handle is marked No. 1 ENGINE and the right handle is marked No. 2 ENGINE. A light on the instrument panel and a light in either the No. 1 or No. 2 engine fire emergency T-handle will illuminate in event of a fire in the corresponding engine compartment. To test the engine fire detector system, place the spring-loaded switch in the (up) FIRE TEST position. The fire warning lights on the instrument panel and in the fire emergency T-handles should go on. The switch will return to the OFF position when released and the lights will go out.

## AUDIBLE FIRE WARNING SYSTEM.

An audible fire warning system is installed in addition to the visual one. When the fire warning light button is depressed, the audible signal is silenced but the light remains on until the temperature in the affected compartment cools below 301.7°C (575°F). Power for the system is supplied by the ac essential bus at 115-volts ac through the fire detection 1-ENG-2 circuit breakers, on the pilot's circuit breaker panel.

### APU FIRE WARNING LIGHT.

The APU fire warning capsule, marked FIRE WARNING, contains a red light with two bulbs and is located on the APU control panel. To test the APU fire extinguisher system, place the spring-loaded fire test switch in the FIRE TEST position. The warning light on the APU control panel should go on.

## FIRE EXTINGUISHER SYSTEM.

The fire extinguisher system for the helicopter are of HRD (High-Rate-Discharge), Bromotrifluoromethane ($CF_3Br$) type.

**WARNING**

$CF_3Br$ is highly volatile and is not easily detected by odor. It is not toxic and is like freons and carbon dioxide, causing danger primarily by reduction of oxygen. Do not allow liquid to contact the skin as it may cause frostbite or low temperature burns because of its low boiling point.

### ENGINE FIRE EXTINGUISHER SYSTEM.

The engine compartment fire extinguisher system consists basically of two charged containers of Bromotrifluoromethane ($CF_3Br$), discharge nozzles, an overboard thermal discharge tube, discharge indicator, circuit breakers, electrical wiring, and the necessary controls. The $CF_3Br$ containers are located in the aft main gear box fairing structure. They are charged with 2.5 pounds of $CF_3Br$ plus a nitrogen charge to propel the extinguishing agent into the engine compartment. The forward fire extinguisher discharge tube is mounted on the center fire wall near the compressor section of the engine. The rear fire extinguisher discharge tube is mounted on the fire wall, aimed at the power turbine section of the engine. A safety outlet in each container is connected to the red thermal discharge indicator located on the lower left side of the fuselage. In the event container pressure becomes excessive due to high temperatures, the safety outlet opens, the thermal discharge indicator seal is ejected, and the container is discharged overboard. A pressure gage on each container facilitates a preflight pressure check. When the spheres are properly charged, the pressure gages should indicate the value within the range shown on the decal adjacent to the gages. Power for the fire extinguisher system is supplied by the dc essential bus system through the fire extinguisher circuit breakers located on the overhead control panel.

**NOTE**

Although designed primarily for combating an engine compartment fire during flight, the fire extinguishing system may be used on the ground if other fire fighting equipment is ineffectual or unavailable. Be sure all ground personnel are clear before using the system.

#### Engine Fire Emergency Selector Handles. (Fire Emergency T Handles).

Two T-shaped handles, below the decal marked FIRE EMER SHUT-OFF SELECTOR, are located on the overhead switch panel (figure 1-13). The handle marked NO. 1 ENGINE is for the No. 1 engine compartment and the handle marked NO. 2 ENGINE is for the No. 2 engine compartment. When either handle is pulled down, dc power from the essential bus actuates the shut-off valve, which closes the fuel lines to the respective engine, selects the engine compartment to which the fire extinguisher fluid is to be directed, and also energizes the circuit to the fire extinguisher switch. The ends of the handles house fire detector warning lights.

#### Engine Fire Extinguisher Switch.

An engine fire extinguisher switch, marked FIRE EXT, located on the overhead switch panel (figure 1-13) in the pilot's compartment, has marked positions RESERVE, OFF, and MAIN. The fire extinguisher switch will return to the OFF position when released. The lock lever type switch is operative only after one of the fire emergency shut-off selector handles has been pulled. When the engine fire extinguisher switch is placed in the MAIN position, after the fire emergency shut-off selector handle has been pulled, the contents of the fire extinguisher sphere is discharged into the corresponding engine compartment. When the engine fire extinguisher switch is placed in the RESERVE position, after fire emergency shut-off selector handle has been pulled, the contents of the opposite fire extinguisher sphere is discharged into the last selected engine compartment. If fires should occur in both engines, pull the No. 1 engine fire emergency selector handle and switch the fire extinguisher switch to MAIN, then, pull No. 2 engine fire emergency selector handle and switch the fire extinguisher switch to RESERVE. This procedure will permit fire extinguisher agent to enter both engine compartments.

