# 407
# Pilot Ground and Flight Procedures

The Bell Training Academy
Est. 1947

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Bell Helicopter
A Textron Company

# General Description

THE BELL TRAINING ACADEMY

Est. 1947

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## General Description

# GENERAL DESCRIPTION

### Helicopter Description

The Bell Model 407 is a single engine, seven place helicopter designed to takeoff and land on any reasonably level terrain. Standard configuration provides for one pilot and six passengers. The helicopter may be flown Day or Night, VFR conditions only and in NON-Icing conditions.

### Airframe

The fuselage consists of three main sections: the FORWARD SECTION, which extends from the cabin nose to the bulkhead aft of the passenger compartment; the INTERMEDIATE SECTION, which extends from the bulkhead aft of the passenger compartment to the tailboom; and the TAILBOOM SECTION.

The airframe utilizes an aluminum honeycomb carbon graphite structure for the forward section, a semi-monocoque structure with 4 longerons, and carbon fiber composite side skins for the intermediate section, and an aluminum full monocoque tail boom.

The basic helicopter landing gear is the low-skid type. Alternative landing gear configurations, including high-skid or emergency flotation gear, are available as kits.

Cowlings and fairings enclose the various roof and tail boom mounted assemblies. Access doors and inspection windows allow for preflight and inspections. Cowlings manufactured from composite or aluminum materials are removed easily for maintenance access.

# GENERAL DESCRIPTION

### Crew Compartment

The crew compartment provides for one or two pilot operation, with the pilot station located on the right side. An instrument panel is mounted on a central pedestal forward of the pilot and copilot seats. The overhead electrical panel places the switches and circuit breakers within easy reach of the pilot.

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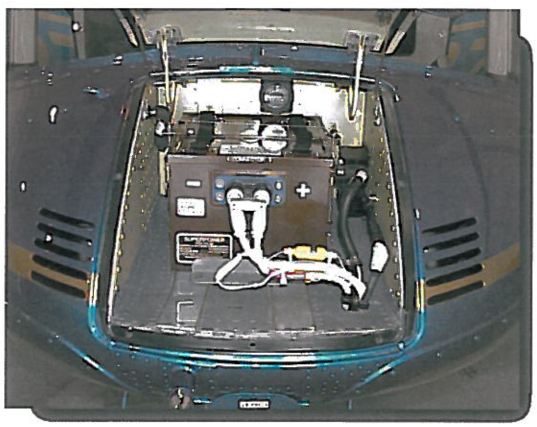

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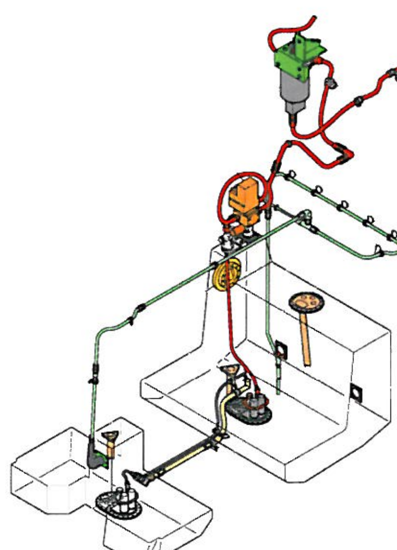
### Electrical System

The helicopter is equipped with a 28 VDC electrical system. A nickel-cadmium battery, a starter/generator and/or an external power source provide aircraft electrical power.

### Fuel

The basic aircraft has a fuel capacity of 130 U.S. Gallons (128.7 U.S.Gallons useable) distributed within two cells. A 19 U.S. Gallon auxiliary tank is also available as optional equipment.



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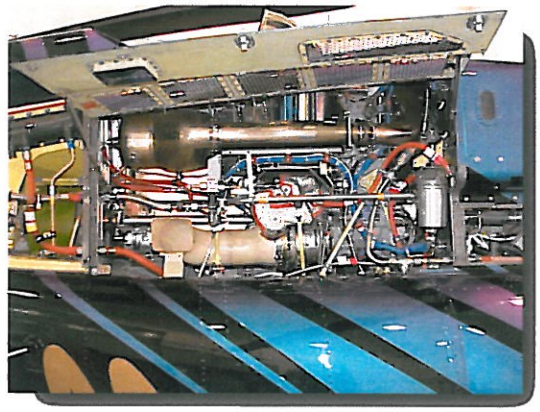
# GENERAL DESCRIPTION

## Powerplant

The Rolls Royce turbo-shaft engine, Model 250-C47B, provides 674 Shaft Horsepower (SHP) for takeoff and 630 SHP for maximum continuous operation. An electronic Full Authority Digital Electronic Control (FADEC) provides additional reliability during engine starts, improved fuel control during flight, and continuous in-flight systems monitoring.



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## Drivetrain and Rotors

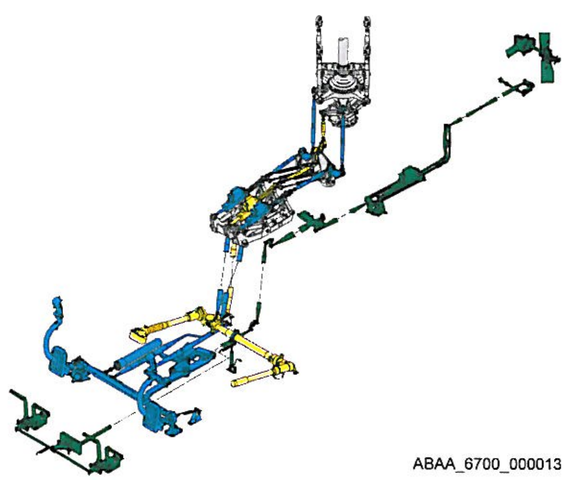
The drivetrain system transmits power from the engine to the main and tail rotor assemblies. The main rotor is a four bladed soft-in-plane, flex beam type hub. The tail rotor is a two bladed teetering rotor that provides directional control. It incorporates a semi-rigid, delta hinged design

## Flight Controls and Hydraulics

The flight controls are mechanical linkages actuated by conventional controls and used to control flight attitude and direction. The cyclic, collective and tail rotor controls incorporate hydraulic servo actuators.

