FUEL BOOST PUMPS.

Two fuel boost pumps are located in the forward end of each fuel tank. Each tank has a No. 1 boost pump and a No. 2 boost pump that are powered by a separate electrical circuits. The No. 1 boost pump in each tank is powered by current from the No. 1 ac generator and normally controlled by the primary bus. The No. 2 boost pump in each tank is powered by the No. 2 ac generator and normally controlled by the primary bus. The boost pumps are protected by circuit breakers marked FWD TANK and AFT TANK under the general heading FUEL PUMPS No. 2, located on the left console circuit breaker panel.

FUEL BOOSTER PUMP SWITCHES.

Four fuel booster pump switches, located on the fuel management panel (figure 1-25), control the fuel flow to the engine. The switches are in sets of two, those for the forward tank marked FWD TANK, while the two for the aft tank are marked AFT TANK. Above each switch is a number 1 or 2 to designate the pump in the tank controlled by that switch. Each switch has two marked positions, PUMP (on position) and OFF. The fuel booster pumps, located in the collector cans in the forward cell of each fuel tank, supply fuel to the engine driven pump when the switches are in the PUMP position (FWD TANK and AFT TANK).

Fuel Booster Pump Failure Warning Lights.

Four pressure switches are installed in the fuel tanks, one in each cell, and are connected to the pressure feed line from each fuel booster pump. The fuel pressure switches are connected by direct current to the essential bus through the warning light circuit breaker, located on the overhead circuit breaker panel and marked WARN LTS, PWR. When the fuel booster pumps are first turned on, or the fuel booster pump switches are tested, the fuel pump failure warning lights will illuminate and then go off. They are illuminated until pressure is built up in the system. The switches close if the pumps fail and the warning lights marked PUMP FAILURE, on the fuel management panel (figure 1-18), will illuminate. Pressure must decrease to approximately 16-1/2 psi to energize the warning circuit.

FUEL LOW-LEVEL CAUTION LIGHTS.

The fuel low-level caution lights are located on the caution panel on the pilot's side of the instrument panel. The lights marked FWD FUEL LOW and AFT FUEL LOW, will illuminate when approximately 210 to 280 pounds per tank remain in a 3 degree nosedown attitude, or between 170 and 200 pounds per tank remain, when in a hover. The caution lights are tested by the master TEST button on the caution panel and operate on direct current from the essential bus and protected by circuit breakers marked LOW LEVEL FWD and AFT, located on the overhead circuit breaker panel.

FUEL QUANTITY GAGES AND TEST SWITCHES.

The fuel quantity gages (figure 1-5), located on the instrument panel, indicate the fuel quantity in each tank in pounds. The tank units capacitance system used in this helicopter is practically unresponsive to volumetric changes resulting from various temperatures. The dielectric between the two electrodes will vary as the fuel varies. The fuel quantity gages are calibrated to measure this voltage differential in pounds of fuel. Refer to figure 1-26 for fuel quantities. The fuel quantity indicating system may be tested by pressing the fuel quantity gages test switches (figure 1-5) marked FUEL GAGE TEST, FWD TANK AFT TANK, located between the fuel quantity gages. Pressing either button-type switch for approximately 10 seconds will induce a current reversal which causes the needle to turn to zero. Upon release of the test switch, the normal current should cause the needle to return to the previous reading. This test shows that the fuel quantity indicating system is operating correctly. The fuel quantity indicating system operates on 115 volts alternating current from either generator or the inverter and is protected by circuit breakers marked QTY under the general heading FUEL, located on the console circuit breaker panel.

MANUAL FUEL SHUTOFF VALVES.

The manual fuel shutoff valves, located under the cabin floor, are turned off when it is necessary to clean the fuel tank's filter elements. An access plate must be removed from the floor to gain access to the manual shutoff valves.

FUEL FLOW INDICATOR.

Two fuel flow indicators (figure 1-5), calibrated in pounds per hour, are located on the instrument panel. The fuel flow indicators provide indication of the fuel consumption of the engines and operate on electrical power from the ac essential bus through circuit breakers marked FLOW, 1-ENG-2, under the general heading FUEL, located on the ac essential circuit breaker panel.

FUEL FILTER BYPASS CAUTION LIGHTS.

The fuel filter bypass caution lights will illuminate whenever fuel bypasses the fuel filters of the respective fuel tank, enabling the pilot to read the markings FWD FUEL BYPASS and AFT FUEL BYPASS. Fuel will bypass the filter whenever grit and particles clog up the filter screen.

PRESSURE FUELING-DEFUELING SYSTEM.

The pressure fueling-defueling system is a single point fueling-defueling system. The pressure re-

FUEL QUANTITY
JP-5 at 15.6°C (60°F)

GRAVITY REFUELING

FUEL TANK	USABLE		UNUSABLE		FULLY SERVICED	
	US GALLONS	POUNDS	US GALLONS	POUNDS	US GALLONS	POUNDS
FWD TANK	344.35	2341.6	2.65	18.0	347.0	2359.6
AFT TANK	350.35	2382.4	2.65	18.0	353.0	2400.4
TOTAL MAIN TANKS	694.70	4724.0	5.30	36.0	700.0	4760.0
*ONE AUX TANK	106.0	720.8	4.0	27.2	110.0	748.0
*TWO AUX TANKS	212.0	1441.6	8.0	54.4	220.0	1496.0

TOTAL USABLE FUEL	WITHOUT AUX TANKS	694.7 GAL.	4724.0 LBS.
TOTAL USABLE FUEL	WITH ONE AUX TANK	800.7 GAL.	5444.8 LBS
TOTAL USABLE FUEL	WITH TWO AUX TANKS	906.7 GAL.	6165.6 LBS.

PRESSURE REFUELING

FUEL TANK	USABLE		UNUSABLE		FULLY SERVICED	
	US GALLONS	POUNDS	US GALLONS	POUNDS	US GALLONS	POUNDS
FWD TANK	338.35	2300.8	2.65	18.0	341.0	2318.8
AFT TANK	341.35	2321.2	2.65	18.0	344.0	2339.2
TOTAL PRES-SURE MAIN TANK	679.70	4622.0	5.30	36.0	685.0	4658.0

TOTAL USABLE FUEL	WITHOUT AUX TANKS	679.7 GAL.	4622.0 LBS.
TOTAL USABLE FUEL	WITH ONE AUX TANK	785.7 GAL.	5342.8 LBS.
TOTAL USABLE FUEL	WITH TWO AUX TANKS	891.7 GAL.	6063.6 LBS.

NOTES

*1. AUXILIARY TANKS REFUELED BY GRAVITY ONLY.
2. USABLE FUEL DETERMINED AT 0 DEGREE FUSELAGE ATTITUDE.
3. FUEL DENSITY JP-5 = 6.8 PPG JP-4 = 6.5 PPG.

Figure 1-26. Fuel Quantity Data

fueling filler cap (figure 1-27) is located on the right side of the fuselage inside the step below the door. The filler cap is marked CAP-FUEL FILLER SINGLE POINT SERVICING. To fill the fuel tanks, the refueling nozzle is connected to the adapter connection, and fuel is pumped through the fuel lines and the fueling and defueling valves, located in each fuel tank. The fueling and defueling valves are normally closed; for refueling, the valves open with pressure at the inlet until the high level shutoffs close the valves. For defueling, the valves open with vacuum at the inlet until the low level shut-offs close the valves.

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Figure 1-27. Pressure Refueling Panel

Pressure Refueling Switches.

The pressure refueling switches marked PRI TEST and SEC TEST are located on a panel (figure 1-27) marked PRESS REFUELING PRECHECK, located next to the fueling-defueling adapter connection. The switches are used to check the reliability of the fuel high level shutoffs. The panel contains information on the refueling precheck that must be followed prior to using the pressure refueling system.

INFLIGHT REFUELING SYSTEM

The inflight refueling system consists of a Wiggins quick disconnect nipple and a fuel filter located on the right side of the fuselage aft of the cabin door. The inflight refueling system is connected directly to the pressure fueling system. The tanks may be filled by attaching the Wiggins refueling nozzle to the quick disconnect nipple or by attaching a Parker refueling nozzle to the pressure refueling adapter. Fuel is pumped throught the fuel lines and fueling and defueling valves located in each tank. The fueling and defueling valves are normally closed; for refueling, the valves open with pressure at the inlet until the high level shutoffs close the valves.

CAUTION

The PRESS REFUELING PRECHECK must be performed prior to using the inflight refueling system. The fuel quantity indicators must be monitored during the pumping phase to avoid overfilling and possible rupturing of the fuel tanks in the event of high level shutoff valve(s) failure.

NOTE

When refueling with the Wiggins nozzle, a slow pumping rate of less than 100 pounds per minute total for both tanks may indicate a clogged fuel filter in the helicopters inflight refueling system.

MAIN FUEL DUMPING SYSTEM.

The fuel dump system from the main tanks provides a means of reducing gross weight during an emergency by dumping fuel at a rate of 170 pounds per minute with four booster pumps operating or 95 pounds per minute with two booster pumps operating and the respective offloading valve open. The system consists of two spring-loaded manually operated valves, one for each fuel tank located overhead in the cabin and a dump port, located on the center line of the hull aft of the tail wheel. The valves are normally in the closed position. For emergency fuel dumping in flight and/or when afloat refer to Section III.

NOTE

When operating the fuel dump system the booster pump failure warning lights and the fuel filter bypass caution lights may light singly or in any combination.

WARNING

Do not open either fuel dump valve unless the crossfeed valve is open and all boost pumps are on to preclude a possible flameout.

AUXILIARY FUEL SYSTEM

The auxiliary fuel system (figure 1-28) is installed to increase the range and endurance of the helicopter. The system does not supply fuel directly to the engines, but functions to replenish fuel into the main fuel tanks. The system consists of the fuel tanks, transfer pumps, pump switches and indicator lights. The tanks are gravity fueled and may be electrically jettisoned. The transfer system operates from the essential bus.

Auxiliary Fuel Tanks.

Each auxiliary fuel tank (figure 1-2) weighs 148 pounds empty and has a capacity of 110 gallons with approximately 106 gallons usable. The tanks, mounted on the left hand and right hand forward cargo sling attaching points, are attached to bomb racks and stablized by sway braces. The tanks may be individually or simultaneously jettisoned electrically. Electrical and self-sealing quick disconnects are attached to each tank. The disconnects will release when approximately 15

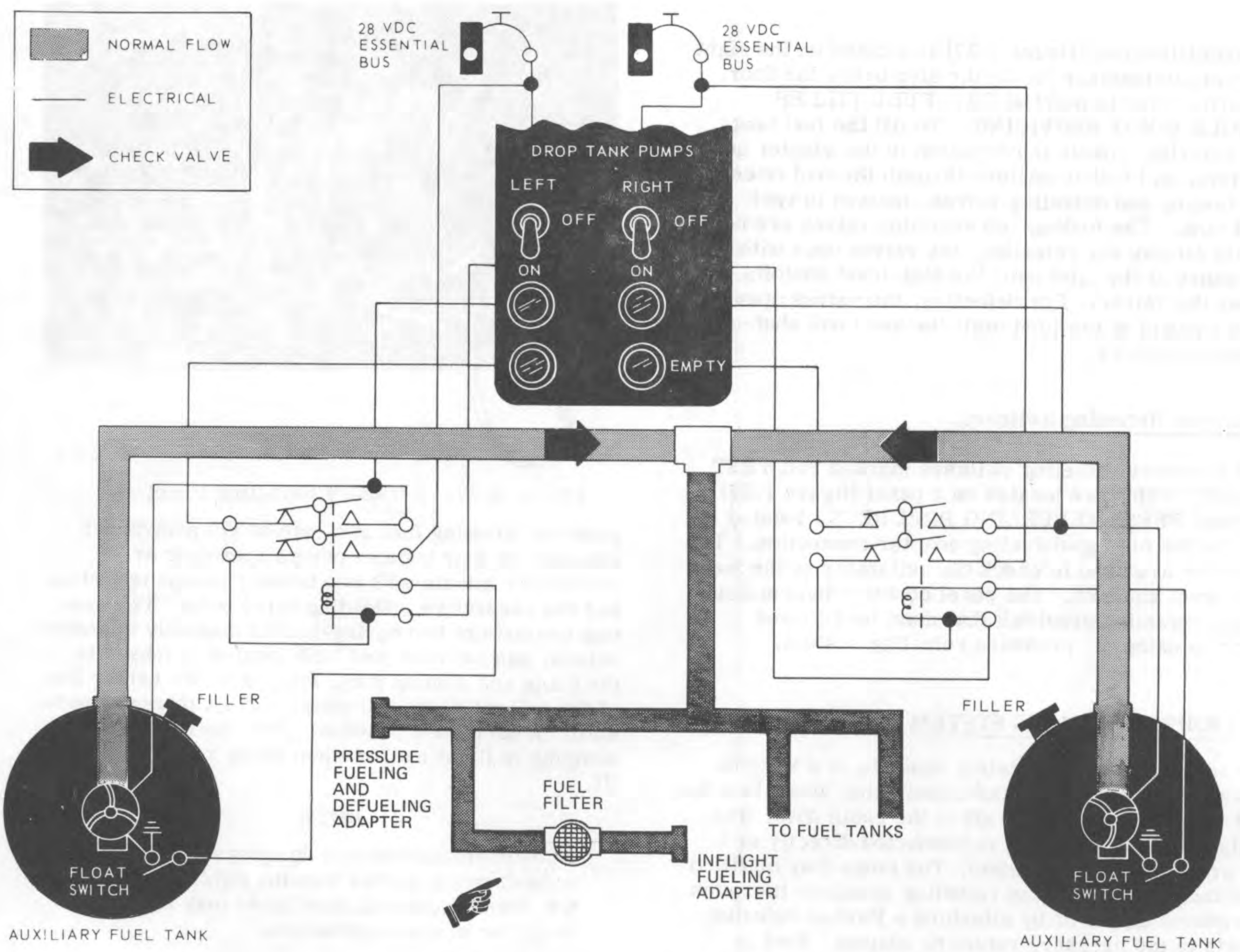

Figure 1-28. Auxiliary Fuel System Diagram

pounds pull is imposed on the tank. Dummy receptacles are located adjacent to each tank so that quick disconnects may be stowed when the fuel tanks are not installed.

Transfer Pump Switches and Indicator Lights.

The transfer pump switches and indicator lights are located on the engine emergency start switch panel. The switches are marked DROP TANK PUMPS, LEFT and RIGHT and have marked positions OFF and ON. When the switch is placed in the ON position, the associated green indicator light illuminates. When the tank is empty, the red EMPTY light illuminates and the transfer pump automatically shuts off. The circuit breakers marked DROP TANK PUMPS, LEFT and RIGHT, located on the overhead circuit breaker panel.

Jettison Switches.

Three guarded jettison switches marked L TK, BOTH TKS and R TK are located on the auxiliary fuel tank jettison panel on the cockpit console. When the L TK or R TK switch guard is lifted and the switch moved forward, the respective auxiliary tank will fall free from the helicopter. When the BOTH TKS switch guard is lifted and the switch moved forward, both tanks will be released. The jettison circuit is wired through the landing gear scissors (squat) switches thereby preventing jettisoning the tanks when the helicopter is on the ground. The jettison system operates from the primary bus and is protected by a circuit breaker marked AUX TANK JETT located on the overhead circuit breaker panel.

ELECTRICAL POWER SUPPLY SYSTEM.

Electrical power is supplied by two basic systems, a 115/200 volt alternating current supply system and a 28 volt direct current power supply system. The alternating current power supply system is the primary system. Direct current is obtained by rectifying alternating current. Power sources, power distribution, and equipment operating from each power source are shown on figure 1-29.

