FOR TRAINING ONLY

FLIGHT MANUAL

USAF SERIES

CH-3B helicopter

AFO(603)64042
F09603-72-D-1406-0078

This change incorporates Safety Supplements T.O. 1H-3(C)B-1SS-19, -24, -25 and -29, and Operational Supplements T.O. 1H-3(C)B-1S-17, -18, -26 and -27.

COMMANDERS ARE RESPONSIBLE FOR BRINGING THIS PUBLICATION TO THE ATTENTION OF ALL AFFECTED PERSONNEL

SEE SUPPLEMENT INDEX T.O. 0-1-1-5 FOR CURRENT STATUS OF SAFETY/OPERATIONAL SUPPLEMENTS AND FLIGHT CREW CHECKLISTS

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AFLC RAFB, GA REPRINT

21 MARCH 1969

Basic And All Changes Have Been Merged To Make This A Complete Publication.

CHANGE 5 – 2 FEBRUARY 1973

LIST OF EFFECTIVE PAGES

Insert latest changed pages; dispose of superseded pages in accordance with applicable regulations.

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CURRENT FLIGHT CREW CHECKLIST

T.O. 1H-3(C)B-1CL-1
22 MARCH 1969
Change 4 — 2 February 1973

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ALL PILOTS READ THIS

SCOPE. This manual contains the necessary information for safe and efficient operation of the CH-3B. These instructions provide you with a general knowledge of the helicopter, its characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and therefore, basic flight principles are avoided.

PERMISSIBLE OPERATIONS. The Flight Manual takes a "positive approach" and normally states only what you can do. Unusual operations or configuretions (such as asymmetrical loading) are prohibited unless specifically covered herein. Clearance must be obtained from the Flight Manual Manager (WRAMA/MMEAH) before any questionable operation is attempted which is not specifically permitted in this manual.

HOW TO BE ASSURED OF HAVING THE LATEST DATA. Refer to T.O. 0-1-1-5, monthly supplements thereto and flyleaf page of latest Safety and Operational Supplements for listing of current Flight Manuals, Checklists, and Safety and Operational Supplements.

SAFETY SUPPLEMENTS. Information involving safety will be promptly forwarded to you by Safety Supplements. Supplements covering loss of life will get to you in 48 hours by TWX, and those concerning serious damage to equipment within 10 days by mail. The title page of the Flight Manual and the title block of each Safety Supplement should be checked to determine the effect they may have on existing supplements. You must remain constantly aware of the status of all supplements - current supplements must be complied with but there is no point in restricting your operation by complying with a replaced or rescinded supplement.

CHECKLISTS. The Flight Manual contains only amplified checklists. Condensed (abbreviated) checklists have been issued as separate technical orders - see the back of the title page for the T.O. number of your latest checklist. Line items in the Flight Manual and checklists are identical with respect to arrangement and item number. Whenever a Safety Supplement affects the condensed (abbreviated) checklist, write in the applicable change on the affected checklist page. As soon as possible, a new checklist page incorporating the supplement will be issued. This will keep handwritten entries of Safety Supplement Information in your checklist to a minimum.

HOW TO GET PERSONAL COPIES. Each flight crewmember is entitled to personal copies of the Flight Manual, Safety Supplements, and Checklists. The required quantities should be ordered before you need them to assure their prompt receipt. Check with your supply personnel - it is their job to fulfill your Technical Order requests. Basically, you must order the required quantities on the Publication Requirements Table (T.O. 0-1-1). Technical Orders 00-5-1 and 00-5-2 give detailed information for properly ordering these publications. Make sure a system is established at your base to deliver these publications to the flight crews immediately upon receipt.

FLIGHT MANUAL AND CHECKLIST BINDERS. Loose leaf binders, stock list numbers 7510-664-5114, and sectionalized tabs, stock list numbers 33-60162-3308, are available for use with your manual. These are obtained through local purchase procedures and are listed in the Federal Supply Schedule (FSC Group 75, Office Supplies, Part 1). Binders are also available for carrying your condensed (abbreviated) checklist. These binders contain plastic envelopes into which individual checklist pages are inserted. They are available in three capacities and are obtained through normal Air Force supply under the following stock list numbers: 7510-766-4268; -4269, and -4270 for 15, 25, and 40 envelope binders respectively. Check with your supply personnel for assistance in securing these items.

WARNINGS, CAUTIONS, AND NOTES. The following definitions apply to "Warnings," "Cautions," and "Notes" found throughout the manual.

WARNING	Operating procedures, techniques, etc., which will result in personal injury or loss of life if not carefully followed.
CAUTION	Operating procedures, techniques, etc., which will result in damage to equipment if not carefully followed.
NOTE	An operating procedure, technique, etc., which is considered essential to emphasize.

YOUR RESPONSIBILITY - TO LET US KNOW. Every effort is made to keep the Flight Manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusion of the latest data in the manual. However, we cannot correct an error unless we know of its existence. In this regard, it is essential that you do your part. Comments, corrections, and questions regarding this manual or any phase of the Flight Manual program are welcomed. These should be forwarded through your Command Headquarters to WRAMA/MMEAH, Robins AFB, Ga., 31098.

GLOSSARY OF TERMS AND ABBREVIATIONS

AC - Alternating current

ACCELERATION - The rate of change of velocity.

ADF - Automatic direction finder.

AIRSPEED

KCAS - Knots calibrated airspeed
KIAS - Knots indicated airspeed
KTAS - Knots true airspeed

ALT - Altitude

ASE - Automatic stabilization equipment

BAR ALT - Barometric altitude controller

BDHI - Bearing distance heading indicator

BIM - Blade Inspection Method

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.

BROC - Best rate of climb.

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
CENT - Centigrade

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

DEG - Degree

DENS - Density

DENS ALT - Density altitude

DG - Directional gyro

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

DRAG DIVERGENCE - 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 requirements in excess of power available.

DUAL PROBE - Pitot static system with dual probes on one side.

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

°F - Degrees Fahrenheit
FAT - Free air, ambient, or outside air temperatures
FOD - Foreign object damage
FPM - Feet per minute
FT - Feet
FT/MIN - Feet per minute
G's - Forces of gravity
GAL - Gallons
GCA - Ground-controlled approach
GSI - Glide slope indicator
GW - Gross weight
HP - Horsepower
HORIZ DIST - Horizontal distance
HR - Hour
HYDROSTATIC ROLL ANGLE - Angle of roll when helciopter is on water.
H-V - Height velocity
IAS - Indicated airspeed
IGE - In ground effect
IN - Inches
INV - Inverter
KN - Knots
KTS - Knots
KVA - Kilovolt-amperes

GLOSSARY OF TERMS AND ABBREVIATIONS (Cont)

LAT - Latitude
LBS/HR - Pound per hour

LEFT AND RIGHT HAND PROBE - Pitot static system with a probe on the right and a probe on the left of the pilot's compartment.

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
MEAN WATERLINE - The mean of the highest and lowest waterline for a given set of conditions, gross weight, sea state etc.
MIN - Minutes
MSL - Mean sea level
NA/Mi - Nautical miles
N_f - Power turbine speed
N_g - Gas generator speed
N_r - Rotor speed

OGE - Out of ground effect
This means hovering at a height of approximately one rotor diameter (72 feet or higher).

P_2 - Compressor inlet total pressure
P_3 - Compressor discharge pressure
PRESS - Pressure
PRES ALT - Pressure altitude
PSI - Pounds per square inch
Q - Torque

ROC - Rate of climb
ROD - 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.

SHP - Shaft horsepower

SL - Sea level

SPEED SELECTOR - Rotor speed selector.

STD DAY - Standard day atmospheric conditions

STD TEMP - Standard day temperature

T_2 - Compressor inlet air temperature

FAT 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
TEE HANDLE - Fuel firewall shut-off handle
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.

UT - Utility hydraulic pressure circuit breaker
UT - Utility
VA - Volt amperes
VAC - Volts alternating current
Vmax - Maximum allowable airspeed

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

NOTE: Airspeeds in this manual refer to indicated airspeeds except when specified otherwise.

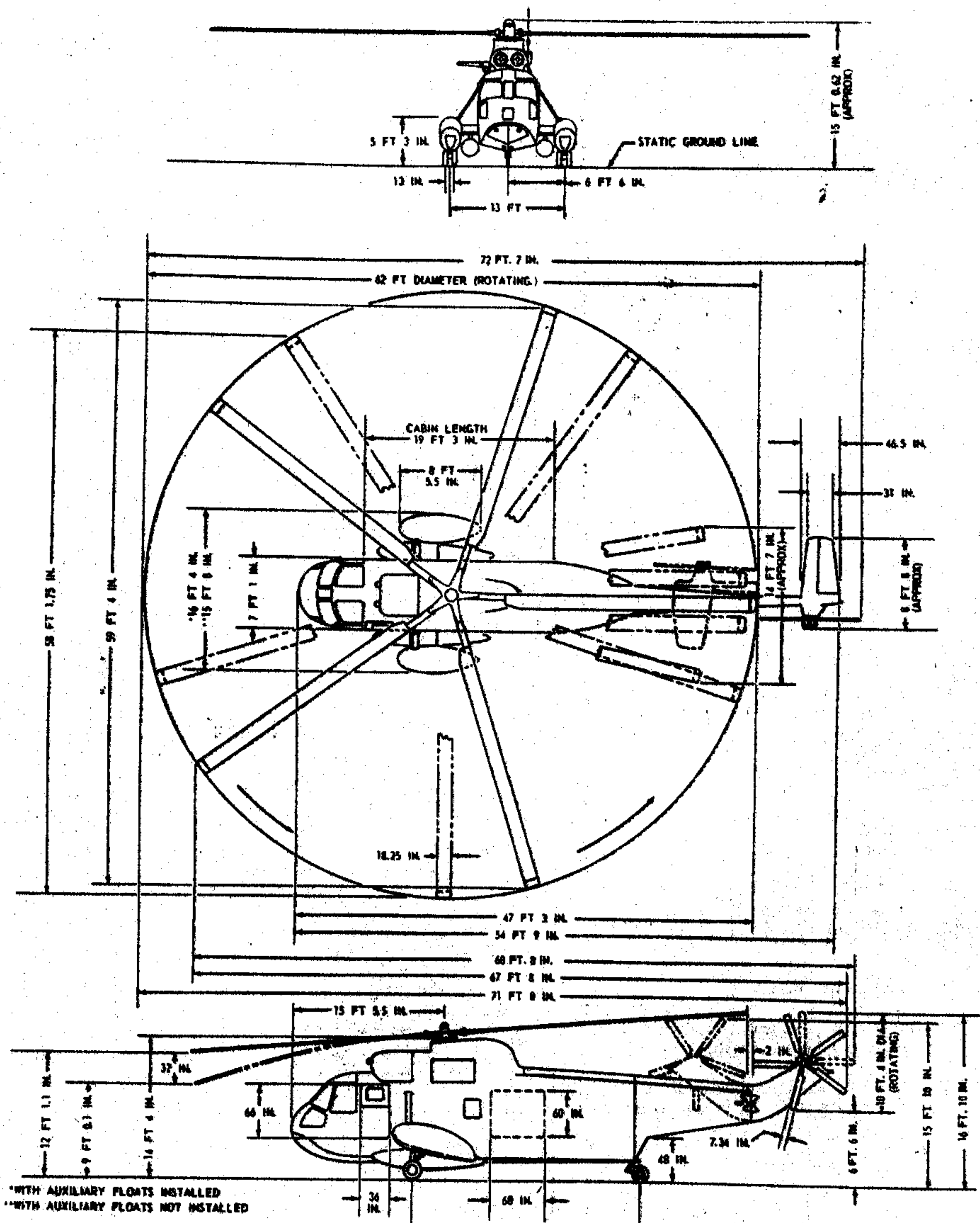

Figure 1-1. Three Dimensions

SECTION I

DESCRIPTION

The function of this section is to describe the helicopter and its systems and controls which contribute to the physical act of flying the helicopter, including all emergency equipment that is not part of auxiliary equipment.