The main rotor controls consist of the swashplate, drive link and pitch links.

The hydraulic system provides pressurized fluid (1000 PSI) to operate the cyclic, collective and tail rotor flight control servo actuators. The hydraulic system switch electrically controls operation of the system.



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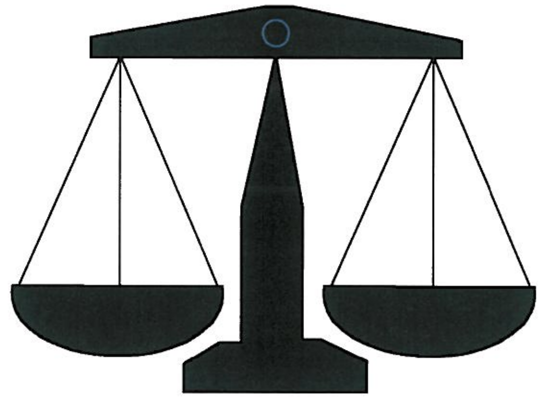
# GENERAL DESCRIPTION

## Weight and Balance

Section 5 of the approved Bell 407 Rotorcraft Flight Manual (RFM) presents the data necessary to compute gross weight and center of gravity for various load configurations. It is the pilot's responsibility to ensure the maximum gross weight and center of gravity limitations are observed throughout each planned flight. Operations outside of the limitations are prohibited and may result in a reduction of aircraft performance, handling qualities, stability, and structural integrity.

Changes in aircraft configuration (example: doors on or off), loading, seating of passengers, positioning of cargo, and fuel consumption are all factors to consider in weight and balance calculations.



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## Performance

The performance section (Section 4) of the Rotorcraft Flight Manual contains the Bell 407 performance information and related charts. All performance charts are based on an engine meeting minimum Rolls-Royce specifications. The data are derived from actual flight tests and are intended to provide information for use in conducting flight operations. These performance data are applicable to the RR 250-C47B engine.

Supplement FMS-28 provides data that allows increased gross weight operation. When operating at takeoff gross weights over 5000 pounds, use performance data from FMS-28.



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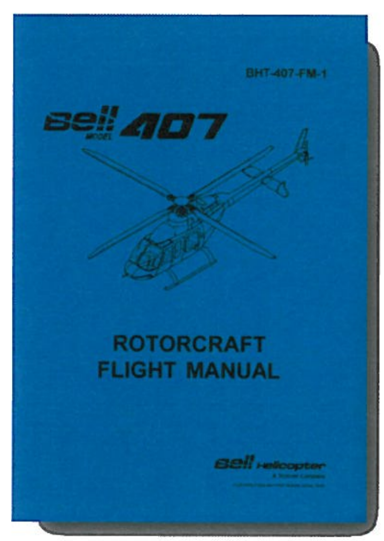
## Limitations

Section 1 of the Rotorcraft Flight Manual (BHT-407-FM-1) contains the Bell 407 limitations, center of gravity, instruments markings, and placards.

Compliance with the limitations section is required by the appropriate operating rules.





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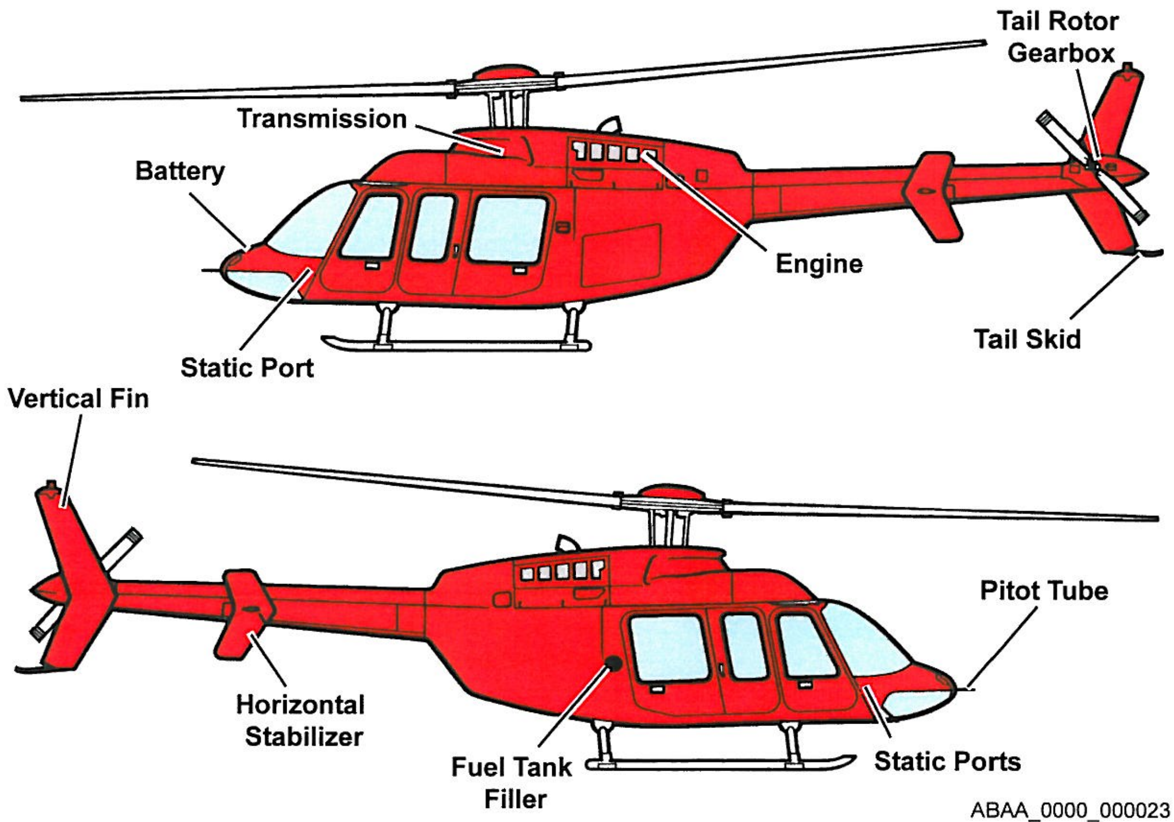
# GENERAL DESCRIPTION