ALTERNATING CURRENT POWER SUPPLY SYSTEM.

The main sources of power for the entire electrical system are two ac generators. Other ac power sources are an ac external power receptacle and a dc-operated inverter used to energize prestart engine instruments.

Alternating Current Power Sources.

Generators. Two 115/200 volt, 3 phase, 400 cps generators (figure 1-3) are mounted on, and driven by, the accessory drive section of the main gear box. The generator output capacity varies with temperature and altitude (approximately 30 KVA at sea level to 22 KVA at 14000 feet). The accessory drive rotor lockout system makes possible the operation of

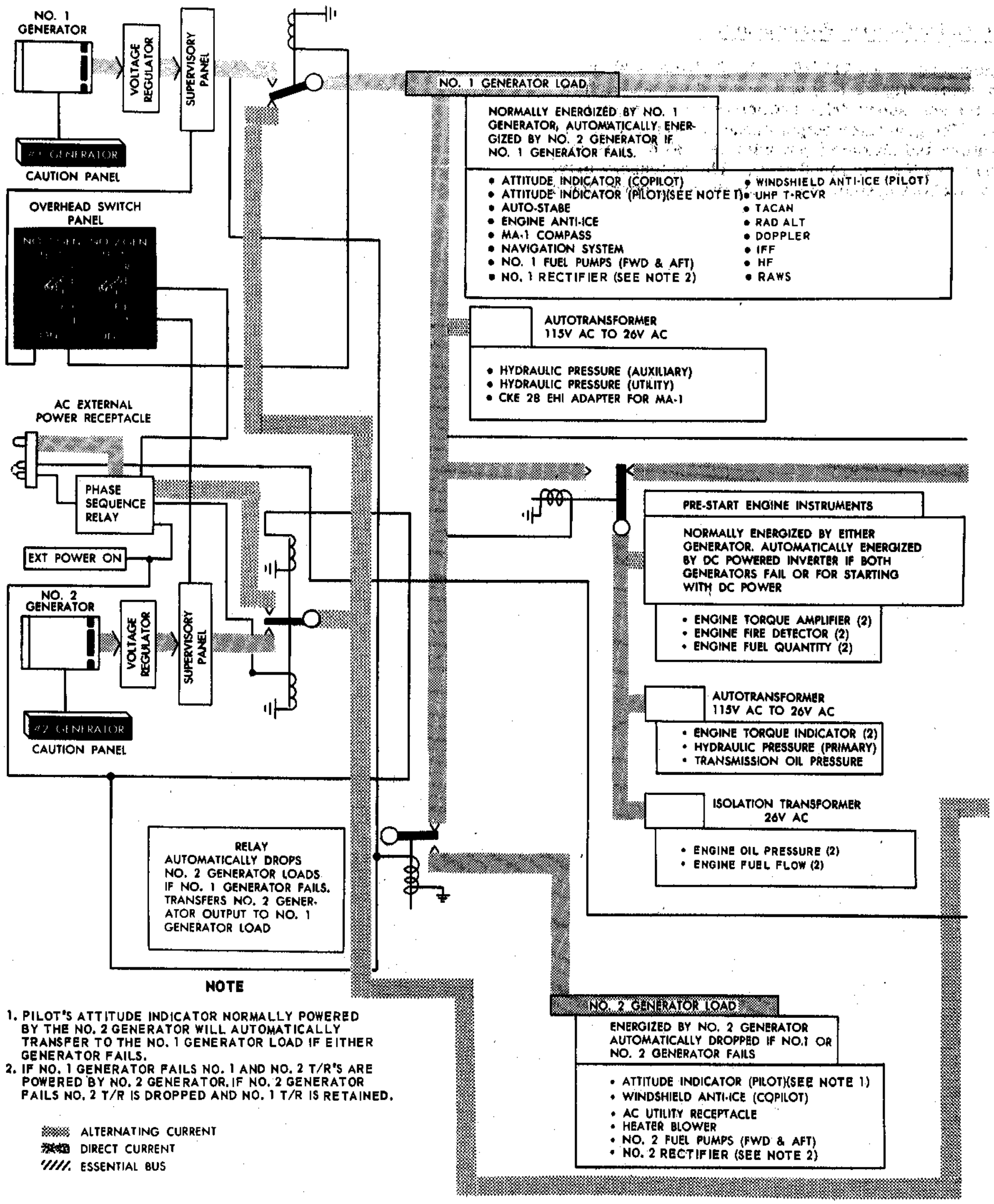

Figure 1-29. Electrical Power Supply System (Sheet 1 of 2)

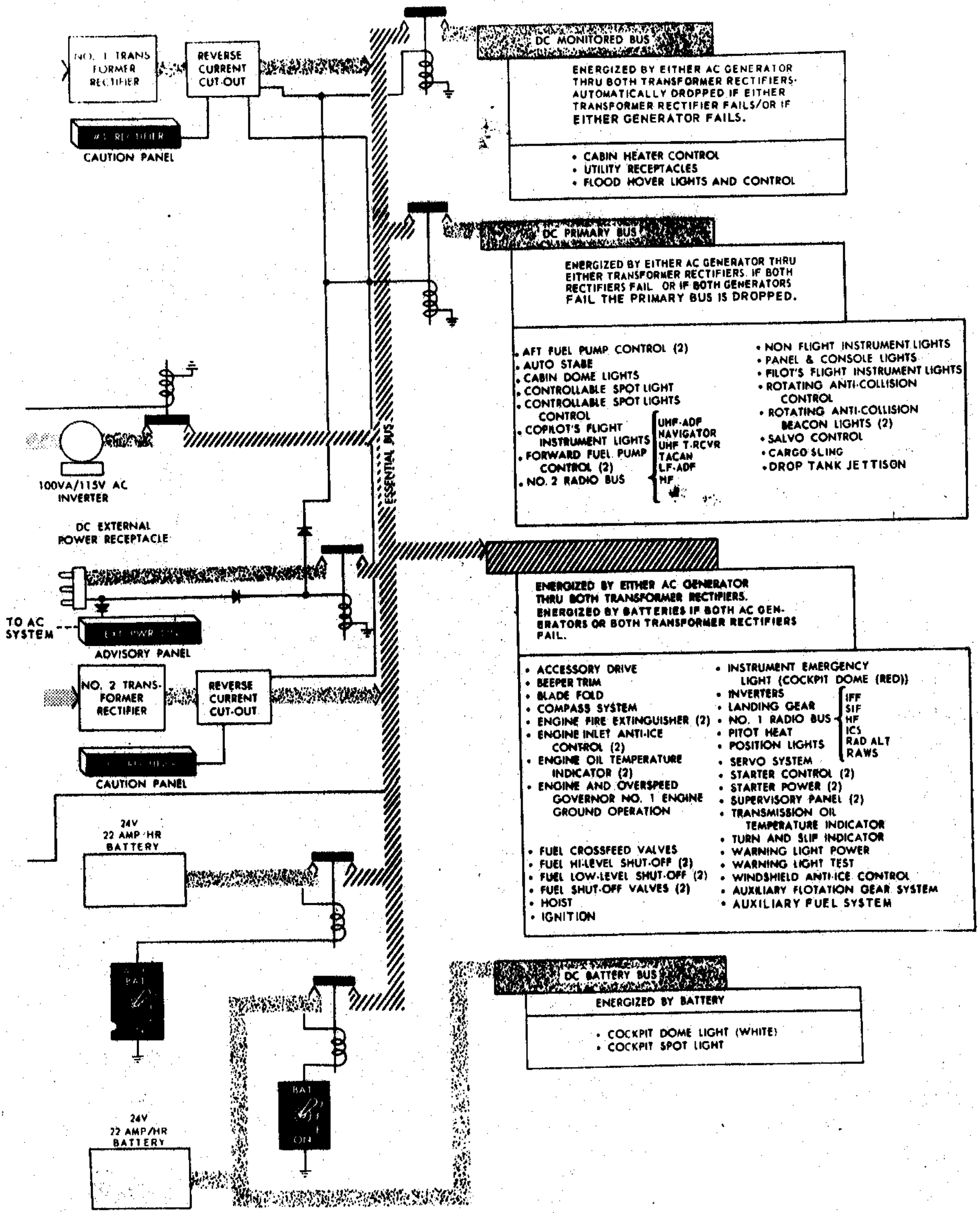

Figure 1-29. Electrical Power Supply System (Sheet 2 of 2)

PAGES 1-30C & 1-30D & FIGURE 1-29A DELETED both generators with the rotor system in a static position. Voltage regulation for each generator is provided by transistorized units, located in the electronics compartment. Two supervisory panels, located in the electronics compartment, provide control and protection of the electrical system from underfrequency, overvoltage, undervoltage, and open phase protection plus automatic switching of loads. When both generator switches are in the ON position, the generators are connected directly to the helicopter's ac system, if the supervisory panel is satisfied that voltage output as well as frequency output are within the prescribed limits during ground operation. Generator cutin occurs at approximately 95.8 through 101.3 percent power turbine speed when operating in accessory drive and should drop out within -2 percent of the cutin value. (These figures assume a 2 percent tachometer system accuracy.) In flight, the underfrequency protection of the supervisory panels is eliminated by action of the microswitch attached to the scissors of each landing gear. In this condition, the generators will remain on line throughout the entire normal rotor speed range.

Inverter. The 100VA, 115V inverter is located on the left-hand side of the electronics compartment. The inverter is energized by dc power from the dc essential bus and supplies power to the prestart engine instruments during starting with dc external power or battery power. In event of double generator failure during flight, the inverter receives power from the battery through the dc essential bus to power the prestart engine instruments. These instruments are the primary hydraulic servo pressure gage engine oil pressure gages, main gear box oil pressure indicator, torque indicators, fuel flow indicators, fuel quantity gages, and the fire detectors. When one or both generators are operating, the inverter is automatically shut off.

AC External Power Receptacle. The 115/200 volt, 400 cycle ac external power receptacle (figure 1-31) is located on the right side of the helicopter below the pilot's window. When an ac external power unit is plugged into the receptacle and turned on, the entire electrical power supply system is energized. The automatic relays which connect the power source to the ac circuits are located in the No. 2 relay junction box, located in the nose section of the fuselage.

NOTE

The type external power cart that should be used for ground operations is an AC, 115 volt, 3 phase, A, B, C phase rotation, 400 cps, a standard square pin plug (WYE connection) with a minimum capacity of 30 KVA.

The minimum rated ac external power unit that may be used for starting is 20 KVA units.

Phase Sequence Relay.

The phase sequence relay is incorporated to sense proper ac external power phase rotation. If, when external power is connected to the helicopter, the phase rotation of the external power cart is incorrect, ac external power will not be applied to the helicopter. However, if the phase rotation is correct, the line contactor relay is energized by direct current from the battery, thus allowing ac power into the helicopter. Therefore, when ac external power is connected and turned on, the battery switch must also be turned on momentarily to provide the necessary dc power for actuation of the line contactor relay. Once ac power is applied to the helicopter's system, the transformer rectifiers will supply the dc power and then the battery should be turned off to prevent overcharging the battery. After the No. 1 engine is started and operating at 104 percent, it will be noted that the No. 1 generator will operate normally, but the No. 2 generator will not. This is due to the manner in which the phase sequence relay is wired into the circuit on the No. 2 generator supervisory panel output. When external power is turned off or removed, the No. 2 generator will operate normally.

Alternating Current Distribution.

No. 1 Generator Load. During ground operation the No. 1 generator load can be energized by connecting ac external power. If dc external power is used instead of ac, or if the battery switch is placed in the ON position, the prestart engine instruments portion of the No. 1 generator load will be energized from the inverter. After the engine is started, in either condition, and the No. 1 generator reaches operating speed, relays will connect the No. 1 generator to the No. 1 generator load. If the No. 1 generator should fail during flight, relays automatically connect the No. 2 generator to the No. 1 generator load and disconnect the No. 2 generator load except for the pilot's attitude indicator and the No. 2 transformer rectifier. If both generators fail during flight, the pre-start engine instruments will be energized from the dc inverter operating on battery power.

No. 2 Generator Load. During ground operation the No. 2 generator load can be energized by connecting ac external power. After the engine is started and the No. 2 generator reaches operating speed, relays connect the No. 2 generator to the No. 2 generator load. If the No. 2 generator fails during flight, the No. 2 generator load is dropped except for the pilot's attitude indicator which is automatically switched to the No. 1 generator load. The No. 2 generator load is also dropped when the No. 1 generator fails and the No. 2 generator is automatically connected to the No. 1 generator load. The pilot's attitude indicator will continue to operate as it will also be connected to the No. 1 generator load.

Generator Switches.

The generator switches are located on the overhead switch panel (figure 1-6) in the pilot's compartment.

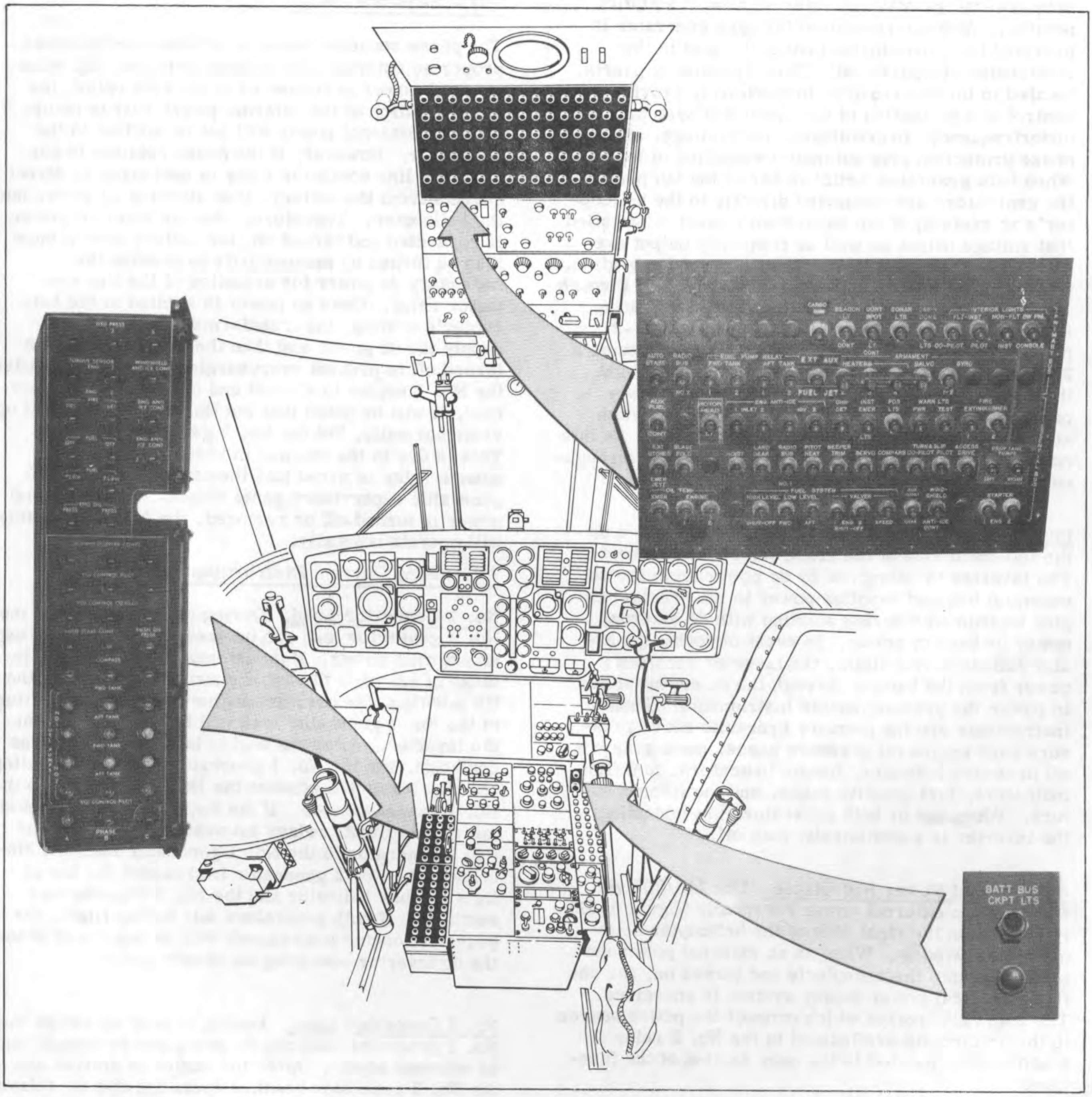

Figure 1-30. Circuit Breaker Panels (Typical)

The switches are marked No. 1 GEN and No. 2 GEN with three marked positions, TEST, OFF RESET, and ON. Placing the switch in the ON position turns the respective generator on if operating at generator speed; placing the switch in the OFF RESET position turns the generator off and resets the cycle. The TEST position is only used for ground maintenance checks by maintenance personnel.

