T.O. 1H-3(C)E-1

FLIGHT MANUAL

USAF SERIES

CH-3E AND HH-3E

HELICOPTERS

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1 SEPTEMBER 1983

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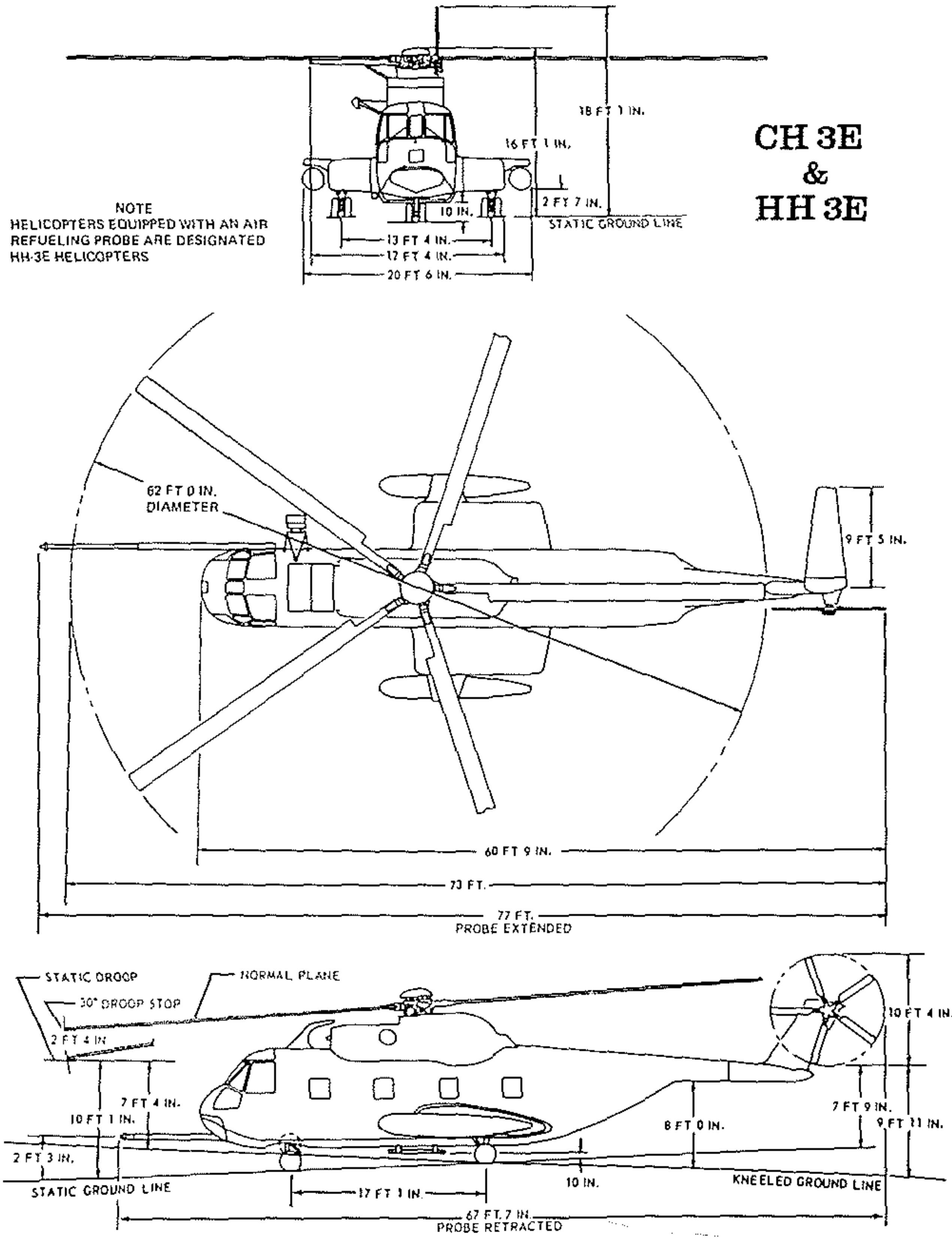

Figure 1-1. Three View and Dimensions

CODE	SERIAL NUMBERS MODEL CH-3E HELICOPTERS
14	65-5697 THRU 65-5700
15	65-12788 THRU 65-12800
16	66-13285
17	66-13291 THRU 66-13293 AND 66-13296
18	67-14703
19	67-14705
20	67-14707
21	67-14718 THRU 67-14720
22	68-8282
23	64-14230 AND 64-14232

CODE	SERIAL NUMBERS MODEL HH-3E HELICOPTERS
24	65-12777, 65-12780, 65-12781, 65-12783, 65-12784 AND 65-12787
25	66-13284
26	66-13286
27	66-13290
28	67-14704
29	67-14706
30	67-14708, 67-14709 AND 67-14711 THRU 67-14717
31	67-14722 THRU 67-14725
32	69-5798 THRU 69-5812

GLOSSARY OF TERMS AND ABBREVIATIONS

AC — Alternating current

ACCELERATION — The rate of change of velocity

ADF — Automatic direction finder

AFCS — Automatic flight control system

AIRSPEED

KCAS — Knots calibrated airspeed

KIAS — Knots indicated airspeed

KTAS — Knots true airspeed

ALT — Altitude

APU — Auxiliary power unit

BAR ALT — Barometric altitude control

BDHI — Bearing distance heading indicator

BLADE TIP STALL — Beginning of blade stall. Occurs at tip of retreating blade due to its high angle of attack and low forward velocity.

BLADE STALL — A stall that begins at the tip of the blade and works progressively inboard as the conditions which cause it increase in severity.

FULL BLADE STALL — Blade stall that is allowed to fully develop causing loss of control and an upward left pitch of the helicopter.

INCIPIENT BLADE STALL — Blade tip stall

BOTTOMING — The engine is considered as bottoming during deceleration whenever a minimum fuel flow to compression-discharge pressure condition is attained.

BUOYANCY — The upward force exerted by water on a floating or immersed body by a fluid.

°C — Degrees Centigrade

CAS — Calibrated airspeed

CDI — Course deviation indicator

CENTER OF GRAVITY (CG) — The center of gravity is the point about which a helicopter would balance if suspended.

CG — Center of gravity

COLLECTIVE — The increasing or decreasing of pitch on all the main rotor blades simultaneously. Also short for collective lever.

CYCLIC — The changing of pitch of each main rotor blade individually as it makes a complete rotation or cycle. Also short for cyclic stick.

DC — Direct current

DG — Directional gyro

GLOSSARY OF TERMS AND ABBREVIATIONS (Cont)

DRAFT — The depth of water the helicopter draws or requires to float.

DRAG DIVERGENCY — Beginning of blade tip stall.

DROOP — Characteristic built into speed control for speed stability and load sharing. When in the governing range steady state N_f will decrease in proportion to engine load at a fixed N_f setting. On this installation the droop is 8.5% N_f from no load to full load conditions.

DECAY — Loss of N_r beyond droop, resulting from a power requirement in excess of power available.

EXCESS BUOYANCY — Buoyancy in excess of that required to float.

°F — Degrees Fahrenheit

FOD — Foreign object damage

FPM — Feet per minute

FT — Feet

FT/MIN — Feet per minute

GAL — Gallons

GCA — Ground-controlled approach

GSI — Glide slope indicator

GW — Gross weight

HR — Hour

HYDROSTATIC ROLL ANGLE — Angle of roll when helicopter is on water.

H-V — Height velocity

IAS — Indicated airspeed

IBIS — Inflight blade inspection system

IGE — In ground effect

IN — Inches

INV — Inverter

KTS — Knots

KVA — Kilovolt-amperes

LAT — Latitude

LB — Pound(s)

LB/GAL — Pound per gallon

LB/HR — Pound per hour

LOAD FACTOR — A factor representing the ratio of weight or pressure of a specified load or force to a standard weight or pressure. The load factor may represent the ratio of the total weight of the helicopter to a weight or pressure imposed by aerodynamic forces, inertia forces, or ground effect.

MAG — Magnetic slaved compass

MAX — Maximum

MEAN WATERLINE — The mean of the highest and lowest waterline for a given set of conditions, gross weight, sea state, etc.

MIN — Minutes/Minimum

MSL — Mean sea level

N_f — Power turbine speed (rpm)

N_g — Gas generator speed

N_r — Rotor speed (rpm)

OAT — Free air, ambient or outside air temperature.

OGE — Out of ground effect (for the CH-3E, and HH-3E, this means hovering approximately 50 feet wheel height or higher)

P_2 — Compressor inlet total pressure

GLOSSARY OF TERMS AND ABBREVIATIONS (Cont)

P_3 — Compressor discharge pressure

PRESS — Pressure

PSI — Pounds per square inch

Q — Torque

R/C — Rate of climb

R/D — Rate of descent

RIGHTING MOMENT — A moment that tends to restore the helicopter to a previous position after an angular displacement on water about one of its axes.

RPM — Revolutions per minute

SEA STATE — Condition of water surface in terms of wind, wave height, wave length, etc.

SERVICE CEILING — Maximum altitude at which a rate of climb 100 FPM can be maintained.

SL — Sea level

STD DAY — Standard day atmospheric conditions

T_2 — Compressor inlet air temperature

OAT may be used in place of T_2 in this manual as T_2 is not indicated in the cockpit.

T_5 — Power turbine inlet temperature

TAS — True airspeed

TEMP — Temperature

TOLD — Takeoff and landing data

TOPPING — A procedure for adjusting engine fuel control to achieve engine performance at maximum operating limits.

TORQUE — Turning force or moment.

TORQUE POWER INDICATION — An indication of power input being delivered to the gear box by the engines.

TRIM ANGLE — The angle at which the helicopter's hull rests on the water.

UTI — Utility

VA — Volt amperes

VAC — Volts alternating current

WATERLINE — The line of intersection between the surface of the water and the side of the helicopter hull when the helicopter is afloat.

WAVE LENGTH — The distance between two successive wave crests.

W_f — Fuel flow

W_f/P_3 — Ratio of weight of fuel flow to be burned to compressor discharge pressure or amount of air available for combustion and cooling.

WL — Water line

XMFR RECT — Transformer rectifier

$\frac{1}{\sqrt{\sigma}}$ — The reciprocal of the square root of density ratio, at the appropriate density altitude. The greek letter sigma (σ) is used to represent the density ratio.

SECTION I

DESCRIPTION

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THE HELICOPTER.

The model CH-3E and HH-3E helicopters are manufactured by Sikorsky Aircraft, Division of United Aircraft Corporation, Stratford, Connecticut. All model helicopters are designed for general purpose operations, and may be equipped for transport of cargo, personnel and litter patients, and air rescue and retrieval of aerial targets. HH-3E and some CH-3E helicopters are configured for search and rescue and combat aircrew recovery. Some helicopters are equipped with titanium armor plate which is installed for protection of the pilot, copilot, crewman, and vulnerable components from small arms fire. Some helicopters are also equipped with three M60 machine guns. General configuration is a single main rotor, twin turbine powered helicopter with amphibious capabilities. The fuselage is all metal, semimonocoque type construction, and is composed of the cockpit, the upper fuselage, the aft fuselage, the pylon, and the lower fuselage. The upper fuselage section contains the cargo compartment, the engine compartment, and the transmission compartment. The cargo compartment may be entered through the personnel door on the right side of the fuselage or through the ramp. Cargo compartment dimensions are 26 feet 2-1/2 inches long, of which 6 feet is ramp area, 6 feet 6 inches wide, and 6 feet high. The cargo compartment is capable of carrying 25 fully equipped troops or 15 litter patients with two attendants. The cargo compartment is also equipped with tiedown rings and skids for transport of cargo. Two gas turbine engines are mounted side by side in the engine compartment which is located above the forward portion of the cargo compartment. The engine shafts extend aft into the main gear box which is located in the transmission compartment. The main rotor assembly, to which the five rotor blades are attached, is splined to the main gear box drive shaft. An auxiliary power unit, used for checkout of equipment, cargo loading, and engine starting, is located aft of the main gear box. The aft fuselage extends from the cargo compartment to the pylon. The lower fuselage contains the electronics-radio compartment in the forward section, the retractable nose gear, and two dual cell fuel tanks. Sponsons are mounted on each side of the lower fuselage. The retractable main landing gear is mounted in the sponsons. The pylon is attached to the rear of the aft fuselage. A horizontal stabilizer is mounted on the upper right-hand side of the pylon. The intermediate gear box is installed in the lower portion of the pylon with a shaft extending upward to the tail rotor gear box at the top of the pylon. The five-bladed tail rotor is splined to the tail rotor gear box. Familiarity with the configuration of the helicopter may be obtained by referring to the exterior and interior general arrangement illustrations at the beginning of this section, and the minimum turning radius and ground clearances diagram (figure 2-4) in Section II.

FOREIGN OBJECT DEFLECTOR.

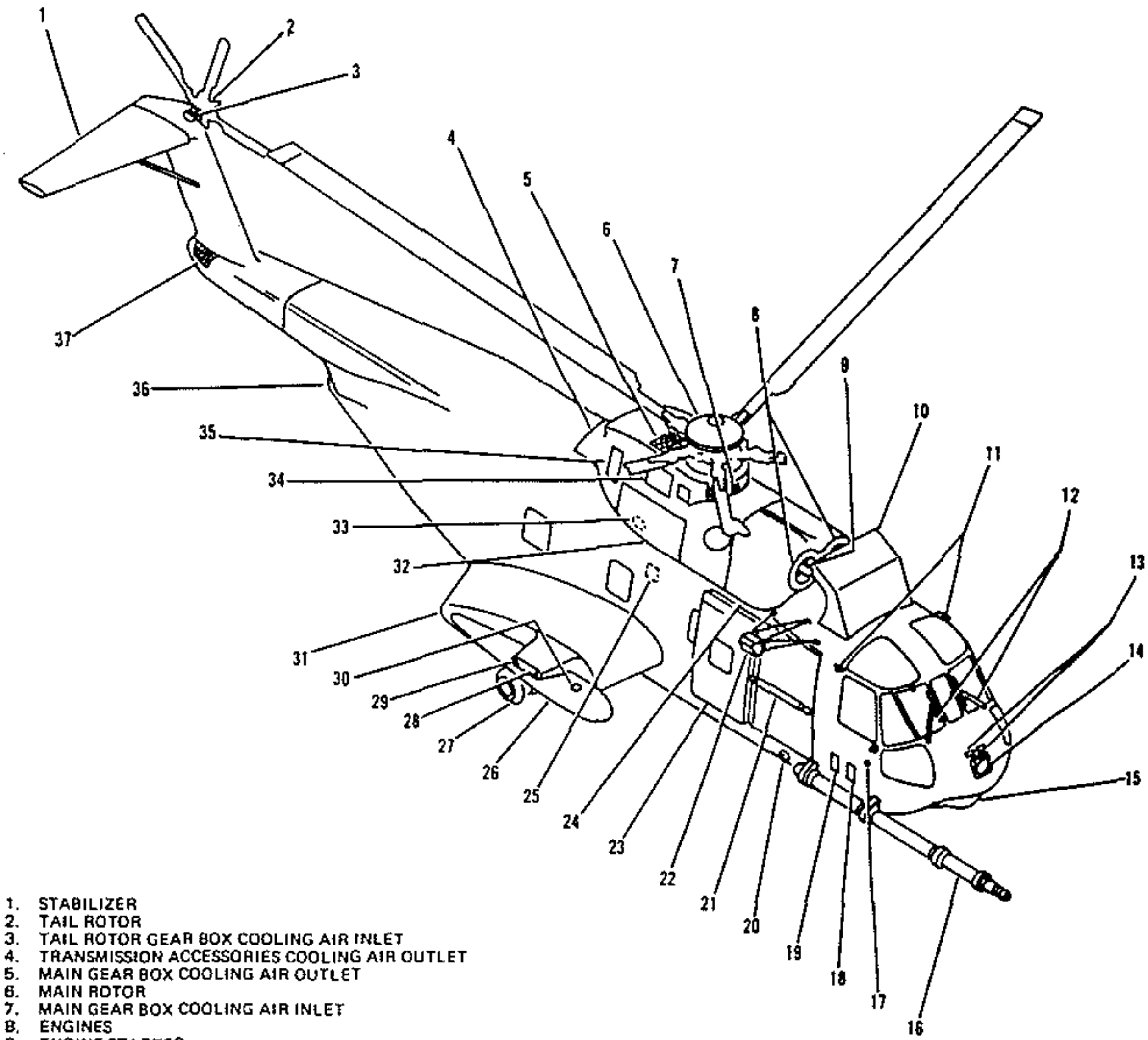
The purpose of the deflector is to inhibit ice or debris from entering the engines. A slight loss of power will result from its use.

DIMENSIONS.

Length.	
Maximum, main rotor blades extended	73 feet 0 inches
Minimum, main rotor blades removed	60 feet 9 inches
Height.	
Maximum to top of tail rotor, blade vertical	
Static	18 feet 1 inch
Kneeled	20 feet 3 inches
Minimum, tail rotor blades removed	16 feet 1 inch
Width.	
Minimum, main rotor blades removed	17 feet 4 inches
Main rotor diameter	62 feet 0 inches
Tail rotor diameter	10 feet 4 inches
Minimum Main Rotor Ground Clearance.	
(Tip clearance - forward sector)	
Static	10 feet 1 inch
Kneeled	7 feet 4 inches
Tail Rotor Ground Clearance.	
Static	7 feet 9 inches
Kneeled	9 feet 11 inches
Tail Pylon Ground Clearance.	
Static	6 feet 5 inches
Kneeled	8 feet 0 inches
Main landing gear tread.	13 feet 4 inches

ENGINES.

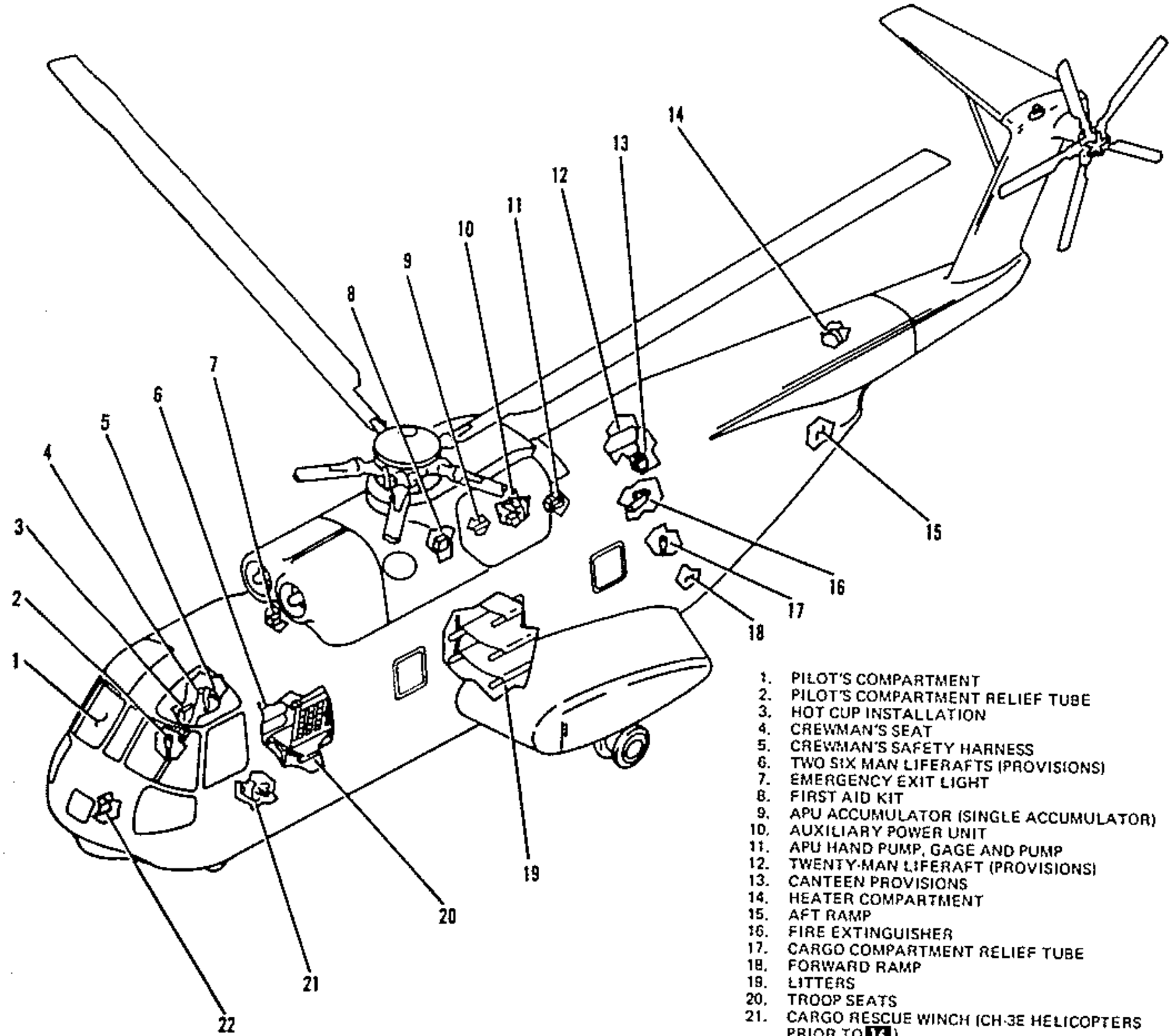
The CH-3E and HH-3E helicopters are powered by two General Electric T58-GE-5 (CT58-140-1) engines. The engines (figure 1-6) are the axial flow gas turbine turboshaft type which incorporate the free power turbine principle. The T58-GE-5 engine develops 1500 shaft horsepower. The engines are located side-by-side above the cargo compartment, forward of the main gear box. Each engine contains the following major components: an axial-flow compressor, combustion chambers, a two-stage gas generator turbine, and a single-stage power turbine that is independent of the gas generator turbine. The gas generator consists of the compressor, annular combustor, and two-stage gas generator turbine. The free turbine principle provides a constant free turbine speed output which results in a constant rotor rpm. Variations in power requirements, to maintain constant free turbine speed, are accomplished by automatic increases or decreases in gas generator speed. A hydromechanical fuel metering unit provides maximum engine performance without exceeding safe engine operating limits. In the normal operating range, power turbine speed is selected by positioning the throttle. The integrated fuel control system delivers atomized fuel in controlled amounts to the combustion chamber. Flow of fuel and air through the combustion chamber is continuous and once the mixture



1. STABILIZER
2. TAIL ROTOR
3. TAIL ROTOR GEAR BOX COOLING AIR INLET
4. TRANSMISSION ACCESSORIES COOLING AIR OUTLET
5. MAIN GEAR BOX COOLING AIR OUTLET
6. MAIN ROTOR
7. MAIN GEAR BOX COOLING AIR INLET
8. ENGINES
9. ENGINE STARTER
10. FOREIGN OBJECT DEFLECTOR
11. PITOT TUBES
12. WINDSHIELD WIPERS
13. BATTERY VENTS
*14. CONTROLLABLE SPOTLIGHT AND BATTERY ACCESS DOOR
15. ELECTRONICS COMPARTMENT ACCESS DOOR
*16. AIR REFUELING PROBE
*17. TRUE AIR SPEED TEMPERATURE SENSOR
18. DC EXTERNAL POWER RECEPTACLE
19. AC EXTERNAL POWER RECEPTACLE
*20. PRESSURE REFUELING ADAPTER
21. PERSONNEL DOOR SAFETY STRAP
22. RESCUE HOIST
23. PERSONNEL DOOR
24. ENGINE ACCESS AND SERVICE PLATFORM
25. INTERNAL AUXILIARY FUEL TANK FILLER CAP
*26 EXTERNAL AUXILIARY FUEL TANK
27. MAIN LANDING GEAR
*28. SPONSON PYLON
*29. SPONSON POSITION LIGHT
*30. EXTERNAL AUXILIARY FUEL TANK FILLER CAP
31. SPONSON
32. TRANSMISSION ACCESS AND SERVICE PLATFORM
33. ALTERNATE LANDING GEAR SYSTEM AIR BOTTLE
34. ACCESS PANEL
35. OIL COOLER ACCESS DOOR
36. RAMP
37. INTERMEDIATE GEAR BOX COOLING AIR OUTLET

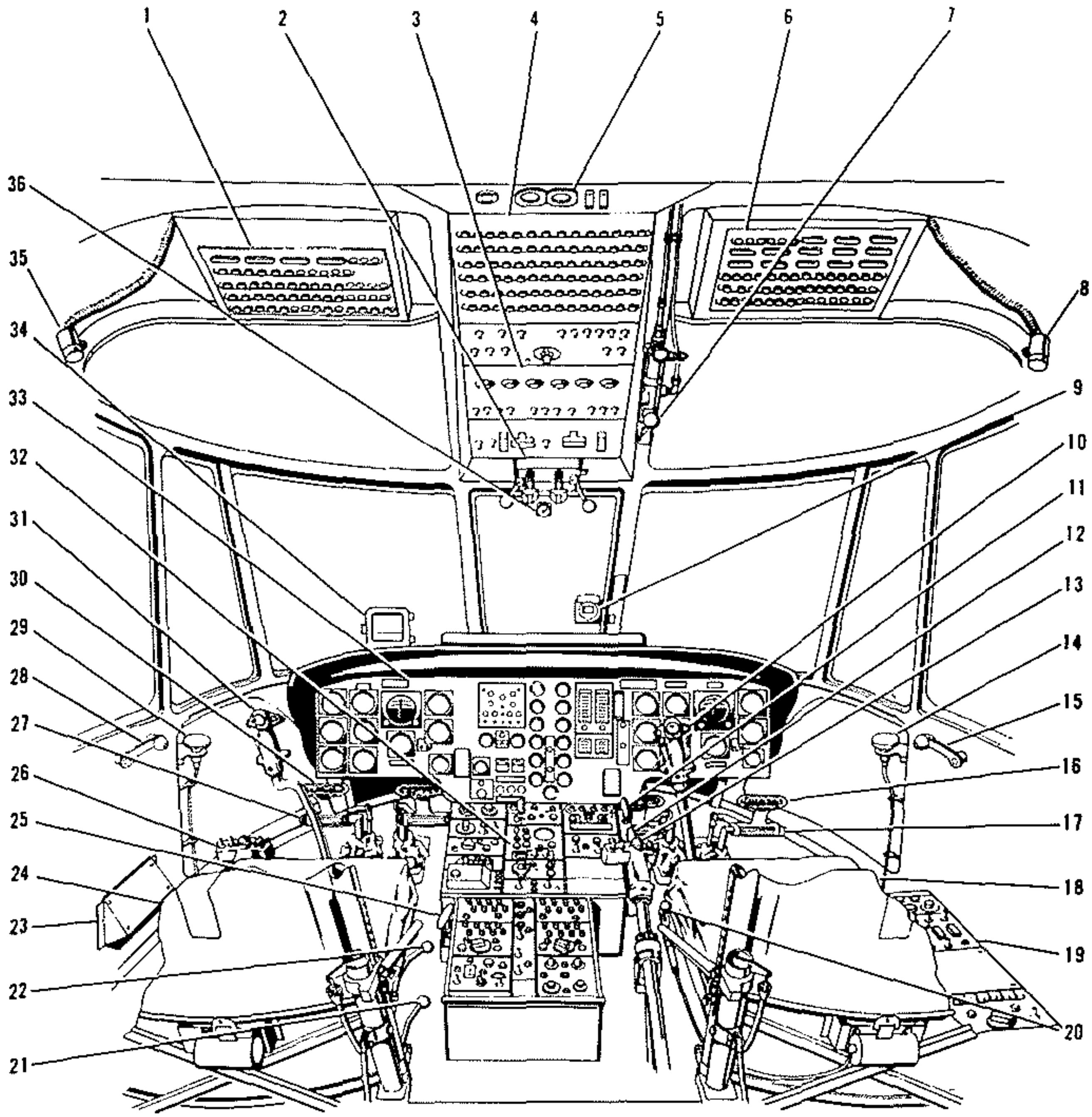
NOTE
ITEMS MARKED * DENOTE CH-3E HELICOPTERS
16 ▶ OR HH-3E HELICOPTERS

Figure 1-2. General Arrangement Exterior Diagram (Typical)



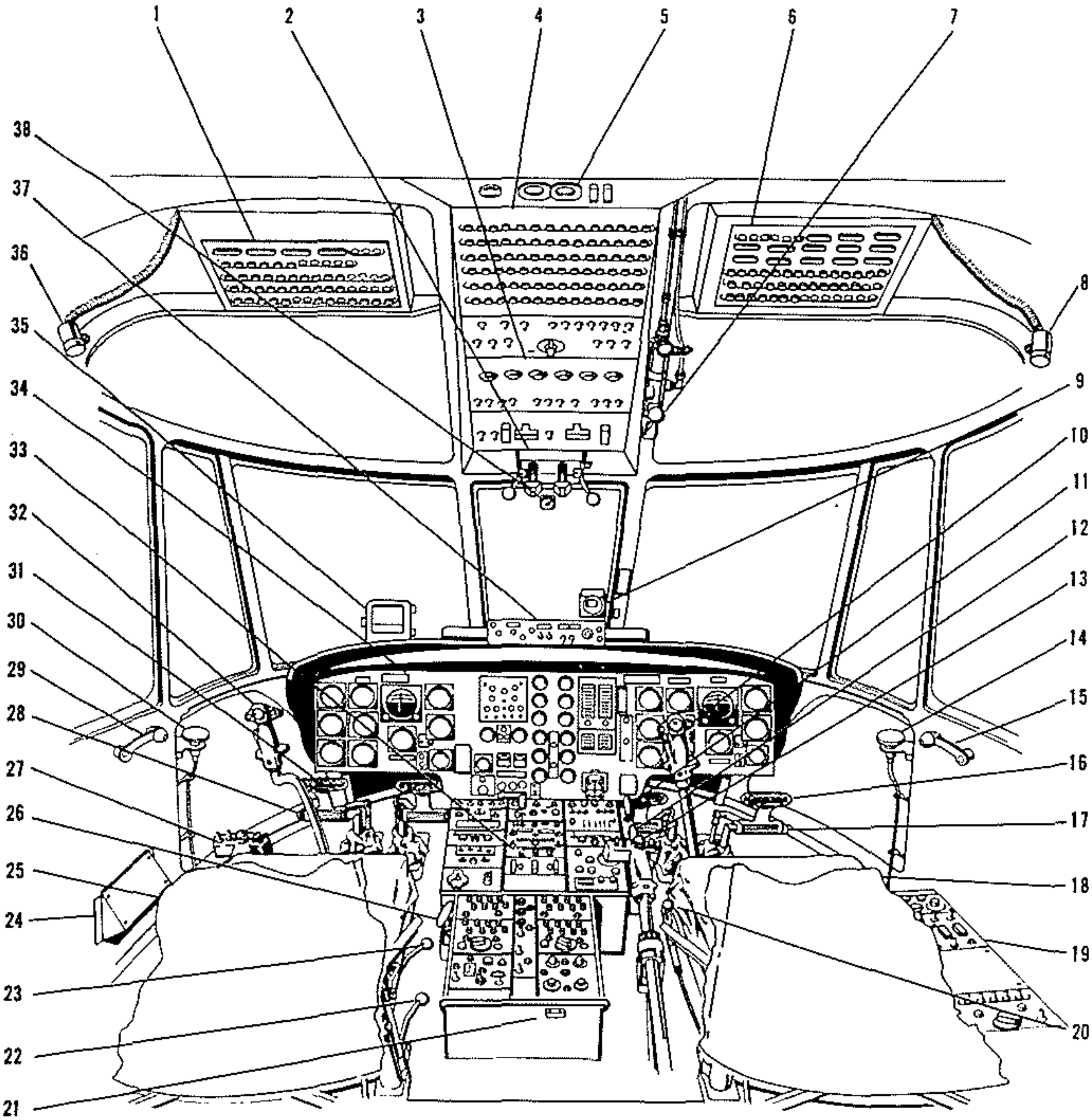
1. PILOT'S COMPARTMENT
2. PILOT'S COMPARTMENT RELIEF TUBE
3. HOT CUP INSTALLATION
4. CREWMAN'S SEAT
5. CREWMAN'S SAFETY HARNESS
6. TWO SIX MAN LIFERAFTS (PROVISIONS)
7. EMERGENCY EXIT LIGHT
8. FIRST AID KIT
9. APU ACCUMULATOR (SINGLE ACCUMULATOR)
10. AUXILIARY POWER UNIT
11. APU HAND PUMP, GAGE AND PUMP
12. TWENTY-MAN LIFERAFT (PROVISIONS)
13. CANTEEN PROVISIONS
14. HEATER COMPARTMENT
15. AFT RAMP
16. FIRE EXTINGUISHER
17. CARGO COMPARTMENT RELIEF TUBE
18. FORWARD RAMP
19. LITTERS
20. TROOP SEATS
21. CARGO RESCUE WINCH (CH-3E HELICOPTERS PRIOR TO 16)
22. BATTERY

Figure 1-3. General Arrangement Interior Diagram (Typical)



1. AC NON-ESSENTIAL CIRCUIT BREAKER PANEL
2. ENGINE CONTROL QUADRANT
3. OVERHEAD SWITCH PANEL
4. OVERHEAD DC CIRCUIT BREAKER PANEL
5. PILOT'S COMPARTMENT DOME LIGHT
6. AC ESSENTIAL CIRCUIT BREAKER PANEL
7. ROTOR BRAKE LEVER
8. PILOT'S SPOTLIGHT
9. MAGNETIC COMPASS
10. PILOT'S CYCLIC STICK
11. PARKING BRAKE HANDLE
12. NOSEWHEEL LOCK HANDLE
13. PILOT'S COLLECTIVE PITCH LEVER
14. PILOT'S TAIL ROTOR PEDAL ADJUSTMENT KNOB
15. PILOT'S WINDOW EMERGENCY RELEASE HANDLE
16. PILOT'S TOE BRAKES
17. PILOT'S TAIL ROTOR PEDALS
18. PILOT'S SEAT
19. PILOT'S RADIO CONSOLE
20. PILOT'S SHOULDER HARNESS LOCK LEVER
21. COPILOT'S SEAT HEIGHT ADJUSTMENT LEVER
22. COPILOT'S SEAT FORWARD AND AFT ADJUSTMENT LEVER
23. COPILOT'S RADIO CONSOLE
24. COPILOT'S SEAT
25. ALTERNATE LANDING GEAR HANDLE
26. COPILOT'S COLLECTIVE PITCH LEVER
27. COPILOT'S TAIL ROTOR PEDALS
28. COPILOT'S WINDOW EMERGENCY RELEASE HANDLE
29. COPILOT'S TAIL ROTOR PEDAL ADJUSTMENT KNOB
30. COPILOT'S TOE BRAKES
31. COPILOT'S CYCLIC STICK
32. COCKPIT CONSOLE
33. INSTRUMENT PANEL
34. COPILOT'S SCROLL CHECKLIST
35. COPILOT SPOT LIGHT
36. FREE AIR TEMPERATURE GAGE