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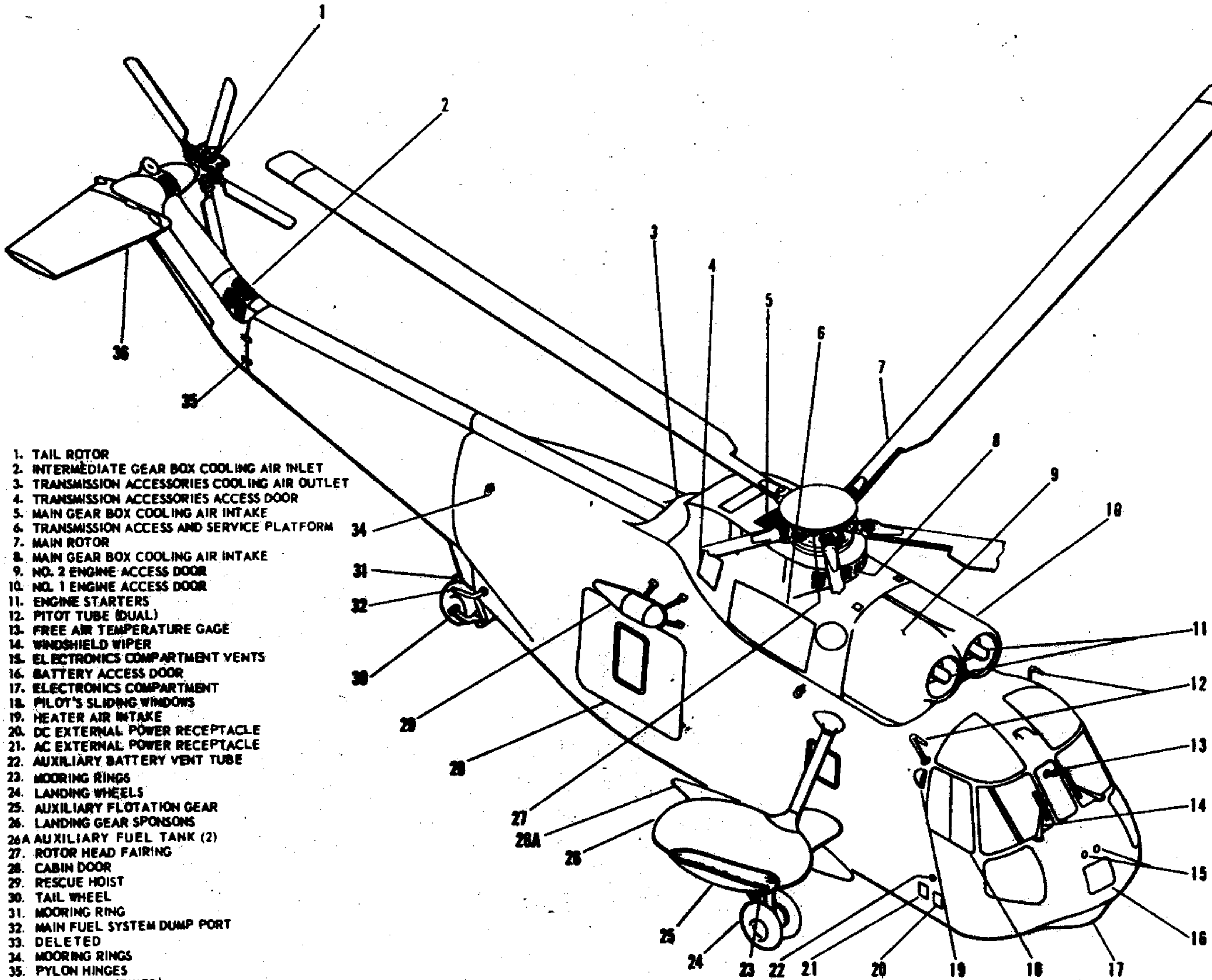
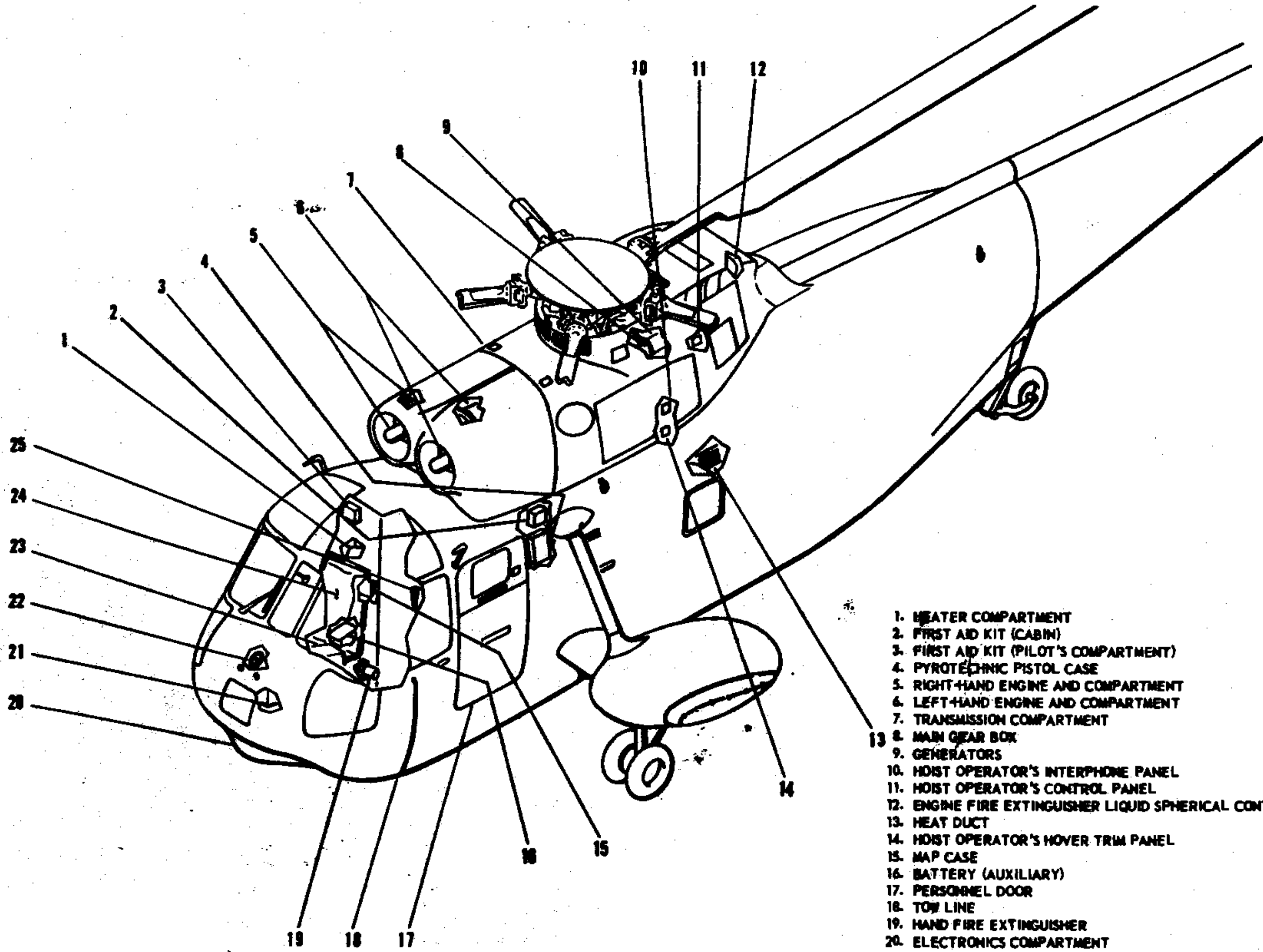
THE HELICOPTER.

The model CH-3B helicopter is manufactured by Sikorsky Aircraft, Division of United Aircraft Corporation, Strafford, Connecticut. The helicopter is designed for long range overwater surface recovery while operating from ship or shore base. A normal crew consists of pilot, copilot, and flight mechanic. Configuration is a single main rotar, twin turbine powered helicopter with emergency amphibious capabilities. The fuselage is all metal semi-monocoque construction, with a boat type hull bottom and two outrigger sponsons. The fuselage is comprised of five sections: (1) the forward fuselage section, (2) the hull, (3) the aft fuselage section, (4) the tail cone section, (5) the pylon. The forward fuselage section and hull consists of pilot's compartment, engine compartment, transmission compartment, cabin and fuel tanks. The electronics-radio compartment is located in the forward portion of the hull, above which is the pilot's compartment which is entered from the cabin. The engine compartment is located above the forward portion of the cabin. The two turbine engines are mounted side by side in the engine compartment, with the engine shafts pointed aft into the main gear box. Directly aft of the engine compartment is the transmission compartment, housing the main gear box. The main rotor assembly to which the five rotor blades are attached is splined to the main gear box drive shaft. Shafting extends aft from the main gear box lower housing to the intermediate and tail gear box, to drive the tail rotor. Directly below the engine and transmission compartments is the cabin. The cabin is twenty-four feet one inch long, six feet six inches wide and five feet ten inches high. The cabin may be entered through the sliding door on the right side of the cabin, or through a hinged personnel door on the left side of the cabin. The two multi-cell fuel tanks which contain six hundred and ninety-five galons of JP-4 or JP-5 are installed in the hull below the cabin floor. The aft fuselage sections and the tail cone sections extend aft from the rear cabin bulkhead. The compass valve and compensator, HF transmitter, HF antenna coupler, and ADF receiver are installed in the tail cone. A horizontal stabilizer is installed on the upper right hand side of the pylon. The intermediate gear box is installed in the lower portion of the plyon with a shaft extending upward to the tail



1. TAIL ROTOR
2. INTERMEDIATE GEAR BOX COOLING AIR INLET
3. TRANSMISSION ACCESSORIES COOLING AIR OUTLET
4. TRANSMISSION ACCESSORIES ACCESS DOOR
5. MAIN GEAR BOX COOLING AIR INTAKE
6. TRANSMISSION ACCESS AND SERVICE PLATFORM
7. MAIN ROTOR
8. MAIN GEAR BOX COOLING AIR INTAKE
9. NO. 2 ENGINE ACCESS DOOR
10. NO. 1 ENGINE ACCESS DOOR
11. ENGINE STARTERS
12. PITOT TUBE (DUAL)
13. FREE AIR TEMPERATURE GAGE
14. WINDSHIELD WIPER
15. ELECTRONICS COMPARTMENT VENTS
16. BATTERY ACCESS DOOR
17. ELECTRONICS COMPARTMENT
18. PILOT'S SLIDING WINDOWS
19. HEATER AIR INTAKE
20. DC EXTERNAL POWER RECEPTACLE
21. AC EXTERNAL POWER RECEPTACLE
22. AUXILIARY BATTERY VENT TUBE
23. MOORING RINGS
24. LANDING WHEELS
25. AUXILIARY FLOTATION GEAR
26. LANDING GEAR SPONSONS
26A AUXILIARY FUEL TANK (2)
27. ROTOR HEAD FAIRING
28. CABIN DOOR
29. RESCUE HOIST
30. TAIL WHEEL
31. MOORING RING
32. MAIN FUEL SYSTEM DUMP PORT
33. DELETED
34. MOORING RINGS
35. PYLON HINGES
36. STABILIZER (FIXED)

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



1. HEATER COMPARTMENT
2. FIRST AID KIT (CABIN)
3. FIRST AID KIT (PILOT'S COMPARTMENT)
4. PYROTECHNIC PISTOL CASE
5. RIGHT-HAND ENGINE AND COMPARTMENT
6. LEFT-HAND ENGINE AND COMPARTMENT
7. TRANSMISSION COMPARTMENT
8. MAIN GEAR BOX
9. GENERATORS
10. HOIST OPERATOR'S INTERPHONE PANEL
11. HOIST OPERATOR'S CONTROL PANEL
12. ENGINE FIRE EXTINGUISHER LIQUID SPHERICAL CONTAINER
13. HEAT DUCT
14. HOIST OPERATOR'S HOVER TRIM PANEL
15. MAP CASE
16. BATTERY (AUXILIARY)
17. PERSONNEL DOOR
18. TOW LINE
19. HAND FIRE EXTINGUISHER
20. ELECTRONICS COMPARTMENT
21. BATTERY (PRIMARY)
22. HEATER REGISTER
23. RADIO CONSOLE
24. PILOT'S SEAT
25. RELIEF TUBE

Figure 1-3. General Arrangement Interior Diagram (Typical) rotor gear box at the top of the pylon. The five-bladed tail rotor is splined to the tail rotor gear box. The five main rotor blades may be folded parallel to the fuselage and the pylon may be folded forward along the right side of the tail cone.

DIMENSIONS.

Length.

Maximum, main rotor blades extended	72 feet 7 inches
Minimum, main rotor blades and pylon folded	47 feet 3 inches

Height.

Maximum to top of tail rotor, blade vertical	16 feet 10 inches
Minimum, tail rotor blades	16 feet 10 inches

Width.

Minimum, main rotor blades and pylon folded	15 feet 8 inches. Auxiliary Floats Not Installed 16 feet 4 inches. Auxiliary Floats Installed
Main rotor diameter	62 feet 0 inches
Tail rotor diameter	10 feet 4 inches

Minimum Main Rotor Ground Clearance.

(Tip clearance - Forward sector)

9 feet 1 inch

Tail Rotor Ground Clearance.

6 feet 6 inches

Main Landing Gear Tread.

13 feet 0 inches

ENGINES.

The model CH-3B helicopter is powered by two General Electric T58-GE-1 axial flow gas turbine engines (figure 1-11) of the turboshaft type, incorporating the free power turbine principal. Each engine develops 1300 shaft horsepower but is limited to 1250 SHP by the main gear box. The engines are located side-by-side above the cargo compartment, forward of the main gear box. Each engine consists of the following major components: an axial-flow compressor, combustion chambers, a two stage gas generator turbine, and a single stage power turbine, which 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 principal 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 hydro-mechanical fuel metering unit provides maximum engine performance without exceeding safe engine operating limits. In the normal operating range engine speed is selected by positioning the speed selector. 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 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.