AC Utility Power Receptacle.

One 115/200 volt ac utility power receptacle is located on the right cabin wall. Power to the receptacle is supplied by the No. 2 ac generator. The receptacle is provided with a cap to prevent entry of foreign material.

Generator Caution Lights.

Two generator caution lights are located on the caution panel (figure 1-43) on the pilot's side of the instrument panel. The lights illuminate whenever a generator fails, enabling the pilot to see the markings No. 1 GENERATOR and No. 2 GENERATOR on the panel.

is plugged into the ac receptacle. The external power advisory light will also illuminate when the dc external power is connected and generating power to the helicopter's electrical system. When the dc external power is plugged in and not producing power, the light will not illuminate.

DIRECT CURRENT POWER SUPPLY SYSTEM.

Two transformer rectifiers are the primary sources of direct current power for the DC electrical supply system. Other sources of direct current power are the batteries and the DC external power receptacle. Each transformer rectifier is powered by a separate AC generator. The No. 1 transformer rectifier is powered by the No. 1 generator and the No. 2 transformer rectifier is powered by the No. 2 generator. The two transformer rectifiers are in parallel and provide DC current regulated between 26 to 29 volts from no load to full load. The DC power supply consists of a four bus system, an essential bus, a primary bus, a battery bus, and a monitored bus. The essential bus provides power to equipment essential for flight under emergency conditions.

The primary bus provides power to equipment necessary for mission completion. The monitored bus provides power to nonessential equipment. The battery bus, connected directly to either one or two batteries, provides power for the pilot's compartment dome light and the cockpit spotlight. Control of the transformer rectifiers is automatic with the generators operating. The transformer-rectifier will supply charging current for the battery when the battery switch is in the ON position. The power sources are connected to the buses through circuit breakers.

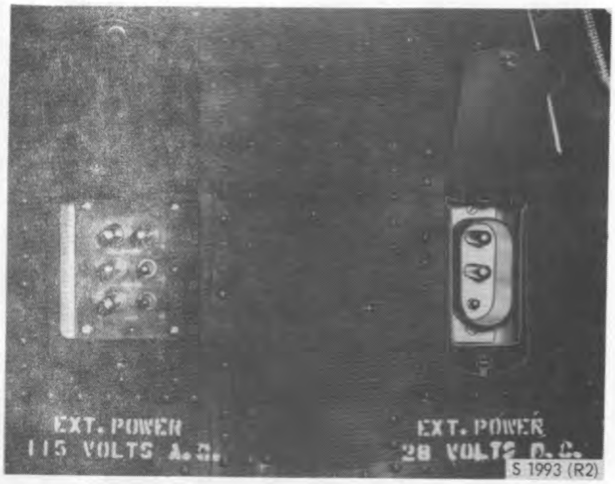

Figure 1-31. AC and DC External Power Receptacles

A 100/115 VAC inverter is provided for engine instruments prior to, and during engine starts.

Direct Current Power Sources.

Transformer-Rectifiers. Two 200 ampere, 28 volt dc transformer-rectifiers, located in the electronics compartment, are tied in parallel and convert generator furnished ac power to dc. During normal operation (generator on) both rectifiers supply all the necessary power. If either rectifier fails, the monitored bus is automatically dropped. If both rectifiers fail, the primary and monitored buses are automatically dropped.

GENERATOR FREQUENCY AT VARIOUS ROTOR AND POWER TURBINE SPEEDS

FLIGHT POSITION

PERCENT N_f or N_r	GENERATOR FREQ.
95	380.0
100	400.5
105	420.0

ACCESSORY DRIVE POSITION

PERCENT N_f	PERCENT N_r	GENERATOR FREQ.
98	0	380.0
104	0	400.0
108	0	420.0

FLIGHT POSITION

NOTES: 1. Underfrequency protection is locked out during flight.

2. Generator should pick up (come on the line) between 92.1 and 96.8 percent N_f/N_r and should drop out within minus two percent of the pickup value. (These figures assume a two percent tachometer system accuracy.)

ACCESSORY DRIVE POSITION

NOTES: 1. During ground operation frequency is set at 377-383 CPS for pickup, 374-380 CPS for dropout.

2. Generator should pick up (come on the line) between 95.8 and 101.3 percent N_f and should dropout within minus two percent of the pickup value. (These figures assume a two percent tachometer system accuracy.)

Figure 1-32. Generator Frequency at Various Rotor and Power Turbine Speeds Table

Figure 1-33. Auxiliary Battery Installation

Primary Battery. The 24 volt, 22 ampere hour battery (figure 1-3), located in the nose section forward of the pilot's compartment, is accessible from outside of the helicopter for maintenance. Battery power is used for limited ground operations, and as an emergency source of power to the essential bus in event of failure of both transformer-rectifiers during flight. The transformer-rectifiers supply charging current for the battery.

Auxiliary Battery. An auxiliary 24 volt, 22 ampere hour battery (figure 1-3) is located in the cabin aft of the pilot's compartment. The auxiliary battery is used with the primary battery in for starts without an external source. Each battery has its own switch.

Battery Switches.

There are two battery switches located on the overhead switch panel, one for the primary battery and the other for the auxiliary battery.

Primary Battery Switch. The primary battery switch is located on the overhead switch panel (figure 1-6) in the pilot's compartment. The switch marked BATT, has two marked positions OFF and ON. When the switch is placed ON and external power is not plugged in, battery power is supplied to the essential and battery buses. When the battery switch is in the OFF position, the battery bus is energized by battery power while all other buses are energized by either external power or rectified ac generator power.

Auxiliary Battery Switch. The auxiliary battery switch is located on the overhead switch panel (figure 1-6) in the pilot's compartment. The switch marked ALT has two marked positions, OFF and BAT. When the switch is placed in BAT and external power is not plugged in, battery power is supplied to the essential bus. When the battery switch is in the OFF position, the battery is taken off the line.

DC External Power Receptacle. The 28 volt dc external power receptacle is located on the right side of the helicopter below the pilot's window. External power can be connected and used for all ground operation until after the engine starts and the generators are in operation. The automatic control relays which connect power sources to the buses are located in the No. 2 relay junction box in the lower right-hand side of the fuselage.

NOTE

The external power cart used for ground operations should be a 28 volt, DC 750 AMPS continuous and 1000 AMPS intermittent.

The minimum rated dc external power unit that may be used for starting is a 400 ampere unit.

Direct Current Distribution.

Power for operation of the dc electrical equipment is distributed through four buses: the essential bus, the primary bus, the monitored bus, and the battery bus.

Essential Bus. The essential bus supplies power for operation of the dc flight instruments and all equipment necessary for safety of flight. The essential bus may be energized from all power sources. When ac external power is plugged in, or either generator is operating, the essential bus is energized by either transformer-rectifier. When dc external power is plugged in, or the primary or both battery switches are placed in the ON position, the essential bus is also energized.

Primary Bus. The primary bus supplies power to equipment necessary for completion of missions. When ac external power is plugged in/or either generator is operating, the primary bus may be energized through either transformer-rectifier. The primary bus may also be energized when dc external power is plugged in.

Monitored Bus. The monitored bus supplies power to equipment not required for safety of flight. When ac or dc external power is plugged in or either generator is operating, the monitored bus may be energized through both transformer-rectifiers.

Battery Bus. The battery bus is energized directly from the primary battery. When the battery switch is placed in the ON position, the battery bus is also energized from either generator or ac external power through either transformer-rectifier. The battery bus circuit breaker, located forward and above the radio circuit breaker panel, marked BATT BUS-CKPT LTS, protects the battery bus relay.

DC Utility Receptacles.

Two capped 28 volt dc electrical utility receptacles are installed in the helicopter. One is located on the pilot's compartment dome light panel (figure 4-23)

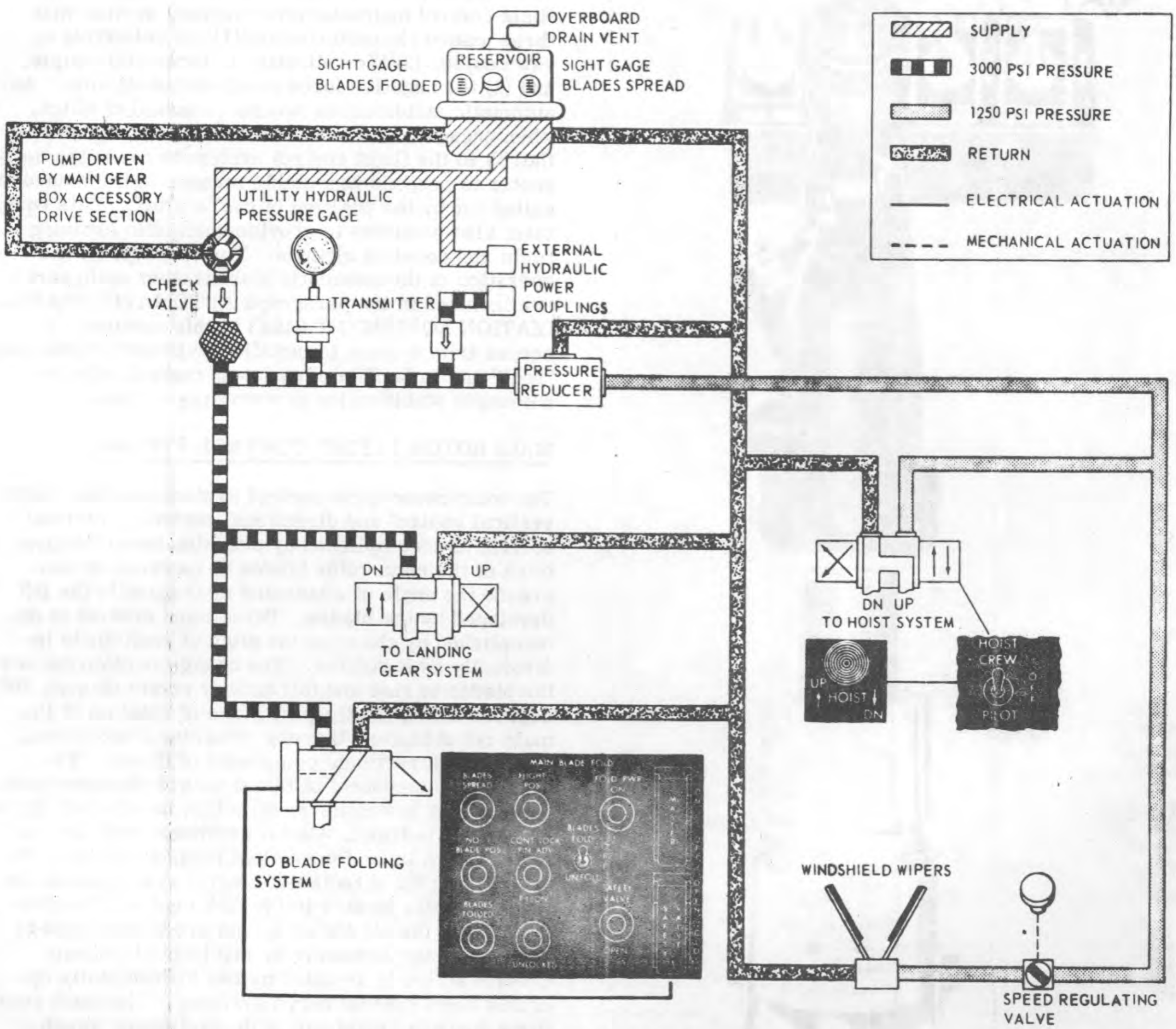

Figure 1-34. Utility Hydraulic System

and the other is located on the right cabin wall. The utility receptacles are powered by dc power from the monitored bus and are protected by a circuit breaker, located on the monitored bus circuit breaker panel.

Direct Current Circuit Breakers.

Circuit breakers protecting the various dc equipment are located on the overhead circuit breaker panel and pilot's console (figure 1-30) in the pilot's compartment. All circuit breakers are marked as to the circuits they protect and are of the push-pull type which may be reset. Any malfunctioning circuit may be isolated from the dc power supply system by pulling its circuit breaker.

Rectifier Caution Lights.

The rectifier caution lights are located on the caution panel (figure 1-43), on the pilot's side of the instrument panel. The lights come on whenever one or both of the transformer-rectifiers fail, enabling the pilot to see the markings, #1 RECTIFIER and #2 RECTIFIER. In normal flight operations with the generator switches on, the rectifiers supply all necessary dc power. When AC input to the transformer-rectifiers fails or is removed by turning off the generators, the rectifier caution lights on the caution panel are illuminated and the primary and monitored buses are automatically dropped. Either battery switch must be on in order for the rectifier caution lights to illuminate during failure of both transformer-rectifiers.

UTILITY HYDRAULIC SYSTEM.

The utility hydraulic system (figure 1-34) provides hydraulic pressure for all hydraulic equipment not included in the flight control servo and ASE systems. The utility hydraulic system reservoir (figure 1-49), located aft of the main gear box, supplies hydraulic

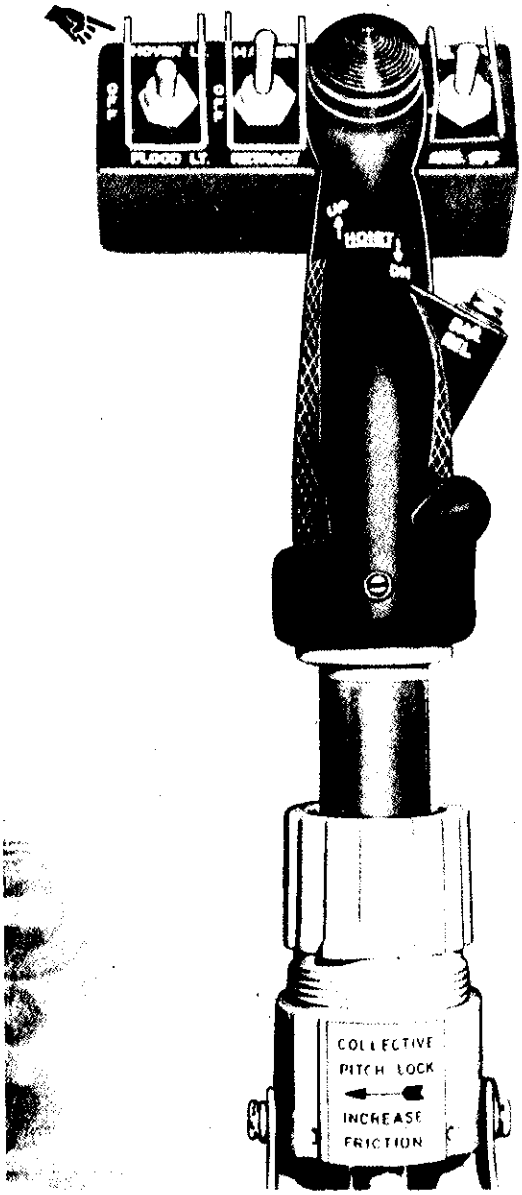

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Figure 1-35. Collective Pitch Lever Grip

fluid to the utility system. The utility hydraulic reservoir has a capacity of 1.09 gallons of hydraulic oil. The utility hydraulic pump, located on the accessory drive section of the main gear box, provides 3000 psi hydraulic pressure. Equipment, operated by the utility hydraulic system, includes the main landing gear, rescue hoist, windshield wipers system, and the automatic blade fold.