Figure 1-4. Pilot's Compartment CH-3E (Typical)



1. AC NON-ESSENTIAL CIRCUIT BREAKER PANEL
2. ENGINE CONTROL QUADRANT
3. OVERHEAD SWITCH PANEL
4. OVERHEAD DC CIRCUIT BREAKER PANEL
5. PILOT'S COMPARTMENT DOME LIGHT
6. AC ESSENTIAL CIRCUIT BREAKER PANEL
7. ROTOR BRAKE LEVER
8. PILOT'S SPOTLIGHT
9. MAGNETIC COMPASS
10. PILOT'S CYCLIC STICK
11. PARKING BRAKE HANDLE
12. NOSEWHEEL LOCK HANDLE
13. PILOT'S COLLECTIVE PITCH LEVER
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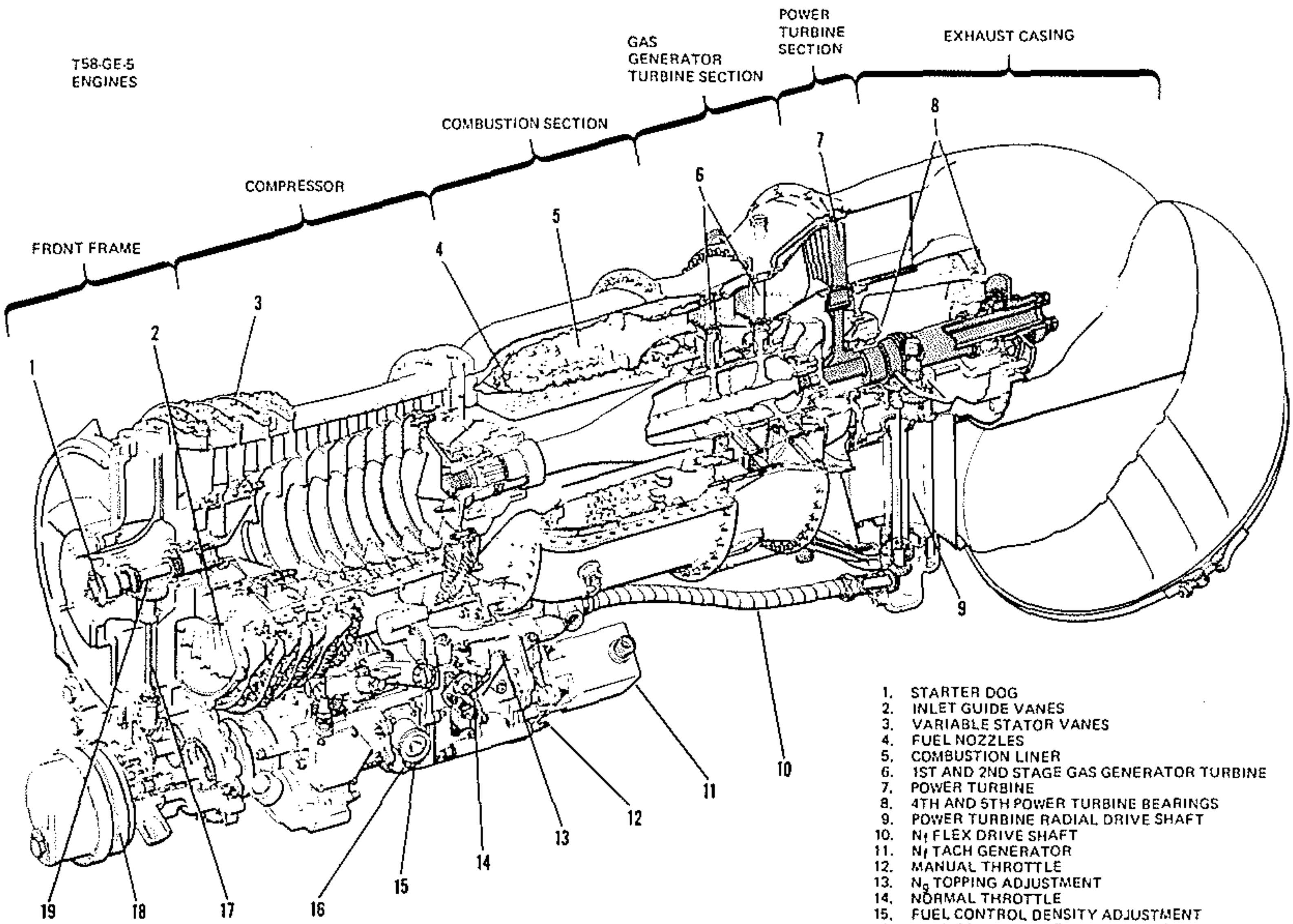
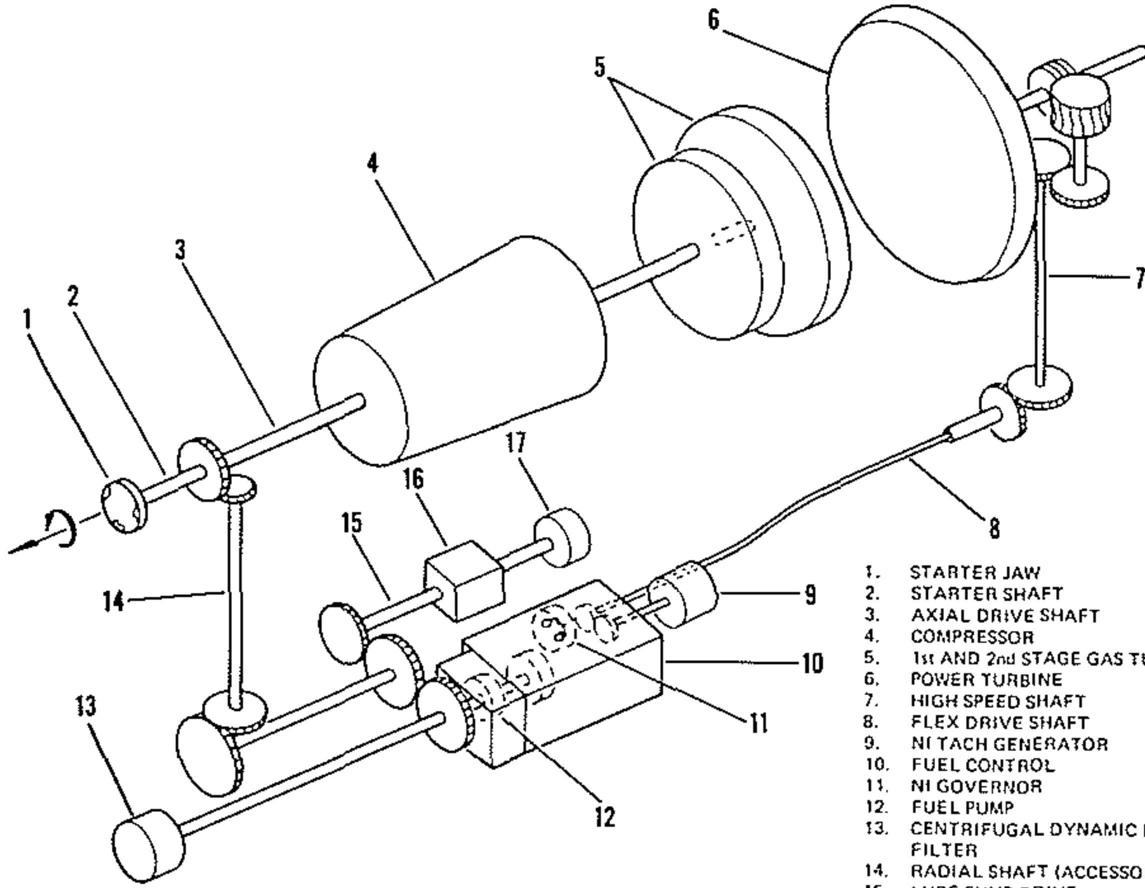
15. PILOT'S WINDOW EMERGENCY RELEASE HANDLE
16. PILOT'S TOE BRAKES
17. PILOT'S TAIL ROTOR PEDALS
18. PILOT'S SEAT
19. PILOT'S RADIO CONSOLE
20. PILOT'S SHOULDER HARNESS LOCK LEVER
21. AUXILIARY FUEL TANK MANUAL RELEASE HANDLE
22. COPILOT'S SEAT HEIGHT ADJUSTMENT LEVER
23. COPILOT'S FORWARD AND AFT ADJUSTMENT LEVER
24. COPILOT'S RADIO CONSOLE
25. COPILOT'S SEAT
26. ALTERNATE LANDING GEAR HANDLE
27. COPILOT'S COLLECTIVE PITCH LEVER
28. COPILOT'S TAIL ROTOR PEDALS
29. COPILOT'S WINDOW EMERGENCY RELEASE HANDLE
30. COPILOT'S TAIL ROTOR PEDAL ADJUSTMENT KNOB
31. COPILOT'S TOE BRAKES
32. COPILOT'S CYCLIC STICK
33. COCKPIT CONSOLE
34. INSTRUMENT PANEL
35. COPILOT'S SCROLL CHECKLIST
36. COPILOT SPOT LIGHT
37. PRESSURE REFUELING CONTROL PANEL
38. FREE AIR TEMPERATURE GAGE

Figure 1-5. Pilot's Compartment - HH-3E (Typical)



1. STARTER DOG
2. INLET GUIDE VANES
3. VARIABLE STATOR VANES
4. FUEL NOZZLES
5. COMBUSTION LINER
6. 1ST AND 2ND STAGE GAS GENERATOR TURBINE
7. POWER TURBINE
8. 4TH AND 5TH POWER TURBINE BEARINGS
9. POWER TURBINE RADIAL DRIVE SHAFT
10. N_f FLEX DRIVE SHAFT
11. N_f TACH GENERATOR
12. MANUAL THROTTLE
13. N_g TOPPING ADJUSTMENT
14. NORMAL THROTTLE
15. FUEL CONTROL DENSITY ADJUSTMENT
16. FUEL CONTROL FUEL INLET FILTER
17. FRONT FRAME RADIAL DRIVE SHAFT
18. CENTRIFUGAL FUEL FILTER
19. AXIAL DRIVE SHAFT

Figure 1-6. Engine Cut-Away View (Typical)



1. STARTER JAW
2. STARTER SHAFT
3. AXIAL DRIVE SHAFT
4. COMPRESSOR
5. 1st AND 2nd STAGE GAS TURBINES
6. POWER TURBINE
7. HIGH SPEED SHAFT
8. FLEX DRIVE SHAFT
9. N1 TACH GENERATOR
10. FUEL CONTROL
11. N1 GOVERNOR
12. FUEL PUMP
13. CENTRIFUGAL DYNAMIC FUEL FILTER
14. RADIAL SHAFT (ACCESSORY DRIVE)
15. LUBE PUMP DRIVE
16. LUBE PUMP
17. Ng TACH GENERATOR

Figure 1-7. Simplified Engine Schematic

is ignited, combustion is self-sustained. Changes in air pressure, air temperature, humidity, helicopter velocity, and rotor operation all affect engine performance. The engine fuel control system automatically maintains selected power turbine speed by changing fuel flow to increase or decrease gas generator speed as required, thus regulating output power to match the load under changing conditions.

COMPRESSOR.

The ten stage compressor consists of the compressor rotor and stator. The compressor rotor is supported by the front frame section and compressor rear frame section. The stator is bolted between the front frame section and compressor rear frame. The primary purpose of the compressor is to compress air for combustion. Ambient air enters through the front frame and is directed to the compressor inlet, passes through ten stages of compression, and is directed to the combustion chambers. The inlet guide vanes (figure 1-6) and the first three stages of the stator vanes (figure 1-6) are variable and change their angular position as a function of compressor inlet temperature and gas generator speed to prevent stall of the compressor.

COMBUSTION CHAMBER.

In the combustion chamber, fuel is added to the compressed air and ignited, causing a rapid expansion of gases toward the gas generator turbine section. As the air enters the combustion section, a portion goes into the combustion chamber where it is mixed with the fuel and ignited. The remaining air forms a blanket between the outer combustion casing and the combustion liner (figure 1-6) for

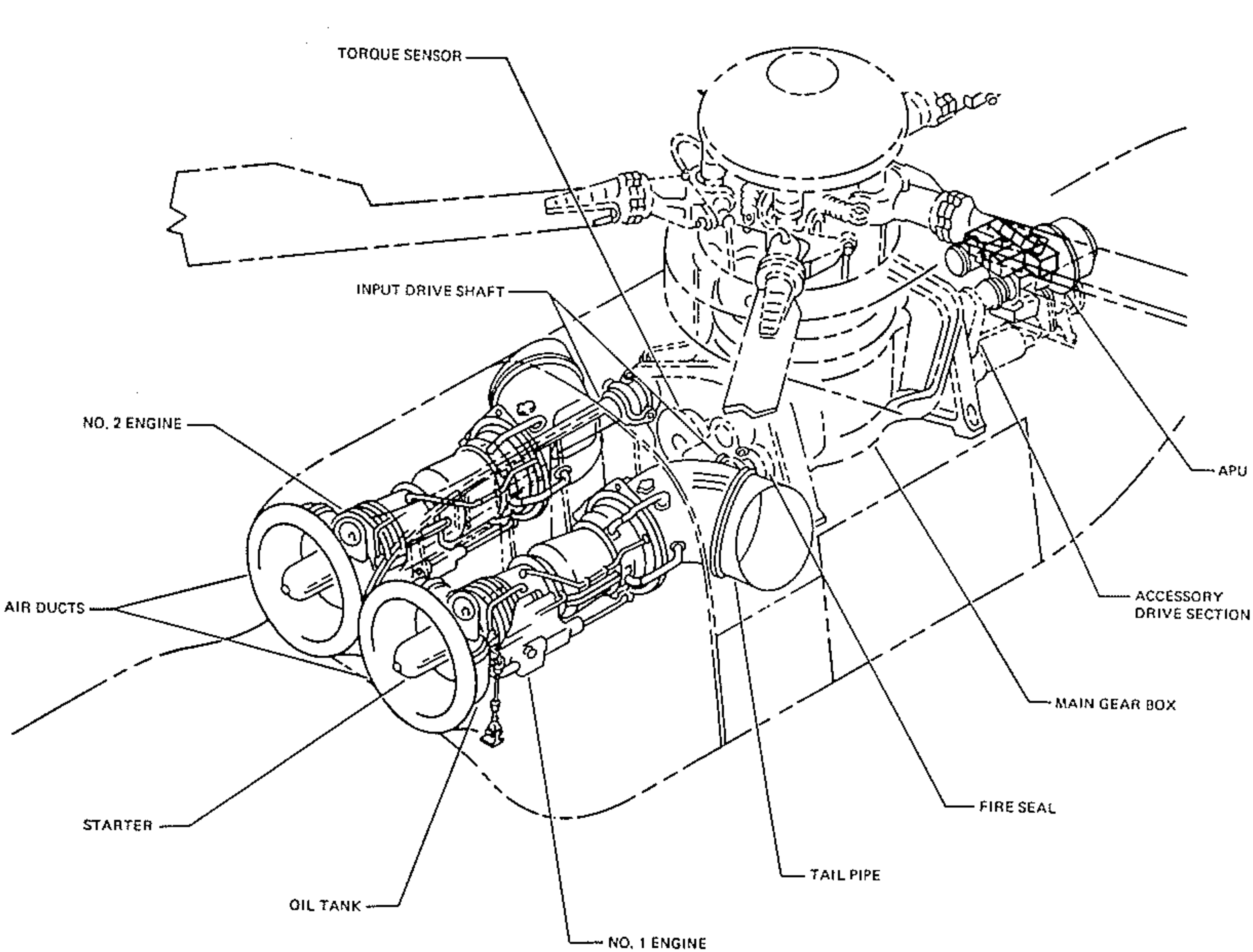


Figure 1-8. Engine, Main Gear Box, and APU Installation

cooling purposes. Once combustion is started by the two igniter plugs, it is self-sustaining. After the air has been expanded and increased in velocity by combustion, it is passed through the first-stage turbine wheel of the gas generator turbine (figure 1-6).

GAS GENERATOR TURBINE.

The two-stage gas generator turbine (figure 1-6) is the rotating component which is coupled directly to the compressor. It extracts the required power from the exhaust gases to drive the compressor. The turbine nozzles that comprise the stator blades direct the exhaust gases to the turbine wheels.

POWER TURBINE.

The power turbine (figure 1-6) is bolted to the rear flange of the second stage turbine casing. The engine utilizes the free turbine principle in which engine output power is provided by the power turbine rotor, which is mechanically independent of the gas generator rotor. This rotor derives its power from the gases which are directed to it by the gas generator turbine nozzles. Within the normal operating range, power turbine speed may be maintained or regulated independent of output power. This principle also provides more rapid acceleration because of the availability of high engine torque at low output speeds.

GAS GENERATOR SPEED (N_g).

Gas generator speed (N_g) is primarily dependent upon fuel flow and is monitored by the engine fuel control unit. The principal purpose of monitoring gas generator speed is to control acceleration and deceleration characteristics, prevent overspeed, and establish a minimum idle setting. Gas generator speed controls mass airflow pumped through the engine and, consequently, the power available to the power turbine.

FREE POWER TURBINE SPEED (N_f).

The free power turbine speed (N_f) is dependent upon engine control input shaft position and rotor load. The principal purpose of monitoring power turbine speed is to regulate fuel flow to maintain an essentially constant power turbine speed for a given engine control input shaft position To prevent destructive power turbine overspeed in the event of a loss of power turbine load, a governor within the fuel control senses power turbine speed and shuts off fuel flow to the engine at a power turbine speed of approximately 120% (N_f). Fuel flow will resume when the power turbine speed drops below the fuel shutoff speed. The engine may or may not relight.

ENGINE FUEL SYSTEM.

The engine fuel systems (figure 1-9), one for each engine, consist of an engine-driven pump, a dynamic filter, a fuel control unit, a static filter, an oil cooler, a flow divider, and a fuel manifold and associated piping. The fuel control unit is supplied fuel from the engine-driven fuel pump. Metered fuel from the engine fuel control unit is piped through an oil-fuel heat exchanger and then enters the flow divider connected directly to the fuel manifold on the engine. For normal flight, rotor speed is selected by positioning the throttles and the engine fuel controls will meter fuel to maintain the selected rotor speed.

Engine-Driven Fuel Pump.

A dual element engine-driven fuel pump, mounted on each engine, consisting of a positive displacement type gear pump and a centrifugal boost pump, is built into a single housing. Power for each pump is furnished from the engine accessory drive section by means of a splined shaft. This shaft drives the fuel pump and simultaneously acts as a link to transmit gas generator speed information to the engine fuel control unit.

Engine Fuel Control Unit.

The engine fuel control units, one located on each engine, are hydromechanical units that regulate engine fuel flow to maintain a constant selected free power turbine speed, and thus maintain a constant helicopter rotor speed. Fuel from the engine fuel pump enters the fuel control unit through the inlet and passes through the fuel filter. The fuel control has a fuel metering section and a computing section. The metering section selects the rate of flow to the combustion chambers, based on information received from the computing sections. The metering section has a metering valve and a pressure regulating valve. The pressure regulating valve maintains a constant pressure across the main metering valve by bypassing excess fuel back to the engine fuel pump inlet. The metering valve is positioned engagement, and with the throttles in the governing range (at or forward of the MIN GOV position), any force attempting to slow the rotor or transmission system, such as increases in collective pitch, will be sensed by the fuel control unit which will attempt to maintain constant rpm by increasing power. The throttles must be placed in the GRD IDLE or SHUT-OFF position before applying the rotor brake, for normal shutdown.

ENGINE SPEED TRIM SWITCHES (BEEPER TRIM SWITCHES).

The engine speed trim switches (beeper trim switches), located on each collective pitch lever grip (figure 1-11), are used to make adjustments to power turbine speed and for engine synchronization. The switches are marked ENG TRIM, 1 and 2, + (plus) and — (minus). The switches provide electrical power to actuators in the overhead control quadrant which are connected to the throttles. The throttles are positioned by the actuators for adjustment to the desired power turbine speed. Moving the ENG TRIM switches forward will cause increases in power turbine speed and moving the switches aft will cause decreases in power turbine speed. When the desired power turbine speed is attained, the switches are released and will return to the spring-loaded center position. Beeper trim system range is approximately 91% to 108% N_f. The ENG TRIM switches receive electrical power from the dc essential bus through circuit breakers, under the general heading ENGINE and marked SPEED TRIM, 1-ENG-2, located on the center overhead dc circuit breaker panel.

NOTE

Pilot's beeper trim switches override the copilot's switches.

EMERGENCY FUEL CONTROL LEVERS.

Two emergency fuel control levers, one for each engine, marked EMER FUEL CONTROL, are located on each side of the engine control quadrant (figure 1-10). The emergency fuel control levers operate independently and are used in case of fuel control unit failure. Each emergency fuel control lever has positive open and close stops and is connected directly by a flexible cable and linkage to the main metering valve in each engine fuel control unit. The primary function of the emergency fuel control lever is to manually override the automatic features of the fuel control. This may become necessary under some starting situations and during any fuel control malfunction that causes erratic engine operation. The emergency fuel control lever must be used with extreme caution as it has a positive influence on fuel flow and misuse can cause engine overspeed or overtemperature. The lever is mechanically connected to a cam within the fuel control which contacts the fuel metering valve. The initial position of the fuel metering valve is dependent upon the automatic features of the control as established by the setting of the throttle. The cam, when actuated by advancing the emergency fuel control lever, contacts the fuel metering valve. Once contact is established, further advancement of the emergency fuel control will manually control fuel flow, which in turn, regulates engine power output. At high power settings, considerable dead band travel will normally be encountered before the emergency fuel control contacts the metering valve lever. This will be felt as a slight restriction in control movement. When this is felt, the control will be very sensitive, and care should be taken not to exceed T_5 and N_g red lines. The emergency fuel control is unable to reduce the position of the metering valve below that called for by the throttles. Control below this point will depend upon the type of malfunction encountered. In all instances of emergency fuel control operation, it must be remembered that the throttles must not be retarded beyond the GRD IDLE position. The fuel stopcock is located downstream of the metering valve and is actuated by the throttles. Placing the throttles in the SHUT-OFF position will stop engine fuel flow, regardless of emergency fuel control lever position.

CAUTION

Rapid movement of emergency fuel control levers may induce compressor stall.

STARTING SYSTEM.

Each engine starting system (figure 1-12) is equipped with an electric starter, located on the front frame section of the engine. In addition, each system is equipped with a start bleed valve. The start bleed valve is automatically opened during the starting cycle, to direct air overboard to reduce

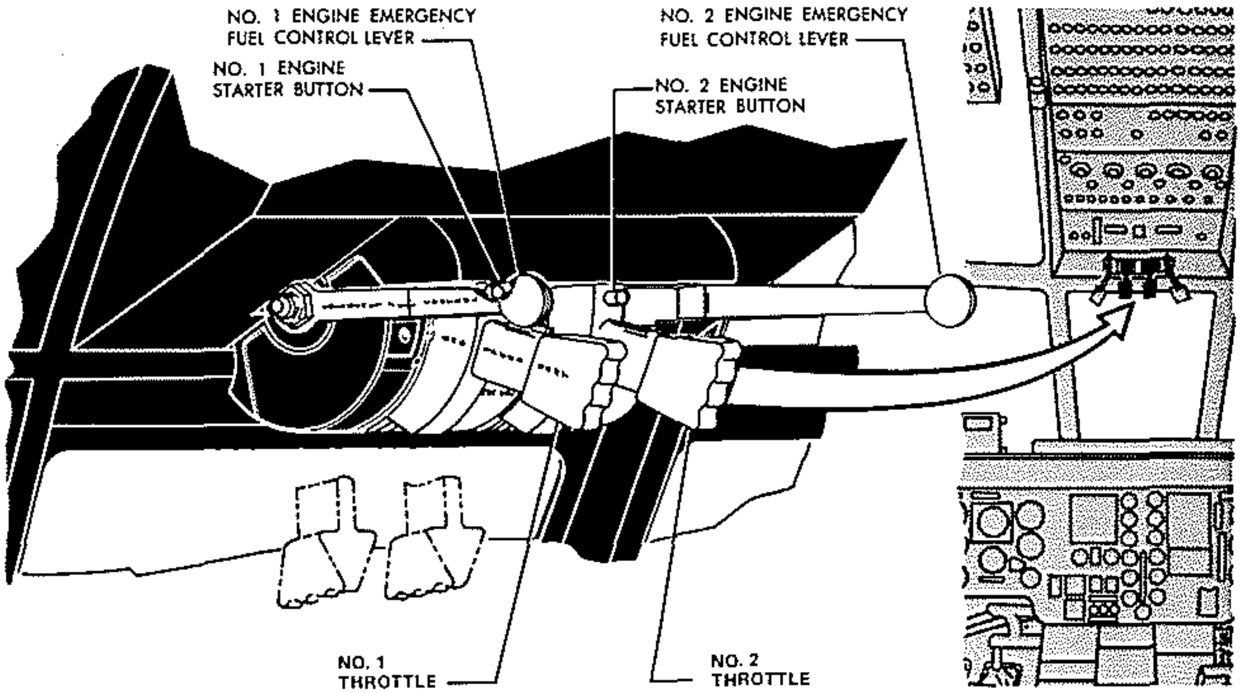


Figure 1-10. Engine Controls

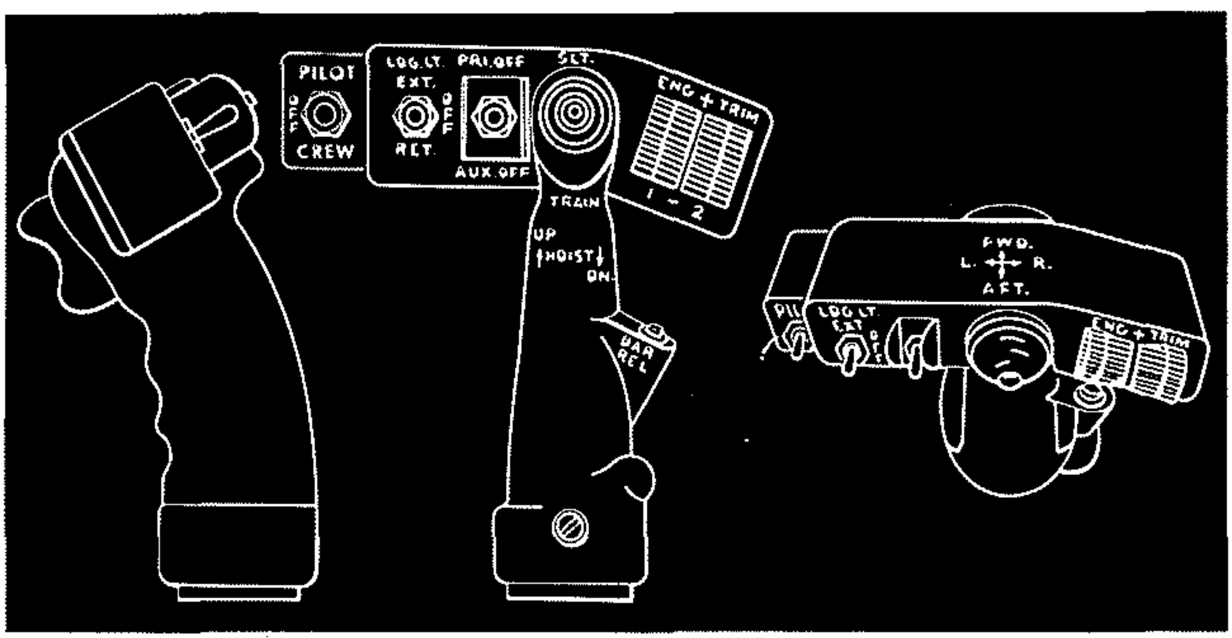


Figure 1-11. Collective Pitch Lever Grip

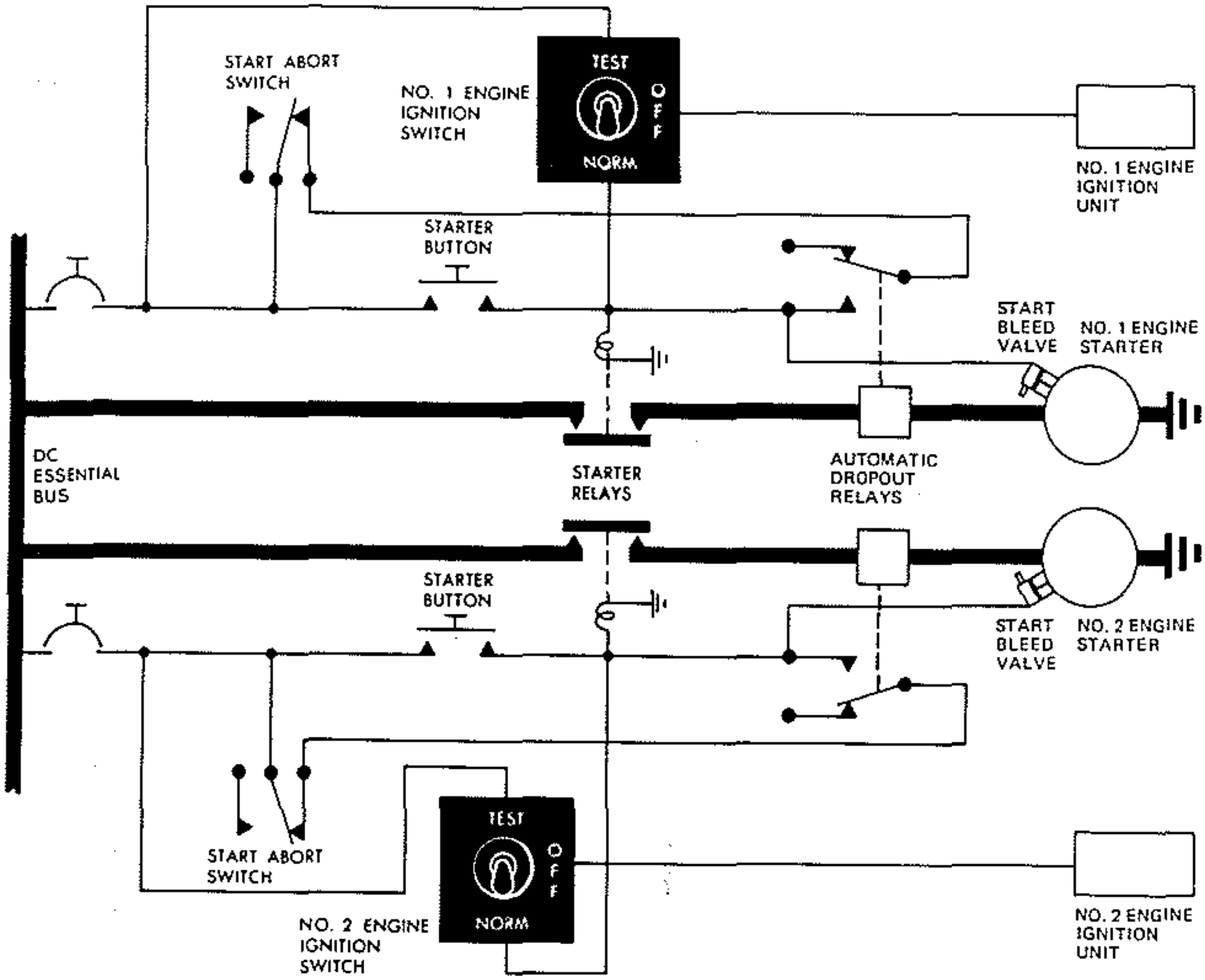


Figure 1-12. Starting System Diagram

back pressure in the compressor and lessen the possibility of engine stall, and closes after the starter has been deenergized (approximately 45 to 53%). The valve bleeds off approximately 6.7% of the compressor discharge air flow, reduces the starter drag, and allows the gas generator speed to increase faster. Engine starts are accomplished by placing the throttle in the SHUT-OFF position, placing the IGNITION switch in the NORM position, and depressing the starter button on the engine throttle. With the relay closed, power is supplied to the starter motor and the start bleed valve. When the starter button is released, a holding circuit in the starter relay, powered through the abort switch, holds the relay contacts closed until the starter amperage falls below 100 ± 15 amperes. With the starter in operation, the throttle may be advanced to the GRD IDLE position to supply the necessary fuel to the engine. After engine lite-off, starter amperage falls below 100 ± 15 amperes and the starter relay drops out. This cuts off power to the starter motor and ignition unit. A start may be aborted prior to engine lite-off by pulling down on the throttle. This opens the normally closed abort switch on the quadrant and deenergizes the starter relay coil. Power for the control circuit of the starter and ignition system is supplied by the essential dc bus at 28 volts dc through the engine starter 1-ENG-2 circuit breakers on the circuit breaker panel. The starter has a duty cycle limited to 30 seconds continuous cranking, a minimum cooling period of 3 minutes between start attempts, and a maximum of three start attempts in any 30 minute period. Before the starter can be energized, the APU must be operating or external power applied, or the battery switch ON. The engine may be motored by using the starter with the ignition switches off.

Starter Buttons.

A starter button is located on each throttle. The starter is energized by holding the engine throttle in SHUT-OFF position and momentarily depressing the starter button which energizes the starter relay and completes the circuit to the starter. After the engine starts and the starter electrical power load decreases, the starter dropout relay automatically disengages the electrical power to the starter and ignition circuit. During the engine start, starter dropout normally occurs between 45 to 53% N_g. Starter dropout can be noted by the magnetic compass swinging to its original heading and the loadmeters being energized.

Starter Abort Switch.

The starter abort switch, located in the engine control quadrant (figure 1-10) and actuated by pulling throttle down, provides the means to abort an engine start prior to engine lite-off. The abort switch interrupts electrical power for ignition and deenergizes the starter relay.

IGNITION SYSTEM.

Each engine ingition system, mounted on the engine, consists of a capacitor-discharge ignition unit, two ignitor plugs, and a control circuit. The system provides ignition for starting only; during engine operation, the flame in the combustion chamber is self-sustaining. When the switch is in the NORM position, the ignition unit operates in conjunction with the starter. When gas generator speed increases and the starter power load decreases, the automatic dropout disengages both the starter and ignition system and combustion is self-sustained. The ignition system operates on current from the dc essential bus through the starter control system.

Ignition Switches.