### Helicopter Component

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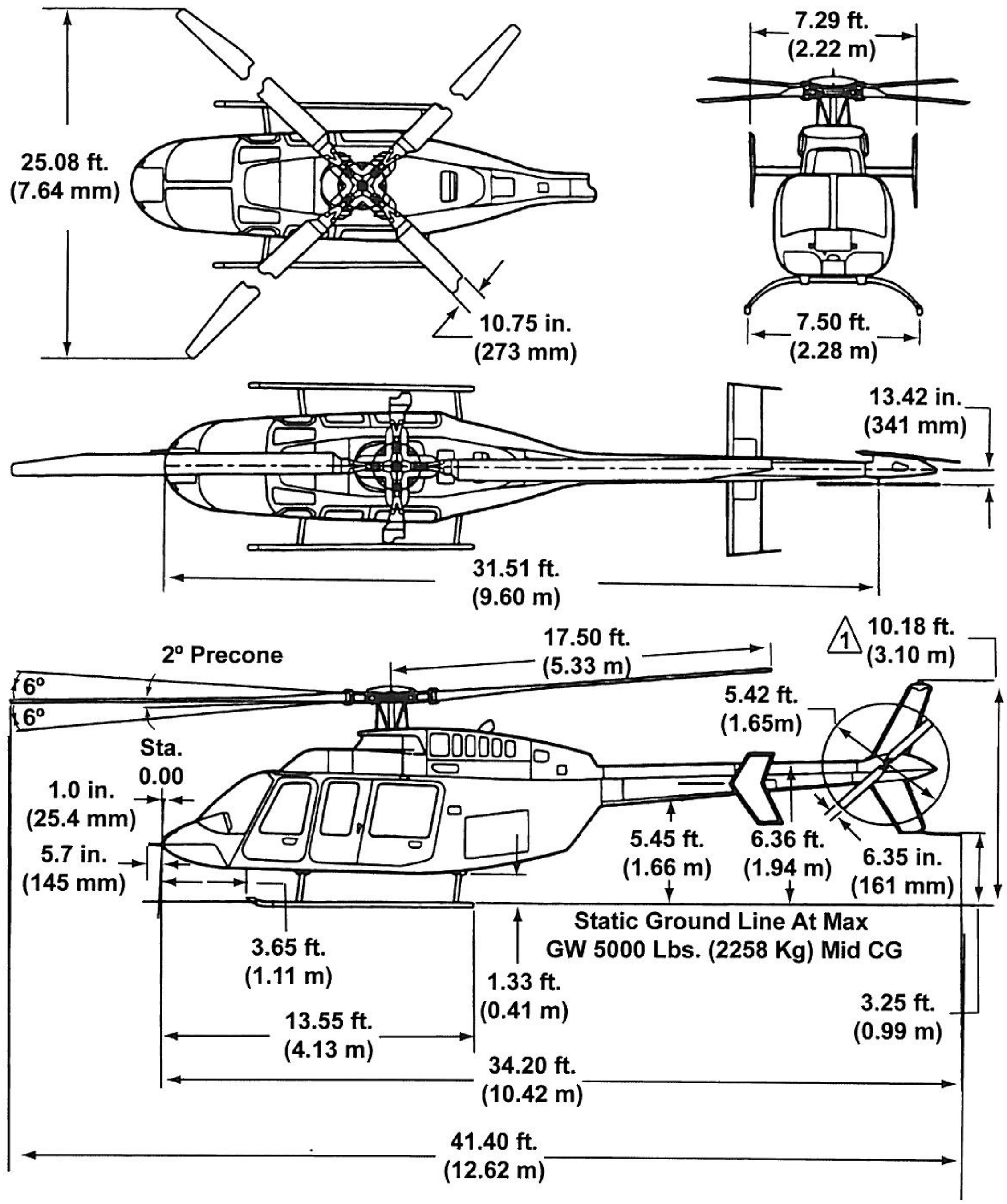
# GENERAL DESCRIPTION

**Principal Dimensions**




**Note**
**When high gear is installed, dimension is 10.91 feet (3.33 m)**

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# GENERAL DESCRIPTION

## General Data Table

### Torque / Horsepower

| | |
|---|---|
| Maximum continuous power | 93.5% Torque/630 Shaft Horsepower |
| Takeoff power (5 minute limit) | 100% Torque/674 Shaft Horsepower |

### Airspeed Limitations (Indicated)

| | |
|---|---|
| Maximum airspeed $V_{NE}$ | 140 KIAS |
| Maximum airspeed for Autorotation | 100 KIAS |
| Maximum airspeed at 93.5 to 100%Torque | 100 KIAS |
| Maximum airspeed with doors off (any combination removed) | 100 KIAS |
| When takeoff loading is in shaded area of the gross weight lateral C.G. limits | 100 KIAS |

### Recommended Airspeeds (Indicated)

| | |
|---|---|
| Minimum rate of descent (autorotation) | 55 KIAS |
| Maximum glide distance (autorotation) | 80 KIAS |
| Best rate of climb | 60 KIAS |
| Hydraulic failure | 70 - 100 KIAS |