UTILITY HYDRAULIC PRESSURE INDICATOR.

The utility hydraulic pressure indicator (figure 1-5), located on the instrument panel, operates on 26 volt ac power from the number 2 autotransformer. The gage marked UT indicates pressure in the utility hydraulic system in psi.

FLIGHT CONTROL SYSTEM.

The flight control system is divided into three systems as follows: the main rotor flight control system, the tail rotor flight control system, and the flight control hydraulic power supply system with three unique characteristics; (1) the collective to yaw couple, (2) the collective to the cyclic couple, and (3) the negative force gradient installation. An automatic stabilization system is installed which, when engaged, provides corrections of limited authority to the flight control system to cause the helicopter to respond in a stable manner to the maneuver called for by the position of cyclic stick. This equipment also functions to provide automatic cruising flight and constant altitude. The description and operation of the automatic stabilization equipment are included in the paragraph AUTOMATIC STABILIZATION EQUIPMENT (ASE) in this section. A beeper trim system is installed to provide cyclic stick "feel" and to facilitate hands-off control with the automatic stabilization system in operation.

MAIN ROTOR FLIGHT CONTROL SYSTEM.

The main rotor flight control system provides both vertical control and directional control. Vertical control is accomplished by changing the collective pitch of the main rotor blades to increase or decrease the angle of attack and consequently the lift developed by the blades. Directional control is accomplished by changing the pitch of each blade individually as it rotates. The change in pitch causes the blades to rise and fall as they rotate through 360 degrees tilting the tip-path plane of rotation of the main rotor blades, thereby obtaining a horizontal, as well as a vertical, component of thrust. The horizontal component of thrust moves the helicopter horizontally in whichever direction the tip-path plane of rotation is tilted. Control motions from the collective pitch lever for vertical control and from the cyclic stick for directional control are combined in a mixing unit, located in the ASE 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 servo systems. The main rotor flight 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 Couple.

When the auxiliary servo is pressurized, there is a proportional but irreversible transfer of collective pitch motion, into the tail rotor blade angle. (Collective pitch motion will act to displace the tail rotor, but tail rotor pedal motion will not affect main rotor collective pitch blade angle.) This coupling provides automatic tail rotor torque change to compensate for collective pitch changes. Tail rotor blade angle changes result from both collective pitch lever and tail rotor pedal inputs. Any combination of collective pitch lever position and tail rotor pedal position, wherein the total would exceed the system limits, is non-attainable during flight. The collective pitch lever has overriding authority and therefore is always free to move within its full travel. If a col-

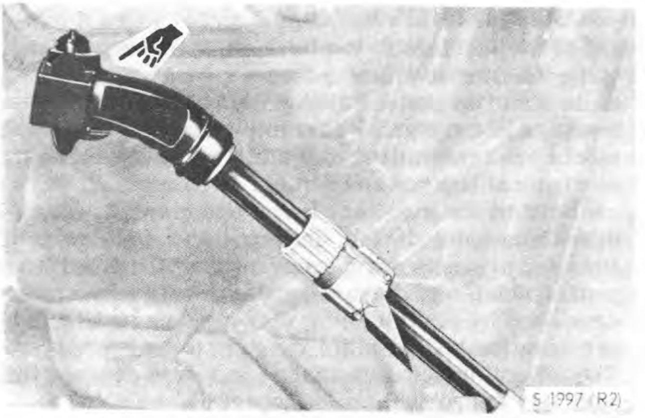

Figure 1-36. Collective Pitch Lever

lective pitch lever position is reached and then added to the tail rotor pedal position which creates a tail rotor blade angle equal to the system limits, additional collective pitch lever motions to exceed the limits is only possible at the sacrifice of tail rotor pedal position; the pedals will be forced to move. With auxiliary servo on, collective pitch lever low, and tail rotor pedal full left, raising the collective pitch lever to high will be accompanied by forward motion of the right pedal. With the collective pitch lever high and tail rotor pedals full right, reducing collective pitch lever to low will be accompanied by aft motion of the left pedal. With the auxiliary servo switch OFF the operation is the same except the irreversibility is not effective. Therefore, when the combination of collective and yaw positions reaches the system limit, additional pedal motion is possible by sacrificing collective pitch; however, the trading of motion will never occur in flight but may be encountered during ground checks. During rapid tail rotor pedal motions on the ground a noticeable noise can be heard aft of the pilot seat when the pedals reach their right or left limits. The sound is created by the system stops and indicates that collective pitch and the pedals have reached the limits of the tail rotor control. *Additional tail rotor* pedal motion is possible by reciprocal motion of the collective pitch lever.

Collective to Cyclic Pitch Couple.

A bias in the collective to cyclic pitch (fore-and-aft) coupling is incorporated in the mixing unit to apply, automatically, a nosedown pitching correction when the collective pitch lever is raised and noseup when the collective is lowered.

Collective Pitch Levers.

Two collective pitch levers (figure 1-36) are located in the pilot's compartment, one to the left of the pilot's seat and the other to the left of the copilot's seat. Both levers operate simultaneously to change the collective pitch of the main rotor blades. A nut 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.

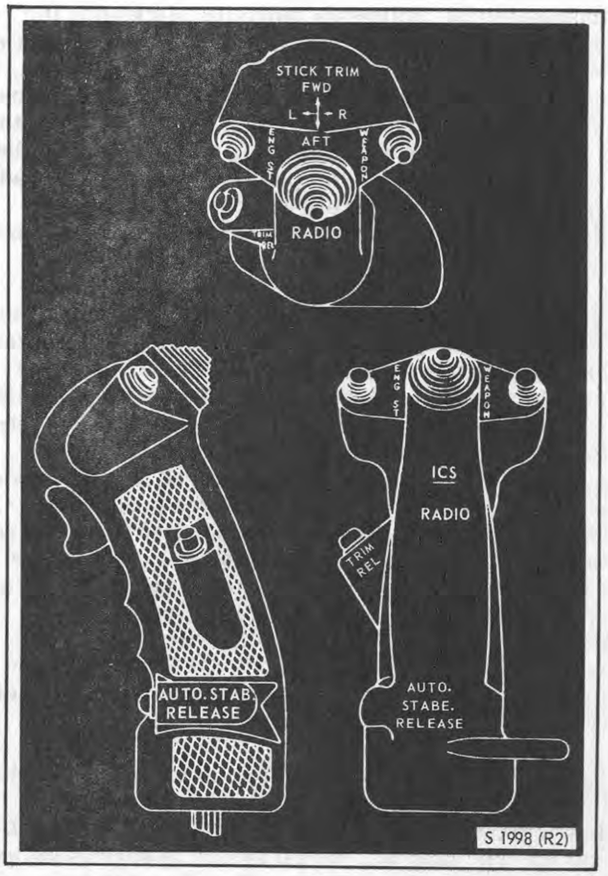

Figure 1-37. Cyclic Pitch Control Stick Grip

Cyclic Sticks.

The cyclic stick, located in front of each seat in the pilot's compartment, provides directional 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 1-37) contains pushbutton and thumb-operated switches for controlling various equipment installed in the helicopter.

Force Gradient - Beeper Trim System.

The force gradient - beeper trim system permits a fine degree of adjustment of the cyclic stick and provides cyclic stick feel. When used with the ASE engaged, the system permits hands-off flight by holding the stick in a selected trim 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, the other actuator positions the cyclic stick fore-and-aft. A master switch, located on the overhead switch panel, marked BEEPER TRIM, ON, and OFF must be placed in the ON position before the beeper trim system will be in operation. The actuators are operated by a four position beeper trim switch mounted on both the pilot's and copilot's cyclic stick grips. To trim the cyclic stick, the beeper trim switch is pushed in the direction of desired cyclic stick movement; the actuators move the stick until the beeper 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. When the pressure on the cyclic stick is released, spring tension returns the stick to the original trim position. The force gradient - beeper trim system will operate as long as there is both dc power to the essential bus and auxiliary hydraulic pressure to the actuators.

Beeper Trim Switch. A beeper trim switch marked BEEPER TRIM, ON and OFF is located on the overhead switch panel (figure 1-6). The switch is the master control for the force gradient - beeper 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. 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 switch is placed in the OFF position, the force gradient system is inoperative and the beeper trim system will not position the cyclic stick.

Beeper Trim. The beeper trim switches, located on the pilot's and copilot's cyclic stick grips (figure 1-37), have four 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 force gradient system will then function about this location of the cyclic stick.

Trim Release Button. The spring-loaded pushbutton switches, located on the pilot's and copilot's cyclic stick grips (figure 1-37), are marked TRIM REL. To change trim position normally (without using the beeper trim), depress the trim release button, move the cyclic stick to the new position, and release the trim release button. The trim release button controls dc essential bus power to the trim actuators. The beeper trim and trim release switches are powered by the dc essential bus and are protected by a circuit breaker marked BEEPER TRIM, located on the overhead circuit breaker panel.

TAIL ROTOR FLIGHT CONTROL SYSTEM.

The functions of the tail rotor flight control system are to compensate for main rotor torque and 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. Gross weight, altitude, rate-of-climb, airspeed and the corresponding power settings, and collective pitch 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 control linkage, prevents abrupt movements of the tail rotor 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 tail rotor 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 without moving the pedals, unless both collective pitch and tail rotor blade angle are at their maximum limits in which case the pedal will be forced back with collective pitch change. The mixing unit coordinates and transfers independent movements of the lateral, forward, aft, and directional (yaw) controls to the primary servocylinders and the tail rotor. The mixing unit integrates collective pitch control movements with the lateral, fore-and-aft, and directional systems, causing the controls to move the three primary servocylinders simultaneously in the same direction, and to change the pitch on the tail rotor blades to compensate for the change in pitch of the main rotor blades.

Tail Rotor Pedals.

The tail rotor pedals (figure 1-4), one set in front of the pilot and the other in front of the copilot, change the pitch and thrust of the tail rotor and, consequently the heading of the helicopter. Pressing the left pedal increases the tail rotor blade pitch which increases thrust, and turns the helicopter to the left. Pressing the right pedal decreases tail rotor blade pitch which decreases thrust, and allows the helicopter to turn to the right. Tail rotor pedal adjustment knobs are used to adjust the pedals for leg length. Electrical switches mounted on the force link assembly cancel the directional signals of the automatic stabilization equipment when approximately 4 to 6 pounds of pressure is exerted on the pedals. Toe brake pedals for the main landing gear wheel brakes are mounted on the pilot's tail rotor pedals.

Tail Rotor Pedal Adjustment Knobs.

Tail rotor pedal adjustment knobs 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 tail rotor pedals. The knobs are rotated to the right, as indicated by the arrow marked FWD for forward adjustments and to the left, as indicated by the arrow marked AFT for aft adjustments. The pilot's tail rotor pedals are adjusted with the knob in the right side of the cockpit and the copilot's tail rotor pedals are adjusted by the knob on the left side of the cockpit.

FLIGHT CONTROL SERVO HYDRAULIC SYSTEMS.

The flight control servo hydraulic systems (figure 1-38) consist of a primary and an auxiliary flight control servo system. The servo systems are required for a power boost by the pilot to operate the controls. The servos also prevent feedback of rotary wing vibratory loads to the control sticks. Both servo systems operate from independent hydraulic systems and both utilize similar servo hydraulic units to vary the main rotor and tail rotor blade pitch through the mechanical linkage of the regular flight control system. Each servo unit consists of a bypass valve and sloppy link, power piston, pilot valve, and the ASE valve in the auxiliary servo units only. The flight control system actuates the pilot valve which admits hydraulic oil into the servo unit. 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 three servo units of the primary servo system are located at the stationary swashplate. All three servo units respond simultaneously and move in the same direction in response to movements of the collective pitch lever. Two of the units (lateral servo units) respond simultaneously but move in opposite directions in response to lateral movements of the cyclic stick. One of the servo units (fore-and-aft servo unit) responds to fore-and-aft movements of the cyclic stick. Since all three movements can occur simultaneously through the action of the mixing unit, the position of any primary servo unit is the result of the combined input of the cyclic stick and collective pitch lever. This results in a primary servo system in which any one servo has an effect on both collective pitch and cyclic (lateral or fore-and-aft) pitch. The three servo units of the auxiliary servo system are located between the mixing unit and the flight controls. Each control input acts independently on the corresponding auxiliary servo. The collective pitch lever positions the collective servo. The cyclic stick positions either, or both, the fore-and-aft servo or the lateral servo. This results in an auxiliary servo system in which only one servo has an effect on collective pitch, one on fore-and-aft cyclic pitch and one on lateral cyclic pitch. The primary servo hydraulic pump is driven by the accessory section of the main gear box. The primary hydraulic system reservoir (figure 1-51), mounted aft of the main gear box, has a capacity of approximately 0.45 gallon of hydraulic oil. The PRI SERVO PRESS caution light will illuminate when the primary servo pressure drops below 1000 psi or is turned off.

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 automatic stabilization corrective signals into the flight control systems and reacts to flight loads in the event of primary servo failure. The auxiliary servo hydraulic pump is driven by the main gear box accessory section. The auxiliary hydraulic system reservoir (figure 1-51), located aft of the primary hydraulic system reservoir, has a capacity of approximately 0.45 gallon of hydraulic oil. The AUX SERVO PRESS caution light will illuminate when the auxiliary servo pressure drops below 1000 psi or is turned off.

FLIGHT CONTROL SERVO SWITCH.

Both the primary and the auxiliary flight control servo systems are controlled by the same three position flight control servo switch, located on the collective pitch grip lever (figure 1-36). The marked switch positions are PRI OFF and AUX OFF. Both servo systems are normally in operation with the switch in the unmarked center (ON) position. To turn off the primary servos, the 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. Stronger rotary rudder pedal forces and the absence of pedal damping will be encountered with the auxiliary servos inoperative. The systems are interconnected electrically in such a way that, regardless of the switch position, it is impossible to turn either one off unless there is 1000 psi in the remaining system for proper operation. The servo shutoff valves operate on direct current from the essential bus.

SERVO HYDRAULIC PRESSURE INDICATORS.

The primary and auxiliary servo hydraulic pressure indicators (figure 1-5) located on the instrument panel, operate in alternating current from the No. 1 generator. The primary servo hydraulic pressure gage is also connected to the inverter. If either servo system malfunctions, the malfunctioning system may be turned off and the helicopter flown on the other servo system. If the pressure in either the primary or the auxiliary system drops below 1000 psi, a pressure switch prevents the other system from being shut off, regardless of the position of the servo switch.

AUTOMATIC STABILIZATION EQUIPMENT (ASE).