An ignition switch, one for each engine, located on the overhead switch panel (figure 1-13), marked IGNITION, 1-ENG-2, has marked positions TEST, OFF, and NORM. When the switch is in the NORM position with the starter engaged, the ignition unit is energized. Holding the switch in the spring-loaded TEST position energizes the ignition unit only. When the switch is placed in TEST position, a clicking noise can be heard. When the switch is in the OFF position, the ignition unit is deenergized. The throttle must be in the SHUT-OFF position before the starter and ignition systems can be energized.

TORQUEMETERS.

Two torquemeters (figure 1-14), one for the pilot and one for the copilot, are located on the instrument panel. Each dual-pointer indicator marked PERCENT TORQUE, contains two pointers, marked 1 and 2, which indicate input torque in percent of maximum engine power output of each engine. The electrically-actuated torquemeter dials, calibrated in percent torque, are graduated in increments of 5 percent from 0 to 150. The torquemeters operate on 26 volts ac and are protected by circuit breakers, marked 1 ENG 2 TORQUE SENSOR, located on the ac essential circuit breaker panel. The torquemeters indicate the amount of torque being applied to the main gear box by the engines. This knowledge serves two purposes; (1) to prevent overstressing the gear box, and (2) to monitor the power output of the engines. The torque sensing cells are located in the main gear box and are hydromechanical in nature, sensing any shift in the helical gear at the input from each engine. Oil pressure within the cells are sensed by pressure transmitters and transmitted electrically to the torquemeters.

ENGINE GAS GENERATOR (N_g) TACHOMETERS.

Two engine gas generator tachometers (figure 1-14), one for each engine, are located on the instrument panel and indicate the speed of the gas generator in percent of total rpm (N_g). Each tachometer has two dials and pointers. The outer dial and pointer indicates gas generator speed from zero to 100 percent, increments of two percent. The small vernier dial and pointer, located in the upper left-hand position of the tachometer, indicates gas generator speed from 0 to 10, in increments of 1 percent. The gas generator tachometer-generator is driven by the engine oil pump shaft. The electrical power produced by the gas generator tachometer-generator is proportional to gas generator rpm (100% N_g=26,300 gas generator rpm).

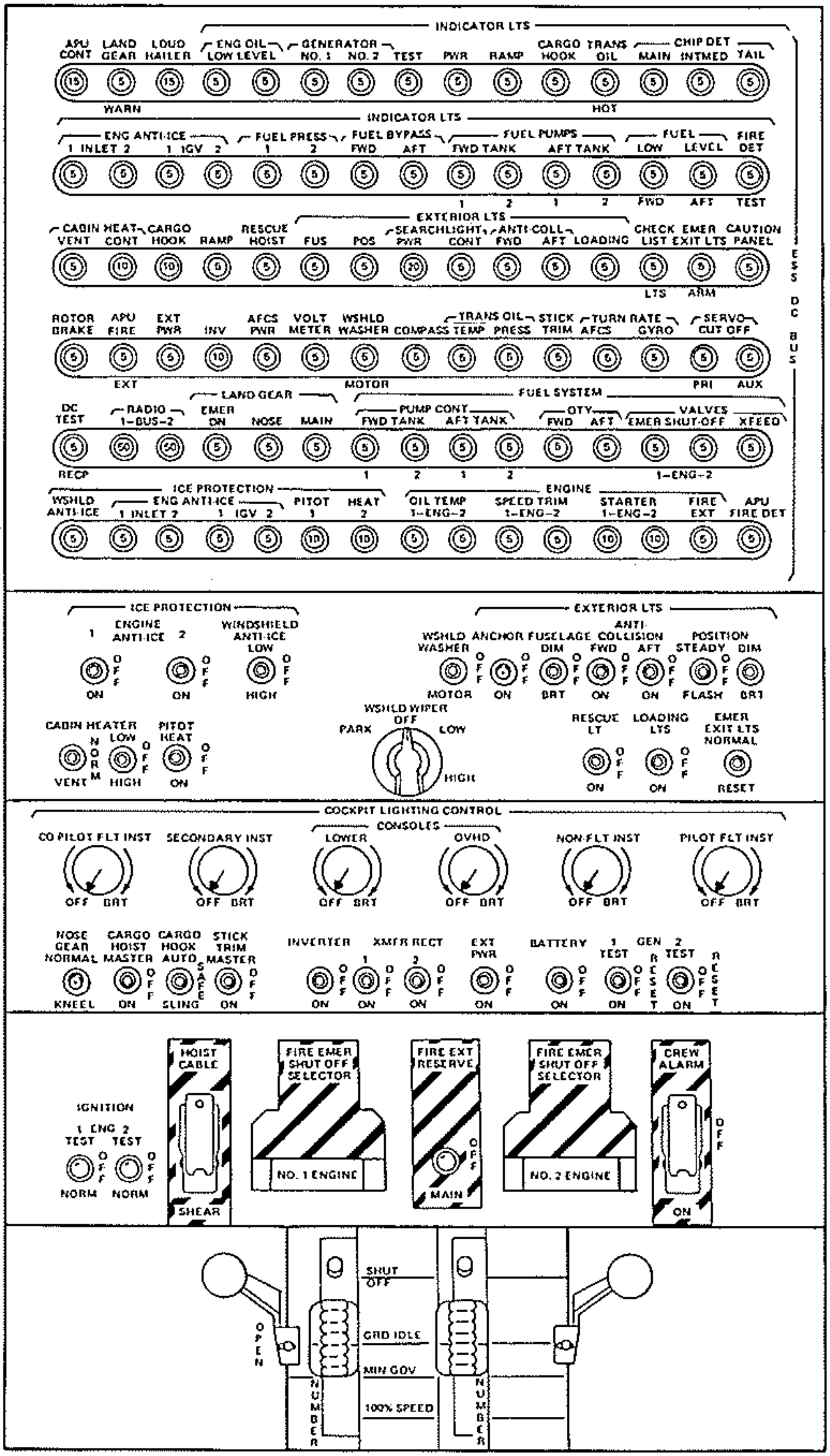
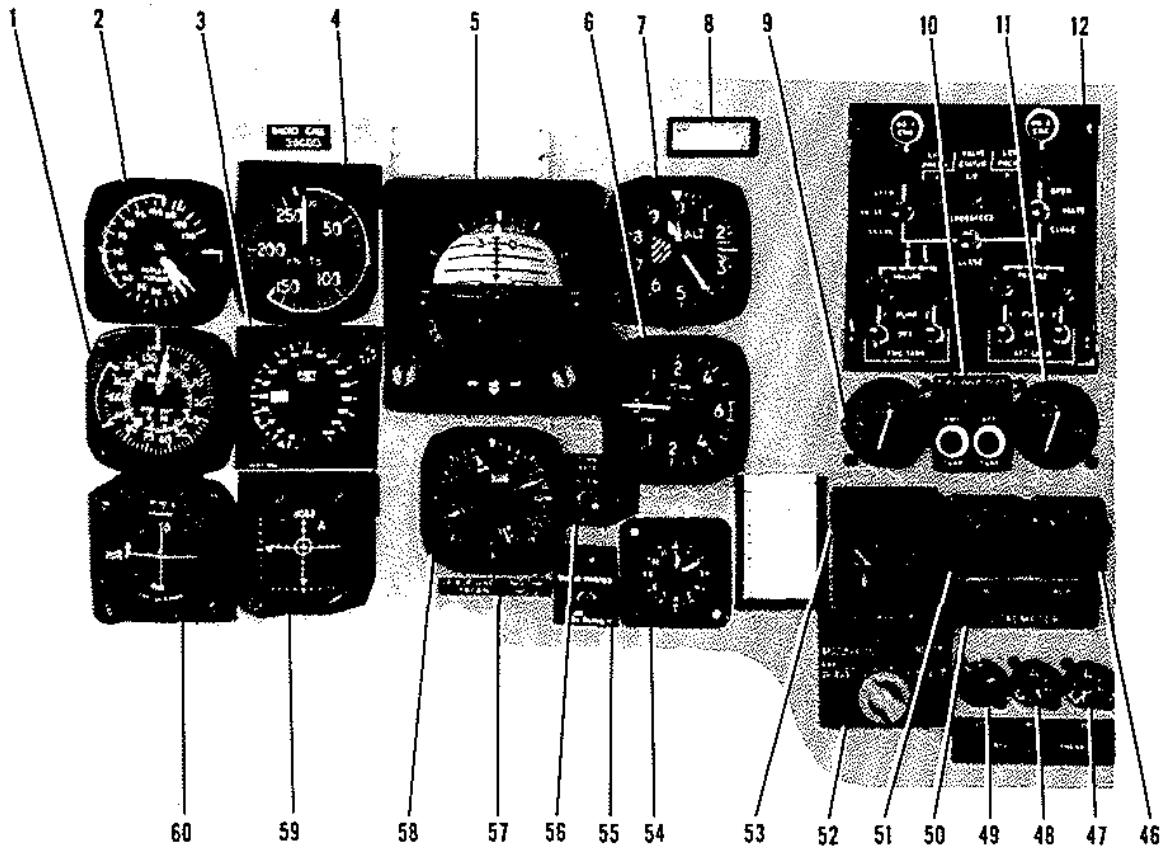
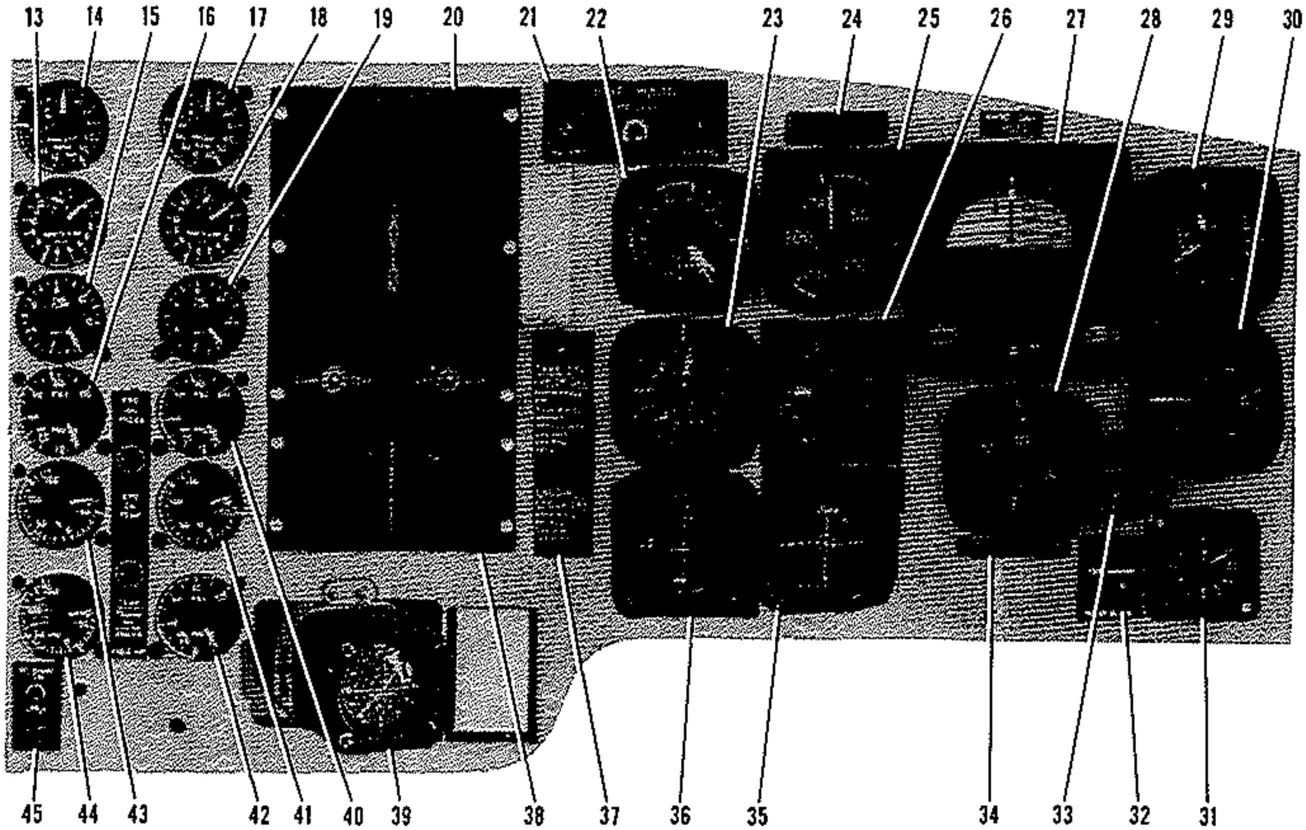


Figure 1-13. Overhead Switch Panel and Engine Control Quadrant (Typical)



1. COPILOT'S TRIPLE TACHOMETER
2. COPILOT'S TORQUEMETER
3. COPILOT'S RADAR ALTIMETER
4. COPILOT'S AIRSPEED INDICATOR
5. COPILOT'S ATTITUDE INDICATOR
6. COPILOT'S VERTICAL VELOCITY INDICATOR
7. COPILOT'S ALTIMETER
8. MASTER CAUTION LIGHT
9. FORWARD TANK FUEL QUANTITY GAGE
10. FUEL QUANTITY GAGE TEST SWITCH PANEL
11. AFT TANK FUEL QUANTITY GAGE
12. FUEL MANAGEMENT PANEL
13. NO. 1 ENGINE POWER TURBINE INLET TEMPERATURE (T_5) INDICATOR
14. NO. 1 ENGINE GAS GENERATOR (N_g) TACHOMETER
15. NO. 1 ENGINE FUEL FLOW INDICATOR
16. NO. 1 ENGINE OIL PRESSURE INDICATOR
17. NO. 2 ENGINE GAS GENERATOR (N_g) TACHOMETER
18. NO. 2 ENGINE POWER TURBINE INLET TEMPERATURE (T_5) INDICATOR
19. NO. 2 ENGINE FUEL FLOW INDICATOR
20. CAUTION LIGHT PANEL
21. FIRE WARNING LIGHTS AND TEST SWITCH PANEL
22. PILOT'S TORQUEMETER
23. PILOT'S TRIPLE TACHOMETER
24. MASTER CAUTION LIGHT
25. PILOT'S AIRSPEED INDICATOR
26. PILOT'S RADAR ALTIMETER
27. PILOT'S ATTITUDE INDICATOR
28. PILOT'S BEARING, DISTANCE, HEADING INDICATOR
29. PILOT'S ALTIMETER

Figure 1-14. Instrument Panel (Typical) (Sheet 1 of 2)



30. PILOT'S VERTICAL VELOCITY INDICATOR
31. PILOT'S CLOCK
*32. PILOT'S VOR/TACAN SELECTOR SWITCH
33. PILOT'S TURN RATE SWITCH
34. PILOT'S BDHI POINTER IDENTIFICATION DECAL
35. PILOT'S AFCS INDICATOR
36. PILOT'S COURSE INDICATOR
37. CHECK OFF LIST
38. ADVISORY LIGHT PANEL
**39. VELOCITY STEERING INDICATOR
40. NO. 2 ENGINE OIL PRESSURE INDICATOR
41. NO. 2 ENGINE OIL TEMPERATURE INDICATOR
42. TRANSMISSION OIL PRESSURE INDICATOR
43. NO. 1 ENGINE OIL TEMPERATURE INDICATOR
44. TRANSMISSION OIL TEMPERATURE INDICATOR
*45. DF RANGE SWITCH
46. NO. 2 LOAD METER
47. PRIMARY HYDRAULIC PRESSURE INDICATOR
48. AUXILIARY HYDRAULIC PRESSURE INDICATOR
49. UTILITY HYDRAULIC PRESSURE INDICATOR
50. LOAD METER IDENTIFICATION PANEL
51. NO. 1 LOAD METER
52. VOLTMETER SELECTOR PANEL
53. DC VOLTMETER
54. COPILOT'S CLOCK
*55. COPILOT'S VOR/TACAN SELECTOR SWITCH
56. COPILOT'S TURN RATE SWITCH
57. COPILOT'S BDHI POINTER IDENTIFICATION PANEL
58. COPILOT'S BEARING, DISTANCE HEADING INDICATOR
59. COPILOT'S AFCS INDICATOR
60. COPILOT'S COURSE INDICATOR

Figure 1-14. Instrument Panel (Typical) (Sheet 2 of 2)

N_f AND N_r TRIPLE TACHOMETERS.

Two triple tachometers (figure 1-14), one for the pilot and one for the copilot, are located on the instrument panel. Each tachometer contains three pointers; the pointers marked 1 and 2 indicate the power turbine speed (N_f) of the No. 1 and 2 engines, respectively, and the pointer marked R indicates the main rotor rpm. The engine tachometers are powered by their own tachometer-generators which are driven by the power turbine through a flex cable, which is routed to the fuel control on which they are mounted. The main rotor tachometer is powered by its own tachometer-generator, located on the accessory section of the gear box, and driven by the accessory gears. The tachometers are read in percent of total rpm (100% N_f=18,966 power turbine rpm and 100% N_r= 203 rotor rpm).

POWER TURBINE INLET TEMPERATURE (T_5) INDICATORS.

Two power turbine inlet temperature indicators (figure 1-14), marked PWR TURB INLET TEMP, are located on the instrument panel. The indicators are graduated in degrees Centigrade and operate from thermocouples, located forward of the power turbine in the second-stage turbine casing on each engine. The indicators are normally powered by 115 volts ac from the ac essential bus, through circuit breakers, marked TURBINE INLET TEMP 1-ENG-2, located on the ac essential circuit breaker panel. However, when the ac essential bus is not energized, the indicators are powered by 115 volts ac from the inverter. The pilot has no direct control for regulating the power turbine inlet temperatures; however, limited control for lowering these temperatures can be achieved by reducing collective pitch or power demand. The maximum power turbine inlet temperature is indirectly controlled by the gas generator maximum speed adjustment of the fuel control.

ENGINE OIL PRESSURE INDICATORS.

Two engine oil pressure indicators (figure 1-14), one for each engine, are located on the instrument panel. The indicators are powered by 26 volts ac from the inverter bus and are protected by circuit breakers, marked OIL PRESS 1-ENG-2, located on the essential circuit breaker panel. Pressure is indicated in psi.

ENGINE OIL TEMPERATURE INDICATOR.

Two engine oil temperature indicators (figure 1-14), one for each engine, are located on the instrument panel. The engine oil temperature bulb, located on each oil inlet line on the bottom of each engine oil tank transmits indications to the respective temperature indicator. The indicators are powered from the dc essential bus and are protected by circuit breakers, marked OIL TEMP 1-ENG-2, located on the overhead dc circuit breaker panel. Temperature is indicated in degrees Centigrade.

FUEL FLOW INDICATORS.

Two fuel flow indicators (figure 1-14), 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 and under the general heading FUEL, located on the ac essential circuit breaker panel.

ROTOR SYSTEMS.

The rotor systems consist of a single main rotor and an anti-torque tail rotor. Both systems are driven by the two engines through the transmission system and are controlled by the flight controls.

MAIN ROTOR SYSTEM.

The main rotor system consists of the main rotor head assembly and the rotor blades. The head assembly, mounted directly above the main gear box, consists of a hub assembly and a swashplate assembly. The hub assembly, consisting of five sleeve-spindle assemblies and five hydraulic dampers clamped between two parallel plates, is splined to the main rotor drive shaft. The root ends of the five rotor blades are attached to the sleeve-spindle assemblies which permit each blade to flap vertically, hunt horizontally, and rotate about their spanwise axis, to change the angle of incidence. Anti-flapping restrainers limit the upward movement of the blades caused by wind pressure and droop stops limit the downward position of the blades. Both are in operation when the blades are stopped or turning at low speed. When speed is increased to approximately 25 percent (50 rpm) rotor speed, centrifugal force automatically releases the anti-flapping restrainers. The droop stops release at approximately 75% (152 rpm) rotor speed. The hydraulic dampers minimize hunting movement of the blades about the vertical hinges as they rotate, prevent shock to the blades when the rotor is started or stopped, and aid in the prevention of ground resonance. The blades are constructed of aluminum allow with the exception of forged steel cuffs which attach the root ends of the blades to the sleeve-spindle assemblies on the main rotor hub. Each blade basically consists of a hollow extruded aluminum spar pressurized with nitrogen, 25 aluminum blade pockets, an aluminum tip cap, an aluminum root cap, a steel cuff, a pressure (IBIS) indicator, an air valve, and an abrasion strip. Vent holes on the underside of each pocket prevent accumulation of moisture inside the blade. Each blade is balanced statically and dynamically within tolerances that permit individual replacement of the blades. In addition, a pretrack number is stenciled on each blade to eliminate the necessity for blade tracking. Balancing and the assignment of a pretrack number is performed at time of manufacture or overhaul. The swashplate assembly consists of an upper (rotating) swashplate driven by the rotor hub and a lower stationary swashplate secured by a scissors assembly to the main gear box to prevent rotation. Both swashplates are mounted on a ballring and socket assembly which keeps them parallel at all times, but allows them to be tilted, raised, or lowered simultaneously by components of the main rotor flight control system that are connected to arms on the lower stationary swashplate. Cyclic or collective pitch changes, introduced at the stationary swashplate, are transmitted to the blades by linkage on the rotating swashplate. The main rotor hub assemblies are equipped with a bifilar absorber assembly to reduce fatigue stress and improve the overall vibration comfort level throughout the helicopter. The bifilar absorber assembly, secured to the main rotor hub, consists of a five pointed, starshaped, aluminum forging with a seventeen-pound weight attached to each star point. Each weight is enclosed by a fairing to reduce drag.

CAUTION

Should an object (door, window, inspection panel, etc.) be inadvertently lost during flight, land at the nearest suitable landing area if a vibration is experienced, or at the nearest airfield if a vibration is not experienced, and inspect the main and tail rotor systems. Possible damage may not be felt in the controls or be visually detected with rotors running.

In-Flight Blade Inspection System (IBIS).

The In-Flight Blade Inspection System (IBIS) visibly indicates in the cockpit that the pressure in one or more main rotor blade(s) has dropped below the allowable limit (figure 1-15).

The IBIS indicator located on the back wall of the spar of each main blade contains a small radioactive source (100 micro curies strontium 90) which is completely shielded (no radiation emitted) when rotor blade is at normal pressure. When the pressure in the rotor blade drops below approximately 6 pounds, the indicator will activate, causing the radioactive source to move to an unshielded position, thereby emitting beta radiation. The detector assembly, located aft of the main rotor shaft, under the transmission cowling, detects the beta radiation and sends a signal to the signal processor. The signal processor causes the BLADE PRESS light on the caution panel to illuminate, indicating a loss of pressure in one or more of the blades. Loss of pressure in the blade spar is also indicated by the IBIS indicator located in the back wall of the spar of each main blade. The indicator has a transparent cover through which a color indication can be observed. If the pressure in the blade spar drops below the minimum permissible service pressure, the indicator will be activated and will show two red stripes. The IBIS system is failsafe, i.e., loss of 115-volt 40 hz power, failure of the detector, and/or failure of the Signal Processor will cause the BLADE PRESS light to illuminate. The system receives electrical power from the 28-volt dc essential bus and 115-volt ac essential bus and is protected by 5-amp circuit breakers located on the pilot's and copilot's circuit breaker panels.

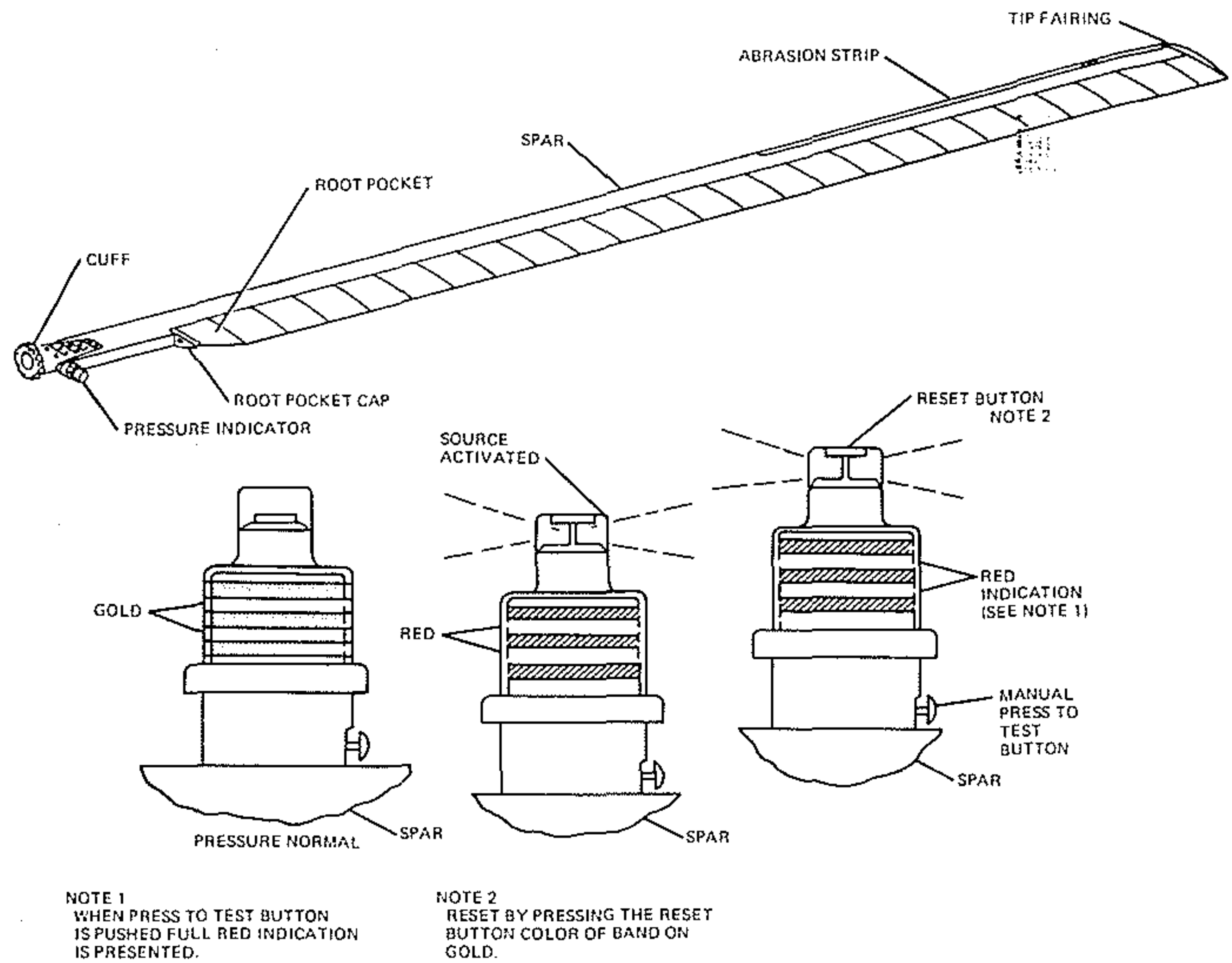


Figure 1-15. IBIS Indicator

TAIL ROTOR.

The tail rotor (figure 1-2) consists of the tail rotor assembly and tail rotor blades. The tail rotor assembly, mounted at the upper end of the pylon, consists of a tail rotor hub and the pitch-changing mechanism. The splined hub is supported and driven by the horizontal output shaft of the tail gear box. Because the tail rotor is directly geared, via transmission shafts, to the main gear box, tail rotor RPM is directly proportional to main rotor RPM. The five tail rotor blades are attached to the tail rotor hub so they are free to flap and rotate about their span-wise axis for pitch variation. The blade pitch-changing mechanism transmits tail rotor control pedal movements to the tail rotor blades through the horizontal output shaft of the tail gear box.

TRANSMISSION SYSTEM.

The transmission system (figure 1-16) consists of three gear boxes that transmit power to the main and tail rotors. The main gear box reduces engine rpm and interconnects the two engines to the rotor head. A freewheeling unit, located at each engine input to the main gear box, permits the rotor head

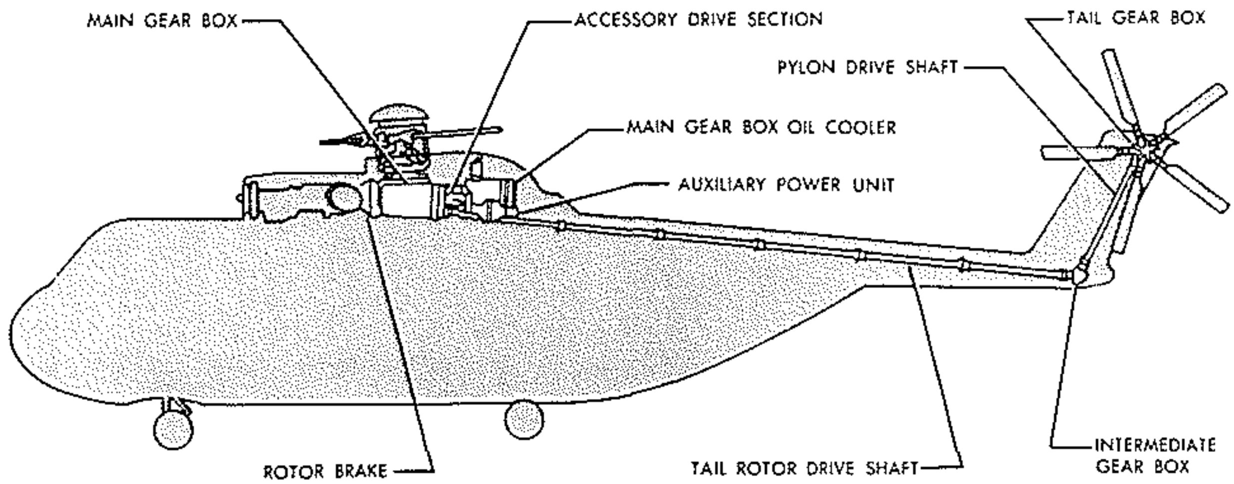


Figure 1-16. Transmission System

to autorotate without engine drag in event of engine (or engines) failure, or when engine rpm decreases below that of the rotor rpm. Engine torque is transmitted through the main gear box to the main rotor drive shaft to drive the main rotor, and aft through a tail rotor drive shaft to the intermediate gear box at the base of the pylon. From the intermediate box, a pylon drive shaft extends upward to the tail gear box to drive the tail rotor. Each of the three gear boxes has a chip detector.

MAIN GEAR BOX.

The main gear box, mounted above the cargo compartment aft of the engines, is a four-stage reduction gear system which reduces engine rpm at a ratio of approximately 93.4 to 1 for driving the rotor head. The main gear box contains a spur, helical bevel gear, and a single planetary gear stage. Shafting extends from the main gear box lower housing to the intermediate gear box and then to the tail rotor gear box to drive the tail rotor. The main gear box accessory section drives the primary, auxiliary, and utility hydraulic pumps, primary and secondary transmission lubrication pumps, torquemeter oil pump, two generators, and the N_r tachometer generator. The freewheeling capability of the tail takeoff freewheeling unit allows the APU to drive the accessory section during ground operations without turning the main rotor system. After rotor speed reaches 100% N_r, the tail takeoff freewheeling unit engages and the accessory section is driven by the main gear box. The APU clutch contains a freewheel unit that enables shutdown of the APU when the main rotor is turning. Should the tail takeoff freewheeling unit fail, the No. 1 engine N_f throughshaft will drive the accessory section whenever the No. 1 engine is running and the rotors are turning. Operation below approximately 92-96% N_f will cause illumination of the generator caution lights during ground operations.

INTERMEDIATE GEAR BOX.

The intermediate gear box, located at the base of the tail rotor pylon, contains a bevel gear direct-drive system to change direction of the shafting that transmits engine torque to the tail gear box. The intermediate gear box is splash-lubricated. Screened air outlets (figure 1-2) in the pylon fairing permit the gear box to be cooled by the rotor downwash.

TAIL GEAR BOX.

The tail gear box, located at the upper end of the tail rotor pylon, contains a bevel gear reduction-drive system to transmit engine torque to the tail rotor. The tail gear box also contains part of the pitch change linkage which extends through the hollow output shaft to the tail rotor hub. The tail gear box is splash-lubricated.

TRANSMISSION CHIP DETECTOR LIGHTS.

Three transmission chip detector lights, marked CHIP LOCATION, MAIN, INTMED, and TAIL, are located on a chip location panel on the cockpit console (figures 1-17 and 1-18). The lights provide a visual indication of metallic chips detected in the main, intermediate, or tail gear boxes. An additional chip detector sensor monitors the system in the main gear box emergency sump. The added chip detector uses the same chip detector light, electrical power source, and protective circuit breaker as the MAIN (primary) chip detector. A caution panel light, marked CHIP DETECTED, illuminates simultaneously with any one of the three lights on the cockpit console. The system operates on current from the dc essential bus and is protected by circuit breakers, marked CHIP DET, MAIN, INTMED, and TAIL, located on the overhead dc circuit breaker panel.

ROTOR BRAKE.

A hydraulically-actuated rotor brake, mounted on a brake shaft forward of the main gear box, stops the rotation of the rotor system and prevents rotation when the helicopter is parked. The rotor brake consists of a hydraulic cylinder and lever, pressure gage, hydraulic brake cylinders, and a brake disc. The rotor brake hydraulic cylinder and lever, located on the pilot's compartment ceiling, operate independently from the hydraulic systems. A spring-loaded accumulator, connected to the rotor brake hydraulic lines at the forward end of the transmission compartment, assures continuous hydraulic pressure when the rotor brake lever is applied. The rotor brake hydraulic cylinder is gravity fed with hydraulic fluid from the rotor brake reservoir. In case of a broken or leaking hydraulic line from the rotor brake reservoir, the rotor brake hydraulic cylinder contains sufficient fluid for braking the rotor system. The hydraulic brake cylinder is located on the supports attached to the main gear box. The brake disc is positioned on the main input shaft of the main gear box.

Rotor Brake Cylinder and Lever.

A rotor brake lever (figure 1-19) is connected directly to the rotor brake hydraulic cylinder, located on the pilot's compartment ceiling to the right and forward of the overhead switch panel. The rotor brake is applied by pulling down and pushing forward as indicated on the decal aft of the lever on the upper structure. The decal is marked TO ENGAGE ROTOR BRAKE PUSH LEVER FORWARD and has an arrow pointing forward. A spring-loaded lock, located at the forward outboard side of the cylinder, automatically locks the brake lever in the applied (forward) position if the pilot places the small handle in the horizontal (forward position). To release the rotor brake, pull out on the lockpin and swing the lever aft and up against the bottom of the cylinder until it snaps into place. The lockpin may be rendered inoperative by rotating until it remains in the OUT position.

Rotor Brake Pressure Gage.