### Altitude

| | |
|---|---|
| Maximum operating altitude | 20,000 $H_D$ Feet or 20,000 $H_P$ Feet (whichever is lower) |
| (FMS-28) 5,001 lbs (2273kg) to 5,250 lbs (2381kg) Internal GW | |
| Maximum altitude for takeoff, landing, and in ground effect (IGE) maneuvers | 11,000 Feet $H_D$ |

### Weights

| | |
|---|---|
| Minimum gross weight for flight | (1202kg) 2,650 Pounds |
| Maximum gross weight (internal) (basic aircraft) | (2268kg) 5,000 Pounds |
| Maximum gross weight (internal) (with FMS-28) | (2381kg) 5,250 Pounds |
| Maximum gross weight (external) | (2722kg) 6,000 Pounds |
| Cargo hook rated capacity | (1202kg) 2,650 Pounds |

# GENERAL DESCRIPTION

## General Data Table (Cont)

### Fuels

| | |
|---|---|
| Maximum capacity | 130 U.S. Gallons (492 Liters) |
| With auxiliary fuel tank installed | 149 U.S. Gallons (564 Liters) |
| ASTM type Jet B (JP-4) | All temperatures |
| ASTM type Jet A or A-1 (JP-5/8) | Above –32ºC (-25ºF) |

### Approved Lubricants

Engine oils must conform to MIL-PRF-7808, MIL-PRF-23699, or DOD-PRF-85734. Transmission and Tail Rotor Gearbox oils must conform to MIL-PRF-7808, or DOD-PRF-85734. Hydraulic fluid must conform to MIL-PRF-5606.

### Capacities:

| | |
|---|---|
| Engine Oil | 6 U.S. Quarts (5.7 Liters) |
| Transmission | 5 U.S. Quarts (4.7 Liters) |
| Tail Rotor Gear Box | .33 U.S. Quarts (.31 Liters) |
| Hydraulic (System & Reservoir) | 1.68 U.S Quarts (1.59 Liters) |

### Main Rotor

| | |
|---|---|
| Number of Blades | 4 |
| Diameter | 35 Feet |
| Chord | 10.75 Inches |
| Twist | -13 Degrees |
| Engine to Main Rotor Gear Ratio | 15.23 to 1 |
| RPM at 100% | 413 |

### Tail Rotor

| | |
|---|---|
| Number of Blades | 2 |
| Diameter | 65 Inches |
| Chord | 6.40 Inches |
| Engine to Tail Rotor Gear Ratio | 2.53 to 1 |
| RPM at 100% | 2500 |

# GENERAL DESCRIPTION

## General Data Table (Cont)

| | |
|---|---|
| **Airframe** | |
| Overall Length | 41 Feet 8.5 Inches |
| Overall Height | 10 Feet 3.8 Inches |
| **Passenger/Cargo Door Opening** | |
| Height | 39 Inches |
| Width | 36 Inches |
| Litter Door | 25 Inches |
| **Cargo Area Volume** | |
| Main Cargo Area | 85 Cubic Feet |
| Baggage compartment | 16 Cubic Feet |

# GENERAL DESCRIPTION

## Terminology

WARNINGS, CAUTIONS, AND NOTES: Warnings, cautions, and notes are used throughout this manual to emphasize important and critical instructions as follows:

**WARNING**

An operating procedure, practice, etc., which could result in personal injury or loss of life if not correctly followed.

**CAUTION**

An operating procedure, practice, etc., which if not strictly observed, could result in damage to or destruction of equipment.

**NOTE**

An operating procedure, condition, etc., which is essential to highlight.

## Use Of Procedural Words

Concept of procedural word usage and intended meaning that has been adhered to in preparing this manual is as follows:

SHALL.................... used only when application of a procedure is mandatory.

SHOULD ............. used only when application of a procedure is recommended.

MAY and NEED NOT.......... used only when application of a procedure is optional.

WILL......... used only to indicate futurity, never to indicate a mandatory procedure.

# GENERAL DESCRIPTION

## Abbreviations and Acronyms

Abbreviations and acronyms used throughout this manual are defined as follows:

| | |
|---|---|
| ADF | Automatic Direction Finder |
| Air Cond | Air Conditioning |
| A/C | Aircraft |
| AGL | Above Ground Level |
| ALT | Altitude |
| A/F | Airframe |
| ANTI COLL LT | Anti-collision light |
| APU | Auxiliary Power Unit |
| ATT | Attitude |
| AUX | Auxiliary |
| BATT | Battery |
| BIT | Built in Test |
| BL | Buttock Line |
| BLO | Blower |
| C | Celsius |
| CAUT | Caution |
| CEFA | Combined Engine Filter Assembly |
| CG | Center of Gravity |
| CKPT COMM | Cockpit Communications |
| CONT | Continuous |
| CWAP | Caution Warning Advisory Panel |
| DC | Direct Current |
| DG | Directional Gyro |

# GENERAL DESCRIPTION

## Abbreviations and Acronyms (Cont)

| | |
|---|---|
| ECS | Environmental Control System |
| ECU | Electronic Control Unit |
| ELT | Emergency Locator Transmitter |
| ENCDG | Encoding Altimeter |
| ENG | Engine |
| F | Fahrenheit |
| FADEC | Full Authority Digital Electronic Control |
| FS | Fuselage Station |
| FT | Foot, Feet |
| FWD | Forward |
| GEN | Generator |
| GPS | Global Positioning System |
| GOV | Governor |
| GW | Gross Weight |
| $H_D$ | Density Altitude |
| $H_G$ | Inches of Mercury |
| HMU | Hydro-Mechanical Unit |
| $H_P$ | Pressure Altitude |
| HYD | Hydraulic |
| ICS | Intercommunication System |
| IGE | In Ground Effect |
| IGNTR | Igniter |
| IMC | Instrument Meteorological Conditions |
| IN | Inch(es) |
| IFL | Inflate |