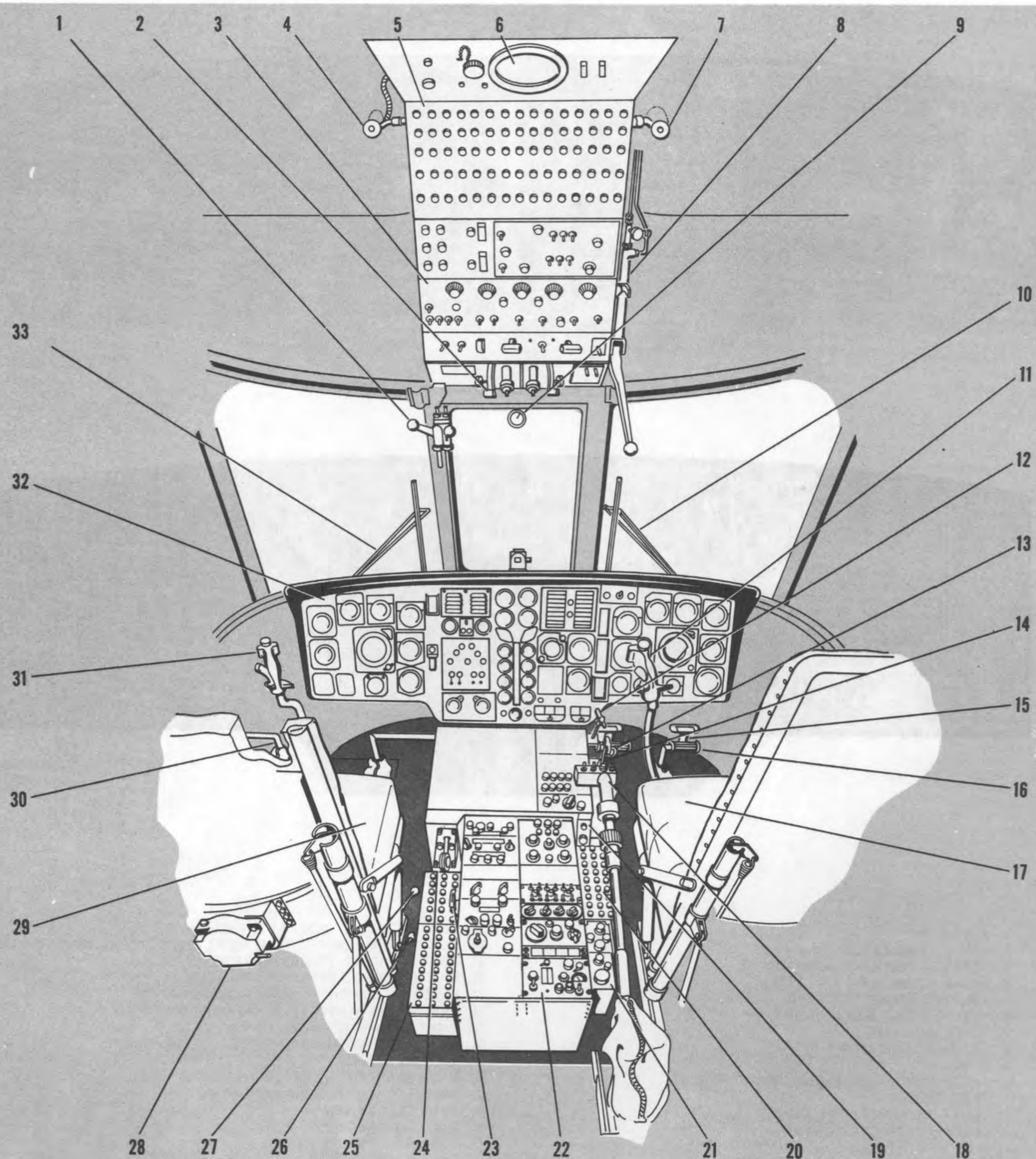
The ten stage compressor consists of the compressor rotor and stator. The compressor rotor is supported by the front frame section and the compressor rear frame section. The stator is bolted between the front frame section and compressor rear frame. The primary purpose of the compressor is to compress 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 and the first three stages of the stator vanes (figure 1-11) 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 CHAMBERS.

The combustion chambers are where 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 for 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.

GAS GENERATOR TURBINE.

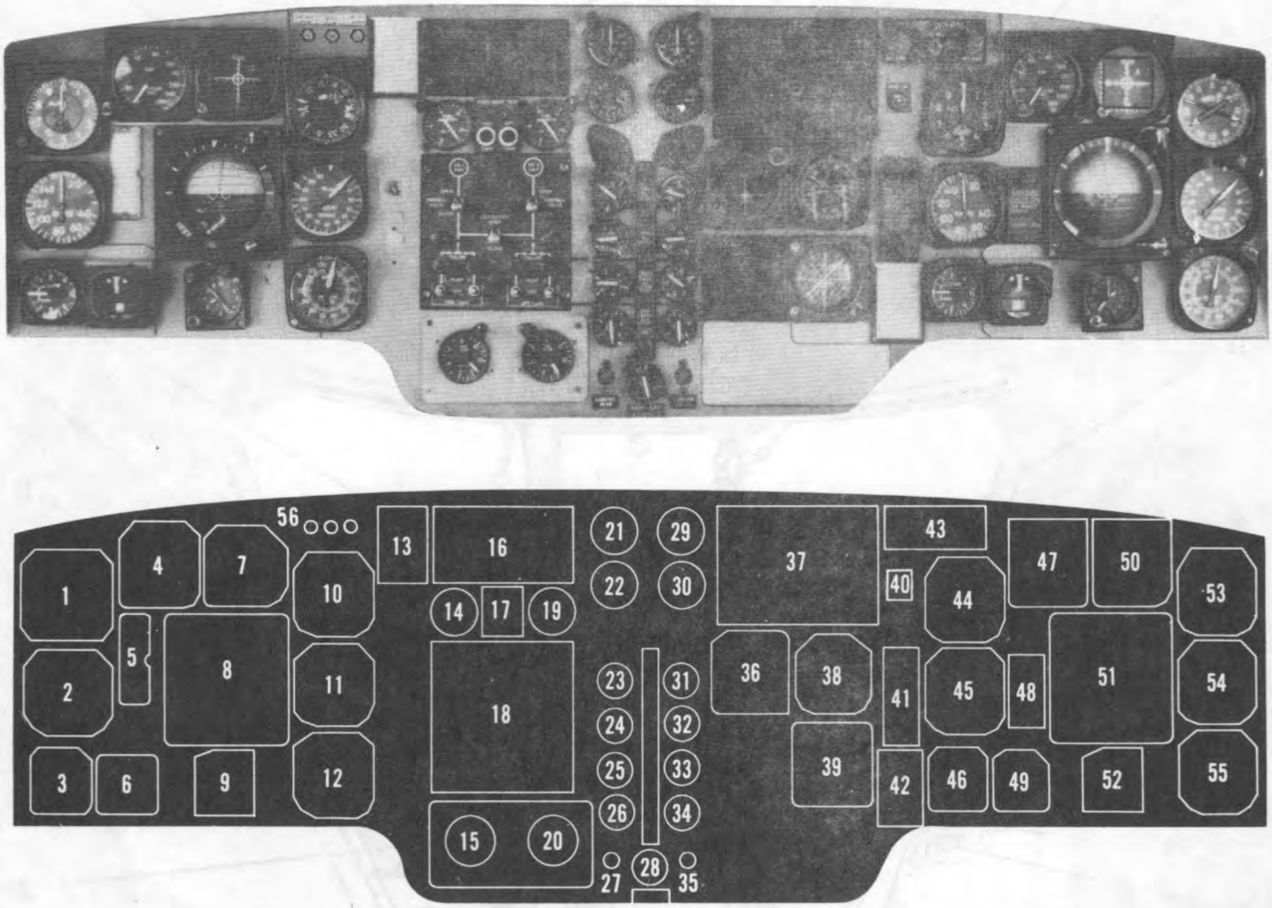
The two stage gas generator turbine is the rotating component which is coupled directly to the compressor. It extracts the required power from the exhaust gases to drive the compressor. The turbine nozzles that comprise the stator blades direct the exhaust gases to the turbine wheels.



1. MANUAL THROTTLES
2. ENGINE CONTROL QUADRANT
3. OVERHEAD SWITCH PANEL
4. COPILOT'S SPOTLIGHT
5. OVERHEAD CIRCUIT BREAKER PANEL
6. COCKPIT DOME LIGHT
7. PILOT'S SPOTLIGHT
8. ROTOR BRAKE
9. FREE AIR TEMPERATURE GAGE
10. PILOT'S WINDSHIELD WIPER
11. PILOT'S CYCLIC STICK
12. PARKING BRAKE HANDLE
13. WINDSHIELD WIPER CONTROL KNOB
14. TAIL WHEEL LOCK HANDLE
15. TOE BRAKE
16. PILOT'S TAIL ROTOR PEDALS
17. PILOT'S SEAT
18. PILOT'S COLLECTIVE PITCH LEVER
19. BATTERY BUS CIRCUIT BREAKER PANEL
20. RADIO CIRCUIT BREAKER PANEL
21. EMERGENCY FLOTATION GEAR CONTROL PANEL
22. COCKPIT CONSOLE
23. EMERGENCY LANDING GEAR RELEASE HANDLE
24. LANDING GEAR CONTROL PANEL
25. CIRCUIT BREAKER PANEL
26. SEAT HEIGHT ADJUSTMENT KNOB
27. SEAT FORWARD AND AFT ADJUSTMENT KNOB
28. COPILOT'S SHOULDER HARNESS INERTIA REEL
29. COPILOT'S SEAT
30. COPILOT'S TAIL ROTOR PEDALS
31. COPILOT'S CYCLIC STICK
32. INSTRUMENT PANEL
33. COPILOT'S WINDSHIELD WIPER

S 1967 (R2)

Figure 1-4. Pilots Compartment (Typical)



1. ALTIMETER
2. AIRSPEED INDICATOR
3. VERTICAL VELOCITY INDICATOR
4. RADAR ALTIMETER
5. AIRSPEED CORRECTION CARD
6. TURN-AND-SLIP INDICATOR
7. HOVER INDICATOR
8. ATTITUDE INDICATOR
9. CLOCK
10. BEARING-DISTANCE-HEADING INDICATOR (BDHI)
11. TORQUEMETER
12. TRIPLE TACHOMETER (N_f AND N_r)
13. COMPASS CORRECTION CARD
14. FUEL QUANTITY GAGE
15. NO. 1 ENGINE FUEL FLOW INDICATOR
16. ADVISORY PANEL
17. FUEL QUANTITY GAGES TEST SWITCHES
18. FUEL MANAGEMENT PANEL
19. FUEL QUANTITY GAGE
20. NO. 2 ENGINE FUEL FLOW INDICATOR
21. NO. 1 ENGINE GAS GENERATOR TACHOMETER (N_g)
22. NO. 1 ENGINE POWER TURBINE INLET TEMPERATURE INDICATOR (T5)
23. NO. 1 ENGINE OIL TEMPERATURE INDICATOR
24. NO. 1 ENGINE OIL PRESSURE INDICATOR
25. MAIN GEAR BOX OIL PRESSURE INDICATOR
26. AUXILIARY SERVO HYDRAULIC PRESSURE INDICATOR
27. LEFT LANDING GEAR POSITION INDICATOR
28. UTILITY HYDRAULIC PRESSURE INDICATOR
29. NO. 2 ENGINE GAS GENERATOR TACHOMETER (N_g)
30. NO. 2 ENGINE POWER TURBINE INLET TEMPERATURE INDICATOR (T5)
31. NO. 2 ENGINE OIL TEMPERATURE INDICATOR
32. NO. 2 ENGINE OIL PRESSURE INDICATOR
33. MAIN GEAR BOX OIL TEMPERATURE INDICATOR
34. PRIMARY SERVO HYDRAULIC PRESSURE INDICATOR
35. RIGHT LANDING GEAR POSITION INDICATOR
36. COURSE DEVIATION INDICATOR (CDI) (ID-387/ARN)
37. CAUTION PANEL
38. RADIO MAGNETIC INDICATOR (RMI) (ID-250/ARN)
39. VELOCITY AND STEERING INDICATOR
40. RAWS TEST SWITCH
41. TAKE OFF CHECKLIST
42. COMPASS CORRECTION CARD
43. ENGINE FIRE WARNING LIGHTS AND TEST SWITCH
44. ALTIMETER
45. AIRSPEED INDICATOR
46. VERTICAL VELOCITY INDICATOR
47. RADAR ALTIMETER
48. LANDING CHECKLIST
49. TURN-AND-SLIP INDICATOR
50. HOVER INDICATOR
51. ATTITUDE INDICATOR
52. CLOCK
53. BEARING-DISTANCE-HEADING INDICATOR (BDHI)
54. TORQUEMETER
55. TRIPLE TACHOMETER (N_f AND N_r)
56. CHIP DETECTOR WARNING LIGHTS

Figure 1-5. Instrument Panel

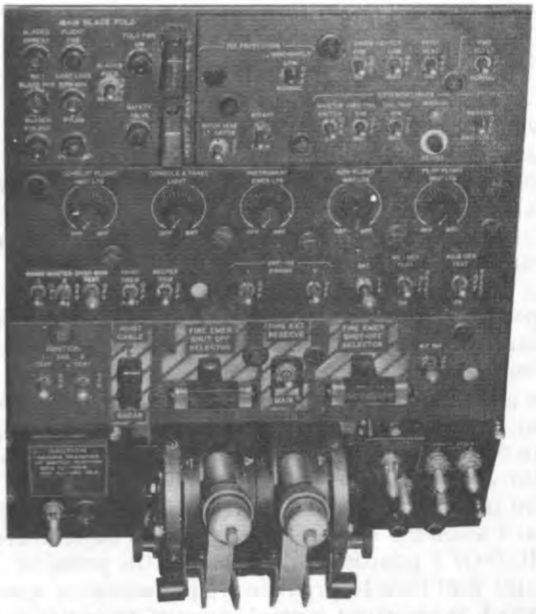


Figure 1-6. Overhead Switch Panel and Engine Control Quadrant (Typical)

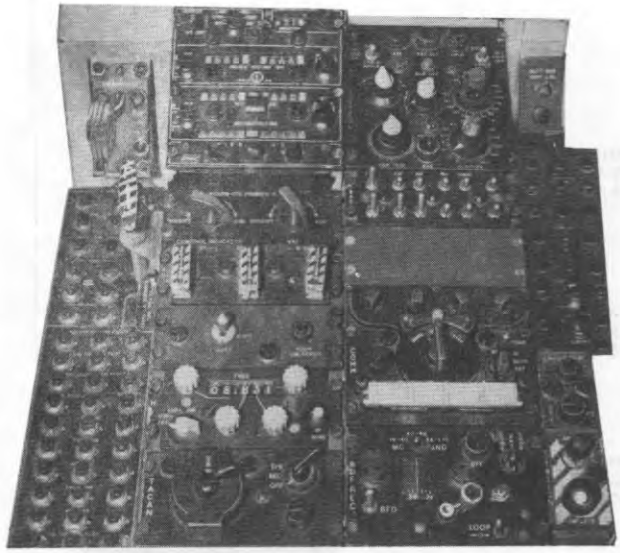


Figure 1-7. Cockpit Console (Typical)

POWER TURBINE.

The power turbine 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 COMPRESSOR SPEED (N_g).

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

FREE POWER TURBINE SPEED (Nf).

The free power turbine speed (Nf) 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.

ENGINE FUEL SYSTEM.

The engine fuel systems (figure 1-13), 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, 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 speed selector and the engine fuel controls will meter fuel to maintain the selected rotor speed.

Engine-Driven Fuel Pump.