A hydraulic actuated rotor brake pressure gage is located to the rear of the rotor brake lever (figure 1-19) on the pilot's compartment ceiling. The reading, indicated by the pointer, indicates psi x 100. A decal, marked ROTOR BRAKE PRESSURE, located adjacent to the rotor brake pressure gage, is marked to identify the operating pressure ranges of the system. The marking ACTUATING RANGE 350-500 psi identifies the system operating pressure range for normal rotor brake application. The marking ENGINE START 320 P.S.I. MIN. identifies the minimum system pressure required before starting engines to ensure the rotors will not turn with both engines operating at ground idle. The marking PARKED POSITION RANGE 250-600 P.S.I identifies the system pressure range maintained for effective rotor brake application when the helicopter is parked with the rotor brake on.

Rotor Brake Caution Light.

The rotor brake caution light, marked ROTOR BRAKE ON, is located on the caution panel (figure 1-20) on the pilot's side of the instrument panel. The light is provided as an aid in the prevention of rotor engagement while the rotor brake is engaged. Whenever the rotor brake hydraulic pressure is 10 $\pm$ 1 PSI or above, the electrical power is supplied to the dc essential bus, the caution light will go on. When the rotor brake pressure drops below 10 $\pm$ 1 PSI, the light will go out. Normally, the rotor brake off the pressure should be zero; however, after the rotor brake is released and pressure, at 10 $\pm$ 1psi or above, is trapped in the system, the caution light will remain on. If the pressure reaches 20 psi, the brake will begin to drag.

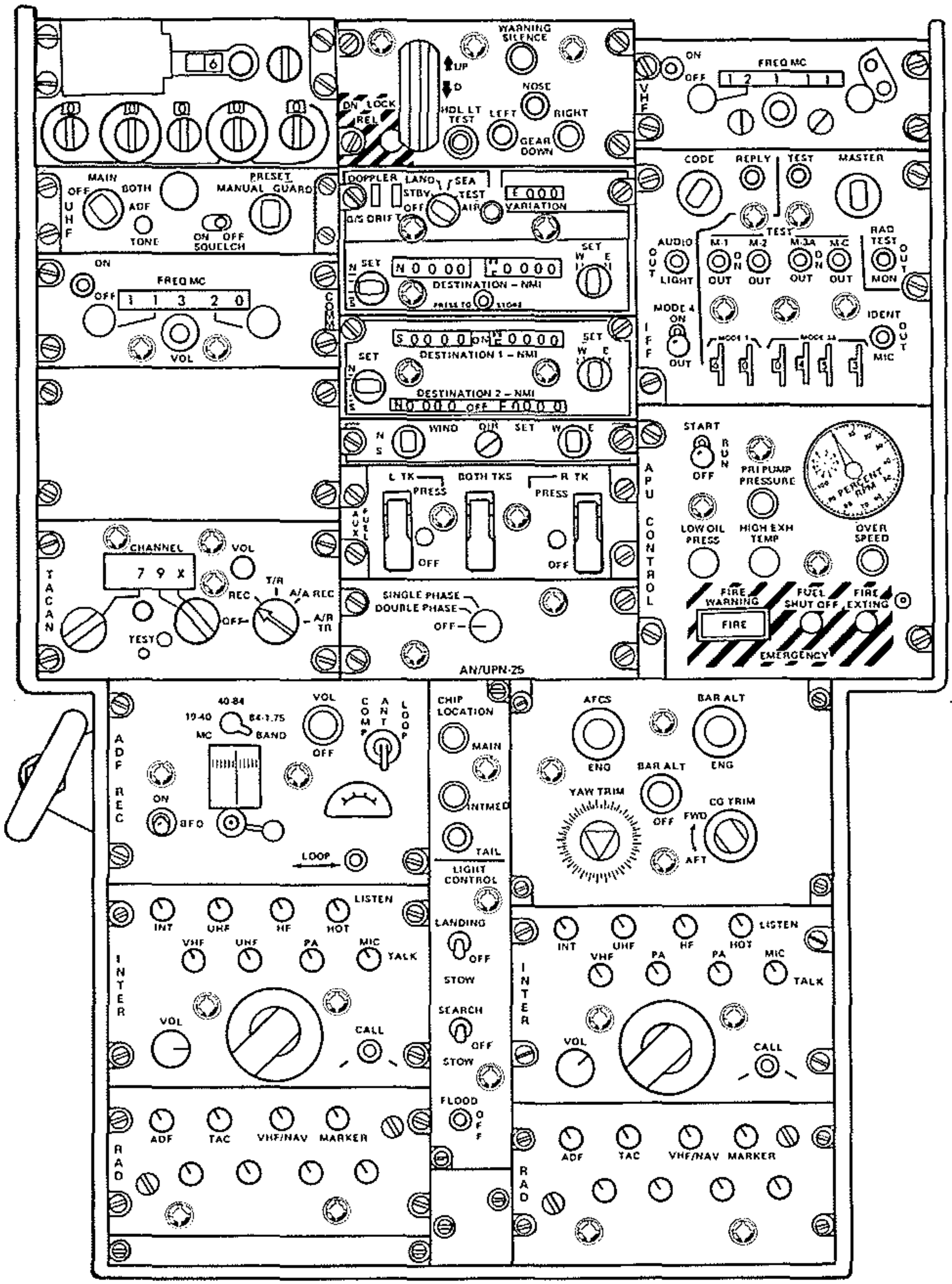


Figure 1-17. Cockpit Console (Typical) (HH-3E Helicopters)

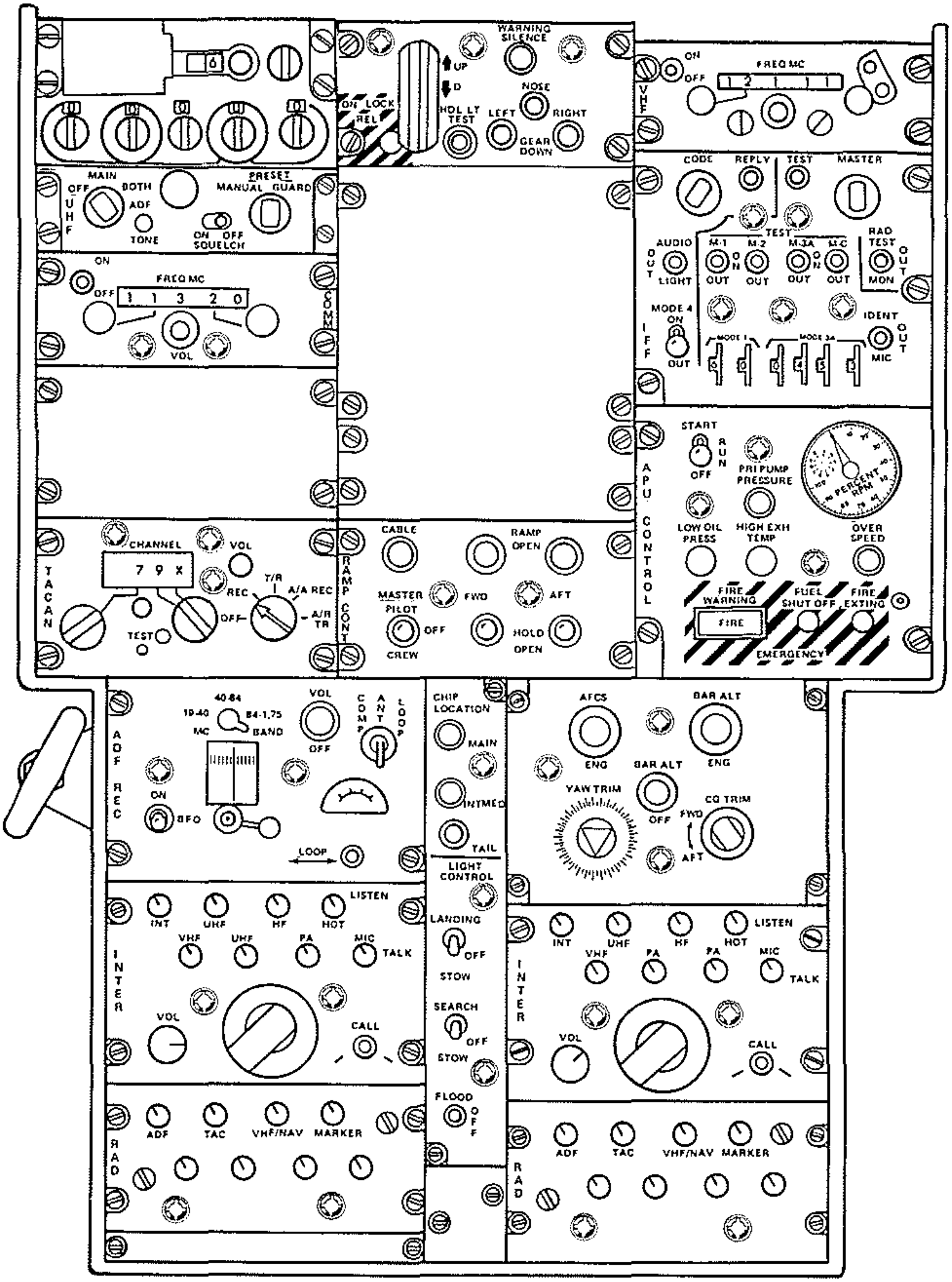


Figure 1-18. Cockpit Console (Typical) (CH-3E Helicopter)

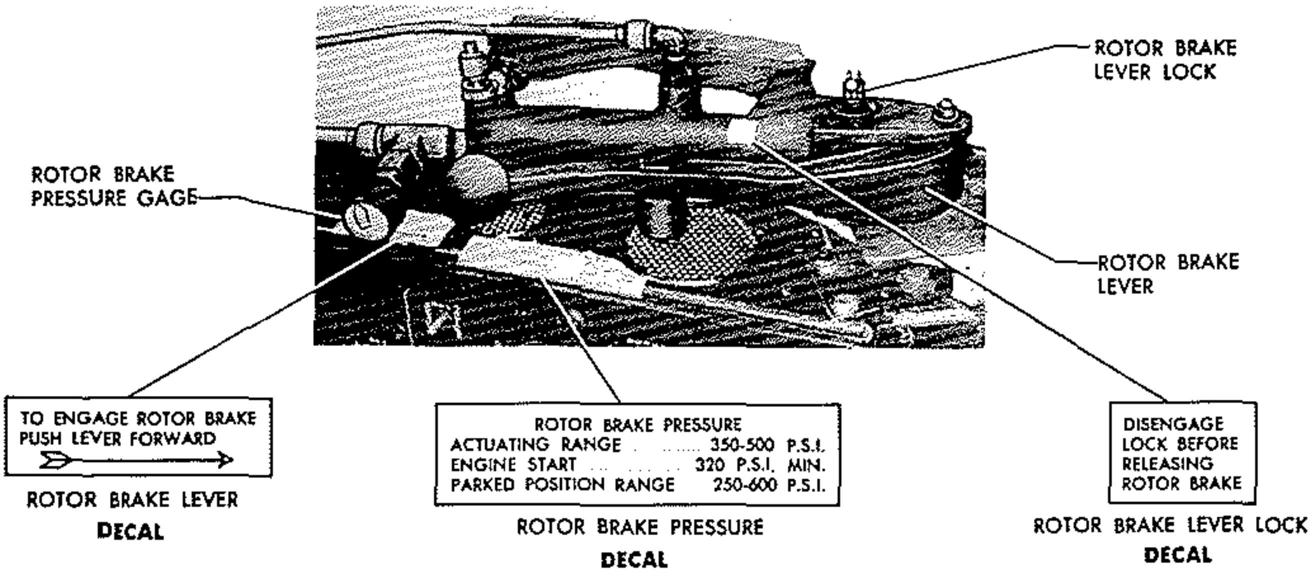


Figure 1-19. Rotor Brake Lever

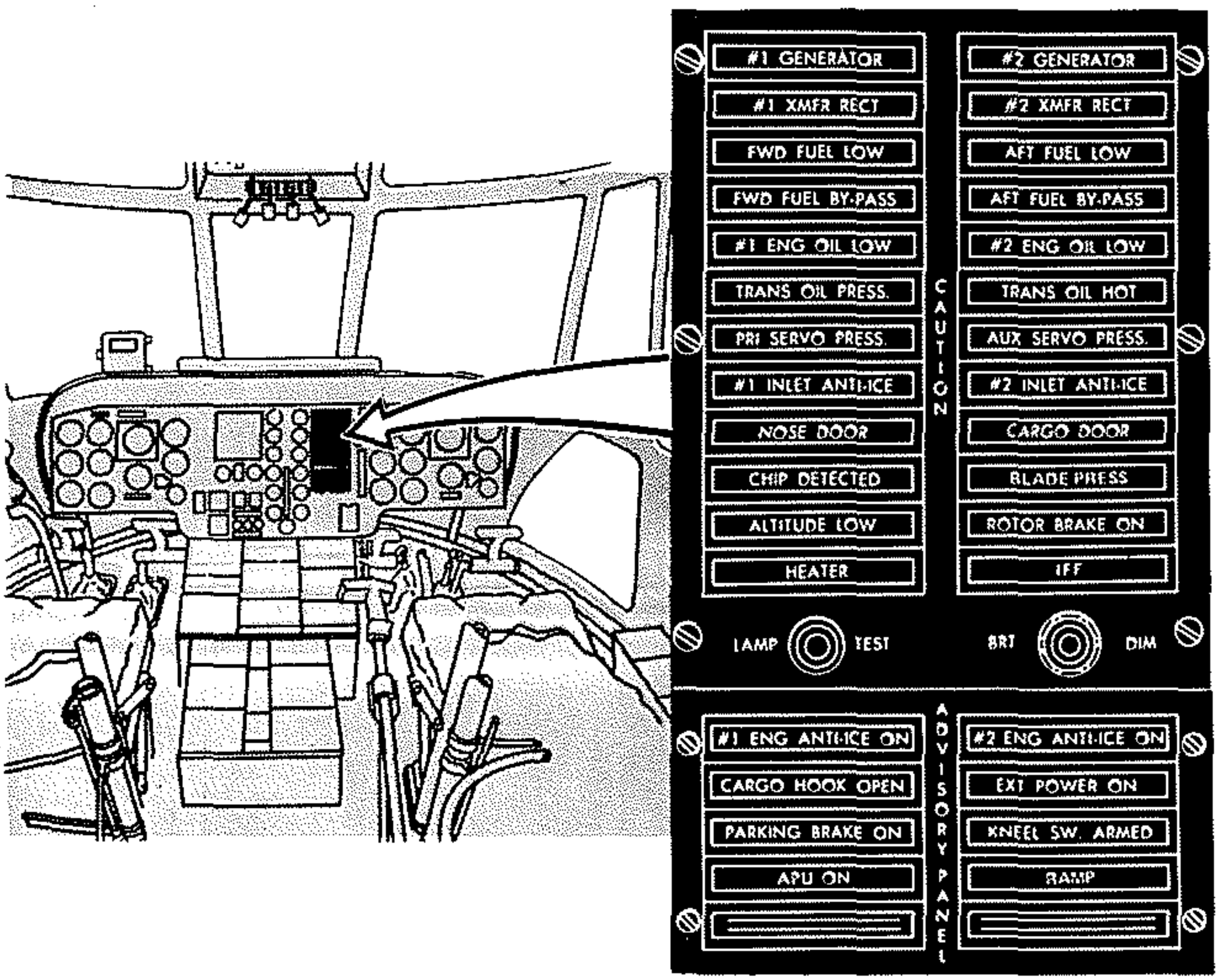


Figure 1-20. Caution and Advisory Panel (Typical)

Figure 1-21 deleted.

OIL SUPPLY SYSTEMS.

The oil supply systems consist of the engine and transmission oil systems. An auxiliary oil system is provided for those helicopters that have the internal auxiliary fuel tanks installed to augment the normal oil tank supply. HH-3E helicopters are equipped with an auxiliary oil system that differs slightly from that provided with the internal auxiliary fuel tanks. CH-3E helicopters 16 are equipped with provisions for installation of the auxiliary oil system used on the HH-3E.

ENGINE OIL SYSTEM.

Each engine has an independent oil tank and dry sump full scavenge oil system. Oil is gravity fed from the tank to the engine drive oil pump, mounted on the forward right-hand side of the engine. The pump distributes the oil, under pressure, through a filter to accessory gears and engine bearings. The oil serves both lubricating and cooling purposes and the system is completely automatic. The scavenge side of the pump returns the oil through an oil cooler to the oil tank. The oil cooler is an oil-to-fuel heat exchanger with an associated oil bypass system. The oil flow through the cooler depends on oil temperature. At low oil temperature, most of the oil bypasses the cooler. Higher oil temperatures close the bypass valve and cause the oil to flow through the cooler. Each engine oil system has a useful capacity of 2.5 US gallons of oil in a 3.5 US gallon tank (1.0 gallon expansion space). The circular tanks are located around the forward section of each engine.

NOTE

> Approximate oil consumption is 1.3 and 1.4 pints per hour at normal and military power, respectively. Maximum allowable oil consumption is 1.4 pints per hour.

Engine Oil Low Level Caution System.

The engine oil low level caution system consists basically of two separate indicating systems, one for each engine. Each system is separately powered through a 5 ampere circuit breaker on the essential dc bus and has a separate caution panel light capsule marked #1 ENG OIL LOW and #2 ENG OIL LOW. A float switch is installed in each engine oil tank. If the oil level in a tank falls 0.6 gallons below full, the float switch contacts close. Closing the switch contacts completes the circuit from the essential dc bus, through the 5 amp circuit breaker and the warning light to ground, causing the light to illuminate. Power for the engine oil low level caution system is supplied by the essential dc bus system through circuit breakers, marked ENG OIL LOW LEVEL, located on the overhead control panel.

AUXILIARY OIL SYSTEMS.

Auxiliary oil systems (figure 1-22) are provided for those helicopters that have the internal auxiliary fuel tanks installed and those helicopters equipped with external auxiliary fuel tanks and an air refueling system. The requirement for an auxiliary oil system is generated primarily by the increased range realized with the auxiliary fuel tanks and the air refueling system. As both systems use the same components and operating procedures, the basic difference being the location of the auxiliary oil tank, they will be discussed under one heading. The auxiliary oil system consists of a tank, hand pump, drain valve, two directional valves, and two TANK FULL lights. The need to replenish the oil supply in the engine oil tanks is indicated by illumination of the appropriate ENG OIL LOW caution light. The appropriate engine oil tank may then be filled to the proper level by opening the applicable direction valve and operating the hand pump until the associated TANK FULL light illuminates. Continued pumping after the TANK FULL light has illuminated will cause the oil to be pumped overboard.

Auxiliary Oil Tank.

The auxiliary oil tank for helicopters equipped with the single internal auxiliary fuel tank installed is mounted on the fuel tank. If the dual internal auxiliary fuel tank system is installed, the tank is mounted on the rear tank. Those helicopters equipped with external auxiliary fuel tanks and an air refueling system have the auxiliary oil tank mounted on the right-hand side of the cargo compartment. The tank has a capacity of 2.6 US gallons and is equipped with a filler cap, drain valve, lights that indicate when the engine oil tanks are full, and the plumbing to carry the oil to the hand pump.

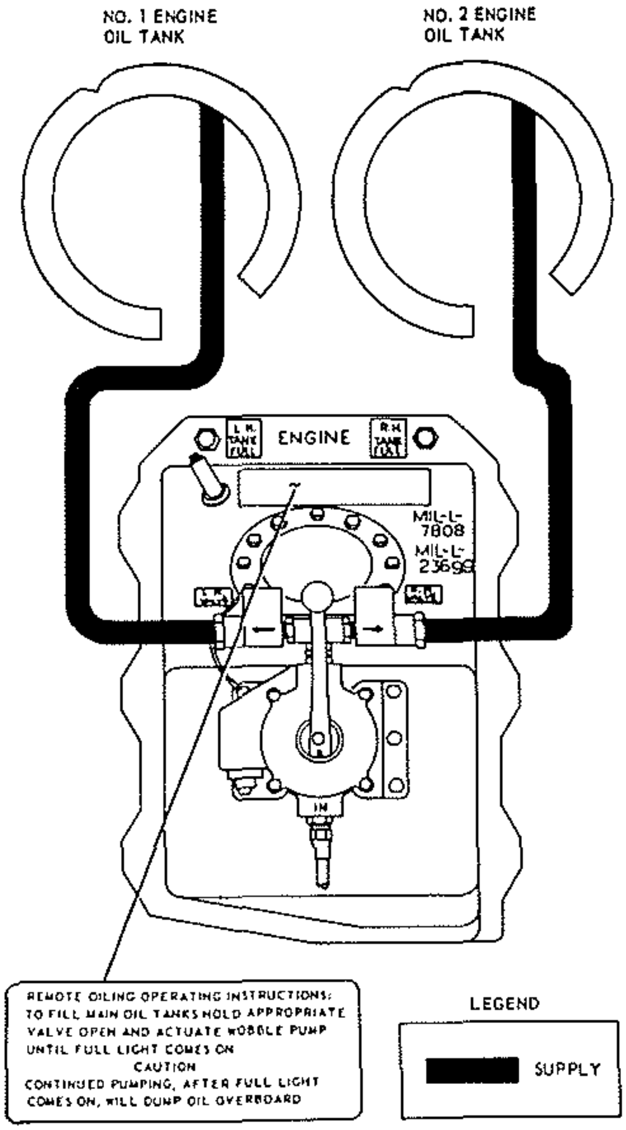


Figure 1-22. Auxiliary Oil System

Hand Pump.

The hand pump, secured to the auxiliary oil tank, is manually operated by moving the handle in a left and right direction.

Directional Valves.

The two directional valves, secured to the auxiliary oil tank, are used to direct the auxiliary oil to a selected engine oil tank. The valve marked LH VALVE directs flow to the No. 1 engine oil tank and the valve marked RH VALVE directs flow to the No. 2 engine oil tank. The valves are opened by depressing plungers that will return the valves to the closed position whenever they are released. There are no provisions for locking the valves in the open position.

Engine Oil Tank Full Indicator Lights.

The engine oil tank full indicator lights, marked LH TANK FULL and RH TANK FULL, are located on the top of the auxiliary oil tank. The lights, one for each engine oil tank, illuminate to indicate that the engine oil tank has been replenished to the full level by the auxiliary oil system. The engine oil tank full indicator lights are powered from the dc essential bus through the INDICATOR LTS PWR circuit breakers located on the overhead circuit breaker panel.

TRANSMISSION OIL SYSTEM.

Each of the three transmission system gear boxes has an individual oil system. The main gear box is pressure-lubricated and the intermediate and tail gear boxes are splash-lubricated.

MAIN GEAR BOX OIL SYSTEM.

A primary oil pump and secondary oil pump circulate oil for main gear box lubrication and cooling. The torque system oil pump is mounted tandem to the primary oil pump and is used to provide lubrication under emergency conditions. Oil is pumped from the gear box sump to an oil cooler located behind the main gear box. Cooling air enters the forward end of the main gear box fairing and is forced through the oil cooler by a blower, driven by belts from the tail rotor drive shaft. After passing through the oil cooler, the oil returns to the main gear box where it is sprayed onto the gears and bearings through jets built into the gear box castings. An oil filler is accessible from the left side of the main gear box fairing. A window in the gear box below the oil filler provides a sight check for the oil level in the main gear box. Normal servicing is 11.6 US gallons. Figure 1-23 shows the main gear box lubricating schematic diagram.

CAUTION

The oil level cannot be seen if the tank is overserviced. Excessive oil temperatures can result from an overserviced main gear box.

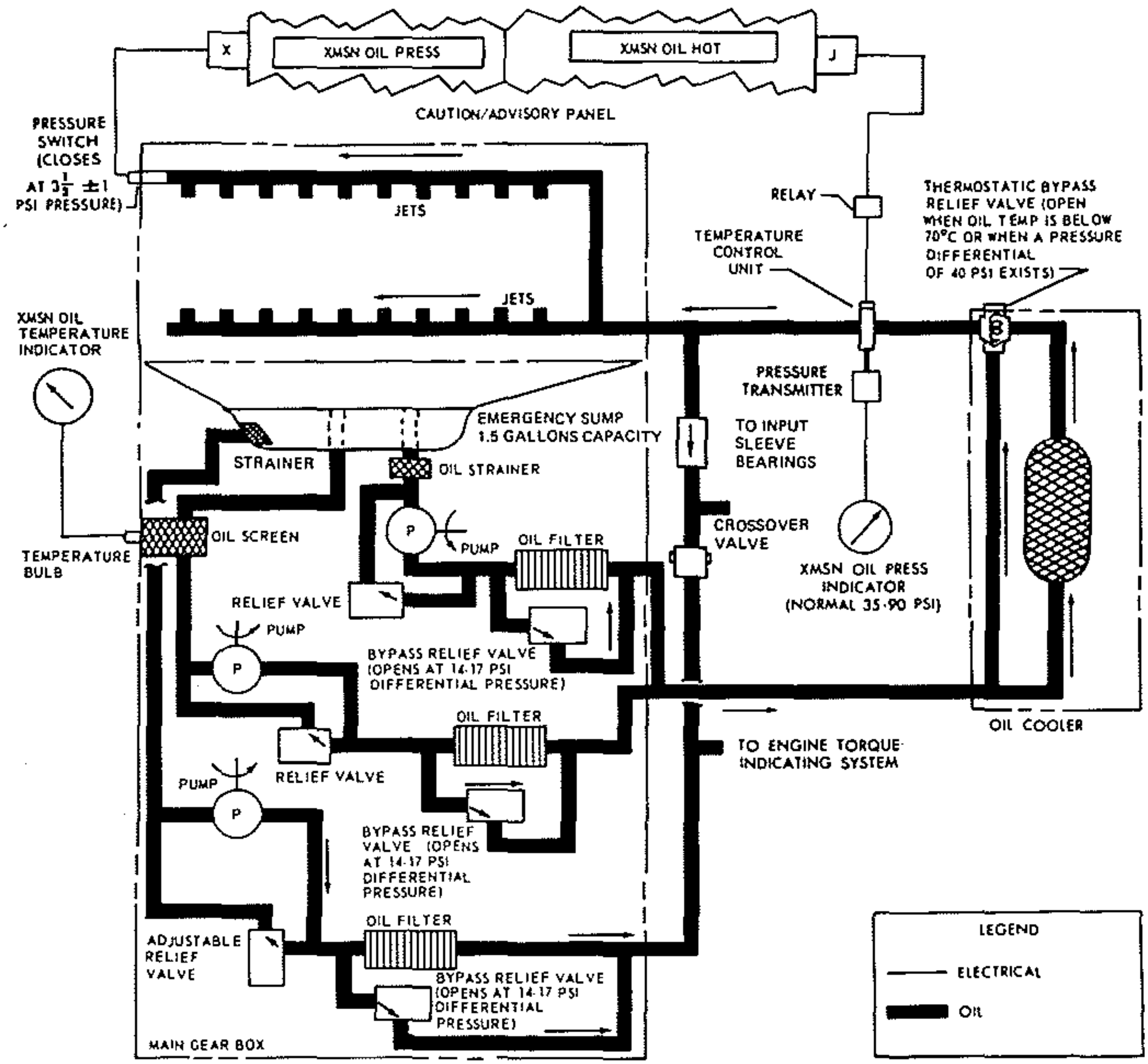


Figure 1-23. Main Gear Box - Lubricating Schematic Diagram

MAIN GEAR BOX EMERGENCY SUMP OIL SUPPLY.

An emergency sump (figure 1-24) in the bottom of the transmission provides a reserve oil supply of approximately 1.5 gallons to lubricate the input sleeve bearings in the high speed section of the main gear box. If a break occurs in the main gear box pressurized lubricating system, the main lubrication pumps will pump most oil overboard, leaving approximately 1.5 gallons in the emergency sump. As the main lubrication pressure decreases to approximately 10 PSI, a crossover valve opens, directing oil from the torquemeter pump directly to each of the four sleeve bearings in addition to providing oil to the torque indicating system.

WARNING

Damage to the bottom of the transmission or torque system oil pump could result in the loss of the emergency sump oil supply, as indicated by no transmission oil pressure indications and a complete loss of torquemeter reading (zero torque on both indicators). This situation requires an immediate autorotation. A power on landing may not be possible.

Main Gear Box Oil Pressure Indicator and Caution Light.

The main gear box oil pressure indicator, marked TRANS OIL PRESS (figure 1-14), is located on the instrument panel. The indicator is graduated in pounds per square inch and is actuated by a pressure transmitter connected to the gear box oil inlet port. The main gear box oil pressure indicator operates on 26 volts ac from the inverter bus and is protected by a circuit breaker marked TRANS OIL PRESS, located on the ac essential circuit breaker panel. The main gear box oil low pressure caution light, marked TRANS OIL PRESS, is located on the caution panel (figure 1-20). The amber caution light operates on direct current from the essential bus and is protected by a circuit breaker, marked CAUTION PANEL, located on the overhead dc circuit breaker panel. The light will come on when the main gear box oil pressure drops below 4 psi as it enters the last oil pressure jet in the gear box.

Main Gear Box Oil Temperature Indicator and Caution Light.

The main gear box oil temperature indicator (figure 1-14), marked TEMP XMSN OIL, located on the instrument panel, is graduated in degrees Centigrade. The indicator, electrically connected to an oil temperature bulb adjacent to the main gear box oil outlet port, receives power from the dc essential bus through a circuit breaker, marked TRANS OIL TEMP, located on the overhead dc circuit breaker panel. The main gear box oil temperature caution light, marked TRANS OIL HOT, is located on the caution panel (figure 1-20). The amber caution light operates on direct current from the essential bus and is protected by a circuit breaker, marked TRANS OIL HOT, located on the overhead dc circuit breaker panel (figure 1-39). The transmission oil temperature caution light will illuminate when the transmission oil temperature reaches 120°C at the main gear box inlet port. The different locations of the temperature sensors for the indicator and caution light allow the pilot to monitor the gear box operation by means of the indicator and the oil cooler operation by means of the caution light. Thus if a malfunction occurs in the oil cooler (blockage, fan belt failure, etc), the caution light will illuminate before the gear box oil temperature rises to a hazardous level.

Intermediate and Tail Gear Box Oil Systems.

Both the intermediate and the tail gear boxes are splash-lubricated from individual sump systems. Internal spiral channels ensure oil lubrication to all bearings. An oil filler plug, and oil level sight gage are located in each gear box casting. When the oil in the intermediate gear box is at FULL on the oil level sight gage, it contains 0.2 gallons. When the oil in the tail gear box is at FULL on the oil level gage, it contains 0.4 gallons.

FUEL SUPPLY SYSTEM.

The helicopters are equipped with two independent pressure-type, fuel systems (figure 1-25) that are joined by a crossfeed system to ensure maximum fuel utilization. An auxiliary fuel system, either an internal or external fuel tank system, is provided to augment the main tank fuel supply. The internal auxiliary fuel tank system may be installed on CH-3E helicopters prior to 16 . CH-3E 16▸ and all HH-3E helicopters are equipped

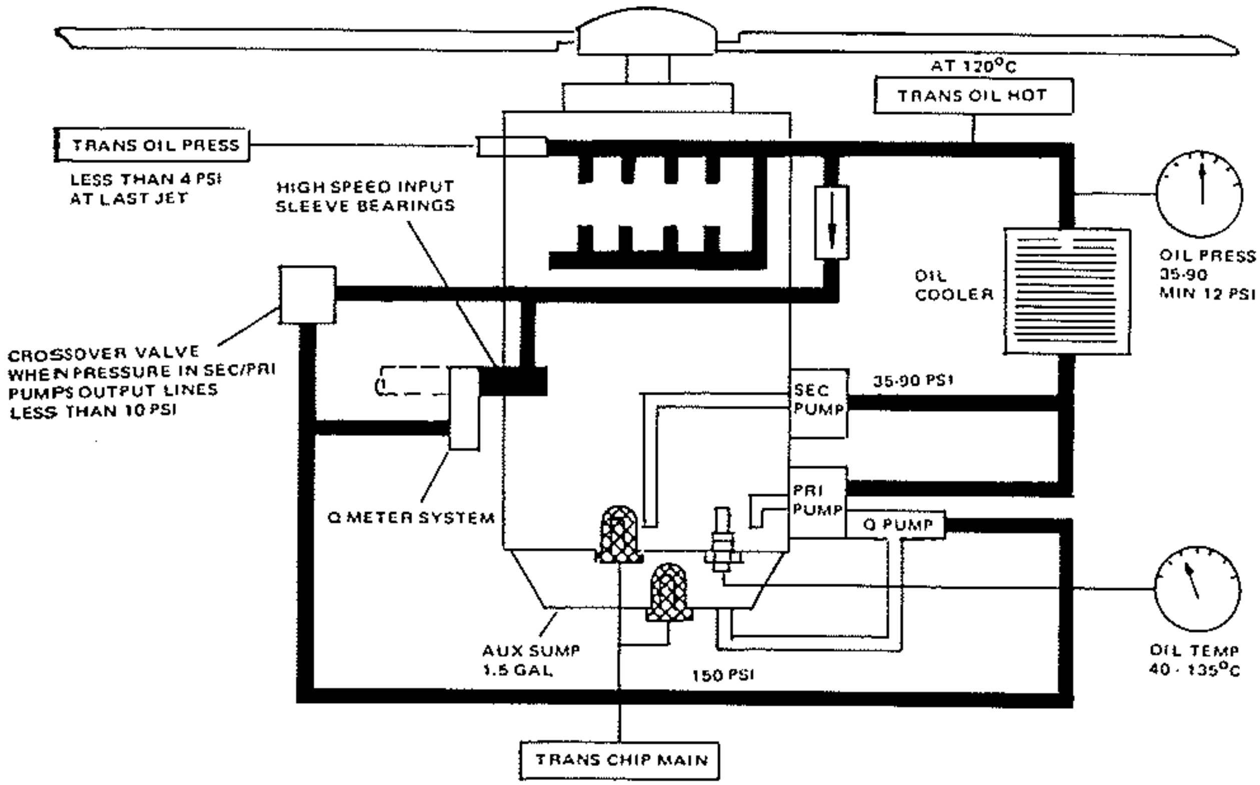


Figure 1-24. Main Transmission with Auxiliary Sump

with external auxiliary fuel tanks. HH-3E helicopters are equipped with a ground pressure and air refueling system and CH-3E helicopters [16]▸ are equipped with provisions for the ground pressure and air refueling systems. Those helicopters equipped with an auxiliary fuel system are also provided with a fuel dumping system. The component installation and procedures for supplying fuel from the main fuel tanks to the engines are the same for all model helicopters. The fuel boost pumps within each tank provide fuel to the appropriate engine and also to the fuel ejector unit within the tank. Fuel passing through the ejector unit creates a venturi effect which draws additional fuel from the tank into the ejector unit. Fuel is then pumped from the ejector unit into a collector can which surrounds the boost pumps within the fuel tank. The fuel ejector unit and boost pump arrangement provides integral fuel transfer within each tank, at all operating altitudes, and a minimum of unusable fuel. The crossfeed system is electrically controlled by a fuel crossfeed valve switch and allows fuel from both systems to be directed to one engine during single-engine operation. The fuel management panel, located on the instrument panel, controls the fuel systems. Fuel for the auxiliary power unit is supplied from the aft fuel tank, and fuel for the heater is supplied from the forward fuel tank.