# GENERAL DESCRIPTION

**Abbreviations and Acronyms (Cont)**

| | |
|---|---|
| INSTR CHK | Instrument Check |
| INSTR LT | Instrument Light |
| KCAS | Knots Calibrated Airspeed |
| KG | Kilogram(s) |
| KIAS | Knots Indicated Airspeed |
| KTAS | Knots True Airspeed |
| L | Liter(s) |
| LB(S) | Pound(s) |
| LCD | Liquid Crystal Display |
| LDG LTS | Landing Lights |
| LT | Light |
| L/FUEL | Left Boost/Transfer Pumps |
| MCP | Maximum Continuous Power |
| MGT | Measured Gas Temperature (TOT/EGT) |
| MM | Millimeter(s) |
| MPH | Miles Per Hour (statute) |
| MSL | Means Sea Level |
| NAV | Navigation |
| NDOT | Rate of NG Speed Change |
| $N_G$ | Gas Producer RPM ($N_1$) |
| $N_P$ | Power Turbine RPM ($N_2$) |
| $N_R$ | Main Rotor RPM |
| NVM | Non Volatile Memory |
| OAT | Outside Air Temperature |
| OBS | Omni Bearing Selector |

# GENERAL DESCRIPTION

**Abbreviations and Acronyms (Cont)**

| | |
|---|---|
| OGE | Out of Ground Effect |
| OVSPD | Overspeed |
| PAF | Pressure after Filter |
| PART SEP | Particle Separator |
| PAX | Passenger |
| PBF | Pressure before Filter |
| PF | Pressure Filter |
| PLA | Power Lever angle |
| PMA | Permanent Magnet Alternator |
| POS LT | Position Light |
| PRESS | Pressure |
| PRCU | Pedal Restrictor Control Unit |
| PSI | Pounds per Square Inch |
| PWR | Power |
| QTY | Quantity |
| RECP | Receptacle |
| RLY | Relay |
| RPM | Revolutions per Minute |
| R/FUEL | Right Boost/Transfer Pumps |
| SHP | Shaft horsepower |
| SL | Sea level |
| SPKR | Speaker |
| SYS | System |
| TEMP | Temperature |
| T/R | Tail rotor |

# GENERAL DESCRIPTION

### Abbreviations and Acronyms (Cont)

| | |
|---|---|
| TRANS | Transmission |
| TRQ | Torque |
| V | Volt(s), Voltage |
| VDC | Voltage Direct Current |
| VFR | Visual Flight Rules |
| VM | Volatile Memory |
| $V_{NE}$ | Never Exceed Velocity |
| VOR | VHF Omnidirectional Range |
| WL | Water Line |
| WRN | Warning |
| XFR | Transfer |
| XMSN | Transmission |
| XPNDR | Transponder |

# Crew Compartment

THE BELL TRAINING ACADEMY
Est. 1947

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## Crew Compartment



Bell Helicopter
A Textron Company

# CREW COMPARTMENT

## Instrument Panel


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# CREW COMPARTMENT

## General

The pilot's station is on the right side of the crew compartment. The left side has provisions to accommodate a passenger or a copilot.

The instrument panel is mounted on a center pedestal forward of the two crew seats and tilts forward at a five degree angle for maximum visibility. Flight instruments are located on the right side of the panel and system instruments are in two rows to the left of the flight instruments. Caution, Warning, and Advisory Panel (CWAP) lights are mounted just below the glare shield across the top of the instrument panel. A "press- to- test" switch for the CWAP lights is located to the right of the light panel.

The pedestal extends aft from the instrument panel between the seats to form a console for the radios.

The overhead console is centrally positioned aft of the windshields on the cockpit ceiling. This console contains most of the circuit breakers and electrical switches.

## Instrument System

The instrument system is divided into four categories: flight, navigation, propulsion, and miscellaneous. All indicators are installed in the instrument panel except the standby compass and hourmeter. The standby compass is on the right side of the cabin slightly forward of the instrument panel. The hourmeter is in the battery compartment.

The flight instruments include the attitude indicator, airspeed indicator, altimeter, turn and slip indicator, and vertical speed indicator.

The navigation instrument on the basic 407 helicopter is the standby magnetic compass.

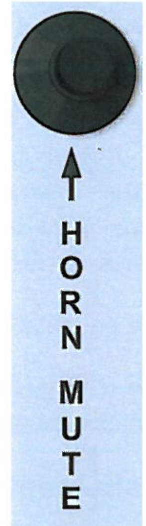
The propulsion instruments are the dual tachometer, gas producer tachometer, engine torquemeter indicator, measured gas temperature indicator, engine oil pressure/temperature indicator, transmission oil pressure/temperature indicator, fuel quantity indicator, fuel pressure indicator, and amps indicator.

The miscellaneous instruments are the clock/OAT/voltmeter and the engine hourmeter.

# CREW COMPARTMENT

## Instrument Panel Switches

**ENGINE OUT and LOW ROTOR RPM HORN MUTE** button located in the upper right hand corner of the instrument panel silences the audio alarms for ENGINE OUT and LOW ROTOR RPM. Pressing the HORN MUTE button temporarily silences both warning tones. The warning horns automatically reset to operate when preset operational parameters are reestablished.

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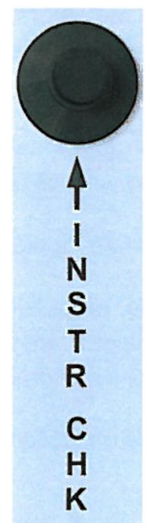
**INSTR CHK** button is for checking and clearing Exceedances. The pilot can acknowledge the exceedance and display the peak value on the analog and digital displays by pushing the INSTR CHK button. The exceedance displays for a maximum of 11 seconds, unless the pilot pushes on the INSTR CHK button. If Exceedances have been recorded on different indicators, each indicator displays its last exceedance. Once the pilot pushes the INSTR CHK button and the exceedance(s) displays, the E disappears from the digital display.