A dual operation 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 hydro-mechanical 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

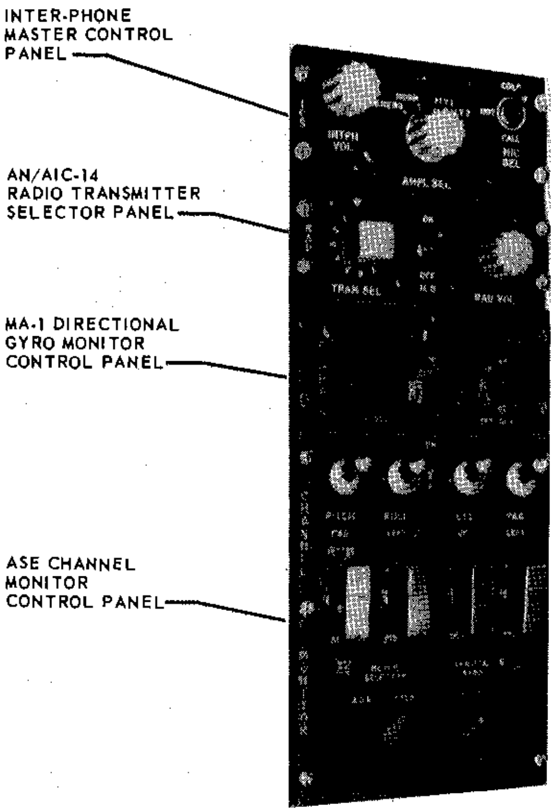


Figure 1-8. Pilot's Console (Typical)

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 in response to various internal operating signals, and meters fuel to the engine as a function of these integrated signals. The engine fuel control unit performs the following functions: prevents compressor stall, turbine overtemperature, rich or lean blowouts, governs gas generator idle and maximum speeds, and schedules inlet guide and stator vane positions to provide optimum compressor performance.

SPEED SELECTORS
(ENGINE SPEED SELECTORS).

Each of the two engine speed selectors located on the overhead engine control quadrant (figure 1-9), regulate an engine fuel control. Individual friction control knobs are provided on the sides of the overhead engine control (figure 1-9) to keep the speed selectors from creeping. Markings on the overhead quadrant are SHUT-OFF, GRD IDLE, and FULL TRVL. When the speed selectors are in the SHUT-OFF position, fuel flow to the fuel nozzles is stopped by means of a stopcock that prevents fuel from entering the combustion chambers during coast down of the engine gas generators or during operation of the fuel booster pumps with the engine shut down. The stopcock is directly actuated by the position of the speed selector and is open whenever the speed selector position is 6 degrees or more from SHUT-OFF and is closed when the selector is 3 degrees or less from the SHUT-OFF position. The GRD IDLE position schedules fuel flow to produce a gas generator speed low enough to transmit a small amount of power to the power turbine. Gas generator idle speed will vary with inlet air temperature. The engine fuel control unit compensates for varying inlet air temperatures by decreasing idle speed at low inlet temperatures. A limit stop at GRD IDLE prevents inadvertent retarding of the speed selectors below the idle speed of the engines. The speed selector may be retarded from the limit stop by exerting a downward pressure on the lever knob. When the speed selector is at the FULL TRVL position, the engine is producing maximum power turbine speed. The knobs, located on the ends of the speed selectors, rotate for vernier engine speed adjustment at any position of the speed selectors.

Speed/Selector Friction Knob
(Engine Speed Selector Friction Control Knob)

A engine speed selector friction control knob (speed selector friction knob) is located on each side of the overhead engine control quadrant (figure 1-9). When either knob is turned forward it tightens a friction brake within the quadrant to restrict movement of its respective speed selector. Two turns of the friction control knob are required from minimum to maximum friction. A friction quick release lever is used to release the friction when required.

Friction Quick Release Levers.

Two friction quick release levers are located on the overhead engine control panel (figure 1-9). There is one quick release lever for each speed selector. After the speed selector friction control knob has been applied for maximum friction to maintain the speed selector in position, the friction quick release lever may be actuated thus releasing the friction allowing the speed selector to be moved. Pulling the release lever to the down position releases the fricition applied by the friction control knob. To avoid excessive wear of the clutch plates, the friction quick release levers should be used only for quick release of the engine speed selector friction control knob. Continuous and excessive use of the fric-

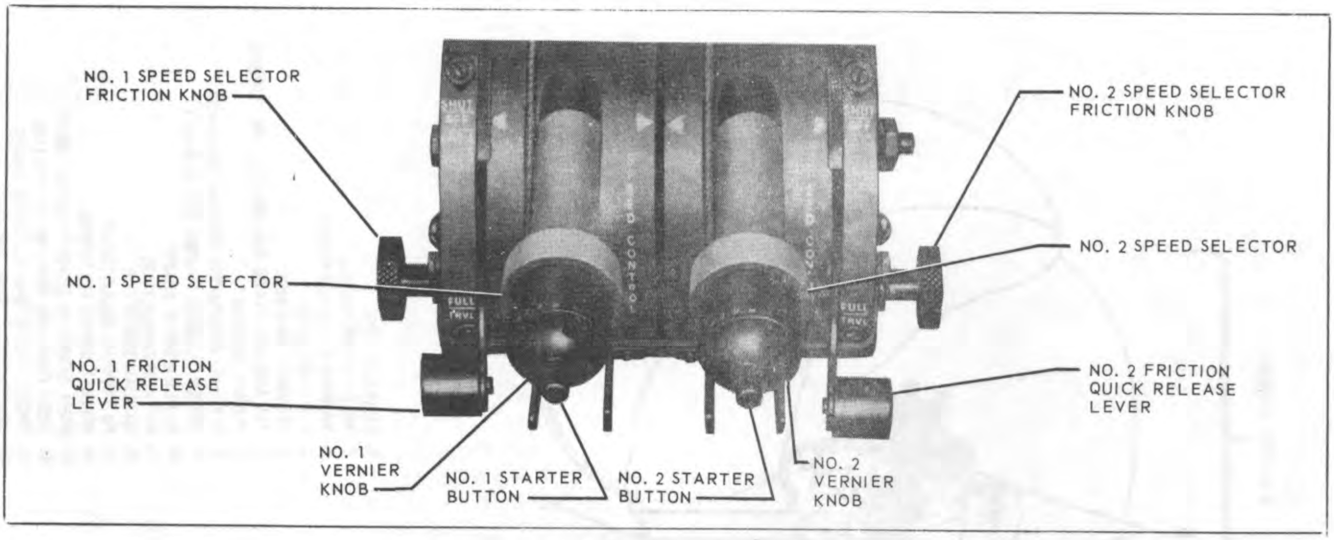


Figure 1-9. Overhead Engine Controls

tion quick release levers in lieu of the friction knob will cause the clutch plates to become gummy or spongy which, in turn, causes binding of the engine speed selectors. For friction, use the speed selector friction control knob. The friction quick release lever for each speed selector is located outboard of its respective engine speed selector.

Vernier Knobs
(Engine Speed Selector Vernier Knobs).

A vernier knob, located at the end of each engine speed selector, provides more vernier control of fuel flow to the engine for finer adjustment of the engine and for synchronization. An arrow marked RPM ADVANCE, located on the knobs, indicates the direction of increased rpm. Rotation of the speed selector vernier knob to the left mechanically increases the flow of fuel and power turbine rpm, rotation to the right decreases fuel flow and power turbine rpm. The vernier mechanism operates against the friction brake within the quadrant. Therefore, the friction control knobs must be sufficiently tight for the vernier control to adjust the engine speed.

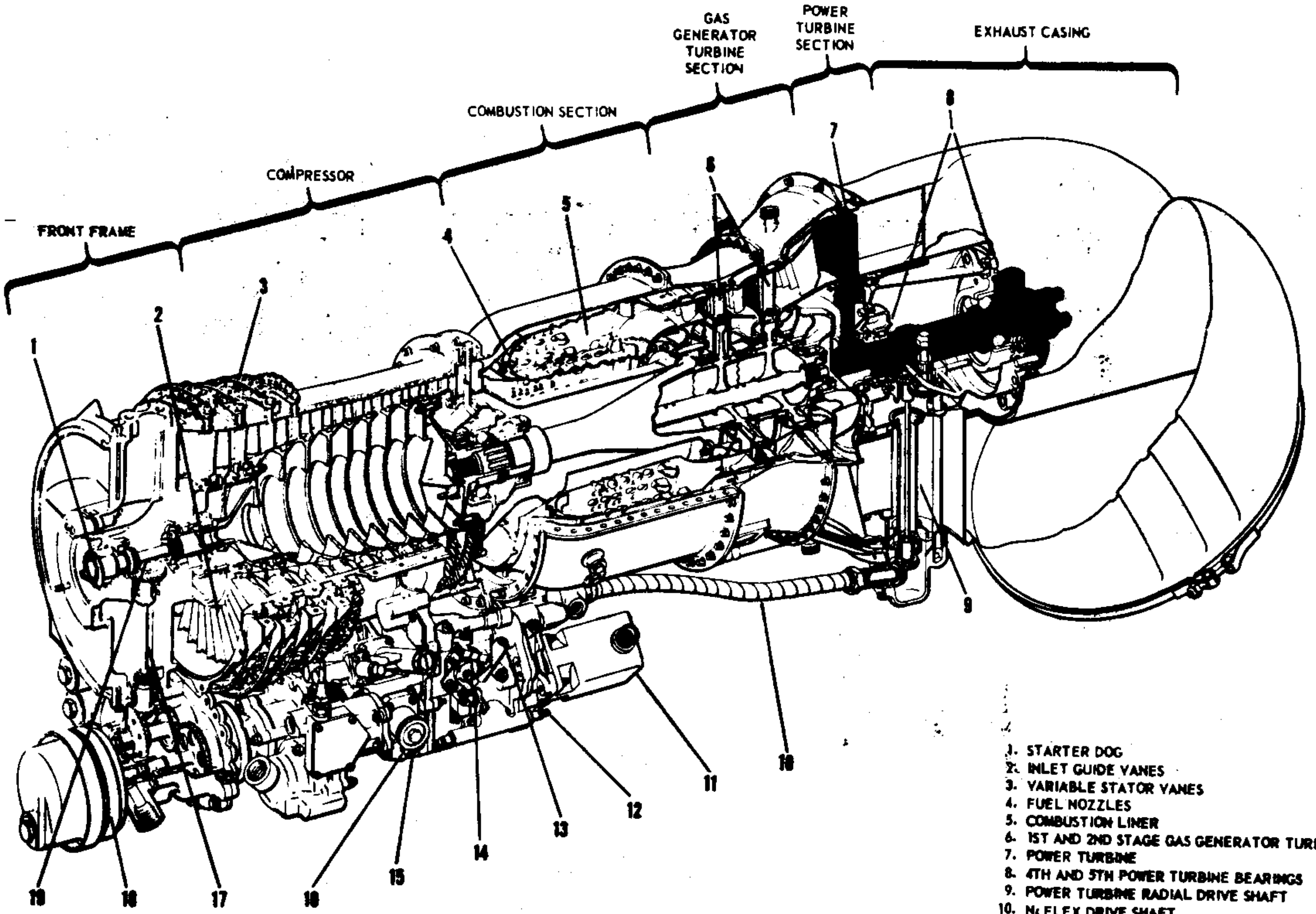
Manual Throttle Levers
(Engine Manual Throttle Levers).

Two manual throttle levers, one for each engine, are located on the left side of the accessory drive panel on the forward overhead control quadrant (figure 1-6). The manual throttle levers operate independently and are used in case of fuel control unit failure. Each manual throttle lever has positive open and closed stops and is connected by flexible cable and linkage directly to the main metering valve in each engine fuel control unit. The manual throttle system does not compensate for compressor inlet temperature or compressor discharge pressure. Consequently, the manual throttle levers must be moved smoothly and cautiously to avoid overtemperature and/or overspeed due to excessive fuel flow. The manual throttle levers meter fuel flow to maintain speeds at or above the speed settings of the speed selectors.

STARTER SYSTEM.

Each engine starting system (figure 1-15) consists mainly of a starter, starter relay, fuel valve, mode selector switch (manual or automatic starts), starter button, and an emergency start switch. The systems operate on current from the 28-volt essential bus and are protected by circuit breakers, marked STARTER 1 ENG 2, on the overhead circuit breaker panel. The engine starting system provides two modes of operation; normal and manual. The normal mode provides automatic starts and automatic starter dropout when engine operation becomes self-sustaining. The manual mode bypasses the automatic starter dropout feature and permits the pilot to motor the starter manually as long as the starter button is depressed. This mode is used primarily for battery starts. An electrically operated fuel valve, to aid in controlling overtemperature conditions during starting, prevents auxiliary start fuel flow from entering the flow divid-

Figure 1-10. Friction Quick Release



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 SPEED SELECTOR
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 1-11. Engine Cut-Away View

er when the valve is closed by the pilot. A series of safety interlocks in the normal or manual control circuit to each engine starter prevents the starter relay from closing should an unsafe condition exist.

The following conditions are required to close the safety interlocks when starting the No. 1 engine:

1. The accessory drive switch in ACCESS DR position.

2. The pylon locked in the flight position.

3. Eigher the blades spread, safety valve CLOSED, fold power master switch OFF, or the rotor brake ON.

The conditions required to close the safety interlocks when starting the No. 2 engine are as follows:

1. Blades spread.
2. Pylon locked in the flight position.
3. Safety valve switch CLOSED.
4. Fold master switch OFF.
5. The auxiliary servo must be pressurized.