The automatic stabilization system (ASE) 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. ASE 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

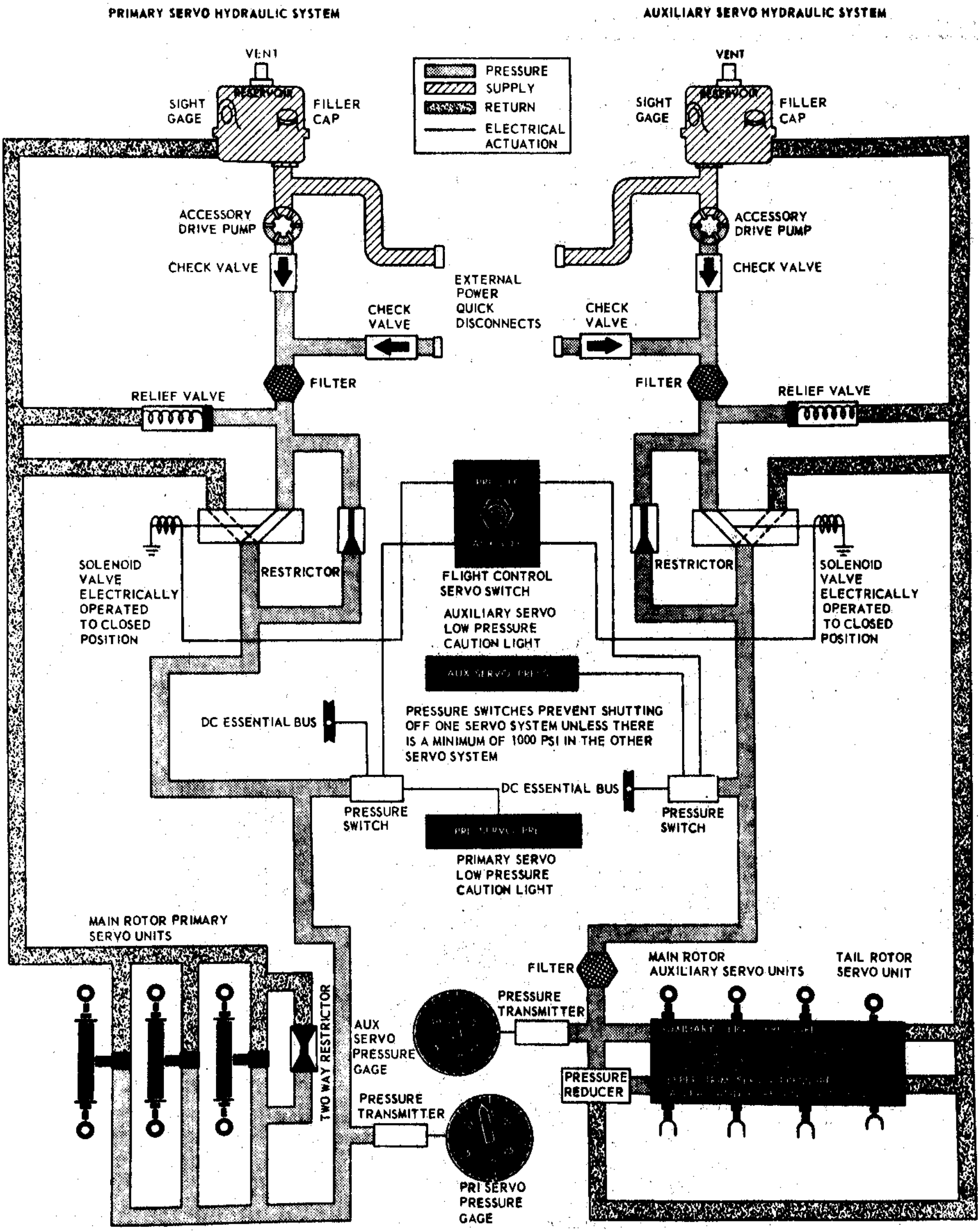

Figure 1-38. Flight Control Servo Hydraulic Systems

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 ASE or any channel, as desired, by means of switches on the ASE control panel, channel monitor control panel, cyclic sticks, and collective pitch levers. The ASE indicator and attitude indicators provide the pilot and copilot with visual indication of all ASE signals. ASE 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. ASE is capable of maintaining the barometric altitude of the helicopter within ± 25 feet or 5% of the altitude, whichever is greater, during straight unaccelerated flight, or when hovering out of ground effect utilizing barometric altitude reference. In the pitch and roll channels, the fuselage attitude is held constant by comparing the actual attitude signal received from the vertical gyro with the reference attitude signal received from the stick position sensor (senses position of cyclic stick). Automatic pitch and roll attitude stability correction occurs any time the helicopter is displaced from the reference attitude. In the yaw channel, the helicopter heading is held constant by comparing actual heading signals received from the MA-1 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.

ASE utilizes both ac power from No. 1 ac GEN, and dc power from the dc primary bus. A thermal time delay relay is incorporated to allow approximately 3 minutes for the vertical gyros to reach a stabilized

Figure 1-39. ASE Control Panel (Typical)

state before dc power is applied to the system. The ASE 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 ASE ENG button must be depressed before the BAR ALT ENG button is depressed. AC power to the ASE is protected by 2 circuit breakers marked AUTO STABE, located on the lower left console circuit breaker panel. DC power to ASE is protected by a circuit breaker marked AUTO STABE, located on the overhead dc circuit breaker panel.

CAUTION

If the ASE channel monitor panel has become water soaked, caution should be exercised for possible ASE hardovers when engaging ASE or BAR ALT.

To Change Altitude If Barometric Altitude Mode is In Use.

1. BAR REL button on collective pitch lever - DEPRESS.
2. Establish new altitude.
3. Stabilize airspeed, altitude, and power.
4. BAR REL button - RELEASE.

AUTOMATIC STABILIZATION CONTROL PANEL.

The automatic stabilization equipment controls are located on the panel marked ASE control (figure 1-39), which is mounted on the control console between the pilots. There are five button switches on the ASE control panel, two of which illuminate when engaged.

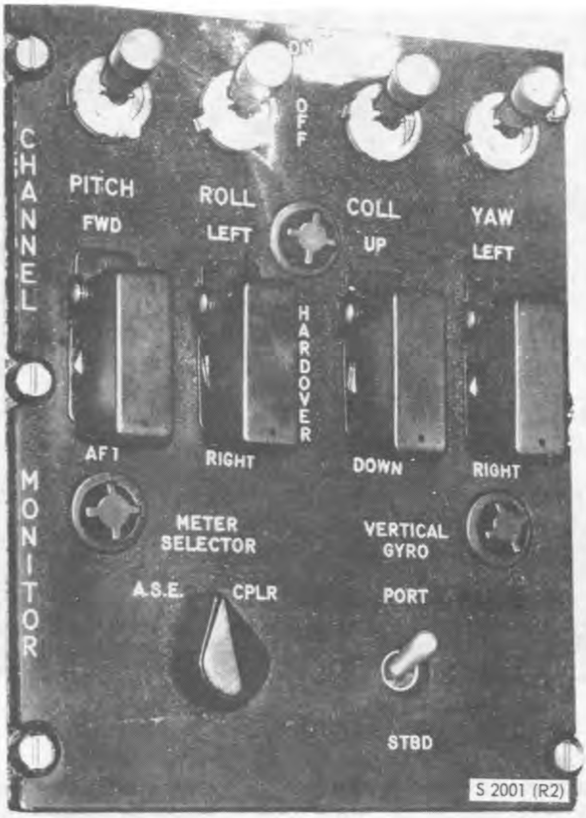

Figure 1-40. Channel Monitor Panel

The three pushbuttons in the top row are marked ASE, BAR ALT, and CPLR. When the ASE button is pressed, the pitch, roll, and yaw channels are operating as indicated by the lighting of the engagement light. Pressing the BAR ALT button engages the barometric altitude controller; the light comes on indicating engagement. The barometric altitude controller is disengaged when the BAR OFF button is pushed. If the barometric altitude controller is engaged, it can be released momentarily when the BAR REL button on the pilot or copilot's collective pitch stick grip is pressed to make possible a change in altitude. The CPLR button is inoperative. The ASE is disengaged by depressing the button marked AUTO STABE RELEASE on the pilot's or copilot's cyclic stick grips. There are five knobs on the ASE control panel. They are marked DRIFT, SPEED, YAW TRIM, CG TRIM, and ALTITUDE. The drift, speed, and altitude knobs are inoperative. The two remaining knobs may be identified by touch. The CG trim knob is clover leaf shaped. The yaw trim knob is triangularly shaped. The CG trim knob changes ASE authority on the longitudinal axis. The yaw trim knob enables the pilot to trim the heading of the helicopter accurately during cruise. One rotation of this knob will turn the helicopter 72 degrees. The electrical power for the ASE control panel is supplied by the primary bus through a circuit breaker marked AUTO STABE, located on the overhead circuit breaker panel.

CHANNEL MONITOR PANEL.

The channel monitor panel (figure 1-40) marked CHANNEL MONITOR is mounted on the pilot's console. There are four toggle switches in the top row, marked PITCH, ROLL, COLL, and YAW, with a marked center position of OFF and forward position of ON. They permit individual disengagement of four channels of the ASE. They are normally left in the ON position except when the pilot wishes to disengage a malfunctioning channel. Four guarded three position switches in the lower row marked HARDOVER are used to check system authority. Each switch may be placed in a position to check the corresponding PITCH, ROLL, COLL, and YAW channels of the ASE. Marked positions of the switches are FWD and AFT, LEFT and RIGHT, UP and DOWN, and LEFT and RIGHT. When the switch guards are closed, the switches are held in the OFF position. The switch guards must be lifted before override checks can be accomplished. A selector switch marked METER SELECTOR has two marked positions, ASE and CPLR. This switch permits viewing on the indicator in the A mode of the total output of the ASE system when in the ASE position. The CPLR position is inoperative. The toggle switch marked VERTICAL GYRO has two marked positions, STBD and PORT. Pitch and roll references for the ASE can be selected from either starboard or port vertical gyros. The preferred position is port so that a gyro failure will not cause the pilot to lose the use of both his attitude indicator and the ASE pitch and roll channels simultaneously. With the vertical gyro switch in the PORT position, the port gyro provides signals for the ASE and the copilot's attitude indicator while the starboard gyro provides signals for the pilot's attitude indicator. With the vertical gyro switch in the STBD position, the starboard gyro provides signals for the ASE and the pilot's attitude indicator while the port gyro provides signals for the copilot's attitude indicator. Electrical power for the channel monitor panel is supplied by the primary bus. A circuit breaker marked AUTO STAB is located on the overhead circuit breaker panel.

CHANNEL MONITOR TEST SWITCH.

The channel monitor test switch marked CHAN MON TEST and OFF is located on the overhead switch panel (figure 1-6). The OFF position disables all the channel monitor panel hardover switches but does not affect any of the other switches on the channel monitor panel. The switch when OFF, breaks the 28 volt dc circuit continuity to the hardover switches, thus preventing hardover inputs to the auxiliary servo if a short in the circuit should arise or if a hardover switch was inadvertently left in the hardover position. The switch is protected by a circuit breaker marked AUTO STABE, located on the overhead circuit breaker panel.

AUTO STABE RELEASE BUTTONS.

ASE is disengaged by depressing the buttons marked AUTO STABE RELEASE which are located on both the pilot's and copilot's cyclic stick grips (figure 1-37).

BAR ALT RELEASE SWITCHES.

The barometric altitude controller is disengaged momentarily, 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 1-35). After stabilizing altitude and airspeed at the new altitude, the BAR REL button is then released and the helicopter will be stabilized at the new altitude.

HOVER INDICATOR.

Two hover indicators (figure 1-5), are installed on the instrument panel. The indicators provide a visual indication of doppler information during automatic cruise flight. 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. (See figure 1-41.) Two movable bars coincide with the center vertical and horizontal axis scale marks of the dial and in a hover intersect at a small circle marked on the dial face. Two arrowhead-type pointers, one located on the left hand side of the indicator move vertically up or down coinciding with the vertical scale, and the other pointer at the bottom of the indicator moves horizontally left or right coinciding with the horizontal scale. A mode selector switch is located at the lower left hand edge of the hover indicator case with marked positions A, C, and D. A mode selector window, located on the dial face of the indicator, operates in conjunction with the mode selector switch and displays one of the letters A, C, or D to indicate the mode of operation. Operation in the A mode connects the hover indicator to the ASE. With the meter selector on the channel monitor panel (figure 1-40) in the ASE position, the hover indicator will operate as a null indicator, indicating the input to the ASE servo valves. The hover indicator horizontal bar is used to monitor the pitch channel; the vertical bar, the roll channel; the vertical pointer, the altitude channel; and the horizontal pointer, the yaw channel. The hover indicator scale factor in this mode is a 2 ma per division of servo valve differential current, with 8 ma considered as hardover. Operation in the D mode connects the hover indicator to the AN/APN-175(V)-1 doppler radar. The horizontal bar indicates heading velocities and the vertical bar indicates drift velocities. Each increment on the hover indicator horizontal and vertical scales indicates 5 knots with a maximum of 20 knots. The vertical pointer indicates vertical velocity, with each increment to 500 feet per minute full scale deflection equal to 2000 feet per minute up or down. To indicate forward flight, the horizontal bar will move downward and to indicate a drift, the vertical bar will move in direction of the drift, therefore, the pilot flies into the bar for correction back to hover position. In both C and D modes the yaw pointer is disconnected and should not move. An OFF flag on the upper dial of the hover indicator is used in all three modes of operation. In the A mode, the flag disappears when the ASE is engaged. In the D mode, the flag disappears when the doppler transmitter is turned on. The C mode or cable mode is no longer operational in this aircraft.

INSTRUMENTS.

MAGNETIC COMPASS.

A magnetic compass is located at the top center of the instrument panel. A compass correction card is located on the pilot's side of the instrument panel.

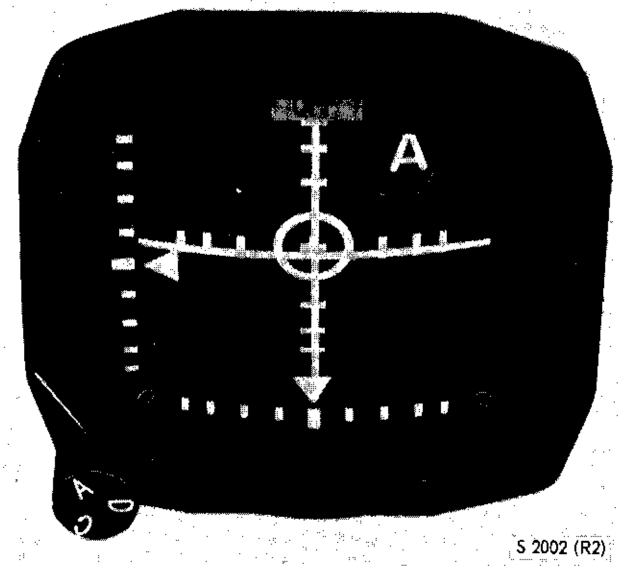

Figure 1-41. Hover Indicator

FREE-AIR TEMPERATURE GAGE.

A bimetallic free-air temperature gage is located on the center line of the helicopter on the windshield glass panel.

CLOCKS.

Two eight day, 12 hour clocks (figure 1-5) 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.

Two electrically heated pitot tubes with static ports are located over the pilot's compartment, one on the right side and the other on the left side of the helicopter forward of the engine air intakes. The lines carrying pressure from both pitot-static tubes are connected together so that the pilot's and copilot's instruments are fed through a common source of plumbing. A restrictor is installed in the static pressure line from both probes to the automatic stabilization barometric altitude sensing unit for filtering transient pressure changes, thereby contributing to a more stable flight path. The static ports are connected by common tubing to the airspeed indicators, the vertical velocity indicators, and the altimeters. Tight fitting sleeves located on both probes (pilot and copilot) are designed to maintain a constant static pressure for all helicopter altitudes and airspeeds, thereby providing a greater degree of accuracy in airspeed indication. Both pitot tube heads may be heated to prevent icing. Refer to PITOT HEATER in Section IV.