FUEL TANKS.

The forward and aft main fuel tanks are located below the cargo compartment floor. The internal and

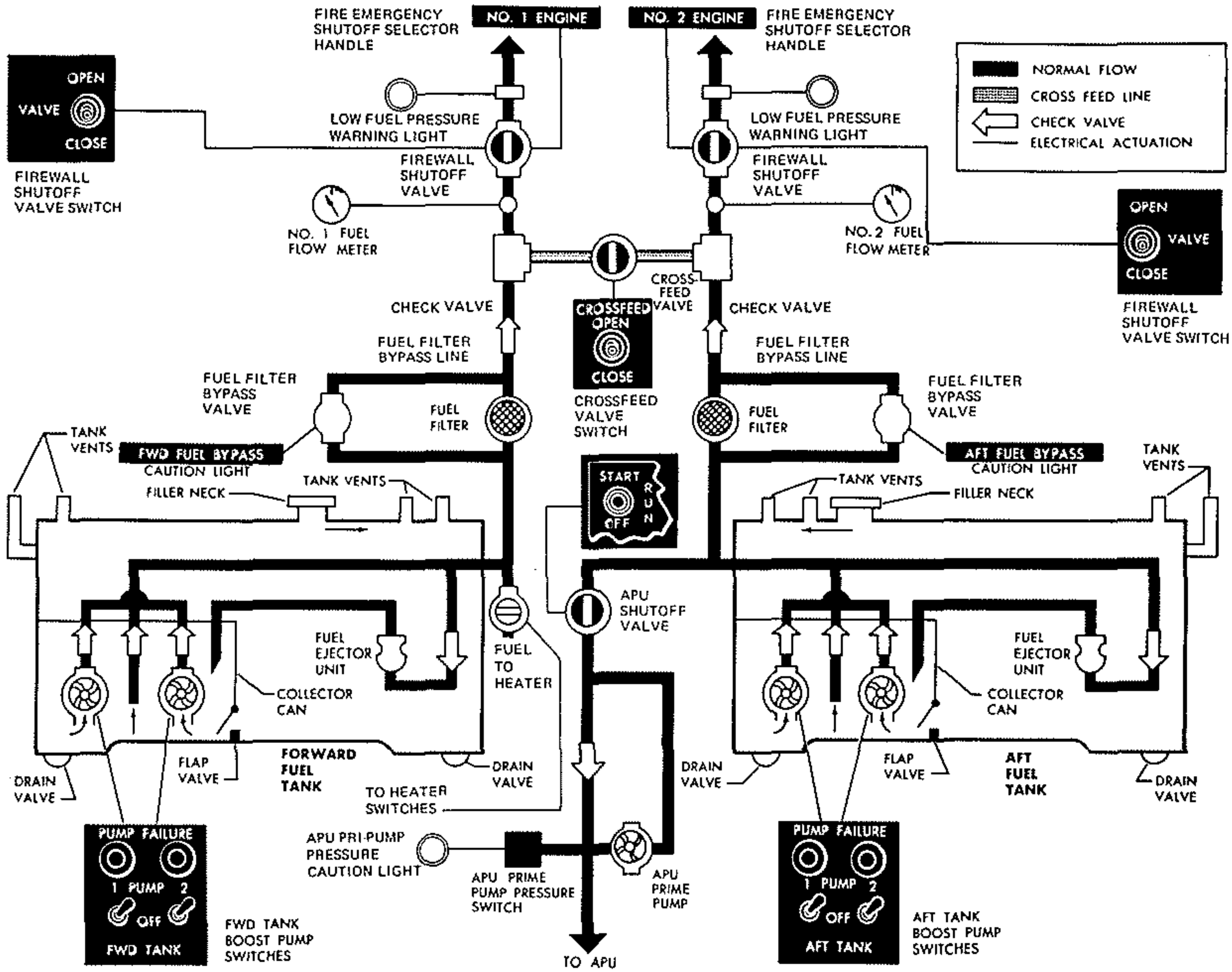


Figure 1-25. Helicopter Fuel System (CH-3E Helicopters Without External Auxiliary Tanks)

external auxiliary fuel tanks are discussed under the heading AUXILIARY FUEL SYSTEMS in this section. Each main fuel tank contains a collector can which surrounds two fuel boost pumps, a fuel ejector unit, vents, and sump drain valves. The main fuel tanks on helicopters prior to CH-3E 16 not modified by T.O. 1H-3(C)C-557 contain two bladder-type fuel cells. CH-3E 16▶ , all HH-3E, or helicopters modified by T.O. 1H-3(C)C-557, are equipped with two self-sealing type fuel cells in each main fuel tank. CH-3E and HH-3E helicopters modified by T.O. 1H-3-609 have polyurethane foam installed in the main fuel tanks. The material is 97 percent void and completely fills the internal volume of the tanks. With the foam material installed, it is impossible to have an explosion occur internally within the fuel tanks, regardless of the ignition source. It must be remembered that foam has no self-sealing capability, and its only function is to prevent internal explosion and to act as a baffle material. With foam installed the dry weight of the aircraft without external tanks is increased approximately 171 pounds. However, the gross weight is increased only 96 pounds on aircraft with external tanks and 22 pounds on aircraft without external tanks as 2.5 percent of the fuel capacity is displaced by the foam. See figure 1-30 for fuel quantity data with explosion suppression foam installed. The main fuel tanks of helicopters equipped with auxiliary fuel systems each contain a float valve. The float valves are actuated by the rising fuel in each tank to shut off the fuel from the auxiliary tanks when the main tanks are full. The float valves regulate the rate of flow from the auxiliary tanks, prevent overfilling the main tanks, and maintain a constant main fuel tank level until the auxiliary tanks are empty. The main fuel tanks of helicopters equipped with, or provisioned for, the ground pressure and refueling system are equipped with the fuel lines and components necessary to support the systems. The tank components installed in each tank for the ground pressure and air refueling systems are a pressure refueling shutoff valve, a high level shutoff sensor, and surge valves. The pressure refueling shutoff valves permit fuel flow when pressure refueling and are shut off by the high level shutoff sensors when the tanks are full. The high level shutoff sensors are actuated by the rising fuel in each tank to shut off fuel flow when the main tanks are full. The high level shutoff sensors operate during ground pressure and air inflight refueling, or when fuel is being transferred from the external auxiliary fuel tanks to the main fuel tanks. The surge valves prevent damage from pressure surges by relieving excessive pressure. The vents are routed through the cargo compartment to minimize vent icing, then to both sides of the helicopter where the cells are vented to the atmosphere. The sump drain valves are manually operated to drain water from the system. The defueling valves provide for complete drainage of the system. Both fuel tanks are gravity filled through two filler caps located on the left side of the helicopter; pressure refueled through the pressure refueling adapter, located on the lower fuselage below the entrance of the cargo compartment; the air refueling probe, located on the right side of the forward fuselage, or from the auxiliary fuel systems.

FUEL SHUTOFF VALVE SWITCHES.

The two fuel shutoff valve switches, marked VALVE, are located on the fuel management panel (figure 1-26). The switch, marked NO. 1 ENG with marked positions OPEN and CLOSE, controls the flow of fuel to the No. 1 engine. The switch, marked NO. 2 ENG with marked positions OPEN and CLOSE, controls the flow of fuel to the No. 2 engine. The switches control the fuel shutoff valves located on top of the cargo compartment before the engine compartment. Placing either switch in the CLOSE position shuts off the flow of fuel to the appropriate engine. The fuel shutoff valves are also actuated to the closed position when the appropriate fire emergency shutoff selector handle is pulled. The fire emergency shutoff selector handles, marked FIRE EMER SHUTOFF SELECTOR, located on the overhead control panel, arm the fire extinguisher circuit in addition to actuating the fuel shutoff valves. In the event of electrical failure, the fuel shutoff valves will remain in the last switch position energized. The fuel shutoff valves are provided with a fail safe capability. The fail safe capability prevents the valves from changing position should a possible malfunction occur whereby both the open and close circuits of the valve are energized simultaneously. The fuel shutoff valves and switches operate on current from the dc essential bus and are protected by circuit breakers marked EMER SHUTOFF 1- ENG-2 under the general headings FUEL SYSTEM and VALVES, located on the overhead dc circuit breaker panel.

FUEL CROSSFEED VALVE SWITCH.

The fuel crossfeed valve switch, marked CROSSFEED, is located on the fuel management panel (figure 1-26). The switch has marked positions

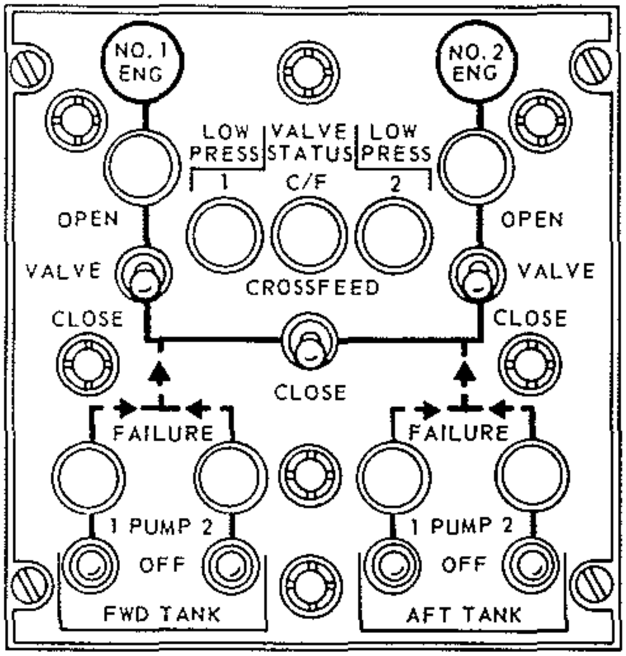


Figure 1-26. Fuel Management Panel

OPEN and CLOSE and controls the fuel crossfeed valve located on top of the cargo compartment between the two fuel systems supply lines in the crossfeed line. With the switch in the CLOSE position, the forward fuel tank supplies fuel to the No. 1 engine and the aft fuel tank supplies the No. 2 engine. When the switch is placed in the OPEN position, the crossfeed valve opens and allows fuel under pressure to be supplied from both fuel tanks to either or both engines. The crossfeed system does not transfer between tanks. The fuel crossfeed valve operates on direct current from the dc essential bus and is protected by a circuit breaker marked X FEED, under the general headings FUEL SYSTEM and VALVES, located on the overhead dc circuit breaker panel.

FUEL SHUTOFF AND CROSSFEED VALVE STATUS LIGHTS.

Valve status warning lights for each shutoff valve and the crossfeed valve are located on the fuel management panel (figure 1-26). The lights, marked 1, 2, and C/F, under the heading VALVE STATUS, illuminate to provide an indication of the status of the shutoff and crossfeed valves. The appropriate light will be illuminated when a valve is being actuated from one position to another, if a protective relay is inoperative when electrical power is applied to the helicopter, or if both the closing and opening circuits should become simultaneously energized. The valve status lights use the same power source and protective circuit as the low fuel pressure warning lights.

LOW FUEL PRESSURE WARNING LIGHTS.

Two low fuel pressure warning lights, marked #1 LOW PRESS, #2 LOW PRESS, are located on the fuel management panel (figure 1-26). The warning lights will illuminate whenever fuel pressure drops below minus 5 psi at the pressure switch, located between the fuel shutoff valve and the engine. The intensity of the lights may be varied by rotating a rheostat under the heading CONSOLES, marked LOWER, located on the overhead switch panel. The warning lights receive electrical power from the dc essential bus through circuit breakers, under the general heading INDICATOR LTS and marked FUEL PRESS 1 and 2, located on the overhead dc circuit breaker panel.

FUEL BOOST PUMPS.

Two fuel boost pumps are located in the forward end of each main fuel tank. Each tank has a No. 1 boost pump and a No. 2 boost pump, powered by separate electrical circuits. The No. 1 boost pump in each tank is powered by current from the ac essential bus, while the No. 2 boost pump in each tank is powered by current from the ac nonessential bus. Control of all boost pumps is provided by switches which operate on current from the dc essential bus. The No. 1 boost pumps are protected by circuit breakers, marked FWD TANK and AFT TANK and under the general heading No. 1 FUEL PUMP, located on the ac essential bus circuit breaker panel. The No. 2 boost pumps are protected by circuit breakers, marked FWD TANK and AFT TANK and under the general heading No. 2 FUEL PUMP, located on the ac nonessential bus circuit breaker panel. The engine may be operated using one, both or no boost pumps.

Fuel Boost Pump Switches.

Four boost pump switches, grouped according to fuel tank designations, are located on the fuel management panel (figure 1-26). The two boost pump switches for the pumps in the forward tank are marked FWD TANK 1 and 2, and those for the aft tank AFT TANK 1 and 2. The number designation, whether 1 or 2, indicates the boost pump controlled by the particular switch. Each switch has marked positions PUMP (ON) and OFF. All boost pump switches are connected to the dc essential bus through circuit breakers, marked FWD TANK 1 and 2, AFT TANK 1 and 2, under the general headings FUEL SYSTEM and PUMP CONT, located on the overhead dc circuit breaker panel. When the switches are placed in the PUMP (ON) position, dc power from the dc essential bus closes relays in the circuit between the appropriate ac essential bus and the respective boost pump. The OFF position deenergizes the relays, cutting off ac power to the respective boost pump.

Fuel Boost Pump Failure Warning Lights.

Each of the four boost pumps is provided with a pressure switch that is connected to the pressure feed line from each boost pump. The fuel pressure switches actuate the boost pump failure warning lights, marked FAILURE, located on the fuel management panel (figure 1-26), when the boost pump pressure falls below a safe operating pressure. Each boost pump is provided with an individual boost pump failure warning light located above the respective boost pump switch. The boost pump failure warning lights should illuminate and then go off when the boost pumps are first turned on, or when the boost pump switches are being tested. They are illuminated until fuel pressure is built up in the system. The pressure switches close if the boost pressure decreases to, or is below, approximately 16 1/2 psi and energizes the respective boost pump failure warning light circuit which lights the boost pump failure warning light. The boost pump pressure switches and failure warning lights operate from the dc essential bus and are protected by circuit breakers, marked FWD TANK, 1 and 2, AFT TANK, 1 and 2 and under the general headings INDICATOR LIGHTS and FUEL PUMPS, located on the overhead dc circuit breaker panel.

FUEL QUANTITY GAGES AND TEST SWITCHES.

The fuel quantity gages (figure 1-14), located on the instrument panel, indicate the fuel quantity in each tank in pounds. Fuel quantities are shown in figures 1-27, 1-28, 1-29 and 1-30. The fuel quantity indicating system may be tested by pressing the fuel gage test switches, marked FUEL GAGE TEST, FWD TANK AFT TANK, located between the fuel quantity gages on the fuel quantity gage test switch panel (figure 1-14). Pressing either button-type switch for approximately 10 seconds will induce a current reversal which causes the pointer to drop below zero. Upon release of the test switch, the normal current should cause the pointer to return to the previous reading. The test shows that the fuel quantity indicating system is operating correctly. The fuel quantity indicating system normally operates on 115 volt ac current from ac essential bus is protected by circuit breakers, marked FWD and AFT and under the general headings FUEL and QTY, located on the ac essential bus circuit breaker panel, but is operated by ac current from the inverter until the ac essential bus is energized.

FUEL LOW LEVEL CAUTION LIGHTS.

The fuel low level caution lights, marked FWD FUEL LOW and AFT FUEL LOW, located on the caution panel (figure 1-20), will illuminate when approximately 140 to 190 pounds of fuel remain in the respective fuel tank depending upon the aircraft attitude. The caution lights operate on current from the dc essential bus, through circuit breakers marked LOW LEVEL, FWD and AFT, and under the general headings INDICATOR LIGHTS and FUEL, located on the overhead dc circuit breaker panel.

FUEL FILTER BYPASS CAUTION LIGHTS.

The fuel filter bypass caution lights, marked FWD FUEL BYPASS and AFT FUEL BYPASS, are located on the caution panel (figure 1-20). The fuel filter bypass caution light will illuminate whenever the respective system filter screen has become clogged and the fuel bypasses the fuel filter of the respective fuel tank. The caution lights are tested by the master TEST button on the caution panel and operate on dc current through circuit breakers, marked FWD and AFT and under the headings INDICATOR LIGHTS and FUEL BYPASS, located on the overhead dc circuit breaker panel.

AUXILIARY FUEL SYSTEMS.

The auxiliary fuel systems consist of an internal auxiliary fuel tank system and an external auxiliary fuel tank system. The internal auxiliary fuel tank system may be installed on CH-3E helicopters

FUEL QUANTITY DATA
(JP-4)

NONSELF-SEALING TANKS

	USABLE		UNUSABLE		FULLY SERVICED	
FUEL TANKS	US GALLONS	POUNDS	US GALLONS	POUNDS	US GALLONS	POUNDS
FWD MAIN	342.0	2223.0	2.0	13.0	344	2236.0
AFT MAIN	345.0	2242.5	1.0	6.5	346	2249.0
TOTAL MAIN	687.0	4465.5	3.0	19.5	690	4485.0
ONE AUXILIARY TANK	437	2840.5	4.00	26.0	441	2866.5
TWO AUXILIARY TANKS	874	5681.0	8.00	52.0	882	5733.0

TOTAL USABLE FUEL	WITHOUT INTERNAL AUXILIARY TANKS	687.0 GAL.	4465.5 LB.
TOTAL USABLE FUEL	WITH ONE INTERNAL AUXILIARY TANK	1124.0 GAL.	7306.0 LB.
*TOTAL USABLE FUEL	WITH TWO INTERNAL AUXILIARY TANKS	1561.0 GAL.	10,146.5 LB.

NOTES

1. USABLE FUEL DETERMINED AT 1 DEGREE NOSEDOWN ATTITUDE.
2. FUEL DENSITY OF 6.5 LB/GAL AT STANDARD DAY TEMPERATURE.
3. THE SINGLE AUXILIARY FUEL TANK INSTALLED WEIGHS 332 POUNDS.
   THE DUAL AUXILIARY FUEL TANKS INSTALLED WEIGH 612 POUNDS.
   *TOTAL USABLE FUEL, WITH TWO AUXILIARY FUEL TANKS IN USE, MUST BE ADJUSTED FOR TAKEOFF GROSS WEIGHT.
4. THE INSTALLATION VARIATIONS OF EACH FUEL TANK MAY CAUSE THE TOTAL CAPACITY OF A TANK TO VARY AS MUCH AS ±3 GALLONS.

Figure 1-27. Fuel Quantity Data

FUEL QUANTITY DATA
(JP-4)

SELF-SEALING TANKS

FUEL TANKS	USABLE US GALLONS	USABLE POUNDS	UNUSABLE US GALLONS	UNUSABLE POUNDS	FULLY SERVICED US GALLONS	FULLY SERVICED POUNDS
FWD MAIN	333.0	2164.5	2.0	13.0	335	2177.5
AFT MAIN	334.0	2171.0	1.0	6.5	335	2177.5
TOTAL MAIN	667.0	4335.5	3.0	19.5	670	4355.0
ONE AUXILIARY TANK	437	2840.5	4.00	26.0	441	2866.5
TWO AUXILIARY TANKS	874	5681.0	8.00	52.0	882	5733.0

TOTAL USABLE FUEL	WITHOUT INTERNAL AUXILIARY TANKS	667.0 GAL.	4335.5 LB.
TOTAL USABLE FUEL	WITH ONE INTERNAL AUXILIARY TANK	1104.0 GAL.	7176.0 LB.
*TOTAL USABLE FUEL	WITH TWO INTERNAL AUXILIARY TANKS	1541.0 GAL.	10,016.5 LB.

NOTES

1. USABLE FUEL DETERMINED AT 1-1/2 DEGREE NOSEUP ATTITUDE.
2. FUEL DENSITY OF 6.5 LB/GAL AT STANDARD DAY TEMPERATURE.
3. THE SINGLE AUXILIARY FUEL TANK INSTALLED WEIGHS 332 POUNDS. THE DUAL AUXILIARY FUEL TANKS INSTALLED WEIGH 612 POUNDS. *TOTAL USABLE FUEL, WITH TWO AUXILIARY FUEL TANKS IN USE, MUST BE ADJUSTED FOR TAKEOFF GROSS WEIGHT.
4. THE INSTALLATION VARIATIONS OF EACH FUEL TANK MAY CAUSE THE TOTAL CAPACITY OF A TANK TO VARY AS MUCH AS ±3 GALLONS.

Figure 1-28. Fuel Quantity Data

ESTIMATED
FUEL QUANTITY DATA
(JP-4)

CONFIGURED FOR EXTERNAL AUXILIARY FUEL TANK INSTALLATION

	USABLE		UNUSABLE		FULLY SERVICED	
FUEL TANKS	US GALLONS	POUNDS	US GALLONS	POUNDS	US GALLONS	POUNDS
FWD MAIN	311.02 (337.52)	2021.16 (2193.9)	3.48	22.6	314.5 (341)	2044.2 (2216.5)
AFT MAIN	313.3 (307.4)	2034.4 (1998.1)	2.60	16.9	315.9 (310)	2051.3 (2015)
TOTAL MAIN	624.32 (644.92)	4056 (4192)	6.08	39.5	630.4 (651)	4095.5 (4231.5)
AUXILIARY TANKS	386.7 (398)	2523.5 (2587)	2	13	388.3 (400)	2523.9 (2600)
TOTAL USABLE FUEL		WITHOUT EXTERNAL AUXILIARY TANKS		624.32 GAL. (644.92)	4056 LB. (4192)	
TOTAL USABLE FUEL		WITH EXTERNAL AUXILIARY TANKS		1011.02 GAL. (1043.32)	6579.5 LB. (6781.6)	

NOTES

1. FUEL CAPACITIES DETERMINED AT 1-1/2 DEGREE NOSEUP ATTITUDE. QUANTITIES SHOWN IN PARENTHESES ( ) ARE FOR GRAVITY REFUELING AND OTHER QUANTITIES ARE FOR PRESSURE REFUELING.
2. USABLE FUEL DETERMINED AT 1 DEGREE NOSEDOWN ATTITUDE FOR BOTH PRESSURE AND GRAVITY REFUELING.
3. FUEL DENSITY OF 6.5 LB/GAL AT STANDARD DAY TEMPERATURE
4. JP-8 FUEL DENSITY OF 6.7 LB/GAL AT STANDARD DAY TEMPERATURE.
5. THE EMPTY EXTERNAL AUXILIARY FUEL TANKS WEIGH 94 POUNDS.
6. AS THE AMOUNT OF FUEL THAT CAN BE RECEIVED IN THE AUXILIARY FUEL TANKS VARIES GREATLY DUE TO THE HIGH LEVEL SHUTOFF SENSING SYSTEM, IT WILL BE NECESSARY FOR THE PILOT TO REQUEST THE TOTAL AMOUNT OF FUEL TRANSFERRED BY THE TANKER TO DETERMINE THE TOTAL AMOUNT OF FUEL AVAILABLE. TO DATE, DATA HAS SHOWN THAT THE AMOUNT OF FUEL RECEIVED VARIES FROM 150 TO 180 GALLONS PER TANK.
7. THE INSTALLATION VARIATIONS OF EACH FUEL TANK MAY CAUSE THE TOTAL CAPACITY OF A TANK TO VARY AS MUCH AS ±3 GALLONS.

Figure 1-29. Fuel Quantity Data (Configured for Auxiliary Fuel Tank Installation)

FUEL QUANTITY DATA
(JP 4)

EXPLOSION SUPPRESSION FOAM INSTALLED

FUEL TANKS	USABLE US GALLONS	USABLE POUNDS	UNUSABLE US GALLONS	UNUSABLE POUNDS	FULLY SERVICED US GALLONS	FULLY SERVICED POUNDS
FWD MAIN	330.2 (325.1)	2146.3 (2113.2)	2.9 (10.9)	18.9 (70.8)	337.7 (336.0)	2169 (2184)
AFT MAIN	297.0 (323.6)	1930.5 (2103.4)	4.4 (14.4)	28.6 (93.6)	303.7 (338)	1974 2197
TOTAL MAIN	627.2 (648.7)	4076.8 (4216.0)	7.3 (25.3)	47.5 (164.4)	637.4 (674)	4143 (4381)
AUXI-LIARY TANKS	386.7 (398)	2513.5 (2587)	2	13	388.3 (400)	2523.9 (2600)

(No foam installed in External Auxiliary Tanks)

		US GALLONS	POUNDS
TOTAL USABLE FUEL	WITHOUT EXTERNAL AUXILIARY TANKS	627.2 (648.7)	4076.8 (4216.0)
TOTAL USABLE FUEL	WITH EXTERNAL AUXILIARY TANKS	1013.9 (1046.7)	6590.3 (6803.0)

NOTES

1. FUEL CAPACITIES DETERMINED AT 1-1/2 DEGREE NOSEUP ATTITUDE. QUANTITIES SHOWN IN PARENTHESES ( ) ARE FOR GRAVITY REFUELING AND OTHER QUANTITIES ARE FOR PRESSURE REFUELING.
2. USABLE FUEL DETERMINED AT 1 DEGREE NOSEDOWN ATTITUDE.
3. FUEL DENSITY OF 6.5 LB/GAL AT STANDARD DAY TEMPERATURE.
4. ONE EMPTY EXTERNAL AUXILIARY FUEL TANK WEIGHS 94 POUNDS.
5. AS THE AMOUNT OF FUEL THAT CAN BE RECEIVED IN THE AUXILIARY TANKS VARIES GREATLY DUE TO THE HIGH LEVEL SHUTOFF SENSING SYSTEM, IT WILL BE NECESSARY FOR THE PILOT TO REQUEST THE TOTAL AMOUNT OF FUEL TRANSFERRED BY THE TANKER TO DETERMINE THE TOTAL AMOUNT OF FUEL AVAILABLE. TO DATE, DATA HAS SHOWN THAT THE AMOUNT OF FUEL RECEIVED VARIES FROM 150 TO 180 GALLONS PER TANK.

Figure 1-30. Fuel Quantity Data (Helicopters Modified by T.O. 1H-3-609)

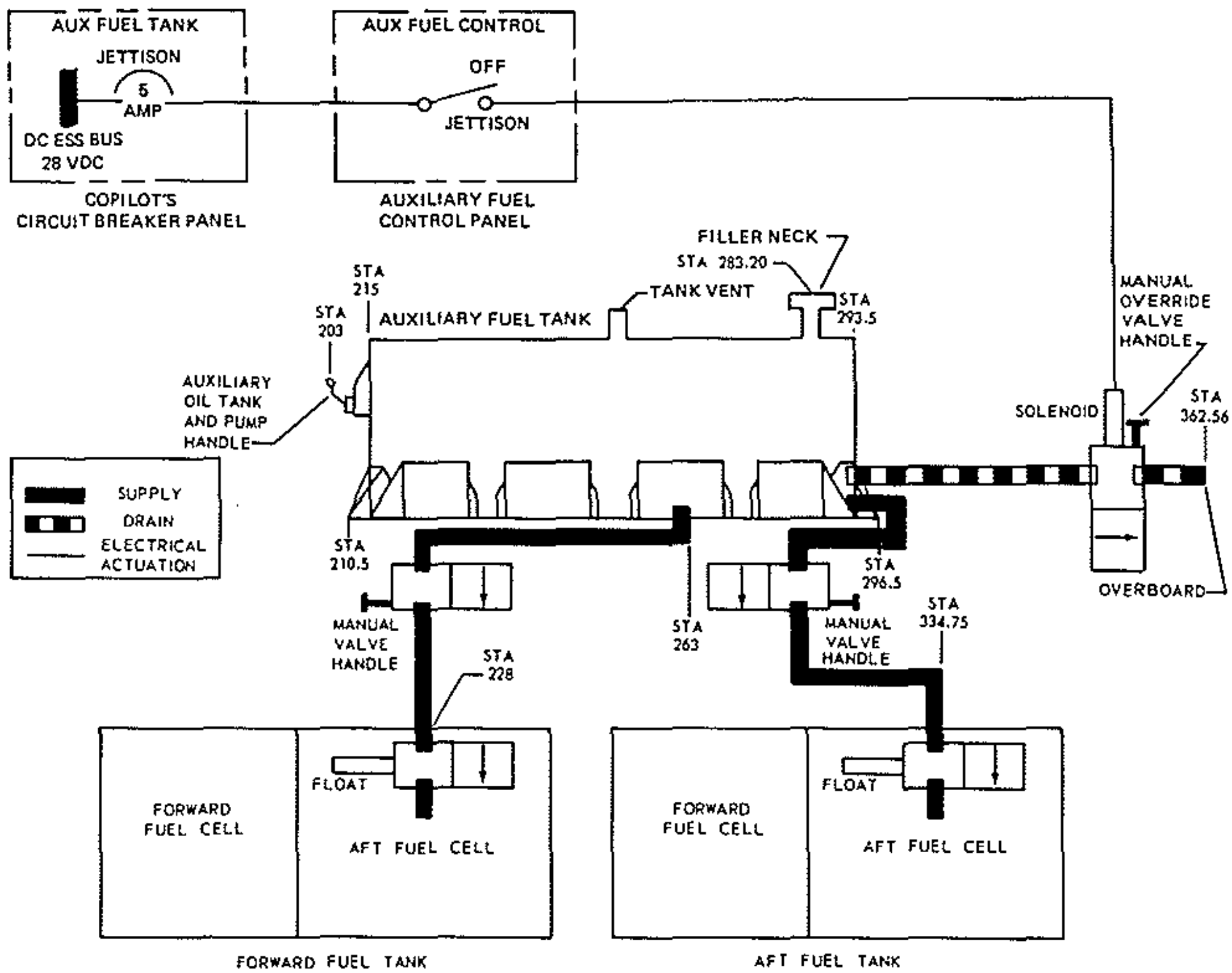


Figure 1-31. Single Internal Tank Auxiliary Fuel System Diagram

prior to 16 . CH-3E 16▸ and all HH-3E helicopters are equipped with the external auxiliary fuel tank system. See figures 1-27, 1-28, 1-29 and 1-30 for fuel quantity data.

Internal Auxiliary Fuel Tank System.

The internal auxiliary fuel tank system, either a single or dual tank system, may be installed to increase the range and endurance of the helicopter. The single tank auxiliary fuel system (figure 1-31) consists of a tank that is mounted on the cargo floor, float valves, an auxiliary fuel filler neck, provisions for fuel venting, and electrical provisions for fuel dumping. A fuel valve, located in each fuel transfer line, allows fuel from the auxiliary fuel tank to be gravity transferred to either or both of the main fuel tanks. The dual tank auxiliary fuel system (figure 1-33) is a combination of two single tank systems with modified plumbing and double the fuel capacity. In the dual tank installation, the forward auxiliary fuel tank is moved forward to allow an auxiliary tank to be installed over each main tank.

Internal Auxiliary Fuel Tanks.

The fiberglass internal auxiliary fuel tanks are the same for either the single or dual installation. The

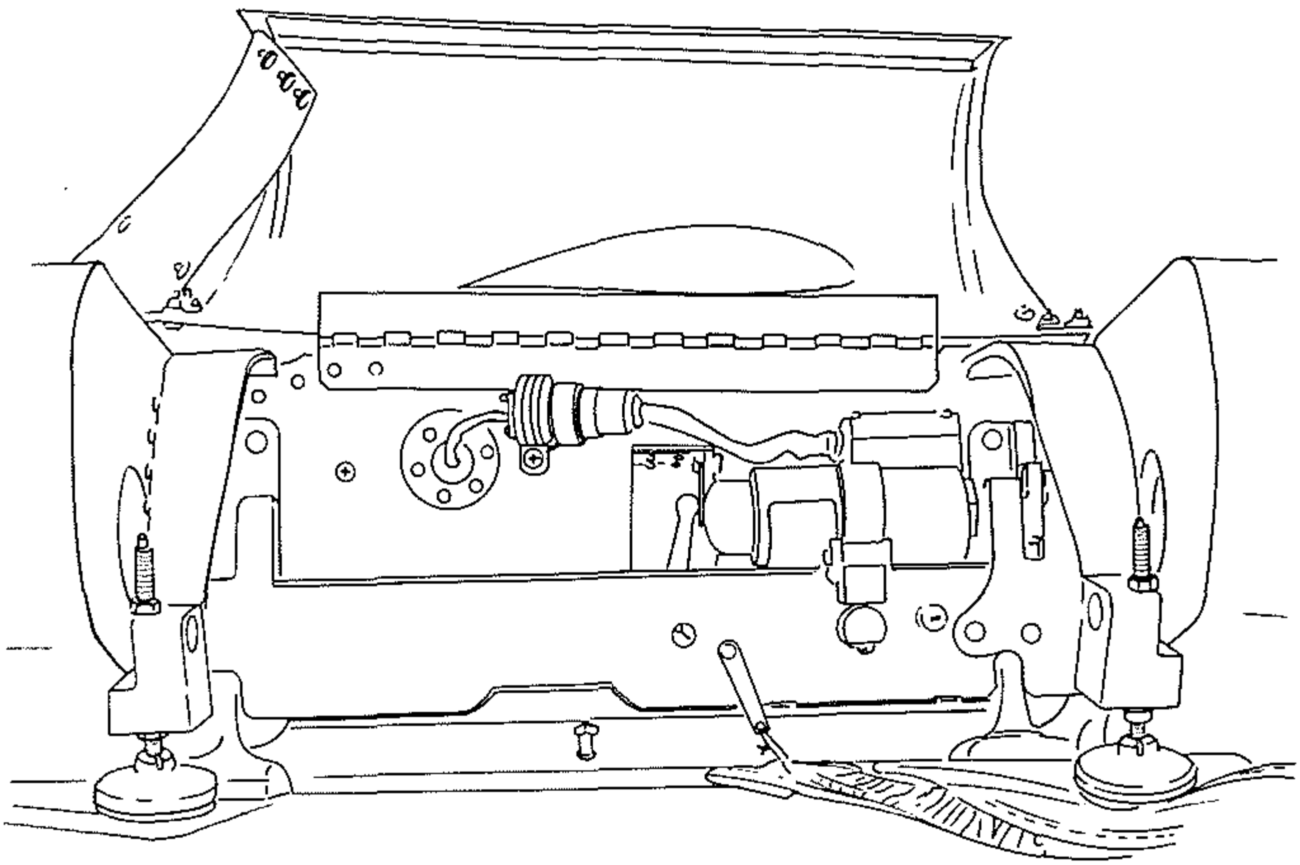


Figure 1-32. Proper External Auxiliary Fuel Tank To Bomb Shackle Installation

dual auxiliary fuel system uses two single tank systems, but only one outlet per tank. When using the single tank system, two fuel transfer lines, each equipped with a float valve, are used to supply fuel to the helicopter's forward and aft fuel tanks. When using the dual tank system, each auxiliary tank is equipped with one fuel transfer line with a float valve. The forward auxiliary fuel tank provides fuel to the helicopter's forward fuel tank and aft auxiliary fuel tank provides fuel to the helicopter's aft fuel tank. Each tank is also equipped with a fuel jettison line and dump valve. Refer to FUEL DUMPING SYSTEMS in the section for procedures to jettison internal auxiliary fuel. A single auxiliary empty fuel tank installed weighs 332 pounds while the dual auxiliary empty fuel tanks installed weigh 612 pounds. An auxiliary tank support displaces a floor area of 7 feet 2 inches by 4 feet 6 inches. The tank, or tanks, are vented to the atmosphere and are filled through an external filler cap located on the right side of the helicopter. The dual tank system is filled through the rear tank only. The auxiliary fuel tanks are secured to the cargo compartment floor tiedown rings, the single tank using 12 tiedowns and the dual tanks using 24 tiedowns.