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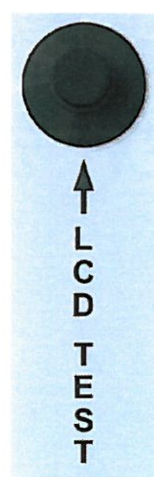
**LCD TEST** button performs a command Built in Test of the propulsions instruments. When the LCD TEST button is pushed, the LCDs on the propulsion instruments turn on so the pilot can verify the LCDs are working. On the dual tachometer indicator, the individual Rotor and Turbine pointers are driven to their respective upper redline limits.

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## CREW COMPARTMENT

**Instrument Panel Switches (Cont)**


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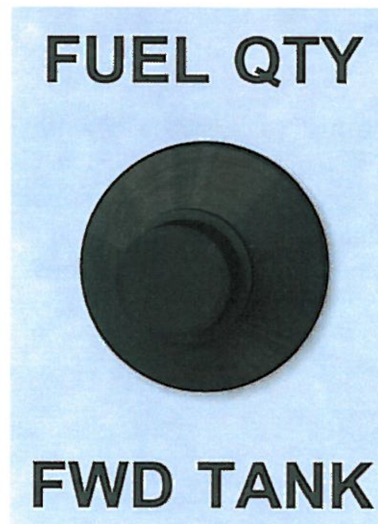
If the fire detection system is installed, a FIRE DET.TEST button will be located at the top L/H side of the instrument panel.

A FUEL QTY FWD TANK switch is located to the left of the fuel quantity gauge. When pushed the fuel quantity gauge will show the total fuel quantity in the forward tank.

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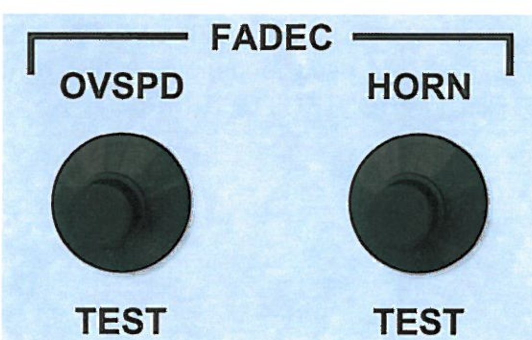


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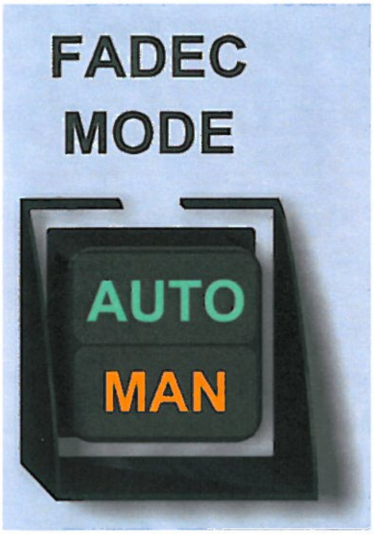
The FADEC OVSPD TEST and the FADEC HORN TEST buttons are located in the lower L/H portion of the instrument panel. The HORN TEST button, when pushed will verify the FADEC Chime works properly. The OVSPD TEST button when pushed, tests the FADEC engine overspeed protection system.

# CREW COMPARTMENT

## Instrument Panel Switches (Cont)

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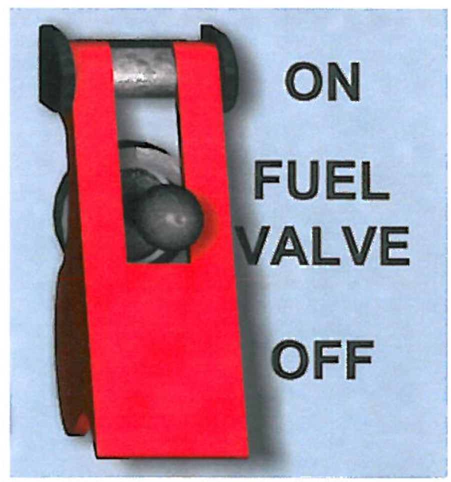
The FADEC MODE switch/annunciator is located in the lower R/H portion of the instrument panel. The FADEC MODE annunciator advises the pilot in which MODE the FADEC is operating. If the FADEC fails to manual, the pilot can push the switch to MUTE the FADEC chime. The igniter also begins to operate and the AUTO RELIGHT light illuminates.

The fuel valve switch is a two-position switch located on the lower right hand side of the instrument panel and electrically operates a motor driven fuel valve providing a means of shutting off fuel to the engine. A cover guards the switch in both the ON and OFF positions.

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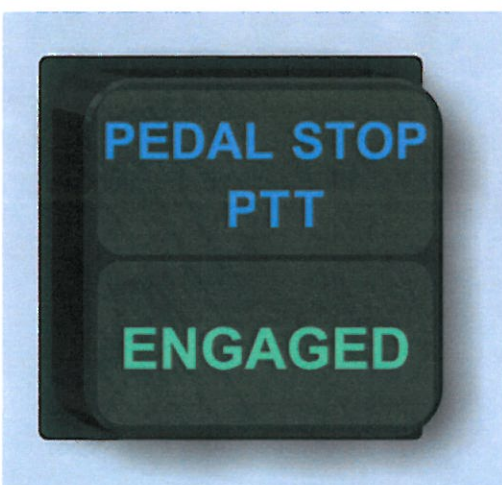