Before starting, check that the 10 amp blade fold circuit breaker, marked BLADE FOLD, is pushed in. Without power to the blade fold interlocks no starting sequence protection exists. For engine starts during flight the manual or normal control circuits may be used to start either engine as all interlocks are deenergized to the closed position when the linear actuator is in the flight position. An emergency start circuit provides a means of bypassing all safety interlocks thus allowing either a manual or automatic start

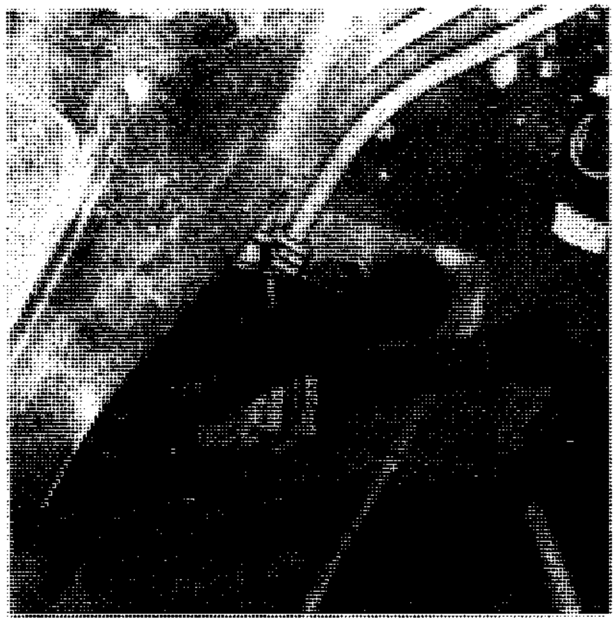


Figure 1-12. Manual Throttle Levers

to be performed. Emergency starts may be accomplished in either the normal or manual mode.

Mode Selector Switch.

The mode selector switch with marked positions MANUAL and NORMAL under the general heading START MODE is on the emergency start switch panel. When the switch is placed in the NORMAL position, the automatic drop-out function of the

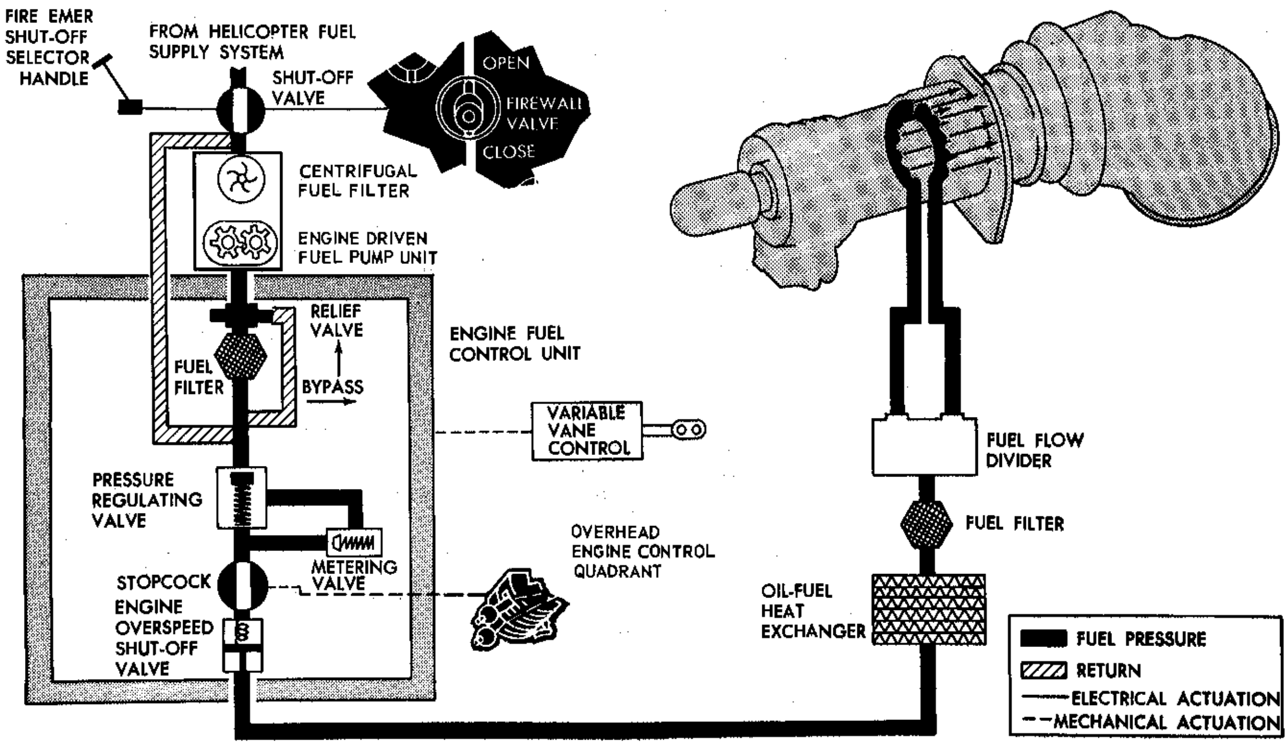


Figure 1-13. Engine Fuel System

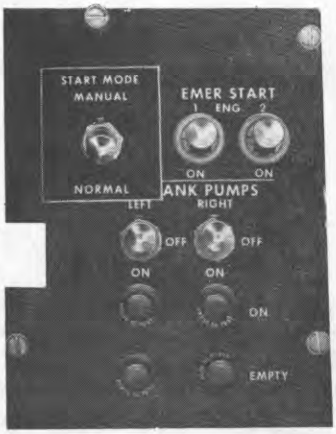


Figure 1-14. Engine Manual and Emergency Start Switches and Auxiliary Fuel Panel

starter relay is energized allowing the starter to remain engaged to approximately 45 percent Ng. When the switch is placed in the MANUAL position, the automatic drop-out feature of the starter relay is bypassed allowing the starter to remain engaged until disengaged by releasing the starter button. The switch operates on 28 volts dc from the essential bus.

Starter Buttons.

Two starter buttons, one for each engine, are on the knobs of the engine speed selectors. For normal starts with the mode selector switch in the NORMAL position, the starter is energized by advancing the engine speed selector approximately 5 degrees, depressing the starter button, returning the engine speed selector to the SHUTOFF position, and then releasing the starter button. When the engine speed selector is moved to the SHUTOFF position and the starter button released, the starter relay is energized from the essential bus, thereby completing the circuit from the essential bus to the starter. After the engine starts and the starter electrical power load lowers, the starter drop-out relay automatically disengages the electrical power to the starter. To abort an engine start prior to engine lite-off, momentarily depress the starter button to disengage the starter. Starter operation and drop-out may be monitored by noting the magnetic compass heading prior to engine start. When the starter is energized, the compass will swing to a new heading. At approximately 45 percent Ng, the compass should swing back to its original heading signifying the starter has dropped out. For manual starts, with the mode selector switch in the MANUAL position, initial starter engagement is accomplished in the same manner. However, the starter button must be held in to keep the starter engaged. The starter is disengaged by releasing the starter button. Starter engagement and disengagement may be monitored in the same manner as for normal starts.

Auxiliary Start Fuel Shutoff Valves.

Two auxiliary start fuel shutoff valves, one for each engine, are located in the engine compartment and installed between the engine fuel control and flow divider. When the valve is actuated during the start cycle of either engine, the flow of auxiliary bypass starting fuel is blocked. This blockage decreases the total amount of fuel flow during starting, thus diminishing the possibility of an overtemperature condition due to excessive fuel flow. The valves may be energized in either normal or manual mode, but will only function when the starter is engaged. The valves operate on 28 volts dc from the essential bus and are protected by the main starting system circuit breakers.

Auxiliary Start Fuel Shutoff Valve Switch.

The pushbutton type auxiliary start fuel shutoff valve switch, marked ENG ST, (figure 1-37) is located on each cyclic stick grip. In addition to depressing the switch, the starter relay for the engine to be started must be closed before the valve will operate. Either the pilot's or copilot's switch will control the operation of both valves. The switch operates on 28 volts dc from the essential bus.

Emergency Start Switch.

Two switches marked EMER START, 1 ENG 2, (figure 1-14) are on the overhead control panel to the right of the speed selectors. The switches have two marked positions, ON and OFF. Normally the switches remain in the OFF position and starting is accomplished through the normal control circuit. When the switches are placed in the ON position, the normal control circuit with its safety interlocks is bypassed and direct current from the essential bus is fed directly to the starter button. Either a normal or a manual mode starting procedure is then followed.

Engine Start with DC Power Source.

When starting No. 1 engine with an external dc power source, the fuel booster pumps will be inoperative due to their ac power source requirement. If all fuel lines to the engine are full, the engine driven fuel pumps will be capable of supplying sufficient fuel for starting.

Engine Start With Dual Batteries.

A second battery (figure 1-33) is installed in the cabin to provide power for engine starting when external power is not available. The dual (primary and alternate) batteries are connected in parallel to the battery and dc essential buses through their respective battery relays. When starting on batteries, both battery switches should be in the ON position.

PAGES 1-12A & 1-12B DELETED

IGNITION SYSTEM.

The ignition system consists of a sealed ignition unit and two igniter plugs on each engine that provide a spark to ignite the fuel-air mixture. The ignition unit is located on the lower right-hand side of the compressor casing. The ignition system provides ignition during starting with the ignition switch in the NORM position. When gas generator speed increases and the starter circuit load drops, the automatic dropout relay releases, de-energizing the starter and ignition systems, and combustion is self-sustained. For manual mode starts, on helicoptors modified by T.O. 1H-3(C)B-593, depressing and holding the starter button keeps the starter and ignition systems energized. After the gas generator increases to 45 percent Ng, releasing the starter button de-energizes the starter and ignition systems, and combustion is self-sustained.

Ignition Switches.

Two ignition switches, one for each engine, located on the overhead switch panel (figure 1-6), are marked IGNITION, 1 and ENG, 2. Each switch has three marked positions TEST, OFF, and NORM. The switches are normally in the NORM position. 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. The TEST position is used (for ground operation only) without the starter to test the ignition circuit. An audible clicking noise can be heard when the switch is placed in TEST. When the switch is in the OFF position, the ignition unit is de-energized. The OFF position is used to motor the engine, using the starter without ignition. The ignition switches are powered by dc current from the essential bus.

TORQUE SENSING SYSTEM.

The torque sensing system determines input torque at the main gear box and transmits this information to torquemeter indicators (figure 1-5) which are located in the pilot's compartment. Each torquemeter indicator presents this information in terms of percent of engine power being delivered to the main transmission. Components of the system include two pressure chambers, two balancing valves, two pressure transmitters, one high pressure oil pump, and two dual-needle torquemeter indicators. The system is designed to measure the oil pressures required to react against the forward displacement of the main gear box second stage helical gear as a result of the input shaft torque of each engine. These oil pressures, required to react against the forward movement of the second-stage helical gear, are sensed by two pressure transmitters which send electrical signals to the torquemeter indicators.

Torquemeters.

Two torquemeter indicators (figure 1-16), 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 electrical pressure torquemeter indicator dials, calibrated in percent torque, are graduated in increments of 5 percent from 0 to 130 percent. The torquemeter indicators operate on 26 volts alternating current and are protected by circuit breakers marked 1 ENG 2 TORQUE SENSOR, located on the ac circuit breaker panel.

ENGINE INSTRUMENTS.

Power Turbine Inlet Temperature (T_5) Indicators.

Two power turbine inlet temperature indicators (figure 1-5) marked EXH TEMP, indicating engine power turbine inlet air temperatures in degrees Centigrade, are located on the instrument panel. The indicators operate from thermocouples, located

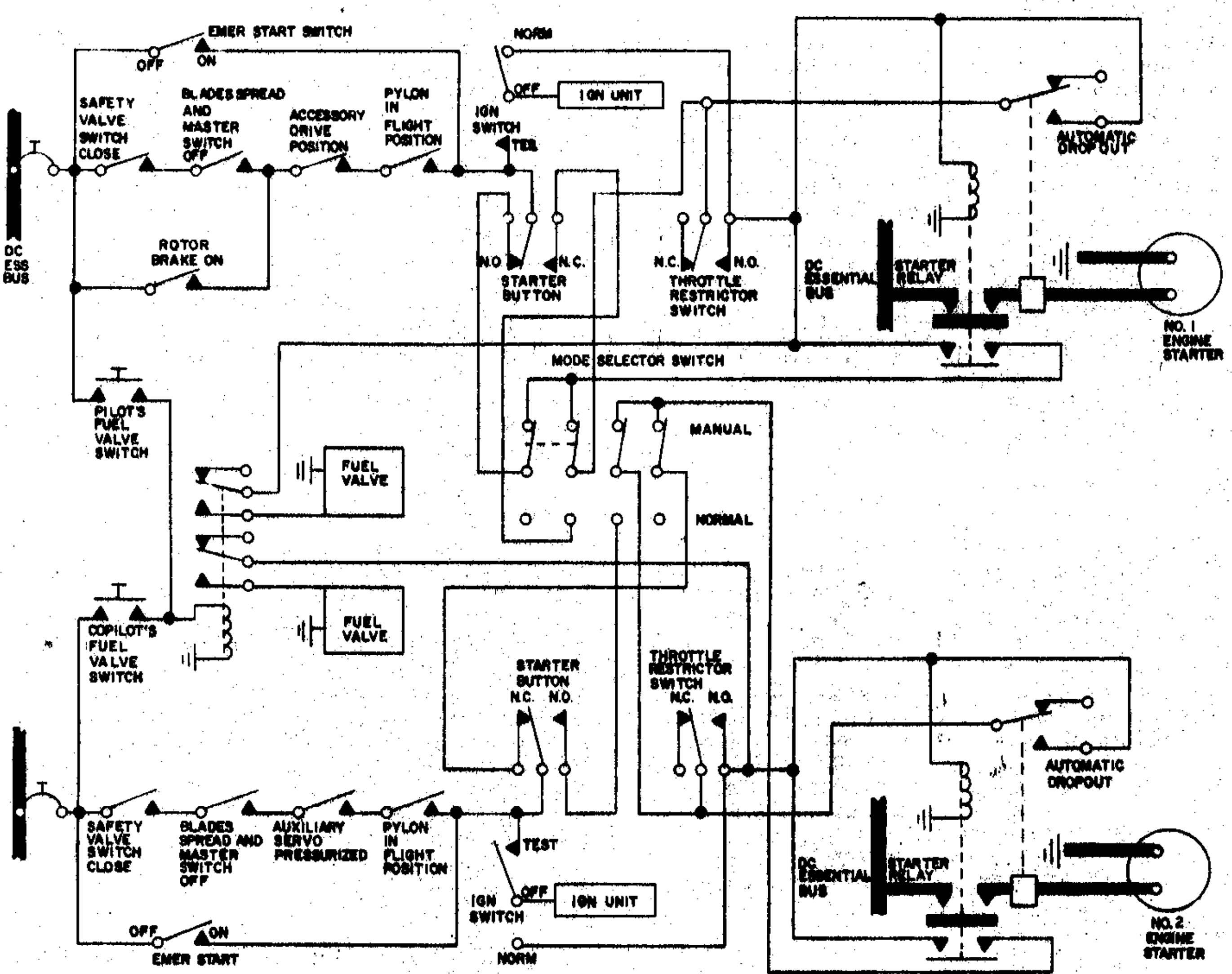


Figure 1-15. Starting System Diagram

FIGURE 1-15A DELETED

Figure 1-16. Torquemeter

forward of the power turbine in the second stage turbine casing, on each engine. When the dissimilar metals of the thermocouples in the engine are heated, an electromotive force (independent of the helicopter's electrical system) is created and a resulting current flow through a known resistance of the thermocouple circuit deflects the indicator pointer which is read in degrees Centigrade. The pilot has no direct control for regulating the power turbine inlet temperatures. Limited control for lowering these temperatures can be indirectly achieved by reducing collective pitch or power demand.