Internal Auxiliary Fuel System Manual Shutoff Valves.

The two auxiliary fuel system manual shutoff valves located between the floor and the auxiliary fuel tank, may be opened any time auxiliary fuel is desired or required.

Float Valves.

The fuel transfer lines are equipped with a float valve which prevents overfilling of the main fuel tanks. The float valves are actuated by the rising

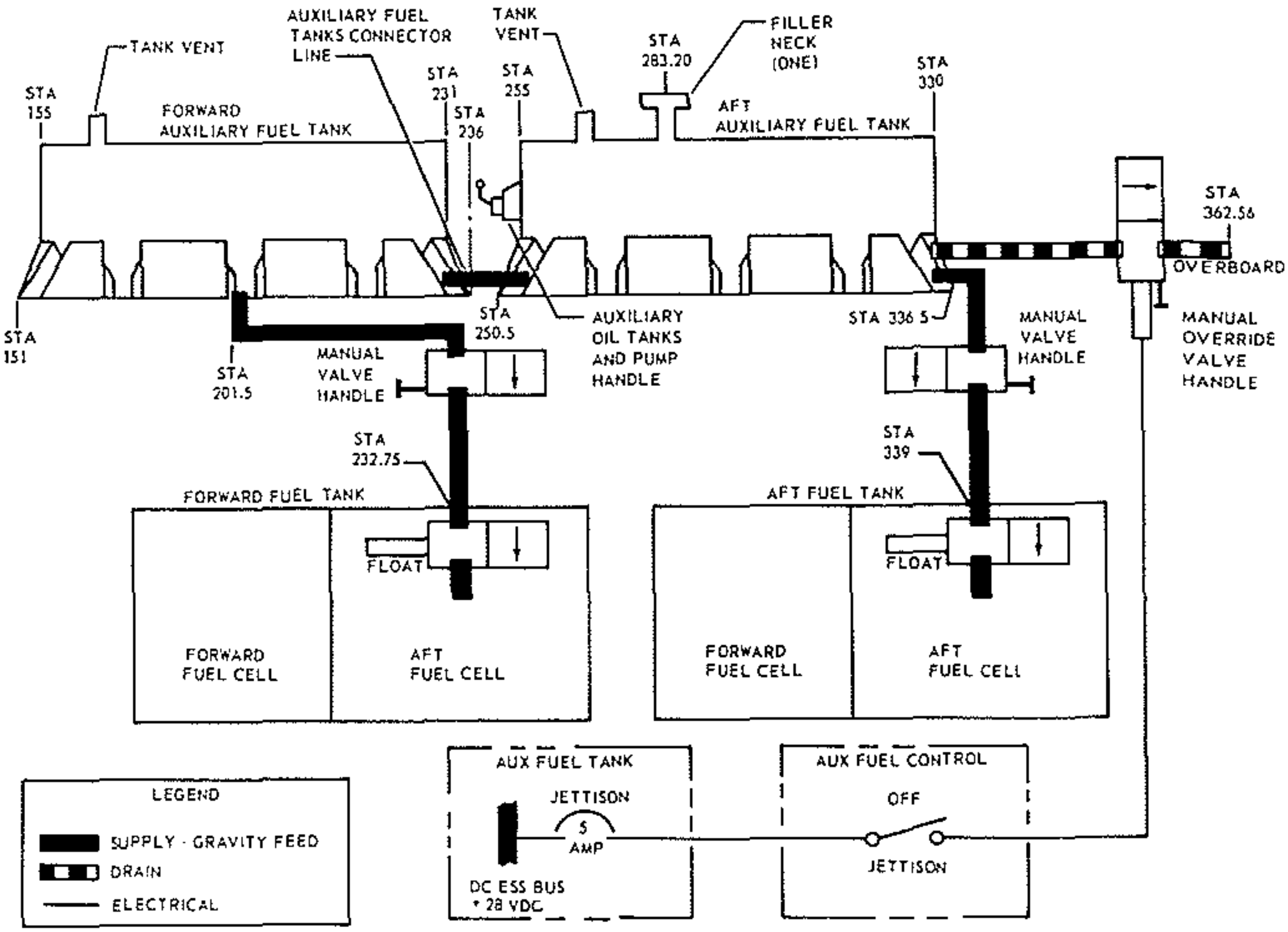


Figure 1-33. Dual Internal Auxiliary Fuel System Diagram

fuel level in the main tanks and shut off when the main tanks are full.

External Auxiliary Fuel System.
(See figure 1-35).

The external auxiliary fuel system is installed to increase the range or endurance of the helicopter. The system does not provide fuel directly to the engine, but functions to replenish fuel into the main fuel tanks. The system consists of an external auxiliary fuel tank, attached outboard of each sponson, an external auxiliary fuel tank pressurization system, a bomb rack with attaching and release mechanisms, and a control panel. The external auxiliary fuel tanks may be electrically or mechanically jettisoned. The tanks may be gravity fueled or serviced through the ground pressure and/or air refueling systems. Fuel is transferred to the main fuel tanks at a rate greater than dual engine consumption by pressurizing the auxiliary fuel tanks with engine compressor bleed air. The tanks may be simultaneously pressurized by either or both engines. The system operates on 28 volts dc from the dc essential bus. The circuit for each tank pressurization system is protected by circuit breakers, marked LH and RH and under the headings BLEED AIR and EXT AUX FUEL TANK, located on the copilot's circuit breaker panel. On CH-3E 16▸ and all HH-3E helicopters, the tank pressurization system and tank fuel valves are protected by circuit breakers, LH and RH, and under the heading FUEL MANAGE, also located on the copilot's circuit breaker panel. The jettison circuits are protected by circuit breakers, marked LH and RH, under the headings JETTISON and EXT AUX FUEL TANK, also located on the copilot's circuit breaker panel.

NOTE

The MASTER POWER switch, located on the pressure refueling panel, must be placed in the OFF position when transferring fuel from the auxiliary to main tanks.

External Auxiliary Fuel Tanks.

Each external auxiliary fuel tank (figure 1-2) weighs 94 pounds empty, has a capacity of 200 gallons, and is interchangeable for use on either sponson pylon. The tanks, one mounted outboard of each sponson pylon, are attached to bomb racks and stabilized by sway braces. When properly installed, the release lever is positioned at an angle toward the electrical solenoid. The manual release striker plate will be positioned between the release lever and the electrical solenoid. The electrical solenoid will be in the cocked position with approximately one inch of solenoid plunger showing. (See figure 1-32.) The mounting attitude of the tanks provides expansion space while refueling in a normal ground attitude as well as maximum usable fuel during cruise. The tanks may be individually or simultaneously jettisoned electrically, and simultaneously jettisoned by the mechanical release handle. Self-sealing quick disconnects are provided at each tank line to seal off the appropriate line when the tanks are jettisoned. The tanks are pressurized by engine compressor bleed air to force the fuel into the main fuel tanks and are refueled through filler caps located on the top of the tanks. When pressure exceeds approximately 13 psi, the tanks are vented through pressure relief valves in the compressor air inlet line. When the auxiliary fuel tanks are empty, compressor bleed air will continue to flow through the auxiliary fuel tanks into the main fuel tanks, where it is vented out the main fuel tank vents until the bleed air shutoff valves are closed. The bleed air shutoff valves may be closed when the auxiliary tanks are empty, as indicated by decreasing main fuel tank levels on the fuel quantity indicating system. Each auxiliary fuel tank is equipped with a thermistor and tank vent valve. The thermistors operate in conjunction with the ground pressure and air refueling systems to indicate when the tanks are full and to then close the auxiliary tank fuel valves.

NOTE

- The external auxiliary fuel tank pressurization should be turned off when tanks are empty to prevent any possible loss of engine performance.
- External auxiliary fuel tanks are not self-sealing or bulletproof and constitute a potential hazard when exposed to small arms fire, and should be jettisoned, particularly when containing unusable fuel.

Fuel Tank Pressurization System.

The auxiliary fuel tanks are pressurized by engine compressor bleed air. The pressurization systems contain check valves, a pressure regulator, shutoff valves, and pressure relief valves. The crossfeed configuration permits both tanks to be simultaneously pressurized by either or both engines. The pressure regulator maintains a pressure of 10 psi to the shutoff valves when either or both engines are operating. Opening the shutoff valves will permit the compressor bleed air to pressurize respective tanks. The pressure relief valve will vent excessive pressures above 13 psi (input compressor air on accumulated tank pressures) overboard. Each shutoff valve is equipped with a vent to relieve all tank pressure when the valve is closed. The shutoff valves are controlled by switches, located on the auxiliary fuel control panel.

Auxiliary Fuel Control Panel.

The auxiliary fuel control panel, located on the cockpit console (figure 1-18), contains all switches for operation of the pressurization system and electrically jettisoning the external auxiliary fuel tanks. The switches that are appropriate to the left tank are under the general heading L TK, and the switches that are appropriate to the right are under the general heading R TK. The respective pressurization switches, with marked positions PRESS and OFF, control the pressurization shutoff valves and auxiliary tank fuel valves. Placing a pressurization switch in the PRESS position will open the appropriate shutoff valve permitting the respective tank to be pressurized and simultaneously open the auxiliary tank fuel valve permitting fuel to transfer to the main tanks. When a switch is placed in the OFF position the bleed air shutoff valve and auxiliary tank fuel valve is closed. The three guarded switches, all marked JETTISON, are used to electrically jettison the auxiliary fuel tanks. When the switch under the general heading L TK is actuated, the left auxiliary fuel tank will be electrically released. When the switch under the general heading R TK is actuated, the right auxiliary fuel tank will be electrically released. Actuating the switch under the general heading BOTH TKS will simultaneously electrically release both tanks.

Auxiliary Fuel Tank Manual Release Handle.

The manual release handle, located on the cockpit console (figure 1-18), is actuated to simultaneously mechanically release both external auxiliary fuel tanks.

FUEL DUMPING SYSTEMS.

The fuel dumping systems consist of an internal auxiliary fuel tank dumping system, a manually controlled fuel dumping system, and a rapid fuel dumping system.

Internal Auxiliary Fuel Tank Dumping System.

Those CH-3E helicopters prior to **16** equipped with internal auxiliary fuel tanks, have the capability of dumping auxiliary fuel at a rate of approximately 560 pounds per minute in level flight. Each auxiliary fuel tank is equipped with a fuel jettisoning (dump) line and an electrically operated dump valve that has a manual override. The dump valve is activated by movement of a toggle switch located on the auxiliary fuel control panel. Fuel is dumped through a dump port, located at station 326 in the right sponson, and exits from the lower rear.

Auxiliary Fuel Jettison (Dump) Switch.

The toggle type auxiliary fuel jettison switch, marked JETTISON and OFF, is located on the auxiliary fuel tank control panel on the cockpit console (figure 1-18). When the switch is placed in the JETTISON position, fuel is dumped overboard through the dump valve. The jettison switch is powered from the dc essential bus through a circuit breaker marked JETTISON and under the general heading AUX FUEL CONTROL, located on the dc circuit breaker panel above the copilot.

Manually Controlled Fuel Dumping System.

Helicopters modified by T.O. 1H-3-505 are equipped with a manually controlled fuel dumping system which permits fuel to be dumped from the forward main fuel tank at a rate of 150 pounds per minute in level flight from 70 knots to V max knots, during descents through full autorotation at 100 knots, and during water taxi. The system consists of manual fuel close line and dump valves, located in the cargo compartment, plus some additional hosing. The dump hosing extends up from the tank through the cargo compartment and out through the outboard tail end of the right sponson. The manual fuel dump system must be used in conjunction with the boost pumps and crossfeed system. All boost pumps must be ON, the crossfeed OPEN, and engines operating, before fuel dumping can be attempted.

CAUTION

The manual fuel dumping system will dump the entire fuel load from the forward tank if not monitored. The system uses the existing fuel boost pumps in the forward tank and does not provide the protection of 500 pounds reserve in each tank that the rapid fuel dumping system provides.

NOTE

- When the manual fuel dump system valve is in operation, fuel boost pump failure lights and fuel filter bypass lights may illuminate. This is a normal condition caused by a drop in prime fuel pressure and the resultant pressure differential across the fuel filters. The fuel boost pump failure lights should go out when the fuel pressure stabilizes upon releasing the manual fuel dump valve.
- Whenever the manual fuel dump system is installed, the decal of instructions must be installed on the instrument panel.

Manual Fuel Close Line Valve.

The manual fuel close line valve (figure 1-34), located on the right side of the cargo compartment, shuts off fuel from the forward fuel tank. Whenever the close line manual valve is closed and the crossfeed is CLOSED, no fuel is flowing to the No. 1 engine.

Manual Fuel Dump Valve.

The manual fuel dump valve, located in the cargo compartment, is manually opened whenever fuel dumping is required. The manual fuel dump valve allows only fuel from the forward tank to be dumped.

Rapid Fuel Dumping System.

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

Fuel Dump Valves.

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

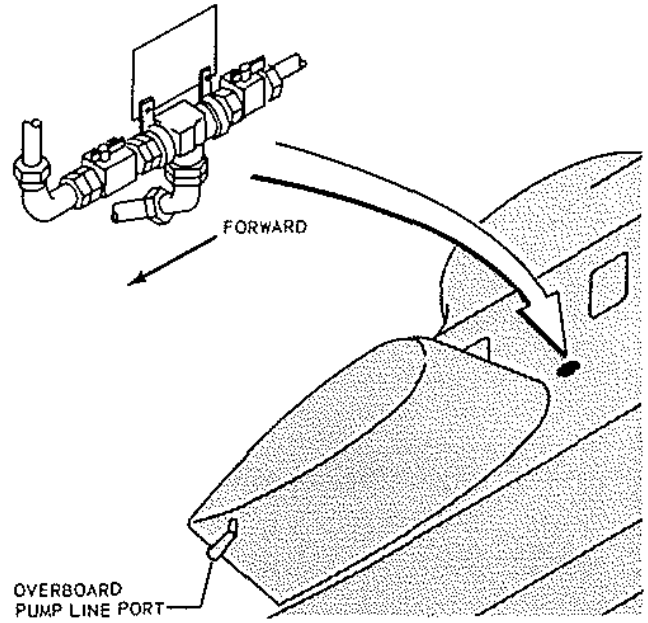


Figure 1-34. Manual Fuel Close Line and Dump Valves

Fuel Dump Pump.

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

GROUND PRESSURE AND AIR REFUELING SYSTEMS.

HH-3E helicopters are equipped with ground pressure and air refueling systems (figure 1-35). CH-3E helicopters 16▸ are equipped with provisions for the ground pressure and air refueling systems. Both systems utilize the same plumbing and system components, except the ground pressure refueling system is refueled through a pressure refueling adapter located on the lower fuselage below the entrance

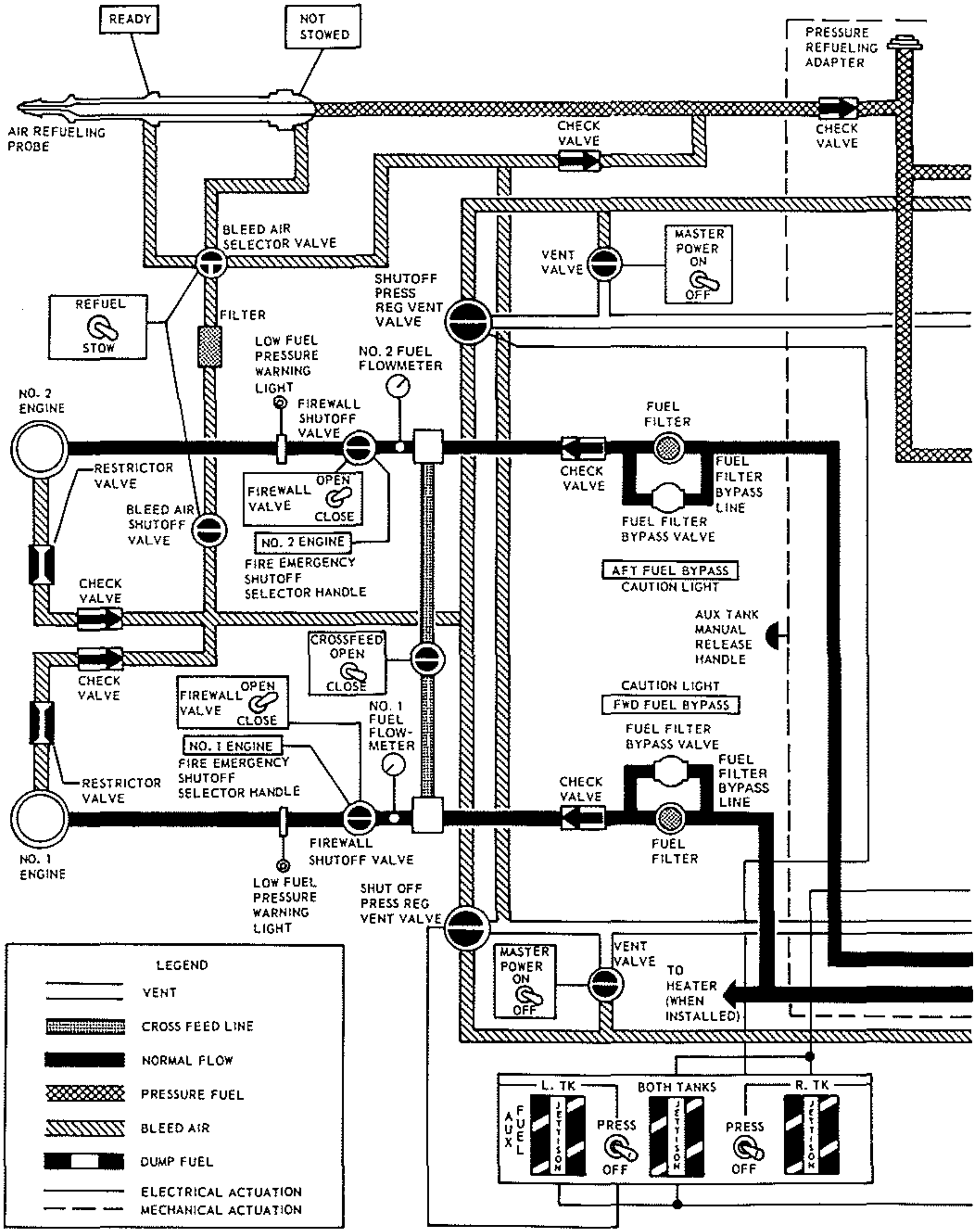


Figure 1-35. Ground Pressure and Air Refueling System (Sheet 1 of 2)

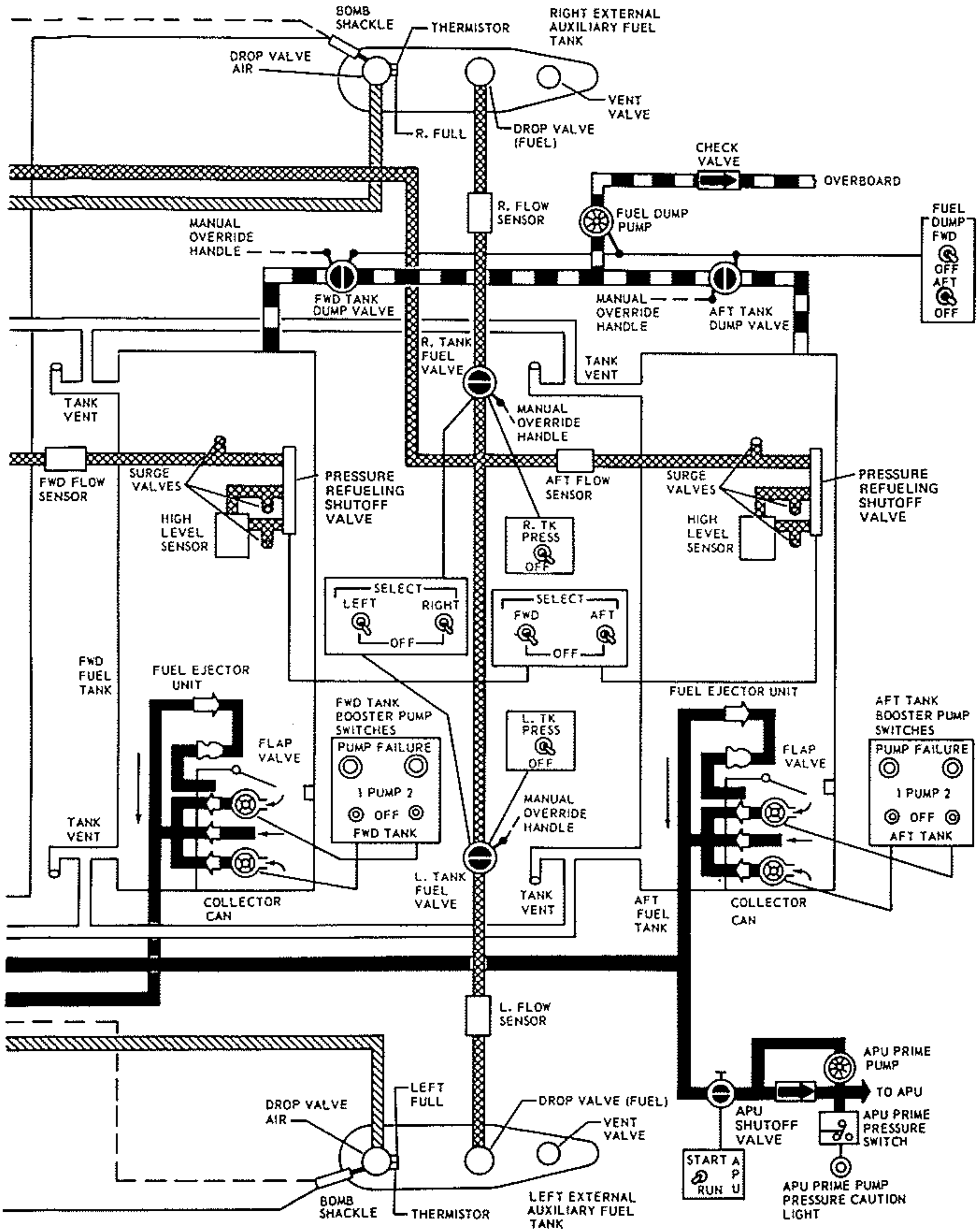


Figure 1-35. Ground Pressure and Air Refueling System (Sheet 2 of 2)

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

Ground Pressure Refueling.

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

Pressure Refueling Adapter.

The pressure refueling adapter (figure 1-2), located on the lower fuselage below the entrance to the cargo compartment, is the single-point hose attachment that is used to simultaneously pressure refuel all fuel tanks.

Air Refueling System.

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

Refueling Probe.

The retractable refueling probe (figure 1-2), located on the forward right-hand side of the helicopter, is extended and retracted by compressor bleed air. The system can either operate on bleed air from either engine, or from both engines simultaneously. Check valves are installed to prevent bleed air from flowing to an inoperative engine during single engine operation. The probe contains lock actuators that lock the probe in the extended or retracted position and cause condition lights to illuminate to indicate the probe condition.

Bleed Air Selector and Shutoff Valves.

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

Flow Sensors.

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

Pressure Refueling Shutoff Valve and High Level Sensor.

The pressure refueling shutoff valve and high level sensor, located in each main tank, controls the flow and fuel level during ground pressure and air refueling operations. The pressure refueling shutoff valves open when fuel pressure is applied and permit fuel to flow to the tanks. The high level sensor senses when the fuel tank has reached its full capacity and causes the pressure refueling shutoff valve to close and shut off the flow of fuel. The high level sensors are also the test components for the PRE-SHUTOFF TEST function.

Pressure Refueling Control Panel.

The pressure refueling control panel (figure 1-36), located on the top and center of the instrument panel, marked PRESSURE REFUEL, contains the control switches and condition lights for the ground pressure and air refueling systems. The fuel tank selector switches and condition lights are used in the same manner for both ground pressure and air refueling operations. The control panel contains the fuel dump switches, the controllable spotlight rheostat, main and external auxiliary fuel tank selector switches and condition lights, and fuel level shutoff test switch, a control panel light test switch, the air refueling probe control switch and condition lights, and the master power switch. The fuel dump switches, under the heading FUEL DUMP, have marked positions FWD and OFF and AFT and OFF, respectively. Placing the switches in the FWD and AFT positions will open respective dump valves to the forward and aft main fuel

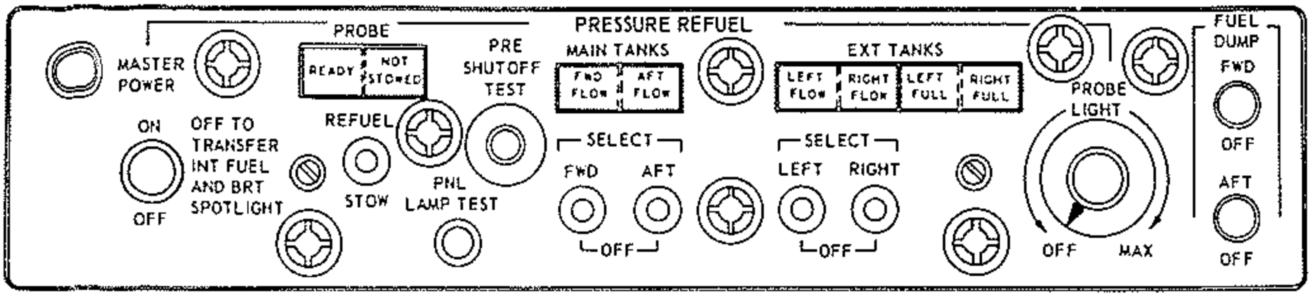


Figure 1-36. Pressure Refueling Control Panel

tanks, energize the fuel dump pump and cause all but 500 pounds of fuel in each main tank to be dumped overboard. The main fuel tanks may be simultaneously or individually selected for fuel dumping. The switches should be placed in the OFF position when 500 pounds of remaining fuel is noted on the fuel quantity indicators. The controllable spotlight rheostat, marked PROBE LIGHT, with marked positions OFF and MAX, is rotated out of the OFF position to vary the intensity of the controllable spotlight (figure 1-2) during night air refueling operations. The external auxiliary fuel tank selector switches and condition lights are under the heading EXT TANKS. The auxiliary fuel tank selector switches, marked SELECT, have marked positions LEFT and RIGHT and OFF. Placing the switches in the LEFT and RIGHT positions will open both auxiliary fuel tank fuel valves and allow fuel to flow to the tanks. The fuel flow condition lights, marked LEFT FLOW and RIGHT FLOW, respectively, will illuminate to indicate that fuel is flowing to the tanks through the flow sensors. The flow lights may flicker on and off any time the probe is in an intermediate position. The fuel level lights, will illuminate to show the tanks are full. When the fuel level lights illuminate, the fuel flow condition lights will go out as the thermistors, one located in each auxiliary tank, also close the fuel valves when they sense that the tanks are full. If the LEFT FULL or RIGHT FULL light remains on after fuel is transferred to the main tanks, reaccomplish the auxiliary tank transfer operation. The external auxiliary tanks may be simultaneously or individually selected for refueling. The main fuel tank selector switches and condition lights are under the heading MAIN TANKS. The main fuel tank selector switches, marked FWD and AFT, have marked positions SELECT and OFF. Placing the FWD and AFT switches in the SELECT position, opens the pressure refueling shutoff valves to allow fuel to flow to the tanks. The fuel flow condition lights, marked FWD FLOW and AFT FLOW, respectively, will illuminate to indicate that fuel is flowing to the tanks through the flow sensors. The FWD and AFT selector switches should be left in the SELECT position at all times unless it is not desired to service a particular tank during a fuel transfer or air refueling operation. When the main tanks are full, the fuel flow condition lights will go out as the high level sensors, one located in each main tank, close the pressure refueling shutoff valves when they sense that the tanks are full. All four fuel tanks, two main and two external auxiliary, may be simultaneously or individually selected for refueling. The fuel level shutoff test switch, marked PRE-SHUTOFF TEST, is depressed to test the integrity of the high level sensors and pressure refueling shutoff valves.

NOTE

Master power switch must be ON to check PRE-SHUTOFF TEST system.

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

NOTE

- The standby compass is not reliable when the pressure refueling panel is energized and operating.
- If the probe has just been stowed, wait approximately one minute to allow the system to bleed before extending the probe again.

ELECTRICAL POWER SUPPLY SYSTEM.

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

ALTERNATING CURRENT SUPPLY SYSTEM.

Two alternating current generators are the primary source of power for the ac electrical supply system (figure 1-37). Other sources of alternating current are the dc powered ground inverter and the ac external power receptacle. An auxiliary power unit is also provided that drives the generators through the main gear box accessory section when the rotor rpm is below 100% N_r.

Generators.

The two 20 KVA, 115/200-volt, three-phase, self-cooled, brushless, self-exciting generators are mounted on, and driven by, the accessory section of the main gear box. Each generator has a prorated capacity of 25KVA at 15,000 feet altitude and a OAT of 10°C. The auxiliary power unit powers the main gear box accessory section to drive the generators when the rotor rpm is below 100% N_r. When rotor speed reaches 100 percent rpm, the accessory section is driven through the main gear box. Each generator output is directed through a respective supervisory panel that provides control and protection of the electrical system from underfrequency, overvoltage, and open-phase protection. The underfrequency protection is not available when the weight of the helicopter is removed from the landing gear wheels. The No. 1 generator normally furnishes power to the ac essential bus. Each generator is controlled by a respective generator switch located on the overhead switch panel in the pilot's compartment. Generator power is supplied to the supervisory panel whenever the generators are operating above the cut-in speed (between 92 and 97 percent rotor speed). The generators are connected directly to the helicopter's system whenever the generator switches are placed in the ON position, provided the supervisory panel is satisfied that voltage output as well as frequency output is within the prescribed limits. Failure of either generator is indicated by failure caution lights, marked No. 1 GENERATOR and No. 2 GENERATOR, located on the caution panel. Both generators will continue to generate power during autorotation.