TURN-AND-SLIP INDICATORS.

Two turn-and-slip indicators (figure 1-5) are installed on the instrument panel, one in front of the pilot and

Figure 1-42. MA-1 Directional Gyro Compass Control Panel

one in front of the copilot. The turn-and-slip indicators give visual indication of the helicopter's rate of turn and coordinated flight. The gyros of the turn-and-slip indicators operate on direct current from the dc essential bus. The pilot's and copilot's turn-and-slip indicators are protected by circuit breakers marked TURN & SLIP, COPILOTS - PILOTS, located on the overhead circuit breaker panel.

MA-1 DIRECTIONAL GYRO COMPASS SYSTEM.

The MA-1 directional gyro compass system provides stabilized compass indications by combining the advantages of the remote indicating magnetic compass with the gyro compass. The oscillations of the magnetic compass and the drift error of the directional gyro are eliminated when operating as a gyro-magnetic compass and an accurate stabilized magnetic heading is indicated. In magnetically unreliable regions, such as encountered in northern latitudes and on carrier decks, the gyro may be unslaved from the compass system to act as a free directional gyro. The system consists of a magnetic flux valve in the tail cone, a directional gyro coupler and amplifier in the pilot's compartment, and a control panel on the cockpit console. Compass headings are indicated by the rotating azimuth card on the course indicators, located on the instrument panel. The system operates on direct current from the essential bus, and ac power from the No. 1 generator. In addition to providing stabilized magnetic headings, the system supplies directional signals to the automatic stabilization system.

MA-1 Directional Gyro Compass Control Panel.

The MA-1 directional gyro compass control panel (figure 1-42), located on the cockpit console, contains all the controls for the operation of the directional gyro compass system. The mode-of-operation switch on the compass control panel has three marked positions: FREE N. LAT, SLAVED, and FREE S. LAT. When the switch is placed in either of the FREE positions, the 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 SLAVED position, the directional gyro is slaved to the magnetic compass heading and the rotating azimuth card on the course indicators will indicate stabilized magnetic headings. The synchronizing indicator is a white pointer visible through a window located directly forward of the mode-of-operation switch. When the pointer is in line with the white arrow on the control panel, the system is in synchronization. A red flag on the synchronizing indicator appears whenever electrical power to the system is turned off or has failed. Synchronization is obtained by pulling out the heading set knob, marked PULL TO SET, and rotating it until the pointer of the synchronizer indicator is in line with the arrow. Two settings of the heading-set knob will cause the synchronizing indicator to line up with the arrow. One is correct and the other will result in an unstable 180-degree ambiguity. The correct setting can be recognized by the relationship between the direction that the heading-set knob is turned and the direction of movement of the synchronizing indicator. When the heading-set knob is turned in a clockwise direction, the synchronizing indicator should approach the arrow from left to right, and when the heading-set knob is turned counterclockwise, the synchronizing indicator approaches the arrow from right to left. One method of insuring correct synchronization and preventing any possibility of the unstable 180-degree ambiguity is to rotate the heading-set knob in a clockwise direction until the synchronizing indicator moves from left to right. After the indicator moves from left to right, the system is synchronized by turning the heading-set knob until the synchronizing indicator lines up with the arrow. The synchronizing indicator will continue to provide a check on the slaving operation during flight; however, the pointer will oscillate about the arrow. When the system is being used as a free gyro, the mode-of-operation switch is set to either FREE N. LAT or FREE S. LAT and the latitude is set on the latitude compensation control marked SET TO LAT to compensate for the drift of the gyro due to the rotation of the earth. The heading-set knob is rotated to set up any desired heading on the rotating azimuth card on the course indicators. 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.

Slaved Gyro Operation.

1. Allow approximately five minutes with ac power in the helicopter for the gyro to reach operating speed.

2. Mode-of-operation switch - SLAVED.

3. Heading-set knob - Synchronize gyro and magnetic heading by pulling knob and rotating until synchronizing indicator is centered.

Free Gyro Operation.

1. Allow approximately five minutes with ac power in the helicopter for the gyro to reach operating speed.

2. Mode-of-operation switch - FREE N. LAT or FREE S. LAT, as required.

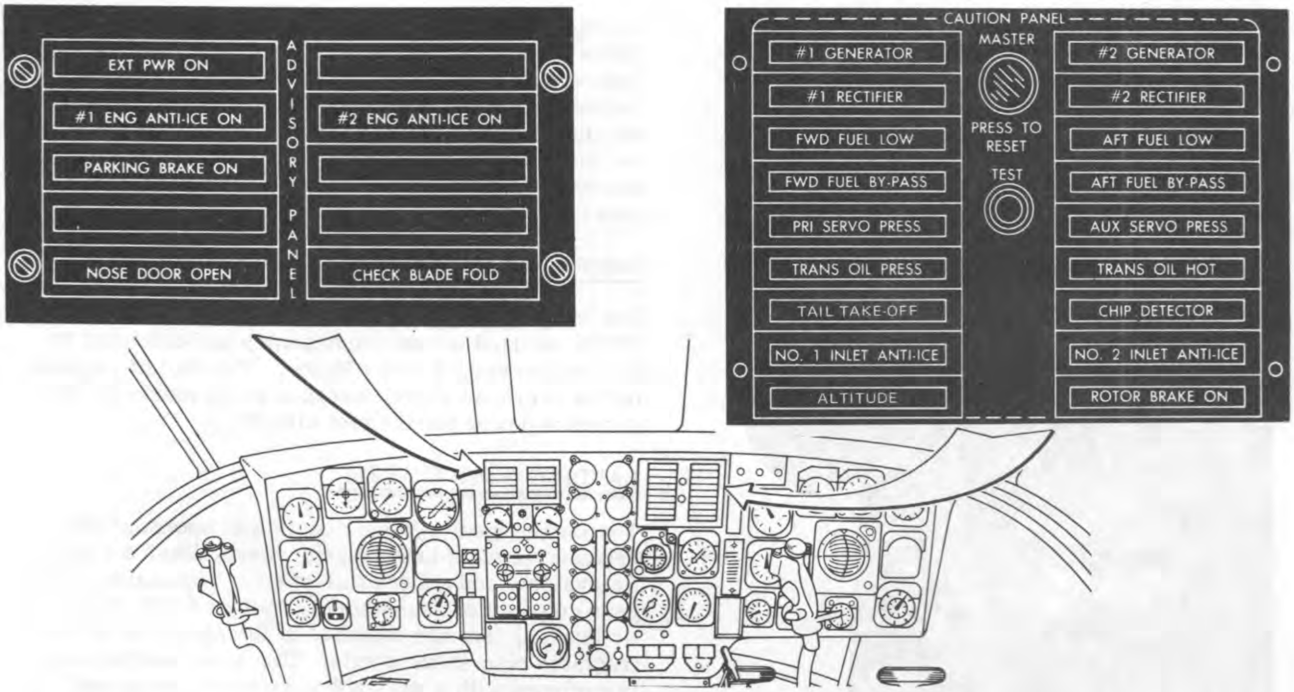

Figure 1-43. Caution and Advisory Panel

3. Latitude compensator control - Set to degree of latitude.

4. Heading control knob - Set to desired heading.

ATTITUDE INDICATORS.

Two attitude indicators (figure 1-5), installed on the instrument panel, one in front of the pilot and one in front of the 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 distinct white horizon line dividing the two colors, white above, black below. A power warning flag marked OFF will appear in the face of the indicator when the indicator is inoperative. The flag will appear under the following circumstances: when ac power has not been applied, for the first 68 seconds after ac power has been applied, and when any unbalance of the three phases of ac power occurs. Two trim adjustment knobs are located on the front of the attitude indicators, one at the upper right 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 band 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 8 and 20 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 4 and 10 degrees climb. The pilot's attitude indicator operates on ac current supplied by the No. 2 generator. The copilot's attitude indicator operates on ac current supplied by the No. 1 generator. Should the No. 2 generator fail, the pilot's attitude indicator will automatically be transferred to the No. 1 generator. The pilot's attitude indicator receives pitch and roll signals from the starboard vertical gyro and the copilot's attitude indicator receives signals from the port vertical gyro regardless of the position of the vertical gyro switch.

WARNING

The attitude warning flag may not be visible with a slight reduction of electrical power or failure of other components within the system. This can result in erroneous or complete loss of pitch and bank presentations without the warning flag appearing.

CAUTION AND ADVISORY PANELS.

ADVISORY PANEL.

The advisory panel (figure 1-43) marked ADVISORY PANEL is located on the copilot's side of 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 which indicates a particular system is in operation or an unsafe condition exists. When a system is in operation or an unsafe 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 unsafe flight condition is corrected. Pressing the switch marked TEST on the caution panel tests lights of both the caution and advisory panels. The following placard-type lights are contained on the advisory panel: EXT PWR ON, #1 ENG ANTI-ICE ON, #2 ENG ANTI-ICE ON, PARKING BRAKE ON, NOSE DOOR OPEN, and CHECK BLADE FOLD.

Figure 1-44. Landing Gear Control Panel

CAUTION PANEL.

The caution panel (figure 1-43) marked CAUTION PANEL is located on the pilot's side of the instrument panel. The caution panel gives the pilot visual indication of failure or unsafe conditions of certain critical power equipment in the helicopter. The caution panel contains placard-type amber caution lights, each having its own operating circuit, which indicates 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 WARN LTS PWR and TEST which are located on the overhead 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. The following placard-type lights are contained on the caution panel: #1 GENERATOR, #2 GENERATOR, #1 RECTIFIER, #2 RECTIFIER, FWD FUEL FLOW, AFT FUEL FLOW, FWD FUEL BY-PASS, AFT FUEL BY-PASS, PRI SERVO PRESS, AUX SERVO PRESS, TRANS OIL PRESS, TRANS OIL HOT, TAIL TAKE-OFF, CHIP DETECTOR, NO. 1 INLET ANTI-ICE, NO. 2 INLET ANTI-ICE, ALTITUDE, AND ROTOR BRAKE ON.

Caution Panel Master Light.

A master light marked MASTER - PRESS TO RESET is located in the center of the panel. The master light illuminates when any of the caution panel lights are energized by a malfunction. The master light will remain on until the malfunction is corrected, or until de-energized by the pilot. Pressing the light de-energizes the master light, permitting the master light to indicate a second malfunction if one should occur while the first malfunction is still present.

Caution and Advisory Lights Test Switch.

The switch, located on the caution panel marked TEST, is used to test the capsules and circuitry to the caution and advisory lights. The circuit breaker on the overhead circuit breaker panel marked TEST provides power for the test circuit.

LANDING GEAR SYSTEM.

The landing gear system consists of sponsons and retractable main landing gear assemblies, a hydraulic system, and a fixed tail wheel. The main landing gear consists of dual wheels, equipped with hydraulic brakes, that are attached to the sponsons by retractable oleo shock struts. The main landing gear is equipped with a one shot pneumatic emergency blow-down feature that permits lowering the main landing gear. The main landing gear hydraulic system (figure 1-45) operates on 3000 psi hydraulic pressure from the utility hydraulic system. The system is provided electric power from the dc essential bus and is protected by a circuit breaker marked LAND GEAR on the overhead circuit breaker panel. The tail wheel, located beneath the tail fuselage, is full-swiveling and self-centering and may be locked in the center position. The sponsons are fixed, hollow, outrigger-type floats attached to the fuselage that enable the helicopter to maintain a level, upright position in the water.

LANDING GEAR CONTROL PANEL.

The landing gear control panel (figure 1-44) marked LDG GEAR CONT is located on the copilot's side of the cockpit console. The landing gear actuating lever, the warning light test button marked HDL LT TEST, and the down lock release marked DN LCK REL are located on the panel.

Landing Gear Actuating Lever.

The landing gear actuating lever on the landing gear control panel marked LDG GEAR CONT has a wheel-shaped knob and two marked positions, UP and DN, with directional arrows. The lever is actuated to raise or lower the main landing gear. A red warning light located in the wheel-shaped knob is on when the landing gear is cycling. An electrically actuated downlock solenoid locks the actuating lever in the DN position when the weight of the helicopter is on the landing gear. After becoming airborne, the lock is automatically released which permits the actuating handle to be moved to the UP position. Should the downlock solenoid electrical circuit become inoperative, a mechanical downlock release marked DN LCK REL can be

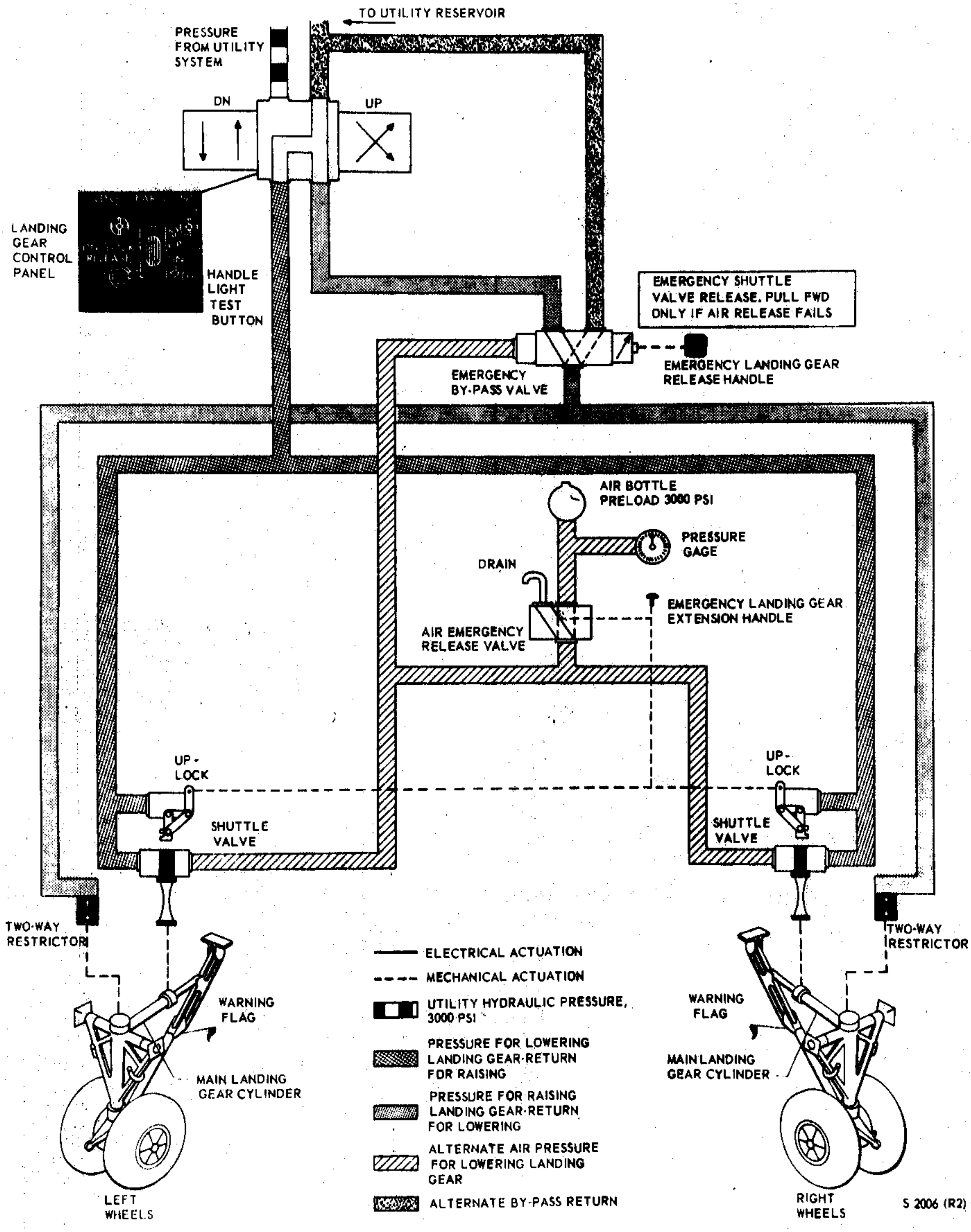

Figure 1-45. Landing Gear Hydraulic System

Figure 1-46. Emergency Landing Gear Extension Handle

actuated to mechanically release the landing gear actuating lever from the DN position. Placing the actuating lever in the UP position opens an electrically operated solenoid valve, allowing hydraulic fluid to pass through a two-way restrictor valve to the main landing gear cylinder, forcing the piston in the cylinder into the up position. The main landing gear, when fully raised, is held in position by a mechanical uplock. A hydraulically operated uplock cylinder or a mechanical release must be actuated to release the mechanical uplock. Placing the actuating lever in the DN position opens an electrically operated solenoid valve, allowing hydraulic fluid to release the mechanical uplock, and to force the piston in the main landing gear cylinder to the down position. A limit switch will be actuated when the main landing gear is fully extended or retracted and the electrically operated solenoid valve will return to the trail position.