Engine Oil Pressure Indicators.

Two engine oil pressure indicators (figure 1-5), one for each engine, are located on the instrument panel and indicate oil pressure in pounds per square inch. The indicators are powered by 26 volts alternating current from either the No. 1 generator or the inverter and protected by circuit breakers marked PRESS under the general heading ENG, located on the console circuit breaker panel.

Engine Oil Temperature Indicators.

Two engine oil temperature indicators (figure 1-5), one for each engine, are located on the instrument panel and indicate engine oil temperature in degrees Centigrade. The engine oil temperature bulb, located in the bottom of the tank, transmits indications to the respective temperature indicators. The indicators are powered by 28 volts direct current from the essential bus and protected by circuit breakers marked ENGINE under the general heading OIL TEMP, located on the overhead circuit breaker panel.

Engine Gas Generator (N_g) Tachometers.

Two engine gas generator tachometers (figure 1-5), one for each engine, are located on the instrument panel. The gas generator tachometer indicates the speed of the gas generator in percent of total rpm. Each tachometer has two dials and pointers. The outer dial and pointer indicate from zero to 100 percent gas generator speed in increments of two percent. The small vernier dial and needle, located in the upper left-hand position of the tachometer, indicate gas generator speed from zero to ten in increments of one percent. The gas generator tachometer-generator is driven by the engine lube pump on which it is mounted. The electrical power produced by the gas generator tachometer-generator is proportional to gas generator rpm. One hundred percent gas generator speed is 26300 rpm.

FUEL FLOW INDICATOR.

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

N_f AND N_r TRIPLE TACHOMETERS.

Two triple tachometers (figure 1-5), one for the pilot and one for the copilot, are located on the instrument panel. Each tachometer contains three pointers: the pointer marked 1 indicates the power turbine speed of the No. 1 engine, and the pointer marked 2 indicates the power turbine speed of the No. 2 engine, while the pointer marked R indicates the main rotor rpm. The engine tachometers are powered by their own tachometer-generators and are driven by the engine fuel control unit on which they are mounted. The main rotor tachometer is powered by its own tachometer-generator geared to and driven by the main transmission output shaft for the Number 1 section of the tail rotor drive. (100% N_f = 18966 power turbine rpm, and 100% N_r = 203 rotor rpm.)

ROTOR SYSTEMS.

The rotor systems consist of a single main rotor and an antitorque 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,

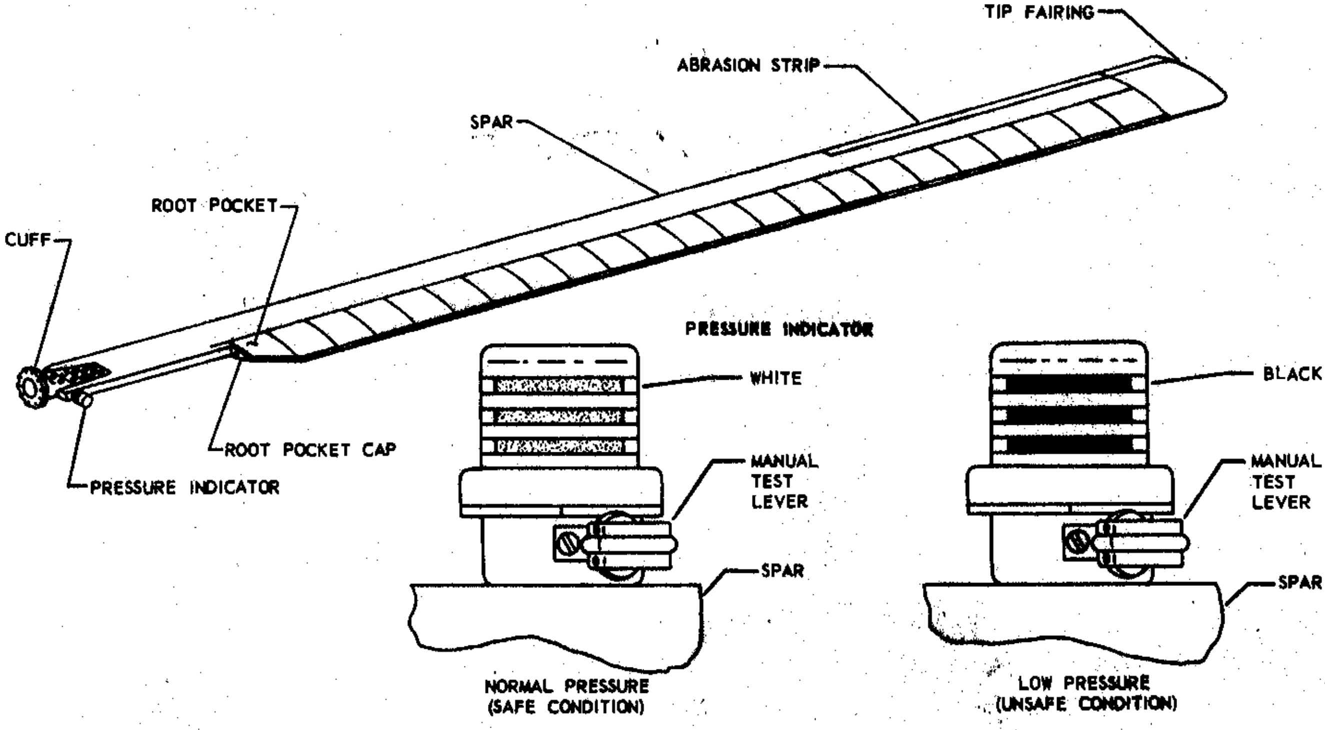


Figure 1-17. BIM Blade Indicators

hunt horizontally, and rotate about their span-wise axis, to change the angle of incidence. Antiflapping restrainers limit the upward movement of the blades caused by wind pressure; 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 antiflapping restrainers. The droop stops release at approximately 75 percent (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 five all metal main rotor blades are of the pressurized spar type, identified as BIM ® blades. The blades are constructed of aluminum alloy 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 consists, basically, of a hollow extruded aluminum spar pressurized with nitrogen, 23 aluminum pockets, an aluminum root cap, a steel cuff, a pressure (BIM) 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 done at manufacturer or overhaul. The swashplate assembly consists of an upper (rotating) swashplate, which is driven by the rotor hub, and a lower (stationary) swashplate, which is secured by a scissors assembly to the main gear box to prevent rotation. Both swashplates are mounted on a ball-ring 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, which connect 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.

BIM ® (Blade Inspection Method) Indicators.

A cylindrical BIM indicator (figure 1-17) is located in the root and plate of each main blade and an air valve is located in the back wall of the spar. The pressure indicator has a transparent cover through which a color indicator can be observed to determine blade serviceability. The indicator which is compensated for temperature changes, compares a reference pressure built into the indicator with the pressure in the blade spar. When the pressure in the blade spar is within the required service limits, indicating the blade is serviceable, three white strips show in the indicator. If an unforeseen combination of events should occur impairing the structural integrity of the spar, or if a seal should leak, nitrogen pressure will decrease. If the pressure in the blade spar drops below the minimum permissible service pressure, the indicator will be actuated and will show three black stripes. To check the integrity of the BIM indicator, depress the manual test lever until a black indication appears and then release the lever.

WARNING

When black is visible in the indicator, it may be an indication of blade damage that is a flight hazard. The cause of the black indication should be determined before flight.

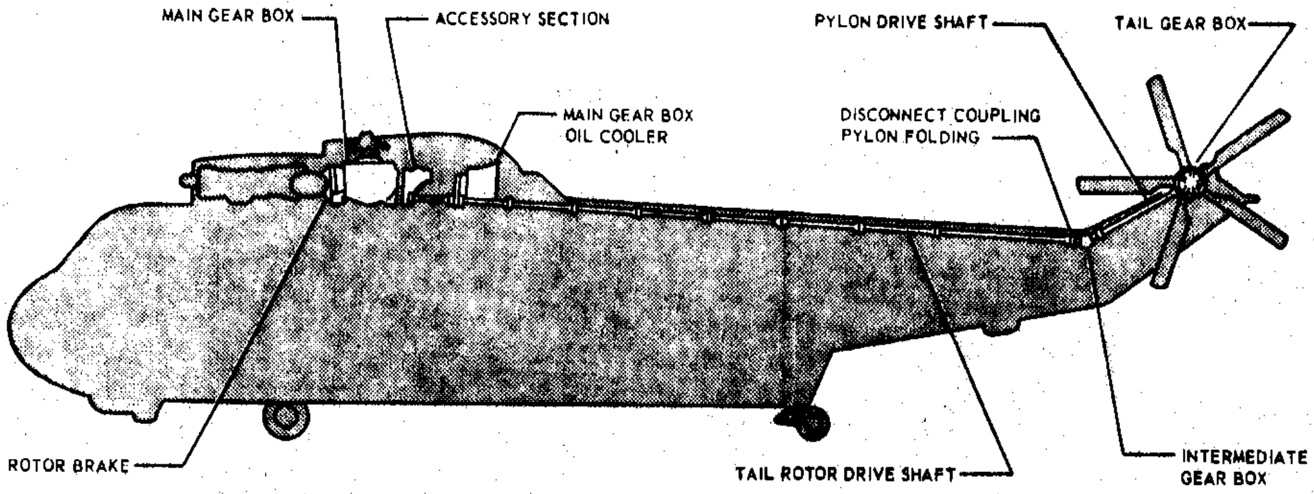


Figure 1-18. Transmission System

TAIL ROTOR.

The tail rotor 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. Since 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 by flapping hinges and spindles so that they are free to flap and rotate about their span-wise axis for pitch variations. The blade pitch changing mechanism transmits tail rotor control pedal movements to the tail rotor blades through the hollow horizontal output shaft of the tail gear box. The five all-metal tail rotor blades are constructed of a one piece wrap around skin bonded to a solid leading edge spar and to a honeycomb core. The 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. The system applies a force to cancel the aerodynamic loads only when the tail rotor is operating at normal speeds. 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 tendency of the pedals is then to go normally to either extreme. Under these conditions, considerable force is required to push the tail rotor pedals from the extreme positions; however, the forces will decrease as the positions approach neutral. The initial force to move the pedals toward the right from a full left position is approximately between 30 and 40 pounds. With the primary servo ON, auxiliary servo OFF, and the main rotor head stationary, pedal motion will cause the collective pitch lever to move up for a right pedal motion and down for a left pedal motion. However, the collective pitch lever can be held in a fixed position by applying sufficient effort to the lever.

TRANSMISSION SYSTEM.

The transmission system (figure 1-18) consists of three gear boxes that transmit power to the main and tail rotors. The three gear boxes are the main gear box, intermediate gear box, and the tail rotor gear box.

MAIN GEAR BOX.

The main gear box, mounted above the cabin aft of the engines, interconnects the two engines through a single planetary gear to the main rotor. The single planetary gear reduces engine rpm at a ratio of approximately 93.4 to 1 for driving the main rotor. Engine rpm is reduced and power is transmitted by the main gear box to the main rotor drive shaft to drive the main rotor, then aft to the intermediate gear box and tail rotor gear box to drive the tail rotor. The main gear box accessory section, located at the rear of the main gear box lower housing, drives the primary, utility, and auxiliary hydraulic pumps, the high pressure torque meter oil pump, and the two generators. Dual oil pumps are installed on the accessory section. These pumps increase reliability through better lubrication and provide a fail-safe function in that flight can be continued if one pump fails. A free wheeling unit, located at each engine input to the main gear box, permits the main rotor to autorotate without engine drag in event of engine (or engines) failure or when engine RPM decreases below that of the main rotor RPM.

Accessory Drive Rotor Lockout System.

The accessory drive rotor lockout system permits the pilot to use engine power to drive the accessory section (hydraulic pumps, oil pumps, generators, etc.) of the main gear box on the ground without rotating the main rotor head. No. 1 engine is used to drive the accessories without turning the main rotor head. A switch allows the pilot to divert power to either the accessory section or the main rotor.