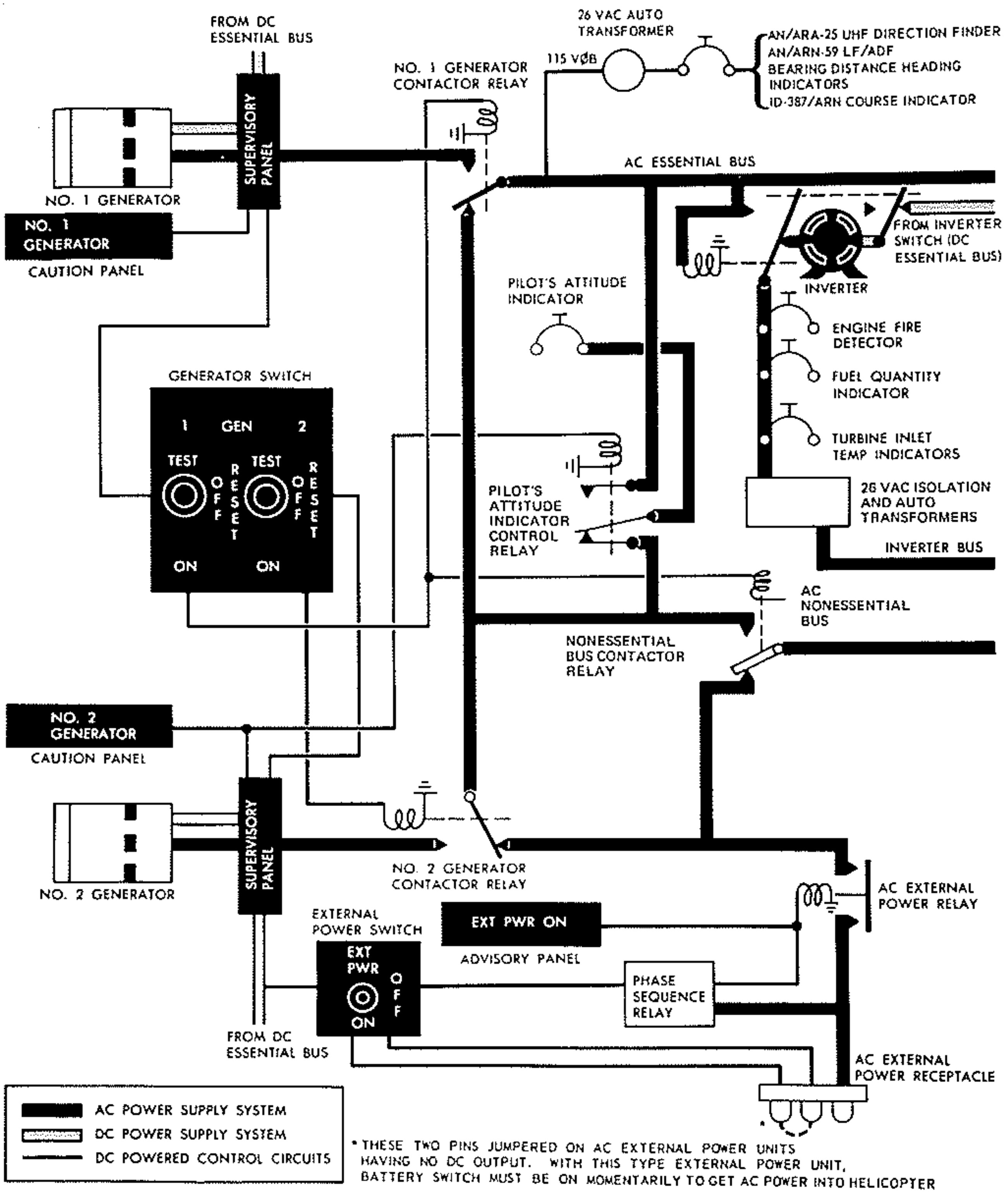


Figure 1-37. AC Electrical Power System (Typical) (Sheet 1 of 2)

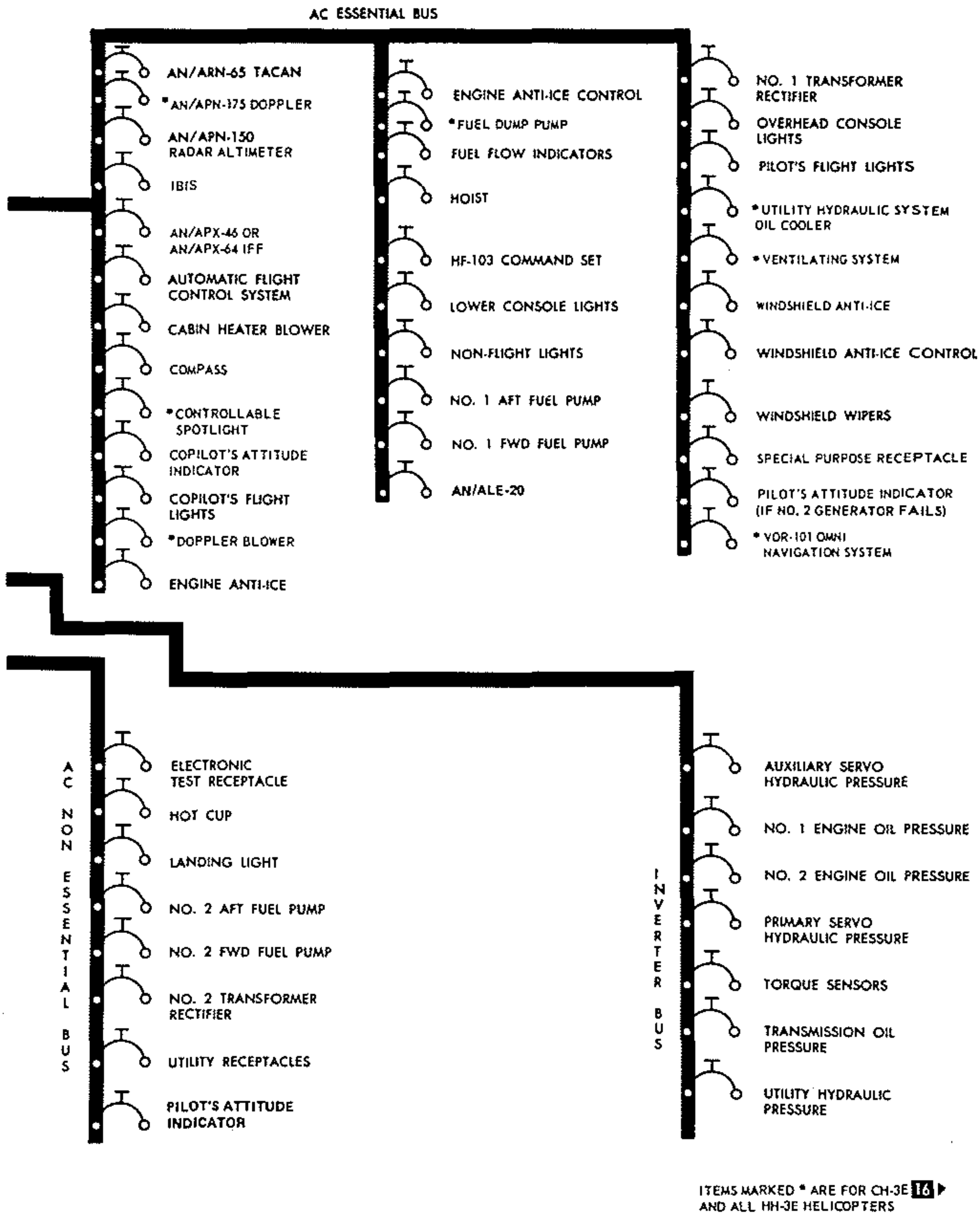


Figure 1-37. AC Electrical Power System (Typical) (Sheet 2 of 2)

Generator Switches.

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

Inverter.

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

Inverter Switch.

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

AC External Power Receptacle, Switch, and Advisory Light.

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

NOTE

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

Alternating Current Distribution.

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

AC Essential Bus.

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

Inverter Bus.

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

NOTE

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

AC Nonessential Bus.

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

Autotransformer.

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

Isolation and Autotransformers.

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

Alternating Current Circuit Breakers.

Alternating current circuit breakers, protecting the various ac circuits, are located on the overhead circuit breaker panels (figure 1-39) above the pilot's and copilot's heads in the pilot's compartment. The circuit breaker panel located above the pilot's head contains circuit breakers that protect the ac essential bus loads, and the circuit breaker panel located above the copilot's head contains circuit breakers that protect the ac nonessential bus loads. All circuit breakers are marked as to the circuit they protect and are of the push-pull type that may be reset. Any malfunctioning circuit may be isolated from the ac power supply system by pulling out its circuit breaker.

DIRECT CURRENT POWER SUPPLY SYSTEM.

Two transformer-rectifiers are the primary sources of direct current power for the dc electrical supply system (figure 1-38). Other sources of direct current power are the battery and the dc external power receptacle.

Transformer-Rectifiers.

Two 200-ampere transformer-rectifiers are located in the electronics compartment. Each transformer-rectifier is powered by a separate generator. The No. 1 generator powers the No. 1 transformer-rectifier through the ac essential bus, and the No. 2 generator powers the No. 2 transformer-rectifier through the ac nonessential bus. Each transformer-rectifier is controlled by a respective transformer-rectifier switch, located on the overhead switch panel in the pilot's compartment. The No. 1 transformer-rectifier is protected by a circuit breaker, marked No. 1 XMFR RECTIFIER, located on the ac essential bus circuit breaker panel, and the No. 2 transformer-rectifier is protected by a circuit breaker, marked No. 2 XMFR RECTIFIER, located on the ac nonessential bus circuit breaker panel. The transformer-rectifiers are connected directly to the helicopter's dc system whenever they are energized by ac power and the transformer-rectifier switches are in the ON position. Failure of the No. 1 transformer-rectifier will be indicated by a light on the caution panel, marked #1 XMFR RECT, and failure of the No. 2 transformer will be indicated by a light on the caution panel, marked #2 XMFR RECT.

Transformer-Rectifier Switches.

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

Battery.

The 24-volt, 22-ampere hour nickel cadmium battery (figure 1-48), located in the nose section forward of the pilot's compartment, is accessible from outside the helicopter for maintenance. Battery power is used for limited ground operations when no external power is available and as emergency source of power to the essential bus if both generators and/or transformer-rectifiers should fail. The battery also powers the battery bus.

Battery Switch.

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

NOTE

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

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

The 28-volt dc external power receptacle (figure 1-2) is located on the right side of the helicopter below the pilot's window. External power is controlled by a switch, marked EXT PWR, with marked positions ON and OFF, located on the overhead switch panel. The circuit is protected by a circuit breaker, marked EXT PWR, located on the overhead dc circuit breaker panel. The external power switch must be in the ON position to utilize an external power source connected to the dc external power receptacle. When the external power relay is energized, a light marked EXT POWER ON, located on the advisory panel, will illuminate. External power will also be provided to energize the nonessential bus relay, thereby allowing both the dc essential and dc nonessential buses to be energized. The external power relay also serves to keep the external power receptacle from being energized when not in use. DC external power, if used, should be used for all ground operations until after the generators are operating.

NOTE

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

Direct Current Distribution.

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

DC Essential Bus.

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

DC Nonessential Bus.

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

Battery Bus.

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

Direct Current Circuit Breakers.

Circuit breakers that protect the various dc circuits are located on the overhead dc circuit breaker panel and on a portion of the ac nonessential bus circuit breaker panel (figure 1-39) in the pilot's compartment. All circuit breakers are marked as to the circuit they protect and are of the push-pull type that may be reset. Any malfunctioning circuit may be isolated from the dc power supply system by pulling out its circuit breaker.

Loadmeters.

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

Voltmeter and Voltmeter Selector.

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

UTILITY HYDRAULIC SUPPLY SYSTEM.

The utility hydraulic system (figure 1-40) operates the main landing gear, nose landing gear, APU start system, and ramp actuating system. On helicopters CH-3E 16▸ , HH-3E, or those helicopters modified by T.O. 1H-3(C)C-561, the utility hydraulic system also powers the rescue hoist. The utility hydraulic system reservoir (figure 1-48), located aft of the main gear box, has a capacity of 3.05 gallons of hydraulic fluid. The utility hydraulic pump is located on the accessory drive section of the main gear box and provides 3000 psi hydraulic pressure. An oil cooler is provided in the hydraulic line to maintain utility hydraulic oil temperature within limits on helicopters CH-3E 16▸ , HH-3E, or those helicopters modified by T.O. 1H-3(C)C-561. The oil cooler blower operates on power from the ac essential bus and is protected by a circuit breaker marked OIL COOLER BLOWER located on the ac essential bus circuit breaker panel. The blower is actuated by power from the dc essential bus

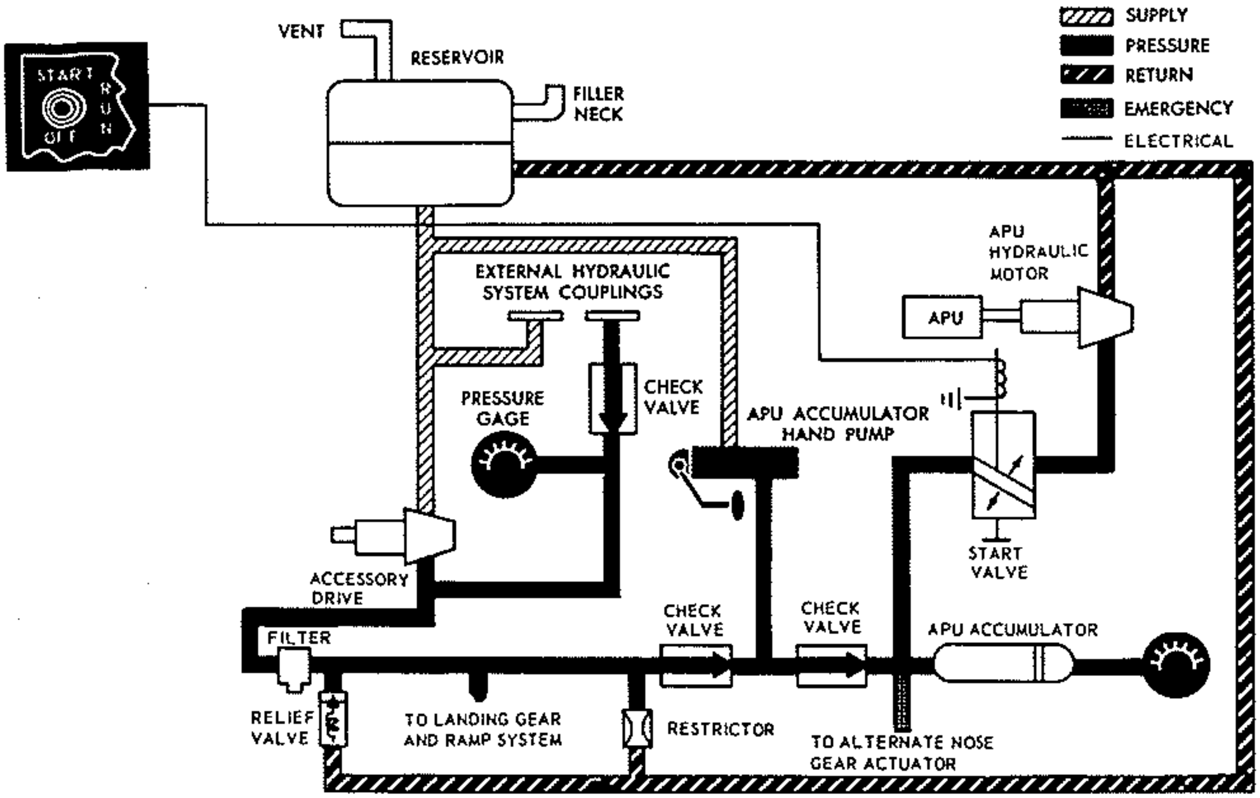


Figure 1-40. Utility Hydraulic System (Typical)

through a circuit breaker marked OIL COOL BLO located on the dc essential bus circuit breaker panel. The blower operates continuously when the buses are energized. Helicopters modified by T.O. 1H-3-581 have a screen-type fan guard over the blower inlet.

WARNING

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

NOTE

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

UTILITY HYDRAULIC PRESSURE INDICATOR.

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

FLIGHT CONTROL SYSTEM.

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

MAIN ROTOR FLIGHT CONTROL SYSTEM.

The main rotor flight control system provides vertical, lateral, and longitudinal control. Control motions from the collective pitch lever for vertical control and from the cyclic stick for lateral and longitudinal control are combined in a mixing unit, located in the AFCS control compartment aft of the pilot's seat, and are transmitted to the main rotor assembly by mechanical linkage. Control action is assisted by two hydraulically-operated flight control servo systems. The main rotor controls terminate at the stationary swashplate of the main rotor head. Control action is transmitted through the rotating swashplate and linkage on the main rotor hub to the blades.

Collective to Yaw Coupling.

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

Collective to Cyclic Pitch Coupling.

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

Collective Pitch Levers.

Two collective pitch levers (figure 1-4) are located in the pilot's compartment. Both levers operate simultaneously to change the collective pitch of the main rotor blades. A friction lock on the pilot's collective pitch lever, marked COLLECTIVE PITCH LOCK with an arrow pointing left, marked INCREASE FRICTION, can be rotated to apply friction to prevent the collective pitch lever from creeping while in flight.

Cyclic Sticks.

The cyclic sticks (figure 1-4) provide lateral and longitudinal control of the helicopter. Moving the cyclic stick in any direction tilts the tip path plane of rotation of the main rotor blades in that direction and moves the helicopter in the same direction. The stick grip (figure 1-41) contains pushbutton and thumb-operated switches for controlling various equipment installed in the helicopter.

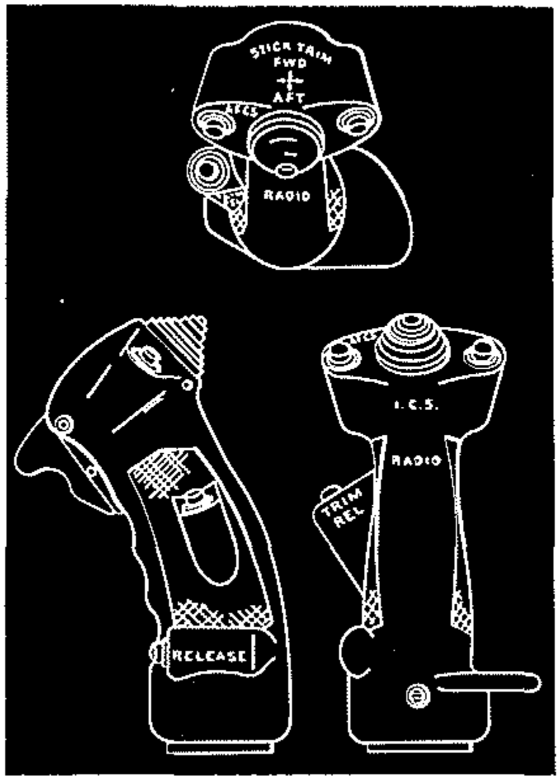


Figure 1-41. Cyclic Stick Grip

Cyclic Stick Trim System.

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

NOTE

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

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

Cyclic Stick Trim Master Switch.

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

Cyclic Trim Switches.

The cyclic trim switches, located on the pilot's and copilot's cyclic stick grips (figure 1-41), have marked positions FWD, AFT, L, and R. The four-way thumb switch is spring-loaded to the center (off) position. When the switch is placed in any of the four positions, hydraulic pressure will drive the cyclic stick in the same direction. When the desired cyclic stick position is obtained, the switch is released. The action of the cyclic stick trim system will then function about this location of the cyclic stick. The cyclic trim switches receive electrical power from the dc essential bus through a circuit breaker, marked STICK TRIM, located on the overhead dc circuit breaker panel.

Cyclic Trim Release Button.

The spring-loaded, pushbutton switches, located on the pilot's and copilot's cyclic stick grips (figure 1-41), marked TRIM REL, are used to change trim position (without using the cyclic trim switch). Cyclic trim position is changed by depressing the cyclic trim release button, moving the cyclic stick to the new position, and then releasing the cyclic trim release button. The cyclic trim system will then hold the selected position of the cyclic stick. The cyclic trim release button controls dc essential bus power to the trim actuators.

TAIL ROTOR FLIGHT CONTROL SYSTEM.

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

Tail Rotor Pedals.

The tail rotor pedals (figure 1-4) change the pitch and thrust of the tail rotor and consequently the heading of the helicopter. Electrical switches, mounted on the force link assembly, cancel the directional signals of the automatic flight control system when approximately 8 pounds of pressure is exerted on either pedal. Toe brake pedals for the main landing gear wheel brakes are mounted on both the pilot's and copilot's pedals.

Tail Rotor Pedal Adjustment Knobs.

Pedal adjustment knobs (figure 1-4) are located on each side of the fuselage, just forward of the ash trays in the pilot's compartment. The adjustment knobs are connected to mechanical linkage that provide for fore-and-aft adjustment of the pedals. The knobs are rotated to the right, as indicated by the arrow marked FWD, for forward adjustment and to the left, as indicated by the arrow marked AFT, for aft adjustments. The pilot's pedals are adjusted with the knob on the right side of the fuselage, and the copilot's pedals are adjusted by the knob on the left side of the fuselage.

NOTE

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

FLIGHT CONTROL HYDRAULIC POWER SUPPLY SYSTEM.

The flight control hydraulic power supply system (figure 1-42) consists of a primary and an auxiliary flight control servo system. The servo systems are required by the pilot for a power boost to operate the controls. The servos also prevent feedback of main rotor vibratory loads to the control sticks. Both servo systems operate from independent hydraulic systems and utilize similar servo hydraulic

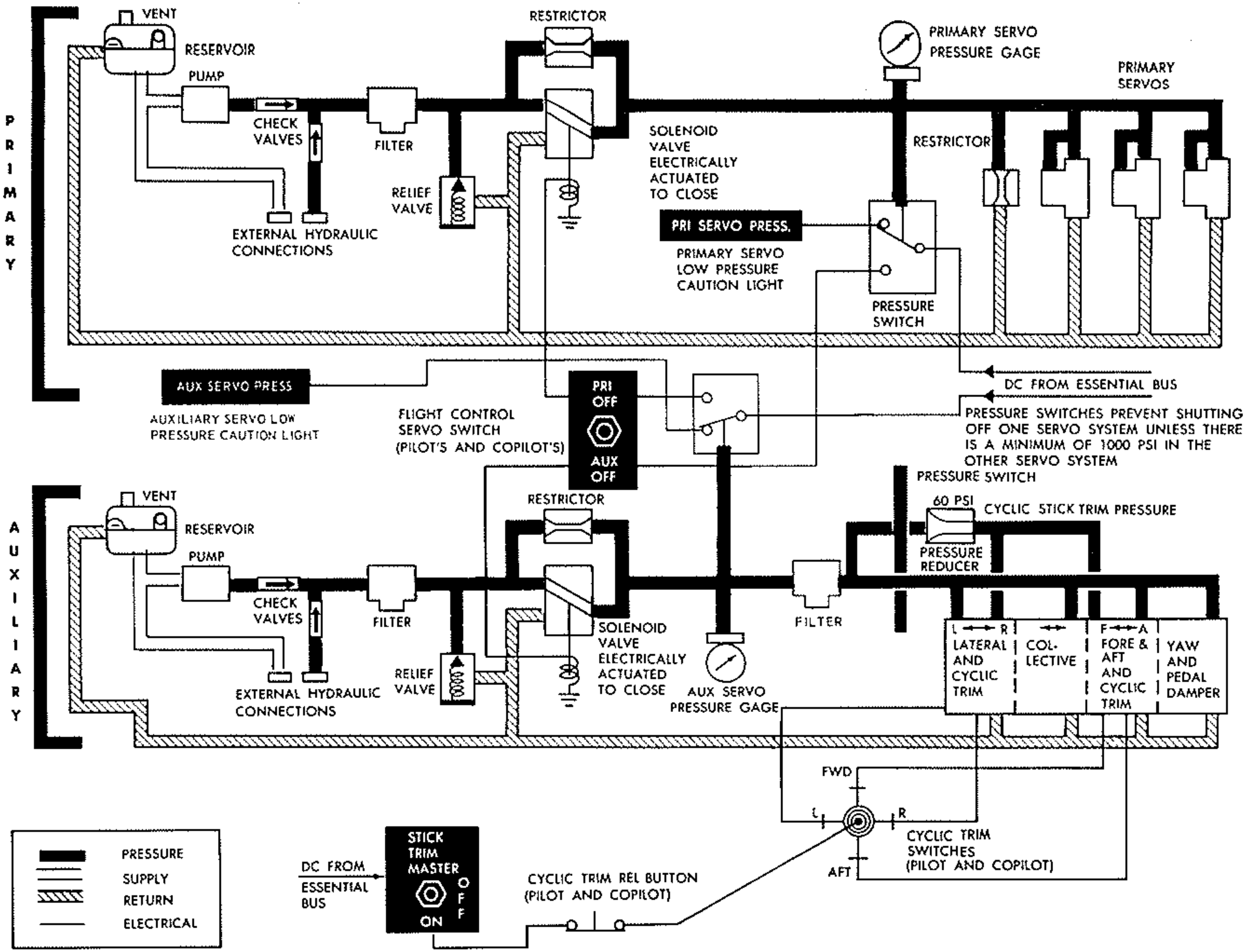


Figure 1-42. Flight Control Servo Hydraulic Systems

units to vary the main and tail rotor blade pitch through the mechanical linkage of the regular flight control system. The servo unit output is connected to the flight control linkage to provide the power boost. The continuity of the direct control linkage is maintained from the controls in the pilot's compartment through the auxiliary and the primary servos to the main rotor blades, except for a slight amount of end play at each servo unit to permit the pilot valves to move before the direct control linkage. Normally, both servo systems are in operation at all times.

Primary Flight Control Servo System.

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

Auxiliary Flight Control Servo System.

The auxiliary flight control servo system, consisting of a bank of four hydraulic servo packages, located below the main rotor flight control system mixing unit, provides the means of introducing AFCS corrective signals into the flight control systems and, in the event of primary servo failure, reacts to flight loads and tail rotor loads. The auxiliary servo hydraulic system reservoir, has a capacity of approximately 0.45 gallon of hydraulic oil. When auxiliary servo pressure fails, the AUX SERVO PRESS light on the caution panel illuminates.

Flight Control Servo Switches (Servo Switches).

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

Servo Hydraulic Pressure Indicators.

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

Primary and Auxiliary Servo Pressure Caution Lights.

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

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS).

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

NOTE

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

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

AUTOMATIC FLIGHT CONTROL SYSTEM CONTROL PANEL.

The AFCS control panel marked AFCS CONT, is located on the cockpit console (figures 1-17 and 1-18) between the pilot and copilot. Controls consist of two engage buttons; one marked AFCS ENG and the other BAR ALT ENG, an off button, marked BAR ALT OFF, a yaw trim knob marked, YAW TRIM; and a center-of-gravity trim knob, marked CG TRIM. The AFCS ENG button is depressed to engage the pitch, roll, and yaw channels, and the BAR ALT ENG button is depressed to engage the barometric altitude controller. Each engage button is equipped with a light that will illuminate to indicate engagment. Once engaged, the entire AFCS can be disengaged by depressing the

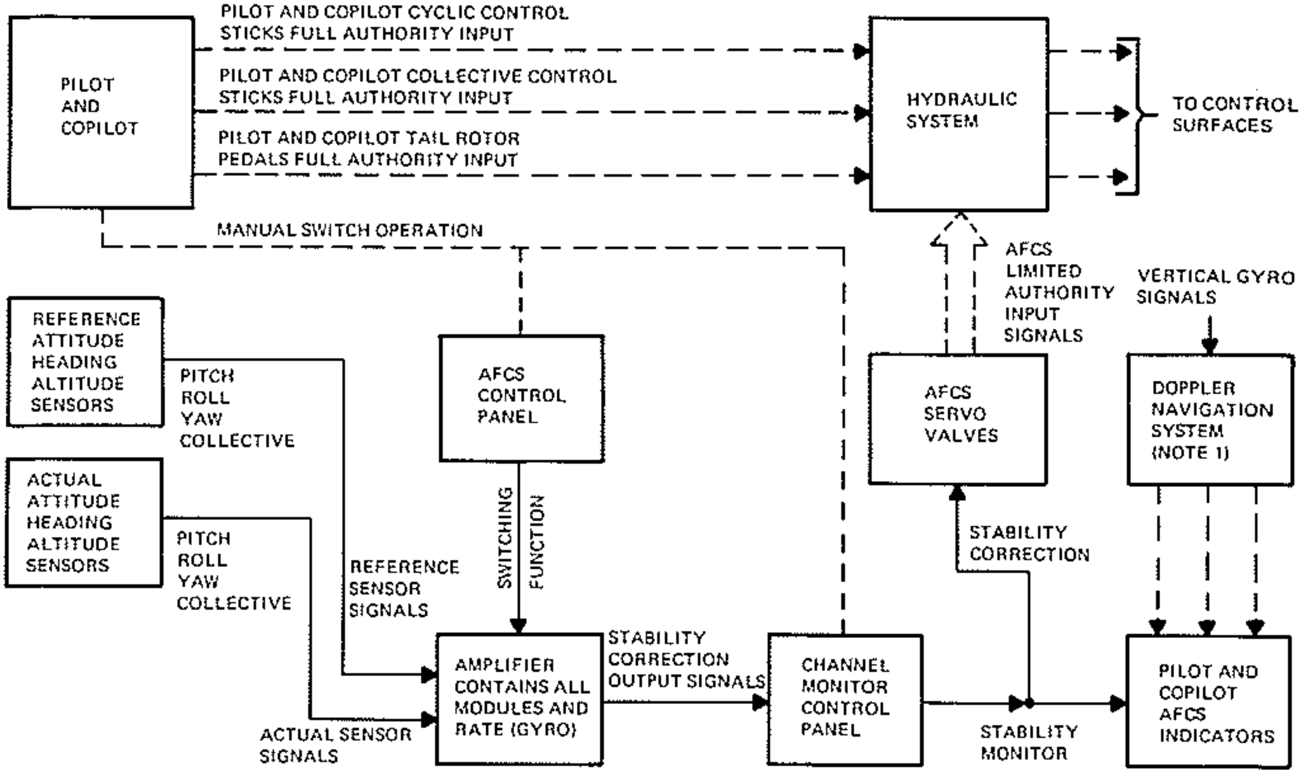


Figure 1-43. AFCS Signal Flow Simplified Block Diagram

button, marked AFCS REL, located on the pilot's and copilot's cyclic stick grips. The barometric altitude controller can be completely disengaged by depressing the BAR ALT OFF button, or released momentarily to make changes in altitude by depressing the BAR REL button located on the pilot's and copilot's collective pitch lever grip. The YAW TRIM and CG TRIM knobs, located on the bottom half of the panel, are designed with characteristic shapes to enable the pilot to easily distinguish between them. The YAW TRIM knob is triangular-shaped and the CG TRIM knob is clover leaf-shaped. The YAW TRIM knob enables the pilot to accurately trim the heading of the helicopter, provided his feet are off the tail rotor pedals. One rotation of the knob turns the helicopter 72 degrees. The CG trim control permits the pilot to recenter the pitch servo valve after a shift in the center of gravity. By a coordinated movement of the CG trim knob and the cyclic stick, the pilot may fly the helicopter at any pitch attitude, and the AFCS will retain its full ability to provide stabilization.

CAUTION

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

CHANNEL MONITOR CONTROL PANEL.

The channel monitor control panel, marked CHANNEL MONITOR is located on the pilot's console (figure 1-44) to the right of the pilot's seat. The controls consist of four toggle switches across the top of the panel, marked PITCH, ROLL, COLL, and YAW, under the general heading CHANNEL DISENGAGE, with marked position ON and OFF. These toggle switches permit individual disengagement of the pitch, roll, collective, and yaw channels of AFCS. They are usually left in the ON position, except when the pilot wishes to disengage a malfunctioning channel. Directly below the four

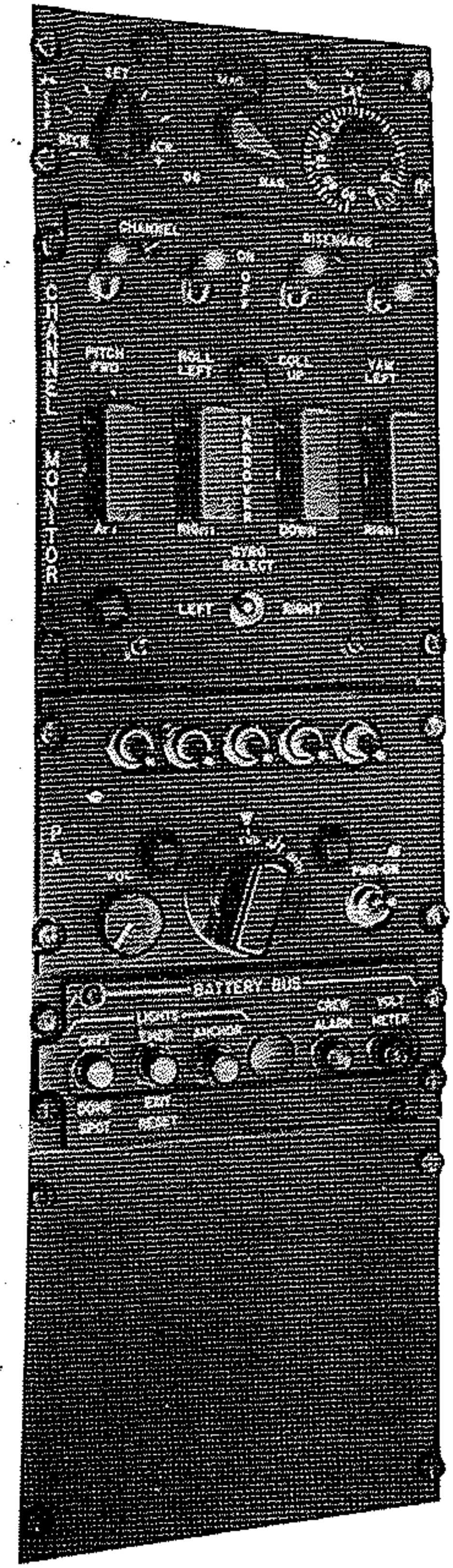


Figure 1-44. Pilot's Console (Typical)

toggle switches are four guarded three-position switches, under the general heading HARDOVER, that are normally used only to check system authority during a ground operational checkout. The PITCH switch has marked positions FWD and AFT. The ROLL and YAW switch have marked positions LEFT and RIGHT. The COLL switch has marked positions UP and DOWN. Each switch may be placed in a position to check the corresponding PITCH, ROLL, COLL, or YAW channels of AFCS. When the hardover switch guards are closed, the switches are held in the OFF position. The hardover switches will transmit a hardover signal to the AFCS indicator and respective servo valve whether the AFCS control is engaged or disengaged and power is supplied to the dc essential bus. The switch guards must be lifted before override checks can be accomplished. The toggle switch on the bottom of the panel is marked GYRO SELECT with marked positions, LEFT and RIGHT. Pitch and roll references for AFCS are selected from either the left or right gyro; however, the normal position is LEFT. With the gyro select switch in the LEFT position, the left gyro provides signals for AFCS and the copilot's attitude indicator while the right gyro feeds the pilot's attitude indicator. If a gyro should fail with the switch in the LEFT position, the pilot will not lose the services of both his attitude indicator and AFCS pitch and roll channels simultaneously.

WARNING

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

AFCS (AUTOMATIC FLIGHT CONTROL SYSTEM) INDICATOR.

Two indicators (figure 1-14) (referred to as AFCS indicator) are installed in front of the pilot and the copilot on the instrument panel. The indicators provide a visual indication of the output signals from the AFCS pitch, roll, yaw, and collective channels. On helicopters equipped with a doppler navigation system, the indicators provide a visual indication of doppler reliability during cruise and display ground speed, drift, and vertical velocity information. Each indicator contains scale increment marks located across the center vertical and horizontal axis and along the left and bottom sides of the dial face. Two movable bars coincide with the center vertical and horizontal axis scale marks of the dial and intersect perpendicularly at a small circle marked on the dial face. There are two arrowhead-type pointers provided, one located on the left-hand side of the indicator which moves vertically, up or down, coinciding with the vertical scale, while the other pointer at the bottom of the indicator moves horizontally, left or right, coinciding with the horizontal scale. On helicopters not equipped with a navigation set, radar, the AFCS indicator has provisions for three modes of operation but utilizes only the A mode for monitoring AFCS output signals. On helicopters equipped with a navigation set, radar, the indicator uses the A mode to monitor AFCS output signals and the D mode to monitor doppler navigation reliability and indications. The C mode is inoperative. The mode selector knob, located on the lower left side of the indicator, is marked A, D, and C. An index mark on the case indicates the mode selector knob position. The mode indicator window in the upper right quadrant of the dial indicates which mode has been selected. The selector should remain in the A position for AFCS indications and the D position for doppler navigation indications. With the AFCS ENG button engaged and mode A selected on the indicator, the AFCS indicator will monitor the input to the AFCS servo valves. AFCS hardovers may be displayed on the AFCS indicator without the AFCS engaged. The horizontal bar is used to monitor the pitch channel, the vertical bar monitors the roll channel, the vertical pointer monitors the altitude channel, and the horizontal pointer monitors the yaw channel. The scale reference marks are spaced to equal zero, 25, 50, 75, and 100 percent of full scale deflection in either direction. All AFCS output signals are routed through the channel monitor control panel to the AFCS indicator. On helicopters equipped with a navigation set, radar, operation in the D mode provides a visual indication of doppler reliability, during cruise, and displays ground speed, drift, and vertical velocity during hover operation at speeds up to 22.5 knots ground speed. When operating in the D mode, the AFCS indicator is connected to the navigation set, radar. Below 22.5 knots ground speed, the horizontal bar indicates forward or rearward velocities and the vertical bar indicates left or right drift. Each increment of the AFCS indicator horizontal and vertical scales indicates 5 knots with a maximum indication of 20 knots. The vertical pointer indicates vertical velocity with each increment equal to 500 feet per minute. Full scale deflection is equal to 2000 feet per minute up and 2000 feet per minute down. To indicate forward flight up to 22.5 knots, the horizontal bar will move downward and, to indicate a drift, the vertical bar will move in a direction opposite to the direction of the drift. Therefore, the pilot flies into the bar intersection for correction. In cruise flight above 22.5 knots ground speed, only the OFF flag and ground speed bar are operative. In the D mode the yaw pointer is disconnected and should not move. An OFF flag on the upper dial face of the AFCS indicator is used in both modes of operation. In the A mode, the flag disappears when the AFCS is engaged. In the D mode, the flag disappears when the doppler transmitter is turned on and doppler return signals are being received. Engage voltage for warning flag operation is supplied from the dc essential bus through a circuit breaker, marked AFCS PWR, located on the overhead dc circuit breaker panel.