If both fire emergency selector handles are pulled and the fire extinguisher switch is placed in either MAIN or RESERVE position, the fire bottle will discharge into the last engine compartment selected.

## AUXILIARY POWER UNIT FIRE EXTINGUISHER SYSTEM.

The fire extinguisher system for the auxiliary power unit consists of a charged container of Bromotrifluoromethane, located adjacent to the APU compartment, with lines, nozzles, and controls similar to the engine compartment extinguisher system. The container holds 2.5 pounds of $CF_3Br$.

### Auxiliary Power Unit Fire Detector and Extinguisher Control Panel.

The auxiliary power unit fire detector and extinguisher control panel is located on the APU control panel on the cockpit console (figures 1-17 and 1-18). The APU fire extinguisher system is energized by first placing the toggle switch, marked FUEL SHUT-OFF and NORM, in the FUEL SHUT-OFF position, then placing the toggle switch, marked FIRE EXTING and OFF, in the FIRE EXTING position.

# EMERGENCY EXITS.

For emergency entrances and routes of escape and exits, (see figures 3-5 and 3-6).

## PILOT'S COMPARTMENT SLIDING WINDOWS.

The pilot's compartment sliding windows are normally opened or closed by actuating the handle located on the bottom of each window. The windows may be opened and will lock in any detent position when the handle is released. The sliding windows can be jettisoned from any position, from open to closed, to provide emergency exits. The manual emergency release handles, marked EMER RELEASE PULL, are located on the lower edge of each window inside the pilot's compartment. The window can be jettisoned outward and downward by pulling the release handle in the direction of the arrow. The windows can also be released from the outside by turning the handle marked EXIT RELEASE-PRESS BUTTON-TURN HANDLE PULL OUT WINDOW. The window can also be jettisoned from the inside by rotating the window emergency release handle upward and pushing out the window from the bottom.

## PERSONNEL DOOR EMERGENCY EXIT.

The personnel door is normally opened and closed by the handle marked TO OPEN TURN AND PUSH, with arrows pointing the direction to turn and push, located on the inside of the door. The handle also locks the door in the open position to facilitate inflight operations that require the door to be open. A release handle is also located on the outside of the door. The door can be jettisoned to provide an additional emergency exit by pulling down on the emergency release handle, marked EMERGENCY EXIT RELEASE, TURN, OPEN-CLOSED, with the direction arrows located at the top of the door. The inside of the personnel door below the window is marked CUT HERE FOR EMERGENCY EXIT.

## RAMP.

The ramp consists of a forward and aft ramp that can be opened hydraulically or manually to provide an exit from the rear of the cargo compartment. The aft ramp can be opened in flight, on the ground, or on the water by placing the master switch, located on the pilot's ramp control panel (figure 4-25), marked RAMP MASTER SW, CREW-OFF-PILOT, in the PILOT position, and placing the switch marked AFT RAMP in the DOWN position. When the ramp master switch is in the CREW position, the aft ramp may be opened by the crewmember, using the crew ramp controls located above the ramp on the right-hand compartment side panel. In the event of an electrical or hydraulic failure, open the aft ramp manually from inside the helicopter by pulling forward on the handle attached to the manual shut-off valve located on the right side of the cargo compartment under a flap of soundproofing, marked EXIT RELEASE HANDLE INSIDE. When the handle is positioned forward, the manual shut-off valve is actuated to vent the aft ramp cylinders, and a cable attached to the handle releases the up-lock to unlock the aft ramp. The ramp will open by its own weight. The rate of opening is controlled by a restrictor. The aft ramp may be opened from outside of the helicopter by removing the handle from the clips beneath the cover on the aft fuselage, marked RAMP EXIT RELEASE HANDLE INSIDE, and pulling the handle down. The outside handle is connected by a cable to the handle attached to the manual shut-off valve. The forward ramp may be opened from inside the cargo compartment by pulling up on the manual override handles on top of the forward ramp cylinders. The ramp will open by its own weight. The rate of opening will be controlled by a restrictor.

**WARNING**

The forward ramp should only be opened on the ground.

## CARGO COMPARTMENT WINDOWS.

Jettisonable windows located on the left front side of the cargo compartment and over each sponson provide access to the ground and to the outer fuselage and sponsons from inside the cabin. This facilitates bilge pump operations, maintenance, and docking during water operations. The windows may be removed from the outside by pulling down on the outside release lever and pulling the window out. The windows may be jettisoned from the inside by rotating the release lever forward and pushing the window out. On aircraft modified for armament, use the forward (red) release lever to jettison the left forward window.

**WARNING**

Prior to removing the left jettison window for access to the left sponson, ensure the HF-103 radio is off. The HF antenna emits high voltage radiation during HF transmission.