Accessory Drive Switch. The accessory drive switch (figure 1-19), located forward of the overhead switch panel, is marked ACCESSORY DRIVE. The lever-lock type switch, with positions FLIGHT and ACCESS DR., must be pulled outward before it can be moved from one position to the other. The accessory drive switch allows the No. 1 engine to drive the accessory section of the main gear box before starting the No. 2 engine and engaging the main rotor head and/or when No. 2 engine rpm is lower than No. 1 engine rpm. Before starting the No. 1 engine, place the switch in ACCESS DR. which positions the rollers in the freewheel unit permitting the No. 1 engine to drive the accessory section of the main gear box. When the switch is in the ACCESS DR. position, the No. 1 engine overspeed protective system is placed in operation. Also at this time, the accessory drive (rotor lockout) warning light will come on and will remain on until the rollers are repositioned to drive the main rotor. When both engines are operating and the rotor is being driven by the No. 2 engine, No. 1 speed selector should be moved to GRD IDLE. When these conditions have been met, placing the accessory drive switch in the FLIGHT position energizes an actuator which repositions the freewheel unit rollers, permitting the No. 1 engine to also drive the rotary wing shaft. An accessory drive limit switch is incorporated in the ground idle range of the speed selector to prevent switching from accessory drive to flight without first retarding the No. 1 engine speed selector to the GRD IDLE range. After the rollers are repositioned, the accessory drive (rotor lockout) warning light will go out. The conditions which will allow the No. 1 engine to be connected to, or disconnected from, the main rotor shaft are: when the No. 2 engine is operating and driving the main rotor and No. 1 engine N_f is reduced to GRD IDLE, or when neither engine is operating. The accessory drive switch circuit operates on direct current from the essential bus and is protected by the circuit breaker marked ACCESS DRIVE, located on the overhead circuit breaker panel (figure 1-19).

Figure 1-19. Accessory Drive Switch Panel

Accessory Drive Warning Light. An accessory drive warning light (figure 1-19) is installed to the right of the accessory drive switch on the overhead switch panel under the general heading of ACCESSORY DRIVE. The light will come on when the rollers in the freewheel unit have been positioned to permit the No. 1 engine to drive the accessory section of the main gear box. When No. 1 engine power is diverted to drive the main rotor shaft, the warning light will go out. The press-to-test light operates on direct current from the essential bus and is protected by the circuit breakers marked PWR and TEST, located on the overhead circuit breaker panel (figure 1-30) under the general heading of WARN LTS.

Tail Takeoff Freewheel Unit Caution Light.

The tail takeoff freewheel unit caution light, marked TAIL TAKE-OFF, is located on the caution panel. The caution light indicates failure of the tail takeoff freewheel unit in the main gear box accessory drive train. When this failure occurs, the accessory section is being driven by the No. 1 engine through shaft at reduced RPM. No. 1 generator output is used to sense this reduction in RPM which results in a reduction in generator frequency. When the generator frequency is reduced to less than that produced through the freewheel unit drive, the caution light will illuminate. Failure of the No. 1 engine subsequent to the illumination of the caution light whould result in loss of the equipment driven by the accessory section. The caution light operates on direct current from the essential bus and is protected by a circuit breaker marked WARN LTS PWR, located on the overhead circuit breaker panel.

INTERMEDIATE GEAR BOX.

The intermediate gear box (figure 1-18), located at the base of the tail rotor pylon, contains a bevel gear direct-drive system to change the direction of the shafting that transmits engine torque to the tail gear box. The intermediate gear box is splash-lubri-

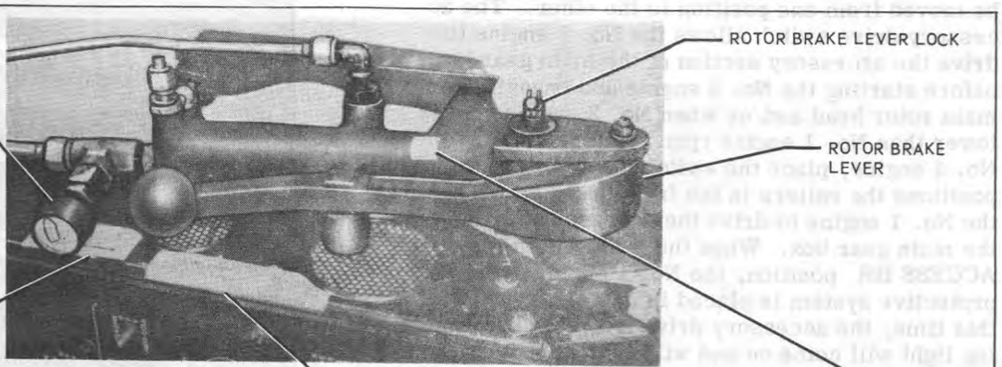


Figure 1-20. Rotor Brake Lever

cated. Screened intermediate gear box cooling air inlets (figure 1-2) in the pylon fairing permit the gear box to be cooled by the main rotor downwash.

TAIL GEAR BOX.

The tail gear box (figure 1-18), 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 horizontal output shaft to the tail rotor hub. The tail gear box is splash-lubricated.

CHIP DETECTOR WARNING LIGHTS

The chip detector warning lights, marked MAIN, INTMED, and TAIL under the general heading TRANS CHIP LOCATORS, are located on the left side of the instrument panel. In addition, a light marked CHIP DETECTOR is provided on the caution panel. The caution light illuminates simultaneously whenever one of the chip locator lights illuminates. The MAIN warning light provides a visual indication that the main gear box magnetic chip detector has picked up and retained metal particles or chips in the oil. The INTMED and TAIL warning lights will illuminate if a chip is detected and or an overheat condition exists in the respective gear box. The presence of these conditions would cause excessive wear and or premature failure of the gear boxes. The system operates on direct current from the essential bus and is protected by a circuit breaker marked CHIP DET, located on the overhead 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 its 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, operates 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 utility hydraulic system reservoir. In case of a broken or leaking hydraulic line from the utility hydraulic system reservoir, the rotor brake hydraulic cylinder contains sufficient fluid for braking the rotor system. The hydraulic brake cylinder is located on supports attached to the main gear box. The brake disc, positioned on the input shaft of the main gear box, has a toothed edge that enables the blade positioner drive unit to engage it to turn the main rotor in the blade positioning cycle.

Rotor Brake Cylinder and Lever.

A rotor brake lever (figure 1-20) is connected directly to the rotor brake hydraulic cylinder 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 forward as indicated on the decal

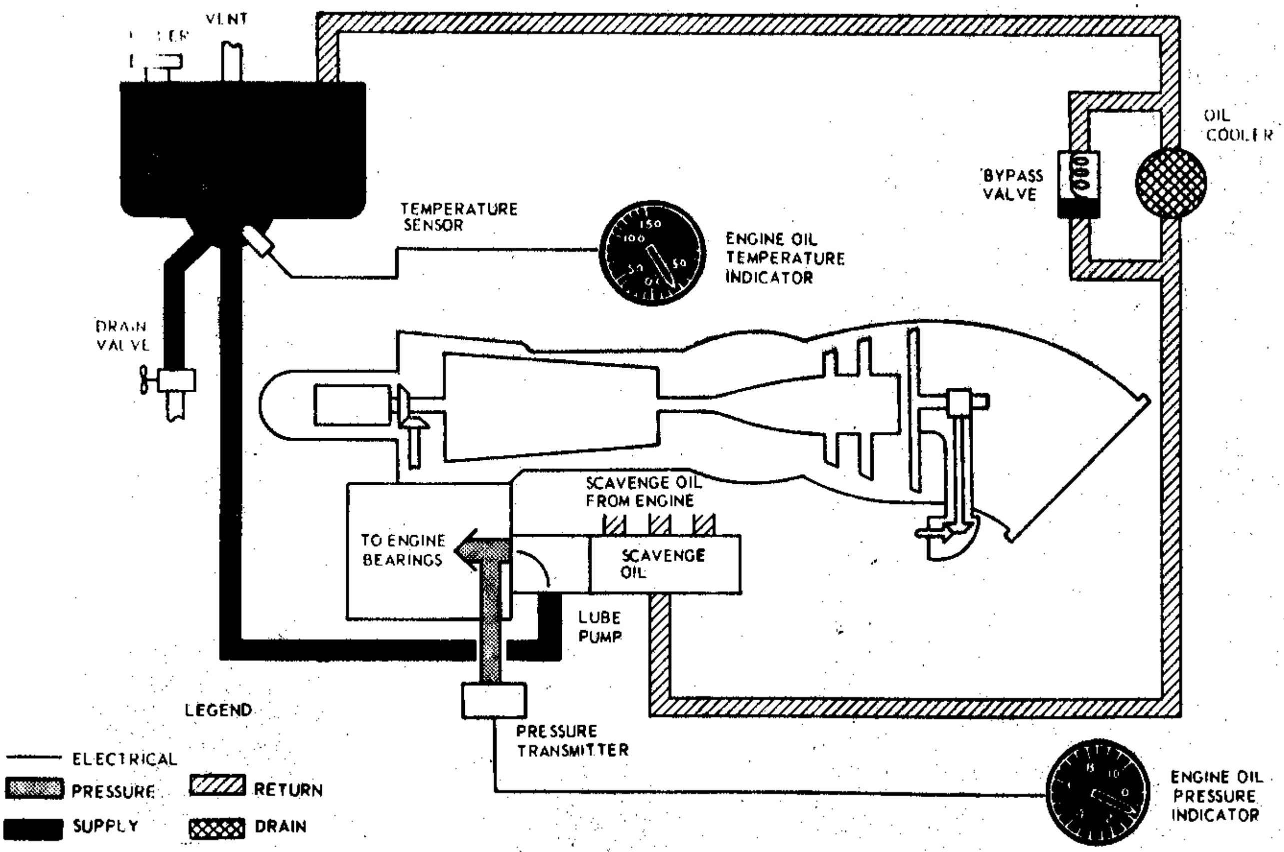


Figure 1-21. Engine Oil System

aft of the lever on the upper structure. The decal is marked TO ENGAGE ROTOR BRAKE PUSH LEVER FORWARD. The decal also has an arrow pointing forward. When actuated, a lock-lever, located at the forward outboard side of the cylinder, locks the brake lever in the applied (forward) position. To release the rotor brake, reposition lock-lever aft, and swing the rotor brake lever aft and up against the bottom of the cylinder until it snaps into place. The lockpin may be rendered inoperative by rotating it until it remains in the OUT position. For normal shutdown the rotor brake should be applied firmly and smoothly. As rotor rpm approaches zero, rotor brake pressure should be reduced in order to ease rotor blades to a stop, precluding any tendency of "whip stopping." When the rotor blades are folded, the rotor brake lever must be on to open interlock for a normal start. For emergency shutdown, the rotor brake lever may be forced forward into the full ON position after closing the engine speed selectors. In event of emergency, the rotor brake, when fully applied, is designed to stop the rotors from 77.3% N_r in 14 seconds with engines at idle and from 91.1% N_r in 20 seconds with engines at idle. A rotor brake caution light, located on the caution panel, will go on when pressure is applied to the rotor brake.

Rotor Brake Pressure Gage. A hydraulic ac'uated rotor brake pressure gage is located to the rear of the rotor brake lever (figure 1-20) on the pilot's compartment ceiling. The reading, indicated by the needle, indicates psi X 100. A decal, marked ROTOR BRAKE PRESS. ACTUATING RANGE 350-500 PSI. ENGINE START 320 PSI MIN. PARKED POS. RANGE 250-600 PSI., is located adjacent to the rotor brake pressure gage. A pressure of 320 psi or more is needed before an engine start can be initiated with the blades folded.

Rotor Brake Caution Light. The rotor brake ON caution light is located on the caution panel. Whenever this light illuminates, it indicates that either the manual or automatic rotor brake is on. However, the automatic rotor brake is not applied until the blade fold master switch is actuated. In either installation the light will go out when the rotor brake is disengaged.

OIL SUPPLY SYSTEMS.

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-driven oil pump, mounted on the forward right-hand side of the engine. The engine-driven pump distributes the oil under pressure through a filter to accessory gears and engine bearings. The oil serves both lubricating and cooling purposes and is a completely automatic system requiring no control action by the pilot. 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 valve. The oil flow through the cooler depends on oil temperature. At lower temperature, the pressure differential across the cooler causes most of the oil to flow through the bypass valve. At higher temperatures, the lower viscosity reduces the pressure differential, which closes the bypass valve and causes all of the oil to flow through the cooler. Each engine oil system has a capacity of between 2.7 to 3.0 US gallons of oil in a 4.5 US gallon (2 gallon expansion space). The circular tanks are located around the forward section of each engine.

TRANSMISSION OIL SYSTEMS.

Main Gear Box Oil System.

(See figure 1-22)

Primary and secondary oil pumps are provided for lubrication. The primary pump is mounted on the lower portion of the rear cover of the main gear box. However, the secondary oil pump is mounted between the utility hydraulic pump and the rear cover mounting pad. This installation enables utilization of a common drive shaft for the two pumps. Oil is pumped from the gear box sump through a hose to an oil cooler located behind the main gear box. Cooling air enters the forward end of the main gear box fairing through a screened main gear box cooling air intake (figure 1-2) and is forced through the oil cooler by a blower driven by belts from the tail drive shaft. The air is then exhausted through a screened transmission accessories cooling air outlet (figure 1-2) at the rear of the fairing. 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, accessible from the left side of the main rotor fairing, is located on the left side of the gear box. A window in the gear box below the oil filler provides a sight check for the oil level in the main gear box. Oil capacity is 15 gallons, normal servicing is 11 gallons.

Main Gear Box Oil Pressure Indicator and Caution Light. The main gear box oil pressure indicator (figure 1-5) 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 26V alternating current from the No. 1 generator or from the inverter through the autotransformer, and is protected by a circuit breaker marked XMSN OIL PRESS., located on the AC circuit breaker panel. The main gear box oil low pressure caution light marked TRANS OIL PRESS., is located on the caution panel (figure 1-43) and is actuated by a pressure switch, located in the forward part of the main gear box. The amber caution light operates on direct current from the essential bus and is protected by a circuit breaker marked WARN LTS PWR, located on the overhead circuit breaker panel. The light will illuminate when the main gear box oil system pressure drops below 12 psi. The different locations of the pressure sensors for the gage and caution light were incorporated to warn the pilot of an oil blockage within the gear box which may not be indicated on the pressure gage.