AFCS RELEASE BUTTON.

AFCS is disengaged by depressing the buttons marked AFCS REL which are located on both the pilot's and copilot's cyclic stick grips (figure 1-41).

BAR ALT RELEASE SWITCHES.

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

INSTRUMENTS.

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

MAGNETIC COMPASS.

A magnetic compass (figures 1-4 and 1-5) is located at the top center of the instrument panel. A standby compass correction card is located on the pilot's side of the instrument panel.

NOTE

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

FREE AIR TEMPERATURE GAGE.

The bi-metallic free air temperature gage (figure 1-5), located in the windshield glass above the instrument panel, is a direct reading instrument that is calibrated in degrees Centigrade.

CLOCKS.

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

PITOT-STATIC SYSTEM.

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

ALTIMETER-ENCODER AAU-21/A.

One altimeter-encoder (figure 1-45) is installed in the pilot's instrument panel. The altimeter-encoder combines a conventional barometric type altimeter, possessing a counter-drum-pointer display, with an altitude reporting encoder in one self-contained unit. The 10,000 and 1000 foot counters and the 100-foot drum provide a direct digital output and readout of altitude in increments of 100 feet, from −1000 to 38,000 feet. The digital output is referenced to 29.92 in H_g and is not affected by changes of barometric setting. The pointer repeats the indications of the 100-foot drum, and serves both as a vernier for the drum and as a quick indication of the rate and sense of altitude changes. Two methods may be used to read indicated altitude on the counter-drum-pointer altimeter: (1) read the counter-drum window, without reference to the pointer, as a direct digital readout in thousands and hundreds of feet, or (2) read the thousands of feet on the two counter indicators, without referring to the drum, and then add the 100-foot pointer indication. The self-contained servo-driven encoder provides altitude encoded in 100-foot increments for automatic transmission when the AIMS/IFF transponder is interrogated on mode C. In case of power loss to the encoder servo system, a CODE OFF flag appears automatically in a window in the upper left portion of the display, indicating that altitude information is no longer being transmitted to the ground. In this condition, the instrument continues to function as a normal barometric altimeter. The barometric pressure is entered by use of a barometric set knob in the lower left front of the instrument case. The altimeter setting appears on counters in the window at the lower right of the display and has a range of settings from 28.1 to 31.0 in H_g. An internal vibrator operates continuously whenever aircraft dc power is turned on. The vibrator minimizes internal mechanical friction, enabling the instrument to provide a smoother display during changing altitude conditions. Should vibrator failure occur, the altimeter will continue to function pneumatically, but a less smooth movement of the instrument display will be evident with changes in altitude.

WARNING

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

NOTE

When each 1000 foot increment is nearly completed, the counter(s) abruptly index to the next digit. The counter-drum-pointer altimeter mechanism may also cause a noticeable pause or hesitation of the pointer due to the additional intermittent friction and inertia loads applied to the mechanism to turn over the thousand-foot counter. This affect may be more pronounced at ten thousand-foot intervals where both counters are turned over simultaneously. This momentary pause is followed by a noticeable acceleration as the altimeter mechanism overcomes the counter wheel load and rolls the dial over to the next digit. The pause occurs during the 9 to 1 portion of the scale. The pause-and-accelerate behavior is normally more pronounced at high altitudes and high rates of ascent and descent. During normal rates of descent or ascent and at low altitudes, the effect will be minimal.

ALTIMETER AAU-27/A.

One altimeter (figure 1-45) is installed in the copilot's flight instrument panel. The instrument is similar to the altimeter-encoder except it does not have an altitude encoder nor the CODE OFF display mechansim. The indicated altitude on the altimeter is from —1000 to 50,000 feet. The altitude display, altimeter setting, and vibrator considerations described for the altimeter-encoder also apply to the altimeter.

ATTITUDE INDICATORS.

Two (Lear 4005T) attitude indicators (figure 1-14), installed on the instrument panel in front of the pilot and copilot give a visual indication of the helicopter's flight attitude. The indicator face consists of a stationary miniature airplane representing the helicopter, a bank angle scale, bank index, and a moving two-colored sphere with a district white horizon line dividing the two colors, white above, black below, and a turn-and-slip indicator, located on the bottom of each attitude indicator that gives visual indication of the helicopter's rate-of-turn and balanced flight. The turn needle is a two minute needle; however, the calibration marks are the same as a four minute needle. For a standard rate turn of 3° per second, one needle width deflection is required. The pilot's attitude indicator receives pitch and roll information from the right vertical gyro and the copilot's attitude indicator receives pitch and roll information from the left vertical gyro. A power warning flag, marked OFF, will appear in the face of the indicator under the following circumstances: (1) when ac power has not been applied, (2) for approximately the first 68 seconds after ac power has been applied, (3) and when any imbalance or failure of the three phases of ac power occurs.

WARNING

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

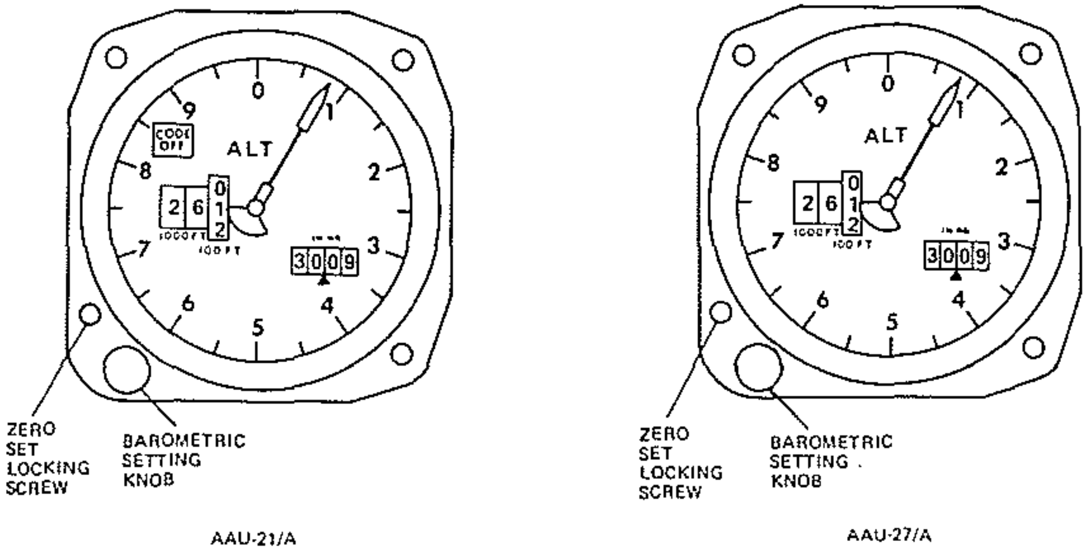


Figure 1-45. Altimeter - Encoder AAU-21/A and Altimeter AAU-27/A

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

TURN RATE SWITCHES.

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

VERTICAL VELOCITY INDICATOR.

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

J-4 COMPASS SYSTEM.

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

J-4 Compass Control Panel.

The J-4 compass control panel, located on the pilot's console (figure 1-44), contains all the controls for operation of the compass system. The function selector switch has marked positions DG and MAG. When the function selector switch is placed in the DG position, the compass system will function as a free directional gyro, with either north or south latitude corrections for the drift effect of the rotation of the earth. When the switch is placed in the MAG position, the gyro is slaved to the earth's magnetic field. Directional reference signals are obtained from the remote compass flux valve. In this mode of operation, the gyro is used as the stabilizing element of the compass system. The rotating azimuth card on the BDHIs will indicate stabilized magnetic headings. The annunciator meter, marked MAG on the compass control panel, provides indication of the synchronization of the compass system. In magnetic compass (MAG) operation, the synchronization circuits automatically synchronize the gyro to the remote compass flux valve. Automatic synchronization is established in the first 15 seconds of MAG operation and is maintained during operation of the compass system in this mode. In the directional gyro (DG) mode of operation, synchronization is not established or maintained automatically. Manual synchronization is accomplished by operating the SET synchronizer switch on the compass control panel. When the SET synchronizer switch, with marked positions DECR (—) and INCR (+), is displaced to the right or left, a synchronized signal is provided through the system at the rate and direction required to set the compass card of the pilot's and copilot's BDHIs, as desired. Synchronization of the system is indicated on the annunciator meter when the pointer is in line with the white arrow on the control panel. Reference heading data for the AFCS system is not affected by the synchronization cycles. The BDHIs compass cards can also be repositioned by use of the SET synchronizer switch when the compass is in magnetic compass (MAG) operation. The switch marked LAT is used to set in the proper magnitude of latitude correction during the DG mode of operation. Maximum drift of the directional gyro occurs when the gyro spin axis is at right angles to the earth's spin axis (north and south poles), and zero drift occurs when the two axes are aligned at the equator. When the helicopter is being flown in a northerly or southerly direction, the latitude compensation control should be set periodically to the latitude at which the helicopter is flying. The hemisphere selector switch, marked N and S, provides selection of the hemisphere in which the helicopter is being flown.

CAUTION AND ADVISORY PANELS.

CAUTION PANEL.

The caution panel (figure 1-20), marked CAUTION, is located in the center of the instrument panel. The caution panel gives the pilots visual indication of failure or unsafe conditions of certain critical items in the helicopter. The caution lights, each having its own operating circuit, indicate a particular condition in the helicopter. If a failure or unsafe condition occurs in one of the systems, the caution light for that particular condition remains on until the failure or unsafe condition is corrected. The warning lights operate through two circuit breakers, marked PWR and TEST, located on the overhead dc circuit breaker panel. The circuit breaker marked PWR provides electrical power for the normal operation of the warning lights, and the circuit breaker marked TEST provides power for the test circuit only. For caution light panel indications, refer to Section III, figure 3-1.

Master Caution Light.

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

NOTE

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

ADVISORY PANEL.

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

CAUTION AND ADVISORY LIGHTS TEST SWITCH.

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

DIM-BRIGHT SWITCH.

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

LANDING GEAR SYSTEM.

The tricycle configured landing gear system consists of dual retractable main landing gear assemblies, a single retractable nose gear assembly, and a hydraulic system (figure 1-46). The landing gear hydraulic system operates on 3000 psi hydraulic pressure from the utility hydraulic system. A landing gear warning system is installed to alert the pilots that the landing gear is not down and locked when at airspeeds of approximately 60 knots. The necessary electrical power is provided from the dc essential bus, through circuit breakers, under the

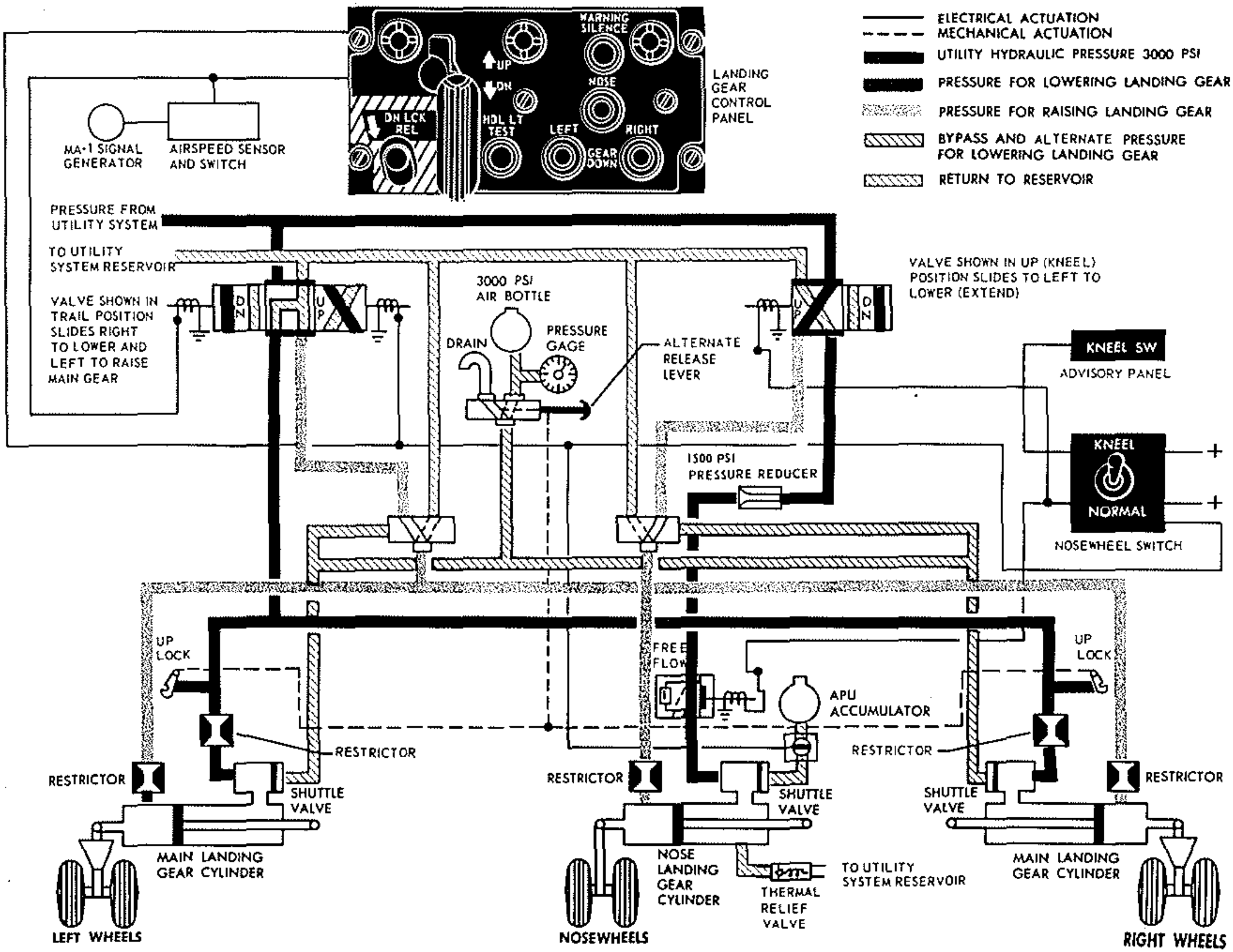


Figure 1-46. Landing Gear Hydraulic System

general heading LAND GEAR and marked EMER DN, NOSE, MAIN, located on the overhead dc circuit breaker panel. The landing gear warning system is powered by the dc essential bus through a circuit breaker, marked LAND GEAR WARN, located on the dc overhead circuit breaker panel. The main landing gear system is equipped with a one-shot pneumatic, alternate extension system, and the APU accumulator is utilized as an emergency extension power source for the nose gear assembly. The nose gear assembly retraction system can be further utilized on the ground to kneel the helicopter to facilitate cargo handling. The landing gear control panel, marked LANDING GEAR CONTROL, is located on the cockpit console (figures 1-17 and 1-18). The watertight hull, plus sponsons, provide the helicopter with amphibious capabilities. Helicopters CH-3E 16▸ and HH-3E 25▸ are equipped with provisions for a fixed landing gear.

MAIN LANDING GEAR.

The two main landing gear assemblies (figure 1-2) are located below the sponsons and retract forward and upward into the sponsons. Each main landing gear is equipped with dual wheels and hydraulic brakes, a retracting cylinder, a pneudraulic strut, attaching drag links and supports, and up and down lock mechanisms.

NOSE LANDING GEAR.

The single nose landing gear (figure 1-48), mounted vertically at the centerline of the helicopter, is free to rotate 360 degrees about the strut centerline. All nose gear motion, including shock stroke, kneeling, jacking, and retraction, is vertical. The nose gear assembly is equipped with dual wheels, a retracting cylinder, a pneudraulic strut and shimmy damper, and attaching drag links and supports. The entire pneudraulic strut acts as a piston which is lowered or raised for retracting, jacking, and kneeling. The nose gear may be kneeled (retracted) 14.8 inches so as to alter the ground clearance of the tail section to facilitate cargo handling. The nose gear assembly is hydraulically locked in the extended, retracted, or kneeled position. The shimmy damper is utilized to minimize and offset nose gear vibration encountered during forward motion on the ground. A centering cam centers the nose gear assembly when the helicopter is airborne. A nosewheel lock is installed to improve ground stability of the helicopter on uneven terrain and for use during rotor shutdown and engagement.

Nosewheel Lock Handle.

The nosewheel lock handle (figures 1-4 and 1-5), marked PARK LOCK, is located below a plate marked NOSE GEAR PULL TO LOCK, on the pilot's side of the cockpit console. The nosewheel is locked by pulling the lock handle aft and up, and unlocked by pushing aft and down.

LANDING GEAR ACTUATING SYSTEM.

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

Landing Gear Control Handle Down-Lock Release.

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

Nose Gear Switch and Caution Light.

Kneeling (figure 1-47) is accomplished by placing the switch, marked NOSE GEAR, NORMAL, KNEEL, located on the overhead switch panel (figure 1-13), in the KNEEL position. After the helicopter is loaded, the nose gear is jacked (hydraulically extended) to the static position by placing the switch, marked NOSE GEAR, located on the overhead switch panel, in the NORMAL position. An advisory light, marked KNEEL SW ARMED, located on the pilot's advisory panel will illuminate when the kneel switch is in the KNEEL position. The green nose gear down light will be OUT and the red warning light in the landing gear handle will be on when the nose gear is kneeled.

Landing Gear Position Lights.

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

LANDING GEAR WARNING SYSTEM.

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

1. Airspeed is increased above the actuating range of approximately 60 knots.
2. The landing gear is extended and locked.
3. The warning silence button is depressed.

NOTE

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

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

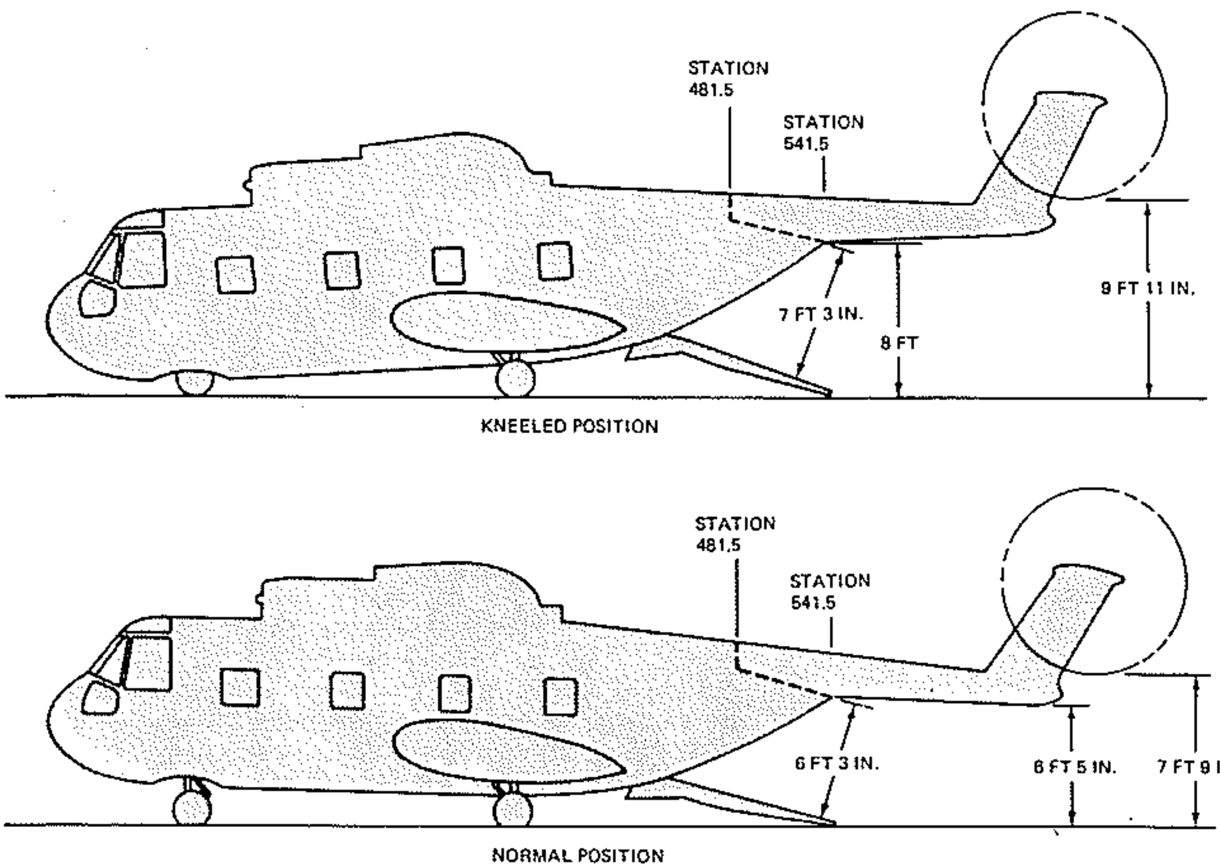


Figure 1-47. Landing Gear Kneeled

Landing Gear Warning Signal Light and Test Button.

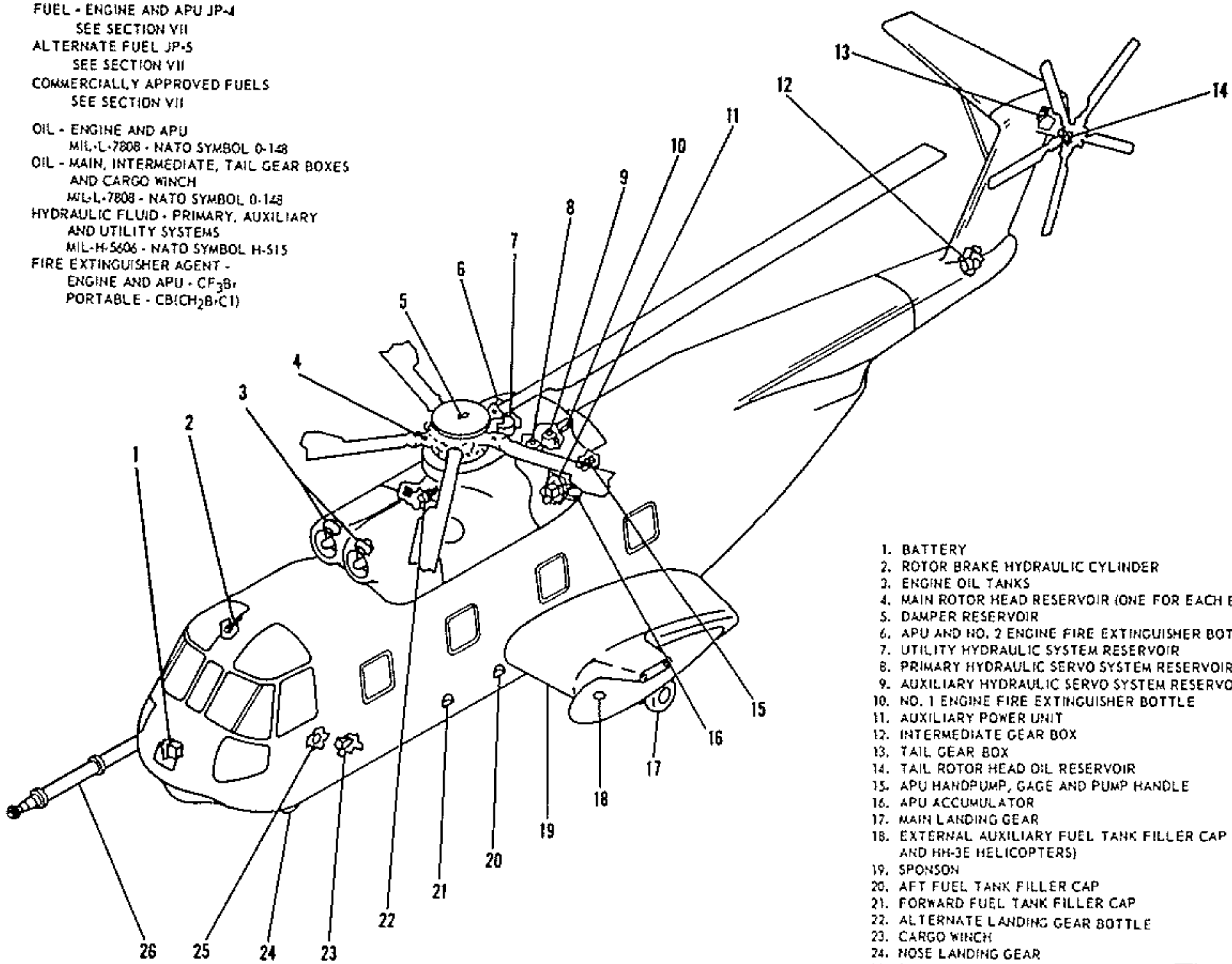
The landing gear warning signal light, located in the handle of the landing gear control handle lever, provides the pilots with a visual warning that the landing gear is up or not locked down safely when at airspeeds below the actuating range. The signal light is the flasher type located in the landing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

Landing Gear Warning System Silence Button.

The landing gear warning system silence button, marked WARNING SILENCE, is located on the landing gear control panel above the landing gear position lights. Whenever the warning system has been silenced, the system will automatically reset when the airspeed of the helicopter exceeds the

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

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



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

Figure 1-48. Servicing Diagram

actuating range of approximately 60 knots. The landing gear warning system operates on current from the dc essential bus through the circuit breaker, marked LAND GEAR WARN, located on the overhead dc circuit breaker panel.

LANDING GEAR ALTERNATE SYSTEM.

An alternate gear handle (figure 1-4), located on the left side of the cockpit console, is used to lower the landing gear should an electrical or hydraulic failure occur with the landing gears in the up and locked position. The handle mechanically unlocks the main landing gear uplocks, positions a directional valve, and discharges a 3000 psi preloaded air bottle. The compressed air charge actuates valves that vent the return side of the actuators to the reservoir and then drives the actuator to lower the landing gears. Fluid from the APU accumulator is used to lower the nose gear. The fluid is directed through an electrically-actuated valve to the top side of the nose gear actuating cylinder to lower the nose gear. After actuation of the alternate gear system, the emergency release valves, located on the right side of the main transmission well, must be manually reset before the main gear emergency air cylinder can be recharged and/or the landing gear operated in a normal manner. If the air charge in the cylinder has been depleted, when the alternate landing gear handle is actuated, main landing gear hydraulic pressure is vented back to the utility reservoir, the uplocks are disengaged, and the main landing gears will lower by their own weight.

NOTE

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

BRAKE SYSTEM.

The main landing gear wheels are each equipped with hydraulic brakes that are operated by toe brakes located on the pilot's and copilot's pedals. A parking brake system is also provided. The parking brake handle (figures 1-4 and 1-5), marked PARKING BRAKE, and a decal, marked ON- DEPRESS TOE BRAKE THEN PULL-OFF-DEPRESS TOE BRAKE, is located on the side of the cockpit console. The parking brakes are applied by depressing the brake pedals, manually pulling the parking brake handle to the PARK position, and then releasing the brake pedals. Depressing the left brake pedal will release the parking brakes, causing the parking brake handle to return to the OFF position.

EMERGENCY EQUIPMENT.

PORTABLE FIRE EXTINGUISHERS.

One hand-operated fire extinguisher (figure 3-7) is located on the bulkhead behind the pilot's seat. A second fire extinguisher (figure 3-7) is located on the left-hand side panel above the ramp. The fire extinguishers are filled with bromochloromethane (CB). The extinguishers are held in place by a bracket with a tight-fitting quick-release spring.

WARNING

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

NOTE

Bromochloromethane is highly corrosive to helicopter metal, paint, and plexiglass. If the fire extinguisher starts to leak, the extinguisher should be inverted and the control valve depressed (cover the nozzle with a cloth or aim into a can, etc., to catch any liquid which may be discharged). This will release the stored charge of pressurizing gas with a minimum discharge of fluid and render the extinguisher harmless. The extinguisher should be returned to its bracket and be reported for replacement in Form 781.

FIRST AID KITS.

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

CREW ALARM BELL.

The crew alarm bell (figure 3-7), located overhead in the cargo compartment, is used to warn crew and passengers in an emergency. The crew alarm bell operates on direct current from the battery bus and is protected by a circuit breaker, marked CREW ALARM, located on the battery bus circuit breaker panel. The alarm bell can be sounded regardless of the position of the battery switch.

Crew Alarm Bell Switch.

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

PYROTECHNIC KIT.

Provisions are made on the bulkhead behind the copilot's seat for mounting a pyrotechnic kit (figure 3-7) that contains a pistol and 12 cartridges. The pyrotechnic kit is a water-tight floatable container in which pistol and cartridges are stowed and is easily removed. On CH-3E helicopters Serial Nos. AF66-13285 and subsequent, and all HH-3E helicopters, provisions are made to stow the pyrotechnic kit on the upper right-hand side of the entrance to the pilot's compartment.

CRASH AXE.

One crash axe (figure 3-7) is installed below the step at the pilot's compartment entrance. The axe is secured by a bracket and a strap.

LIFERAFTS.

There are provisions for three liferafts that can be secured by straps to fittings in the ceiling of the cargo compartment. Two 6-man liferafts (figure 3-7) can be installed above the cargo compartment escape hatch, and one 20-man liferaft (figure 3-7) can be installed above the aft ramp on the right-hand side of the cargo compartment.

FIRE DETECTION SYSTEM.

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

ENGINE FIRE WARNING LIGHTS AND TEST SWITCH PANEL.

The red engine fire warning lights and test switch panel (figure 1-14) are located on a plate, marked FIRE WARN, on the instrument panel. The lights are marked No. 1 ENG and No. 2 ENG. The switch has two marked positions, FIRE TEST and OFF. The lights are pushbutton type that have a manual warning reset capability. The light(s) are depressed when it is desired to deenergize the audible fire warning system. The copilot's spring-loaded toggle switch, located on the upper part of the copilot's instrument panel, also deenergizes the audible fire warning system. In addition, four red engine fire warning lights, two for each engine, are installed in the engine fire emergency T-handles, located below the decal marked FIRE EMER SHUTOFF SELECTOR on the overhead switch panel. The left handle is marked No. 1 ENGINE and the right handle is marked No. 2 ENGINE. A light on the instrument panel and a light in either the No. 1 or No. 2 engine fire emergency T-handle will illuminate in event of a fire in the corresponding engine compartment. To test the engine fire detector system, place the spring-loaded switch in the (up) FIRE TEST position. The fire warning lights on the instrument panel and in the emergency T-handles should go on. The switch will return to the OFF position when released and the lights will go out.

AUDIBLE FIRE WARNING SYSTEM.

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

APU FIRE WARNING LIGHT.

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

FIRE EXTINGUISHER SYSTEM.

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

WARNING

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

ENGINE FIRE EXTINGUISHER SYSTEM.

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

NOTE

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

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

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

Engine Fire Extinguisher Switch.

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

AUXILIARY POWER UNIT FIRE EXTINGUISHER SYSTEM.

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

Auxiliary Power Unit Fire Detector and Extinguisher Control Panel.

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

EMERGENCY EXITS.

For emergency entrances and routes of escape and exits, see figure 3-6.

PILOT'S COMPARTMENT SLIDING WINDOWS.

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

PERSONNEL DOOR EMERGENCY EXIT.

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

RAMP.

The ramp consists of a forward and aft ramp that can be opened hydraulically or manually to provide an exit from the rear of the cargo compartment. The aft ramp can be opened in flight, on the ground, or on the water by placing the master switch, located on the pilot's ramp control panel (figure 4-28), marked RAMP MASTER SW, CREW-OFF-PILOT, in the PILOT position, and placing the switch marked AFT RAMP in the DOWN position. When the ramp master switch is in the CREW position, the aft ramp may be opened by the crewmember, using the crew ramp controls located above the ramp on the right-hand compartment side panel. In the event of an electrical or hydraulic failure, open the aft ramp manually from inside the helicopter by pulling forward on the handle attached to the manual shutoff valve located on the right side of the cargo compartment under a flap of soundproofing, marked EXIT RELEASE HANDLE INSIDE. When the handle is positioned forward, the manual shutoff valve is actuated to vent the aft ramp cylinders, and a cable attached to the handle releases the uplock to unlock the aft ramp. The ramp will open by its own weight. The rate of opening is controlled by a restrictor. The aft ramp may be opened from outside of the helicopter by removing the handle from the clips beneath the cover on the aft fuselage, marked RAMP EXIT RELEASE HANDLE INSIDE, and pulling the handle down. The outside handle is connected by a cable to the handle attached to the manual shutoff valve. The forward ramp may be opened from inside the cargo compartment by pulling up on the manual override handles on top of the forward ramp cylinders. The ramp will open by its own weight. The rate of opening will be controlled by a restrictor.

WARNING

The forward ramp should only be opened on the ground.

CARGO COMPARTMENT WINDOWS.

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

WARNING

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

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

CAUTION

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

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

PILOT'S AND COPILOT'S SEATS.

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

Seat Height Adjustment Lever.

The seat height adjustment levers (figure 1-4) are the rear levers at the right of the pilot's and copilot's seats. The levers are pulled up to release the height adjustment lockpins.

Seat Foreward and Aft Adjustment Lever.

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

Shoulder Harness Lock Lever.

A two-position shoulder harness inertia reel lock lever (figure 1-4) is located at the left side of each seat. When the lever is in the unlocked (aft) position, the shoulder harness cable will extend to allow the occupant to lean forward; however, the inertia reel will automatically lock if an impact force between two and three g's in any direction is encountered. When this occurs, the inertia reel will remain locked until the lever is moved to the locked position and then to the unlocked position. When the lever is placed in the locked (forward) position, the shoulder harness cable is locked so that the occupant is prevented from leaning forward. The locked position is used to provide an added safety precaution when a crash landing is anticipated, or when desired during critical operations.

CREWMAN'S SEAT.

The crewman's seat (figure 1-3), located at the entrance to the cockpit and aft of any between the pilot's and copilot's seats, may be folded against the entrance wall to facilitate cockpit entrance and exit. The seat is equipped with a safety belt.

AUXILIARY EQUIPMENT.

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

Heating System

Anti-Ice Systems

Communication and Associated Electrical Equipment

Lighting Equipment

Auxiliary Power Unit

Cargo Compartment