Landing Gear Warning Light. The main landing gear warning light is located in the landing gear actuating lever knob. The light in the knob comes on whenever the actuating handle is moved and the landing gear is in transit to the up or down position. The intensity of the landing gear warning light is controlled by the pilots flight instrument lights rheostat on the overhead switch panel. When the rheostat is in the OFF position the light will be illuminated at full strength, but as the rheostat is moved from the OFF position the landing gear warning light will go to the dim position. When the landing gear is locked in either the UP or DOWN position, the light will go out. The warning light operates on electrical power from the dc essential bus and is protected by a circuit breaker marked LAND GEAR, located on the overhead circuit breaker panel.

Landing Gear Warning Light Test Button. A landing gear actuating lever warning light test button HANDLE LT TEST is located on the landing gear control panel on the pilot's compartment console. Pressing the button tests the warning light in the landing gear actuating lever knob. The warning light test button operates on electrical power from the dc essential bus and is protected by a circuit breaker marked WARN LTS TEST, located on the overhead circuit breaker panel.

LANDING GEAR POSITION INDICATORS.

The landing gear position indicators (figure 1-5) are located on the instrument panel. The indicators operate on direct current from the essential bus and are protected by a circuit breaker marked LAND GEAR, located on the overhead circuit breaker panel. The indicators read UP only if the landing gear wheels are in the up and locked position, and show pictures of landing gear wheels only if the wheels are in the down and locked position. During landing gear extension or retraction (whenever the landing gear is neither up and locked nor down and locked) and whenever electric power is not available, the indicators show black and white diagonal lines.

EMERGENCY LANDING GEAR SYSTEM (LANDING GEAR ALTERNATE SYSTEM).

The landing gear emergency system is used to lower the main landing gear pneumatically if the hydraulic system fails. The landing gear emergency system is a pneumatic system that consists of a 50 cubic inch, 3000 psi maximum capacity air bottle, located in the controls enclosure (broom closet), located immediately aft of the pilot, an emergency landing gear extension handle, located on the copilot's side of the cockpit console, and an emergency landing gear release handle, located on the side of the controls enclosure. A gage, visible through a window on the controls enclosure door, indicates the air pressure in the air bottle. An air bottle filler cap is located behind a hinged panel marked ALTERNATE LANDING GEAR RELEASE AIR CHARGING CONNECTION, on the side of the control enclosure.

Emergency Landing Gear Extension Handle (Alternate Uplock Release Handle).

The emergency landing gear extension (figure 1-46) painted with orange-yellow and black diagonal stripes is located on the copilot's side of the cockpit console. The emergency landing gear extension handle is used to lower the main landing gear in case the normal system fails. The handle must be rotated 90 degrees and pulled to withdraw the alternate uplock release pins and to release air into the landing gear emergency system, forcing the landing gear to the down and locked position. The air chamber release valve, displaced by the air charge, actuated by the emergency landing gear release handle, must be manually reset before the landing gear can be retracted or the air bottle recharged. An instruction plate marked EMER LG EXTENSION, TURN THEN PULL is located beside the handle. When using the landing gear emergency system, the warning light, located on the actuating lever, marked LDG GEAR CONT., will not illuminate unless the actuating lever is placed in the down position. The position indicator will function normally as long as electrical power is available to the dc essential bus.

Emergency Landing Gear Release Handle (Alternate Landing Gear Release Handle).

The emergency landing gear release handle (figure 1-47) marked EMERGENCY SHUTTLE VALVE RELEASE, PULL FWD, ONLY IF AIR RELEASE FAILS is located behind the pilot's seat on the broom closet. Pulling the emergency landing gear release handle actuates a two position shuttle valve, allowing hydraulic fluid trapped in the up lines to be vented to the hydraulic system. Gravity will then lower the landing gear to the down position. The emergency landing gear release handle should be pulled only after the emergency landing gear extension handle has been pulled and the landing gear still does not lower. Do not attempt to reset the handle after it has been placed in the forward or emergency position.

TAIL WHEEL LOCK HANDLE.

A handle (figure 1-4), located next to a decal marked "TAIL WHEEL LOCK - PULL TO LOCK" is located on the right side of the cockpit console. Pulling the handle out permits a spring-loaded lockpin to engage at the swivel joint after the tail wheel is centered. Pushing the handle in releases the lock and allows the tail wheel to swivel. A button in the center of the handle must be pressed in to release a ratchet-type lock before the tail wheel can be unlocked. The tail wheel should only be locked for straight takeoffs and landings. During maneuvers on the ground, the tail wheel should be unlocked to reduce strain on the pylon and the possibility of shearing the tail wheel lockpin. If any side loads are imposed on the tail wheel, the lockpin will not disengage.

EMERGENCY FLOTATION GEAR SYSTEM.

The emergency flotation gear system provides the helicopter with stability on the water with the rotor stopped. The system consists of two inflatable bags, four air cylinders, and a control panel. The system operates on 28 volt dc current from the essential bus and is protected by a circuit breaker marked AUX FLOAT GEAR, located on the overhead circuit breaker panel.

EMERGENCY FLOTATION GEAR BAGS.

The emergency flotation gear bags are located on the outboard chine of each sponson stowed in a bungee cord laced canvas enclosure. Each bag is sub-

Figure 1-47. Emergency Landing Gear Release Handle

divided into two chambers having a combined displacement of 35 cubic feet. The chambers are inflated by individual air cylinders. Although each chamber is inflated by a single air cylinder, all chambers will inflate simultaneously upon actuation of the INFLATE switch. The floats are made of neoprene coated nylon and are scuff-resistant for durability during adverse conditions. The two chamber air bags provide a fail-safe function in that one chamber per side inflated will provide considerably improved stability.

EMERGENCY FLOTATION GEAR AIR CYLINDERS.

There are two emergency flotation compressed air cylinders mounted horizontally inside each sponson. Access to these cylinders is through the inspection ports on the top of the sponson. The cylinders are electrically discharged through solenoid operated valves. A pressure gage is also mounted on each valve. The gage is graduated from 0 to 3500 psi in 100 psi increments. The gage is green lined for 2650 to 3000 psi. If the pressure is not within the minimum and maximum limits, service the air cylinders.

EMERGENCY FLOTATION CONTROL PANEL.

The emergency flotation control panel (figure 1-48) marked EMERGENCY FLOTATION is located on the lower right-hand side of the radio console. The panel consists of a rotary selector test switch, indicating light, a two position lever-lock-type switch, and a guarded pushbutton. The rotary selector test switch with marked positions OFF, L1, L2, R1, and R2 provides a means of checking the respective circuit continuity to the air cylinders. When the switch is placed in either L1, L2, R1, or R2 position the green light will illuminate if the circuit is functioning properly. The lever-lock switch has marked positions OFF and ARMED. When the switch is placed

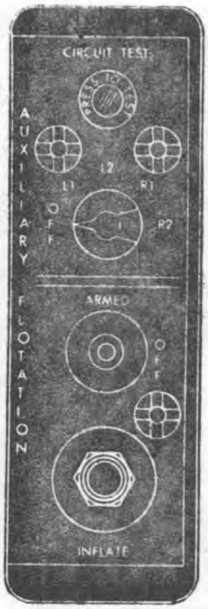

Figure 1-48. Emergency Flotation Gear Control Panel

in the ARMED position, electrical power from the dc essential bus is supplied to the system. When the switch is placed in the OFF position, all electrical power to the inflation portion of the system is removed. The pushbutton type switch is marked INFLATE. When the switch is depressed, all four air cylinders simultaneously discharge air into the flat chambers, if the OFF-ARMED switch is on ARMED. Normally the bags will take approximately 6 to 7 seconds to inflate. After the system has been activated, the bags can only be deflated by maintenance personnel on the ground.

SEA ANCHOR.

The sea anchor provides the helicopter with a bow into the wind attitude while adrift. During light winds 5 to 10 knots, the anchor does not completely bring the bow into the wind but it does considerably reduce the rate of drift. In higher winds, the bow points directly into the wind and the drift rate is effectively reduced. The sea anchor is stowed in a case mounted on the back of the copilot's seat and consists of a spring-loaded 96 inch drogue parachute, a 20 foot nylon rope, and a 10 foot rip cord and spring-loaded hook. All these items are contained in a single package.

WHEEL BRAKE SYSTEM.

The main landing gear wheels are equipped with hydraulic brakes operated by the pilot's toe pedals, located on the pilot's tail rotor pedals. A parking brake handle operates a hydraulic valve to lock the wheel brakes. A light on the advisory panel comes on whenever power is applied to the helicopter, enabling the pilot to see the marking, PARKING BRAKE ON.

TOE BRAKE PEDALS.

The main landing gear wheels are individually braked by depressing the corresponding pilot's toe brake pedals (figure 1-4), mounted above the pilot's tail rotor pedals (figure 1-4). The brakes operate on hydraulic pressure developed by depressing the brake pedals.

PARKING BRAKE HANDLE.

The parking brake handle (figure 1-4) marked PARKING BRAKE is located on the right-hand side of the cockpit console. A decal, located adjacent to the parking brake handle, is marked ON-DEPRESS TOE BRAKE THEN PULL, OFF-DEPRESS TOE BRAKE. The parking brake is applied by first depressing the toe brake pedals and then pulling the parking brake handle out. Pressing the left brake pedal will release the parking brake, causing the parking brake handle to return to the OFF position.

WARNING

The parking brake can be set for one wheel without the other. This could be hazardous during shipboard operations. Insure that both pedals are depressed firmly when setting the parking brake.

EMERGENCY EQUIPMENT.

ENGINE FIRE DETECTOR SYSTEMS.

Two fire detector systems (one for each engine) are installed to warn the pilot of an engine fire. Four continuous-element fire detector cables are located in each engine compartment. They are wired into a closed series loop connected to a control unit that turns on warning lights in the pilot's compartment in the event of a fire. The fire detector systems operate on alternating current from the No. 1 ac generator or inverter.

ENGINE FIRE WARNING LIGHTS AND TEST SWITCH.

Two red engine fire warning lights and test switch (figure 1-5), are located on a plate marked FIRE WARN, on the pilot's side of the instrument panel. The lights are marked NO. 1 ENG and NO. 2 ENG. The switch has two marked positions FIRE TEST and OFF. In addition, four red engine fire warning lights, two for each engine, are installed in the engine fire emergency shutoff selector handles marked FIRE EMER SHUT-OFF SELECTOR NO. 1 ENGINE, NO. 2 ENGINE, located on the overhead switch panel. A light on the instrument panel and a light in either the No. 1 or No. 2 engine fire emergency shutoff selector 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 shutoff selector handle will illuminate. The switch will return to the OFF position when released, and the light will go out.

Thermal Discharge Indicator.

A safety outlet in each engine fire extinguisher container is connected to a red THERMAL DISCHARGE

INDICATOR, located on the lower left side of the fuselage. In the event pressure becomes excessive within the container, a safety outlet opens, the THERMAL DISCHARGE INDICATOR seal is ejected and the container's contents are discharged overboard. On preflight check the pressure of the container in relation to the pressure/temperature chart located on the inside of the inspection plate.

ENGINE FIRE EXTINGUISHING SYSTEM.

A liquid Bromotrifluoromethane (CF_3Br) fire extinguisher system is installed to enable the pilot to extinguish an engine fire in either engine compartment during flight. The liquid is stored under pressure in two fire extinguisher liquid spherical containers (figure 1-49), mounted in the aft section of the transmission compartment. Each spherical container has two valves that contain a disc which, when broken by an explosive cartridge actuated by the engine fire extinguisher switch, empties its contents into the pre-selected engine compartment. Choice of engine compartments and spherical containers is made by pulling one of the engine fire emergency selector handles. Tubing extends from one valve on each container to the No. 1 engine compartment and from the other valve on each container to the No. 2 engine compartment. Within each engine compartment the tubing divides into four nozzles which extend along the inboard side of the engine. The extinguishing liquid, when released through the nozzles turns into a vapor that smothers the fire. The spherical containers are equipped with a pressure gage and a thermal discharge valve which will discharge overboard outside of the helicopter if the temperature of the sphere reach 93.3° to $104.4^{\circ}C$ (208° to $220^{\circ}F$). A thermal discharge indicator is located on the outside of the fuselage to the rear of the left cabin window. When the spheres are properly charged, the pressure gages should indicate the value within the range shown on the decal adjacent to the gages. The engine fire extinguishing system operates on direct current from the essential bus. Although designed primarily for combating an engine fire during flight, the fire extinguishing system may be used on the ground if other fire fighting equipment is ineffectual or not available. Be sure all ground personnel are clear before using the system.

WARNING

CF_3Br is very volatile and is not easily detected by odor. It is nontoxic and can be considered to be about the same as other freons and carbon dioxide, causing danger primarily by reduction of oxygen. The liquid should not be allowed to contact the skin, as it may cause frostbite or low temperature burns because of its low boiling point.

Engine Fire Emergency Shutoff Selector Handles (Engine Tee Handles).

The T-shaped handles marked FIRE EMER SHUT-OFF SELECTOR are located on the overhead switch panel (figure 1-6). 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, ac power from the essential bus actuates the fuel shutoff valve which closes the fuel lines to the respective engine and 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.

WARNING

After actuation of the first bottle, check and reset the circuit breaker, if necessary, prior to releasing the reserve bottle.

An engine fire extinguisher switch marked FIRE EXT, located on the overhead switch panel (figure 1-6) in the pilot's compartment, has three marked positions, RESERVE, OFF, and MAIN. The guarded switch is operative only after one of the fire emergency shutoff selector handles has been pulled. When the engine fire extinguisher switch is held in the MAIN position, after the fire emergency shutoff 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 held in the RESERVE position, after fire emergency shutoff selector handle has been pulled and the switch returned from the MAIN position, the contents of the opposite fire extinguisher sphere are discharged into the selected engine compartment. Pulling both engine fire emergency selector handles, and placing the fire extinguisher switch in MAIN, discharges the contents of each fire extinguisher's sphere into the corresponding engine compartments. When this occurs there is no reserve of fire extinguishing fluid. The switch will return to the OFF position when released.

PORTABLE FIRE EXTINGUISHER.