On aircraft modified for armament the left front cabin window may be removed in flight by forward rotation of the aft (black) release lever. Hold the handle at the top of the window while unlocking the release lever and remove the window by pulling the bottom handle inward. Insert the window by placing the tangs on the top of the window in their respective slots, push the bottom of the window in place and lock by rotating the aft (black) release lever toward the rear.

**CAUTION**

If the procedures outlined are not followed when removing or inserting jettisonable windows while in flight, the windows may be blown overboard. It two crew members are available, one should hold the window handles while the other operates the release levers.

After removal the window may be secured by a strap on the left cabin wall bracket between the second and third windows. The cargo compartment windows forward and aft of the sponsons are permanently installed and are not designed as emergency exits. These windows are attached to the inside and outside by aluminum retainers to preclude loss of the windows during flight.

### PILOT'S AND COPILOT'S SEATS.

The pilot's and copilot's seats are located side-by-side in the pilot's compartment. The pilot's seat is on the right. The track-mounted seats are designed to accommodate back-type parachutes and seat-type pararafts. Both seats have a 5-inch range of height adjustment, and a 5-inch forward and aft adjustment and are equipped with cushions that are interchangeable with the pararaft and parachute. However, forward and aft adjustment will be restricted to less than 5 inches on aircraft equipped with armor plating.

#### Seat Height Adjustment Lever.

The seat height adjustment levers (21, figure 1-5) are the rear levers at the right of the pilot's and copilot's seats. The levers are pulled up to release the height adjustment lockpins.

#### Seat Foreward and Aft Adjustment Lever.

The seat fore-and-aft adjustment levers (22, figure 1-5) are the front levers on the right side of the pilot's and copilot's seats. The levers are pulled up to release the forward and aft seat adjustment lockpins. The levers must be held up while the seat is moved on the racks, forward or aft, as desired. The lockpins will automatically engage in any of eight positions when the levers are released, except when restricted by armor plating.

#### Shoulder Harness Lock Lever.

A two-position shoulder harness inertia reel lock lever (20, figure 1-5) is located at the left side of each seat. When the lever is in the unlocked (aft) position, the shoulder harness cable will extend to allow the occupant to lean forward; however, the inertia reel will automatically lock if an impact force between two and three g's in any direction is encountered. When this occurs, the inertia reel will remain locked until the lever is moved to the locked position and then to the unlocked position. When the lever is placed in the locked (forward) position, the shoulder harness cable is locked so that the occupant is prevented from leaning forward. The locked position is used to provide an added safety precaution when a crash landing is anticipated, or when desired during critical operations.

### CREWMAN'S SEAT.

The crewman's seat (4, figure 1-4), located at the entrance to the cockpit and aft of any between the pilot's and copilot's seats, may be folded against the entrance wall to facilitate cockpit entrance and exit. The seat is equipped with a safety belt.

**CAUTION**

To facilitate aircrew egress in the event of an emergency, the crewman's seat will not be occupied and will be in the stowed position during air refueling operations.

## AUXILIARY EQUIPMENT.

The following major systems and items are covered in section IV:

Heating System

Anti-Ice Systems

Communication and Associated Electrical Equipment

Lighting Equipment

Auxiliary Power Unit

Cargo Compartment

Winch Installation

Rescue Hoist

External Cargo Sling

Troop Carrying Equipment

Casualty Carrying Equipment

Miscellaneous Equipment

Windshield Wiper System

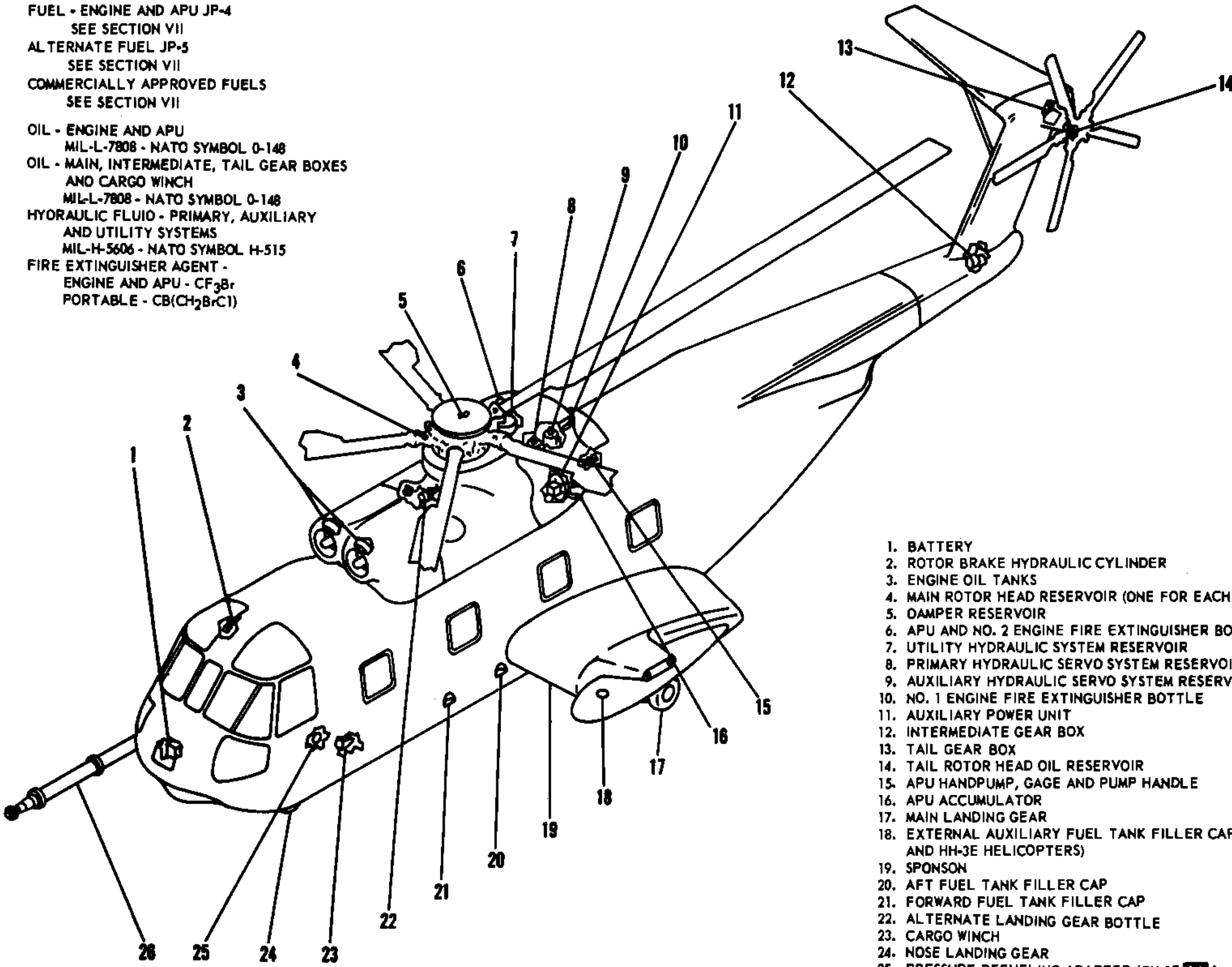
Bilge Pump

Load Adjuster

Weapons System

FUEL - ENGINE AND APU JP-4
SEE SECTION VII
ALTERNATE FUEL JP-5
SEE SECTION VII
COMMERCIALLY APPROVED FUELS
SEE SECTION VII

OIL - ENGINE AND APU
MIL-L-7808 - NATO SYMBOL 0-148
OIL - MAIN, INTERMEDIATE, TAIL GEAR BOXES
AND CARGO WINCH
MIL-L-7808 - NATO SYMBOL 0-148
HYDRAULIC FLUID - PRIMARY, AUXILIARY
AND UTILITY SYSTEMS
MIL-H-5606 - NATO SYMBOL H-515
FIRE EXTINGUISHER AGENT -
ENGINE AND APU - $CF_3Br$
PORTABLE - $CB(CH_2BrCl)$




1. BATTERY
2. ROTOR BRAKE HYDRAULIC CYLINDER
3. ENGINE OIL TANKS
4. MAIN ROTOR HEAD RESERVOIR (ONE FOR EACH BLADE)
5. DAMPER RESERVOIR
6. APU AND NO. 2 ENGINE FIRE EXTINGUISHER BOTTLES
7. UTILITY HYDRAULIC SYSTEM RESERVOIR
8. PRIMARY HYDRAULIC SERVO SYSTEM RESERVOIR
9. AUXILIARY HYDRAULIC SERVO SYSTEM RESERVOIR
10. NO. 1 ENGINE FIRE EXTINGUISHER BOTTLE
11. AUXILIARY POWER UNIT
12. INTERMEDIATE GEAR BOX
13. TAIL GEAR BOX
14. TAIL ROTOR HEAD OIL RESERVOIR
15. APU HANDPUMP, GAGE AND PUMP HANDLE
16. APU ACCUMULATOR
17. MAIN LANDING GEAR
18. EXTERNAL AUXILIARY FUEL TANK FILLER CAP (CH-3E 16 ▶ AND HH-3E HELICOPTERS)
19. SPONSON
20. AFT FUEL TANK FILLER CAP
21. FORWARD FUEL TANK FILLER CAP
22. ALTERNATE LANDING GEAR BOTTLE
23. CARGO WINCH
24. NOSE LANDING GEAR
25. PRESSURE REFUELING ADAPTER (CH-3E 16 ▶ AND HH-3E HELICOPTERS)
26. AIR REFUELING PROBE

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Figure 1-44. Servicing Diagram