Main Gear Box Oil Temperature Indicator and Caution Light. The main gear box oil temperature indicator (figure 1-5) marked XMSN OIL TEMP, located on the instrument panel, is graduated in

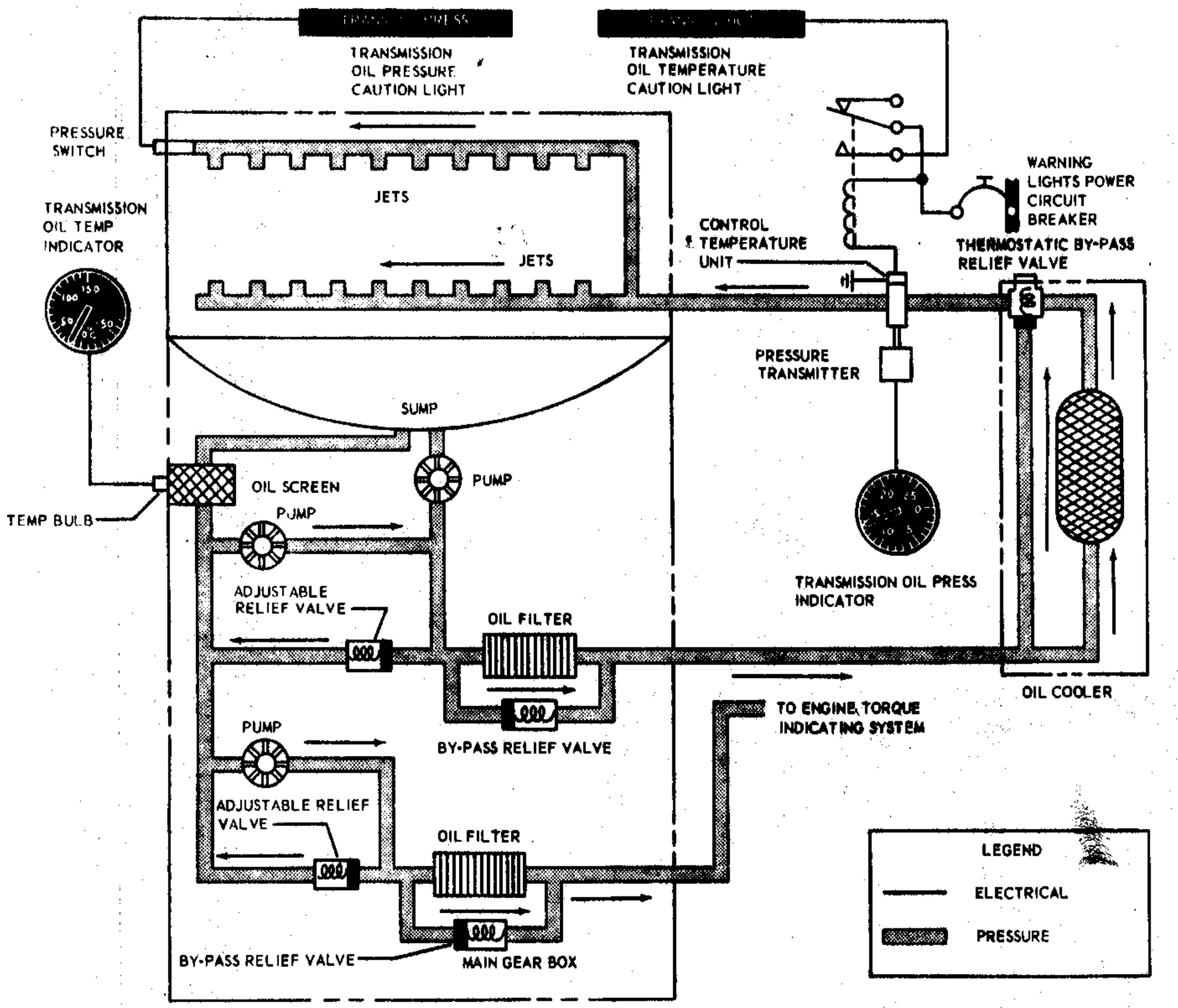


Figure 1-22. Main Gear Box Oil System

degrees centigrade. The indicator is connected by direct current from the essential bus to an oil temperature bulk adjacent to the main gear box oil outlet port and is protected by a circuit breaker marked XMSN OIL TEMP, located on the overhead circuit breaker panel. The main gear box oil temperature caution light is located on the caution panel and actuated by a temperature sensor located at the outlet of the oil cooler. The amber caution light operates on direct current from the essential bus and is protected by a circuit breaker marked WARN LTS PWR, located on the overhead circuit breaker panel (figure 1-30). The transmission oil temperature caution light will come on when the transmission oil temperature reaches 120°C (248°F). The different locations of the temperature sensors for the gage and light allow the pilot to monitor the gear box operation by means of the gage 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 dangerous 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 insure oil lubrication to all bearings. An oil filler plug, drain plug, and oil level window are located in each gear box casting. Oil capacity for the intermediate gear box is about 0.2 gallon and for the tail gear box 0.4 gallon.

FUEL SUPPLY SYSTEM.

The helicopter is equipped with two independent pressure-type fuel systems and/or may have an auxiliary fuel system incorporated for longer range and

FIGURE 1-23 DELETED.

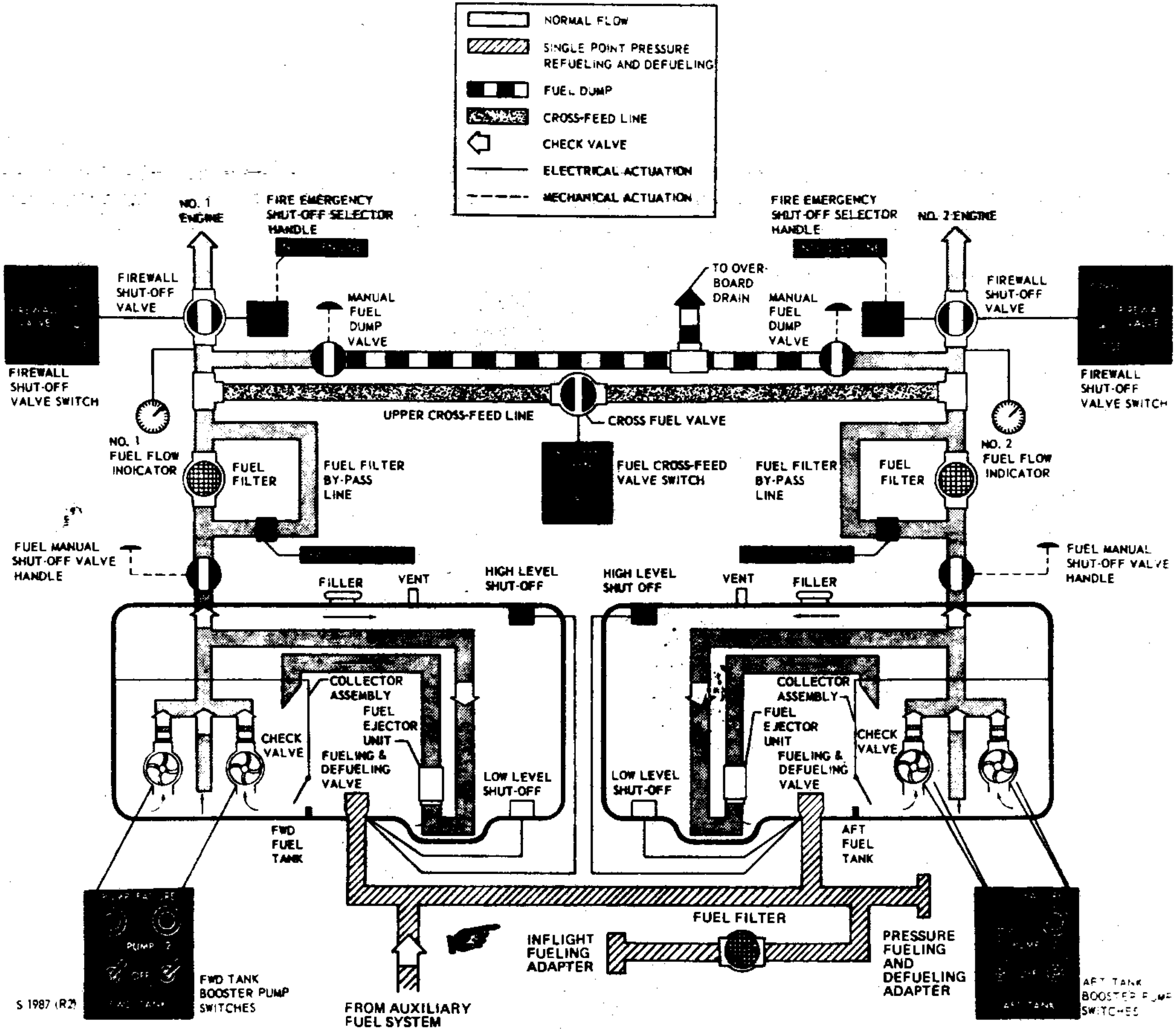


Figure 1-24. Helicopter Fuel System

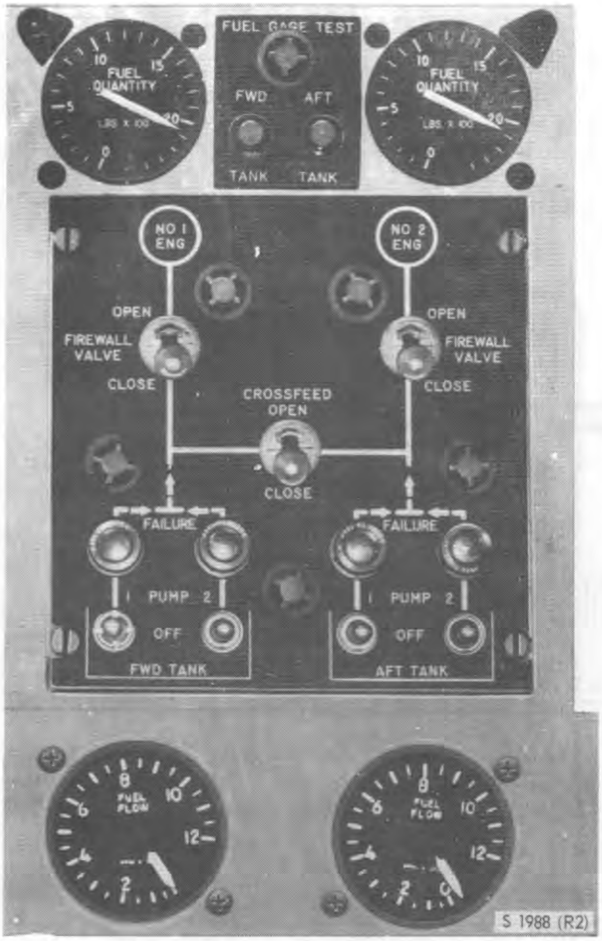


Figure 1-25. Fuel Management Panel and Indicators

endurance missions. The systems are joined together by a crossfeed system. Each fuel system consists of a fuel tank with two bladder-type cells and collector cans in which two fuel booster pumps are located. A fuel ejector system within each tank pumps fuel from the cell into the collector can. Fuel is pumped to the main line from the collector cans by booster pumps. The ejector system and booster pump arrangement provides for a minimum of unusable fuel. The crossfeed system is electrically controlled by a crossfeed switch. The crossfeed system allows fuel from both systems to be directed to one engine during single engine operation, or fuel from one system to supply both engines if the other system fails. The fuel management panel, (figure 1-25) located on the instrument panel, contains the four fuel booster pump switches, the fuel booster pump failure warning lights, the crossfeed switch, and the two engine fuel firewall shutoff valve switches. The tank may be filled by either a pressure refueling system or the conventional filler neck.

FUEL TANKS.

This helicopter may operate with two main fuel tanks and two externally installed auxiliary fuel tanks.

Main Fuel Tanks.

There are two main fuel tanks, a forward fuel tank and an aft fuel tank. Each tank consist of two cells. The two cells interconnect at flanged holes in adjacent cell walls and are equipped with manually operated drain valves which are accessible from the outside bottom surface of the hull. Each cell is connected by fittings and tubes to vent fittings on the upper and lower left and right sides of the cabin wall. The forward fuel cell contains a collector, high level shutoff valve, low level shutoff valve, fueling and defueling valve, and a fuel quantity tank unit probe. The aft fuel cell contains a fuel ejector unit and fuel quantity tank unit probe. One access cover for each fuel cell is in the cabin floor over the cells. A filler cap and adapter is secured to the scupper, at the forward left side of the forward cell.

Auxiliary Fuel Tanks.

An external auxiliary fuel tank may be mounted on the left hand and right hand forward cargo sling attaching points to increase the range and endurance of the helicopter. Refer to AUXILIARY FUEL SYSTEM in this section.

FIREWALL VALVE SWITCHES (FUEL FIREWALL SHUTOFF VALVE SWITCHES).

Two fuel firewall shutoff valve switches (firewall valve switches) marked **FIRE WALL VALVE**, with two marked positions **OPEN** and **CLOSE** are located on the fuel management panel (figure 1-25). The switches control the fuel shutoff valves, located on top of the cabin before the engine compartment. Placing the switches in the **CLOSE** position, shuts off the flow of fuel to the engines. In case of electrical failure, the valves will remain in the last energized position. The switches and valves are operated by dc power from the essential bus and are protected by circuit breakers on the overhead circuit breaker panel.

FUEL CROSSFEED SWITCH (FUEL CROSSFEED VALVE SWITCH).

A fuel crossfeed valve switch, (fuel crossfeed switch) marked **CROSS-FEED** is located on the fuel management panel. The switch has two marked positions, **CLOSE** and **OPEN**, and actuates a valve which operates on direct current from the essential bus and is protected by a circuit breaker on the overhead circuit breaker panel marked **X FEED**. The crossfeed valve is located in the crossfeed line. Normally, the switch is in the **CLOSE** position. With the switch in the **CLOSE** position, the No. 1 engine receives fuel from the forward fuel tank and the No. 2 engine receives fuel from the aft fuel tank. Placing the switch in the **OPEN** position electrically opens the crossfeed valve which connects the fuel systems. The crossfeed system may be used to supply fuel under pressure from both tanks to any one or both engines. The crossfeed system does not transfer fuel between tanks.