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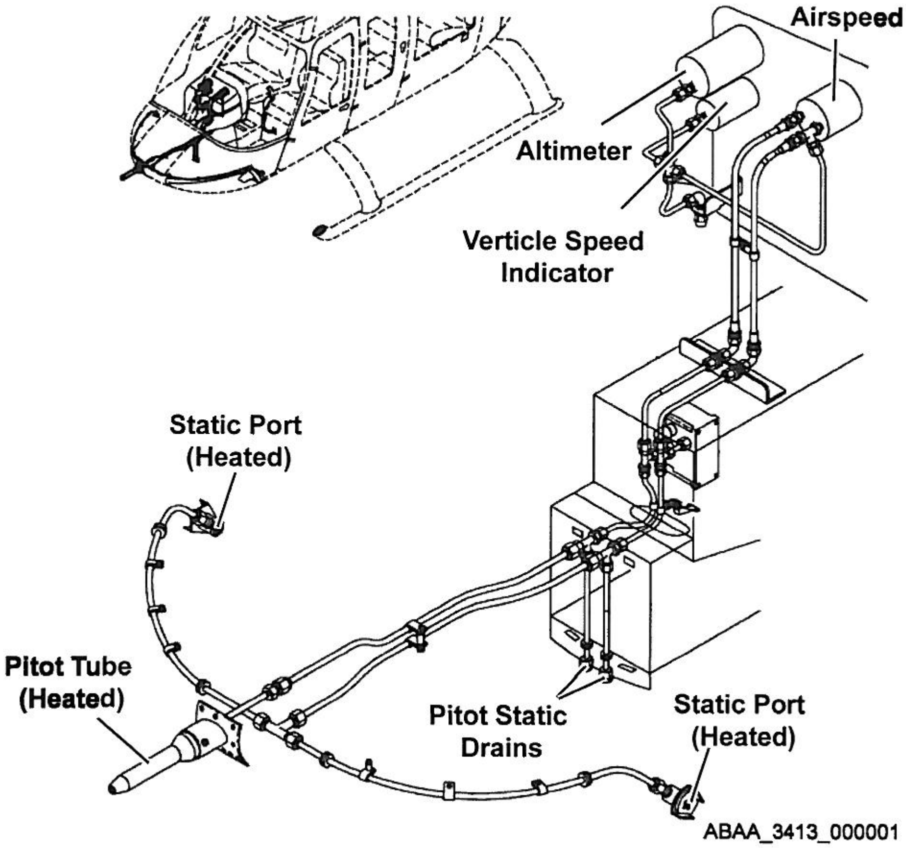
The PEDAL STOP Annunciator/switch provides a visual signal to the pilot as to the position of the PEDAL STOP solenoid. The switch is used to perform a test of the PEDAL RESTRICTOR CONTROL UNIT (PRCU).

# CREW COMPARTMENT

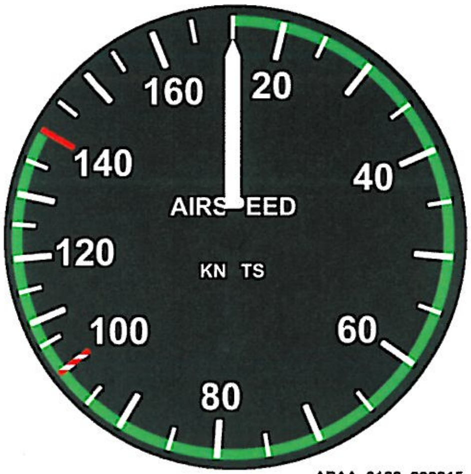
## Pitot /Static System




The pitot tube is mounted on the most forward part of the cabin nose structure. The tube supplies impact air to the airspeed indicator. Two static ports on the left and right sides of the aircraft, aft of the cabin lower windows, provide static air pressure for instrument operation. Turning the PITOT HEAT switch to the ON position activates the heater elements for both the pitot tube and static ports.

# CREW COMPARTMENT

## Flight Instruments

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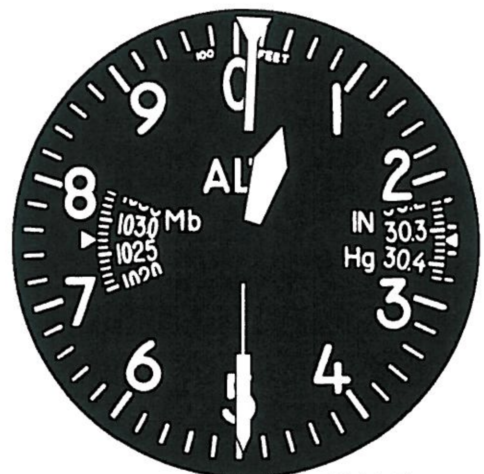
**Airspeed Indicator** - The airspeed indicator is a standard pitot/static instrument. This indicator provides an airspeed reading in knots (KTS) by measuring the difference between impact air pressure from the pitot tube and the static air pressure from the static ports.

The indicator presents airspeed from 0 to 160 knots in 5-knot increments at 20 knots and above.

**Altimeter** - The barometric pressure altimeter presents an altitude reading in feet above mean sea level (MSL) based on the relationship between the static air pressure and the barometric setting on the altimeter. The pilot may adjust the barometric setting to reflect the current barometric pressure corrected to sea level in inches of mercury or in millibars, depending on the instrument installed.