A portable fire extinguisher (figure 1-3) is located at the entrance to the pilot's compartment. The CO_2 fog-type extinguisher is held in place by a bracket with a tight-fitting, quick release, spring steel clamp. When using the extinguisher, the nozzle must be held in close proximity to the source of the fire as the charge has a short duration of approximately 30 seconds.

FIRST AID KITS.

One first aid kit (figure 1-3) is mounted in the pilot's compartment on the control enclosure (broom closet). A second first aid kit is installed in the cabin.

PYROTECHNIC PISTOL.

A pyrotechnic pistol and twelve signal cartridges are stowed in a buoyant case (figure 1-3), located on the right-hand side of the personnel compartment entrance. The case is opened by releasing the catch at the top. The case may be removed from the helicopter by grasping the handle and pushing upward, then forward. Be sure the lid is closed and latched.

SIX/SEVEN MAN LIFE RAFT.

Life rafts may be secured to the tiedown rings in the cabin floor.

EMERGENCY EXITS.

For emergency routes of escape and exits, see figure 3-8.

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 windows 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.

PERSONNEL DOOR EMERGENCY EXIT.

The lower door of the personnel door is normally opened and closed by the handle marked OPEN and LOCKED with direction arrows. The upper door of the personnel door is normally opened and closed by the handle marked OPEN and CLOSED with a direction arrow pointing to open. The upper door can be jettisoned by pulling down on the yellow knobbed handle marked EMERGENCY RELEASE, TURN with direction arrow at lower aft corner of the upper door, and pulling lower end of the support tube from the fitting on the forward portion of the fuselage, and pushing outward. The panel also may be released from the outside by turning the handle marked EXIT RELEASE, TURN with a direction arrow, located to the bottom and right of the window. The lower half cannot be jettisoned.

CABIN DOOR.

The cabin door is normally opened and closed by the handle marked OPEN with a direction arrow. The handle also friction locks the door in any open position to facilitate inflight operations that require the door to be open. The cabin door window can be removed to provide an additional emergency exit by actuating the emergency release handle, marked EMERGENCY RELEASE PULL with a direction arrow below the cabin door window. Around the window is a solid yellow line marked CUT FOR EMERGENCY RESCUE.

CABIN WINDOWS.

The two cabin windows, one on each side of the cabin may be pushed out for emergency exit by sharply striking the corner of the window. Each window is marked EMERGENCY EXIT WINDOW PUSH OUT. Cabin windows are not suitable as bail out windows.

PILOT'S, COPILOT'S, AND FLIGHT MECHANIC'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 (PK-2). Both seats have a 5-inch range of height adjustment, a 6 1/2 inch forward and aft adjustment, and are equipped with safety belts and shoulder harnesses. The seats are also equipped with cushions that are interchangeable with the pararaft and parachute.

SEAT HEIGHT ADJUSTMENT LEVER.

The seat height adjustment levers (figure 1-4) are the rear levers at the right side of the pilot's and copilot's seat. The lever is pulled up to release the height adjustment seat lockpins. The seats aided by spring-loaded bungees can then be adjusted for height by varying the weight upon them. The lockpins will automatically engage in any of 12 positions.

SEAT FORWARD AND AFT ADJUSTMENT LEVER.

The seat forward and aft adjustment levers (figure 1-4) are the front levers on the right side of the pilot's and copilot's seat. The lever is pulled up to release the forward and aft seat adjustment lockpins. The lever must be held up while the seat is moved on the tracks forward or aft as desired. The lockpins will automatically engage in any of eight positions when the lever is released.

SHOULDER HARNESS LOCK LEVER.

A two position shoulder harness inertia reel lock lever is located on the left side of each seat. When the lever is in the unlocked (aft) position, the shoulder harness 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

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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.

FLIGHT MECHANIC'S SEAT.

The flight mechanic's seat is installed on the ASE control compartment (broom closet) wall, behind the pilot's. The seat is removable but is normally folded against the ASE compartment wall. When the seat is in use it extends in front of the pilot's compartment entrance. The back and bottom of the seat is made of canvas supported by metal tubing. A safety belt is attached to the seat.

AUXILIARY EQUIPMENT.

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

Heating System

Anti-Icing Systems

Communication and Associated Electrical Equipment

Lighting Equipment

Cabin (Cargo Compartment)

Rescue Hoist

External Cargo Sling

Troop Carrying Equipment

Miscellaneous Equipment

Windshield Wiper System

Windshield Washer System

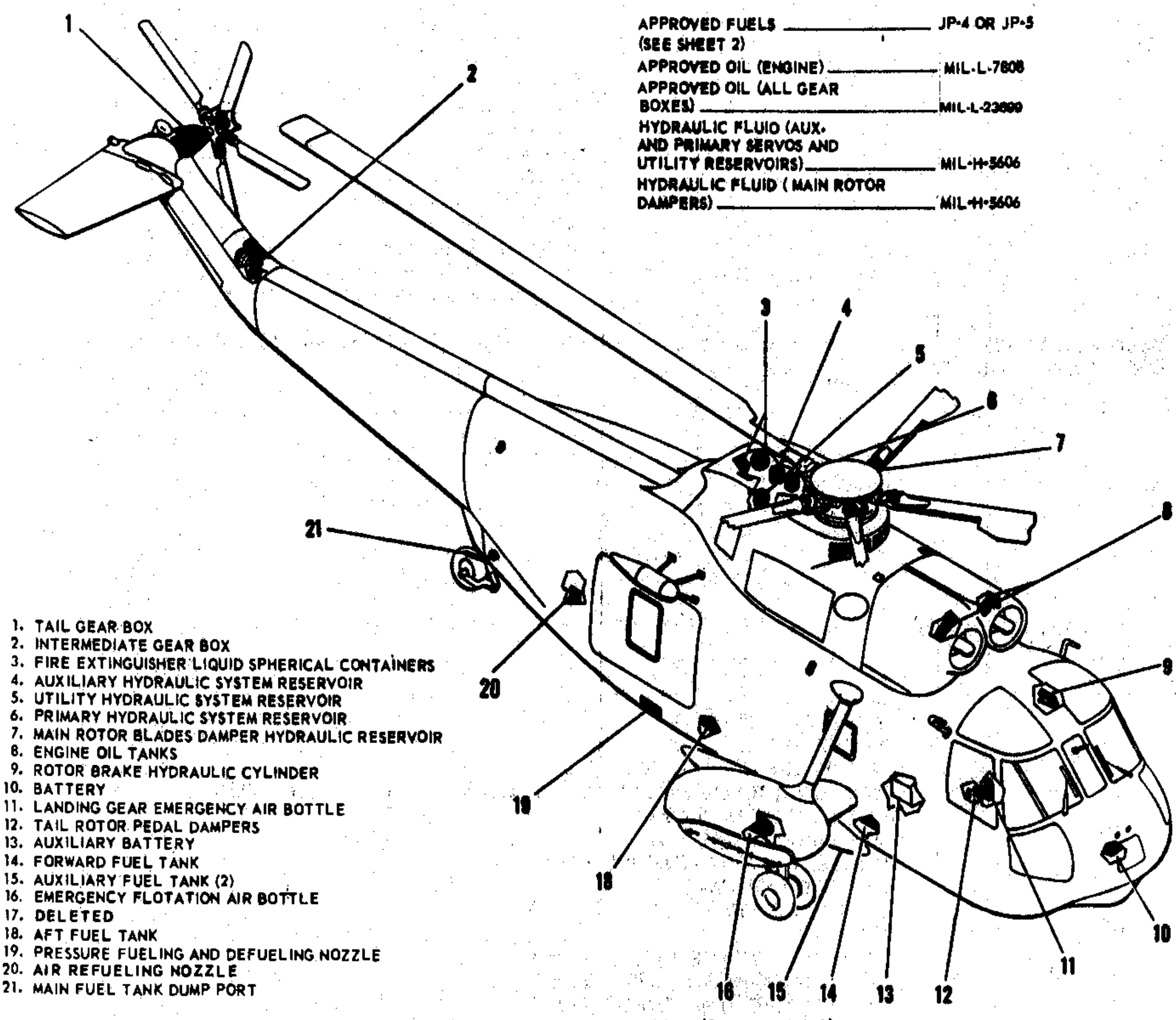

Figure 1-49. Servicing Diagram (Sheet 1 of 4)

SERVICING DIAGRAM

FUELS

SPECIFICATION FUEL JP-4, MIL-J-5624

FUEL TYPE	MILITARY SPECIFICATION	NATO SYMBOL	COMPARABLE COMMERCIAL DESIGNATION
RECOMMENDED FUELS			
Wide Cut Gasoline	MIL-J-5624 JP-4	F-40	JET B
Kerosene	MIL-J-5624 JP-5	F-44 F-42	JET A-1

THE RECOMMENDED FUEL FOR THIS ENGINE IS JP-4 OR JP-5. THE FOLLOWING LISTING GIVES THOSE MILITARY NATO F-40 FUELS WHICH ARE INTERCHANGEABLE WITH JP-4. TURBINE ENGINE FUELS, WIDE CUT GASOLINE TYPE, NATO F-40.

UNITED STATES	MIL-J-5624E	JP-4
GREECE	MIL-J-5624E	JP-4
NETHERLANDS	MIL-J-5624E	JP-4
PORTUGAL	MIL-J-5624E	JP-4
TURKEY	MIL-J-5624E	JP-4
BELGIUM	BA-PT-2	
CANADA	3-GP-22D	
FRANCE	AIR 3407A	
GERMANY	VTL-9130-006 AM 1	
ITALY	AM-C-142F	
NORWAY	D. ENG. R.D. 2486 ISS. 3 AM. W	
UNITED KINGDOM	D. ENG. R.D. 2486 ISS. 3 AM. W	
DENMARK	D. ENG. R.D. 2486 ISS. 3 AM. W	

COMMERCIAL JET FUEL - CONFORM TO ASTM TYPE B, JET B. THESE FUELS LIMIT STARTING TO -65°F. NATO F-40, SPECIFICATION MIL-J-5624, GRADE JP-4.

COMMERCIAL DESIGNATION

CALTEX JET B, CALIF TEXAS OIL CO.
CITIES SERVICE TYPE B, CITIES SERV OIL CO.
ESSO TURBINE FUEL 4, HUMBLE OIL AND REFINERY
PUREJET TYPE B, THE PURE OIL CO.
ATF-4, STANDARD VACUUM OIL CO.
TEXACO AVJET JP-4 TYPE B

COMMERCIAL JET FUELS - CONFORM TO ASTM TYPE A-1, JET A-1. THESE FUELS LIMIT STARTING TO -25°F. NATO F-34.

ARCOJET-1
AMERICAN TYPE A-1
CALTEX JET A-1
440 UNIVERSAL TURBINE FUEL
CHEVRON AVIATION TURBINE FUEL NO. 1
GULFLITE TURBINE FUEL A
ESSO TURBO FUEL 1-A
KEROSENE - AVIATION TYPE
PUREJET TYPE A-1
AEROSHELL 650
AVTUR 50
AVIATION TURBINE FUEL TYPE A
AVIATION TURBINE FUEL TYPE 1
CHEVRON TURBINE FUEL TYPE 1

Figure 1-49. Servicing Diagram (Sheet 2 of 4)

AFT-1A
407 AVJET K-58

COMMERCIAL JET FUELS - CONFORM TO ASTM TYPE A, JET A, THESE FUELS LIMIT STARTING TO -20°F, NATO F-30.

ARCOJET-A-1
AMERICAN TYPE A
CALTEX JET A
CITIES SERVICE TYPE A
CONOCO JET 50
PHILLIPS KEROSENE, GRADE TF
PUREJET A
RICHFIELD TURBINE FUEL A
SINCLAIR SUPERJET FUEL
AEROSHELL 640
AVTUR 40
406 AVJET K-40

REFER TO NATO INTERCHANGEABILITY TABLES, T.O. 42B1-1-15, FOR NATO NATIONAL SPECIFICATIONS FOR ALTERNATE FUELS IF THE NEED REQUIRES FUEL FROM NATO COUNTRIES.

NOTE

A FUEL CONTROL ADJUSTMENT IS REQUIRED WHEN FUEL TYPE IS CHANGED.

ALTERNATE FUELS

WHEN JP-4 OR JP-5 IS NOT AVAILABLE, FUELS LISTED IN T.O. 42B1-1-14 MAY BE USED AS ALTERNATES. THE GROUPS ARE LISTED IN ORDER OF PREFERENCE.

NOTE

MIXING OF FUELS IS NOT RECOMMENDED BECAUSE OF PROBLEMS ENCOUNTERED WITH FUEL CONTROL SETTINGS. (Refer to MIXED FUELS VS. TYPE OF START TO EXPECT CHART, SECTION VII).

NOTE

MANUAL ADJUSTMENT OF FUEL CONTROLS MAY BE NECESSARY WHEN USING CERTAIN ALTERNATE FUELS TO AVOID EXCEEDING RPM & T_5 OPERATING LIMITS.

JP-5 JET FUEL - THESE FUELS LIMIT STARTING TO -55° F, NATO F-44, SPECIFICATION MIL-J-5624. WHEN JP-5 FUEL IS USED, THE FUEL DENSITY ADJUSTMENT ON THE FUEL CONTROL AND FLOW DIVIDER WILL BE CHANGED TO THE JP-5 POSITION, SEE T.O. 2J-T58-2. THE FOLLOWING LISTING GIVES THOSE MILITARY NATO F-44 FUELS WHICH ARE INTERCHANGEABLE WITH JP-5.

CANADA	3-GP-24C
GERMANY	VTL-9130-007 AM. 1 & 9130-010 A
ENGLAND	D. ENG. R.D. 2496 ISS. 1 AM. 1 A

WHEN CHANGING FROM JP-5 TO JP-4 FUEL AND WITH THE FUEL CONTROL AND FLOW DIVIDER NOT PROPERLY ADJUSTED FOR THE CHANGE IN FUEL, HOT STARTS WILL OCCUR. HOWEVER, WHEN CHANGING FROM JP-4 TO JP-5 FUEL UNDER THE SAME CONDITIONS, COLD HANGUPS OR SLOW ACCELERATION WILL BE ENCOUNTERED.

COMMERCIAL JET FUEL - CONFORM TO ASTM TYPE B, JET B. THESE FUELS LIMIT STARTING TO -65°F. NATO F-40, SPECIFICATION MIL-J-5624, GRADE JP-4 AND JP-5, JET A-1.

NOTE

FOR AN ADDITIONAL LISTING OF APPROVED MILITARY AND COMMERCIAL FUELS REFER TO T.O. 42B1-1-14.

OILS

APPROVED OIL (ENGINE)	MIL-L-7808
APPROVED OIL (ALL GEAR BOXES)	MIL-L-23699

Figure 1-49. Servicing Diagram (Sheet 3 of 4)

HYDRAULIC FLUID (AUX. AND PRIMARY SERVOS AND UTILITY RESERVOIRS)	MIL-H-5606
HYDRAULIC FLUID (ROTARY WIND DAMPERS)	MIL-H-5606

EXTERNAL ELECTRICAL POWER REQUIREMENTS

DC 28 VOLTS, 750 AMPS CONTINUOUS, 1000 AMPS INTERMITTENT.

AC 115/200 VOLT 3 PHASE, A, B, C, PHASE ROTATION, 400 CPS, A STANDARD SQUARE 6 PIN PLUG (WYE CONNECTION), WITH A MINIMUM CAPACITY OF 30 KVA.

NOTE

THE MINIMUM RATED EXTERNAL POWER CART THAT MAY BE USED FOR STARTING IS A 115 A.C. VOLT 3 PHASE 400 CPS.

Figure 1-49. Servicing Diagram (Sheet 4 of 4)