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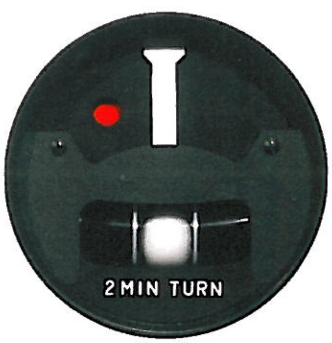


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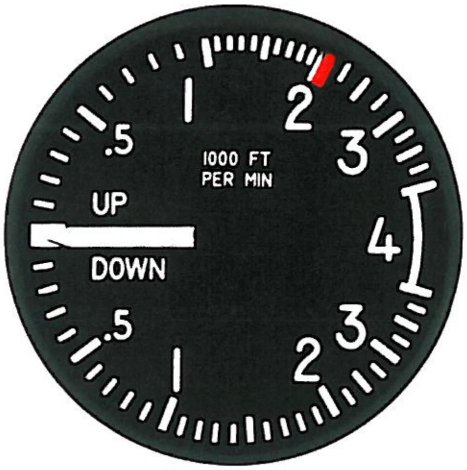
**Turn And Slip Indicator** - The inclinometer is a simple instrument consisting of a curved glass tube, ball, and dampening fluid. The ball indicates when the helicopter is in a coordinated turn or balanced straight and level flight, i.e. "in trim." If the helicopter is in a slip or skid, the ball will move off center. The 28 VDC bus through the TURN circuit breaker switch powers the turn needle. The turn needle indicates the rate at which the aircraft is turning about the vertical axis in degrees per second. A red electrical power failed flag indicates the turn needle is inoperative.

**Flight Instruments (Cont)**

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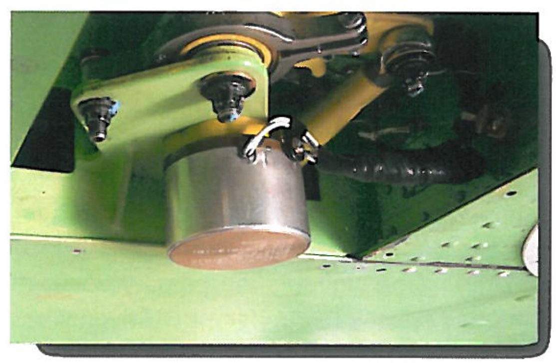
**Vertical Speed Indicator** - The vertical speed indicator presents rate of climb or descent from 0 to 4000 feet per minute. A maximum rate of climb red line is located at 2000 FPM.

## Airspeed Actuated Pedal Restrictor Control System

The airspeed actuated pedal stop system includes a pedal restrictor control unit (PRCU), an actuating rotary solenoid, positioning sensing microswitch, and a press-to-test PEDAL STOP (PTT) switch. The helicopter pitot and static installation interfaces with PRCU. The PRCU calculates airspeed from the pitot static inputs, and when greater than 55 +/- 5 KIAS, drives the solenoid, which rotates a mechanical restrictor and limits the left pedal travel.



**View Looking Up**

ABAA_6731_000015

Upon full extension of the pedal stop, the ENGAGED message will appear on the PEDAL STOP (PTT). When airspeed is 50 +/- 5 KIAS or less, the PRCU de-energizes the solenoid, which rotates the mechanical restrictor and left pedal travel returns to normal. The ENGAGED message on the PEDAL STOP (PTT) will extinguish. A PEDAL STOP warning light is located on the CWAP to advise the pilot of a system malfunction. During a total electrical failure (battery and generator), the solenoid is Fail Safe to the disengaged position. The restrictor is retracted and full left pedal travel is available.

# CREW COMPARTMENT

## Navigation Instruments


ABAA_3423_000001

**Standby Magnetic Compass** - The magnetic compass is a standard, non-stabilized, magnetic type instrument mounted on a support attached to the pilot side of the forward cabin next to the doorpost. A compass correction card is located below the instrument.

## Propulsion Instruments

**Dual Tachometer ($N_P$ / $N_R$)** - The Dual Tachometer is equipped with two pointers, one displays main rotor RPM on the outer scale and one displays engine RPM on the inner scale. All scales are in percent of rated RPM.

The 28 VDC bus powers the $N_P$ (engine) side of the instrument through its own circuit breaker.

An $N_P$ monopole speed pickup mounted on the engine provides two separate signals to the ECU (Electronic Control Unit). One signal is the primary driver of the $N_P$ gauge, and passes through the ECU even when the ECU is not powered. The second signal is the primary input to the ECU.


ABAA_3121_000001

The 28 VDC bus powers the $N_R$ (rotor) side of the gauge through its own circuit breaker. An $N_R$ monopole speed pickup mounted on the transmission lower case provides three identical signal outputs of rotor RPM:

- One to the $N_R$ circuit of the dual tachometer gauge
- One to the ECU
- One to the main rotor RPM sensor switch

### Propulsion Instruments (Cont)

ABAA_3121_000002

**Gas Producer Tachometer ($N_G$)** - The gas producer ($N_G$) displays engine gas producer speed in percent of rated RPM. The 28 VDC bus powers the gas producer tachometer through its own circuit breaker. An $N_G$ monopole speed pickup mounted on the engine accessory gearbox provides two separate signals to the ECU. One signal is for the ECU, the other is for the gauge.

$N_G$ in percent RPM displays on the LCD trend arcs and the digital display. The gauge has the ability to record exceedances.

**Engine Torque Meter (TRQ) Indicator** - The engine torque meter uses engine oil pressure to determine power output of the engine in the form of percent torque. The gauge is powered by the 28VDC bus through its own circuit breaker. Oil pressure is fed to both the ECU and the torquemeter oil pressure transducers. TRQ in percent is displayed on the LCD trend arcs and digital display. The torque gauge has the ability to record exceedances.

ABAA_3121_000003

ABAA_3121_000004

**Measured Gas Temperature (MGT) Indicator** - The measured gas temperature (MGT) indicator displays engine gas temperature in degrees Celsius. Four probes measure gas temperature between the gas producer turbine and the power turbine. MGT displays on the LCD trend arcs and on the digital display. The 28 VDC bus power the MGT gauge through its own circuit breaker. The MGT gauge can determine and record two separate exceedance levels, one for starting, and another for normal operations.

**Engine Oil Pressure/Temperature Indicator** – The engine oil pressure/temperature indicator displays oil pressure in PSI on the left side of the instrument and oil temperature in degrees Celsius on the right side. This information displays on the LCD trend arcs.

The 28 VDC bus, through its own circuit breaker, powers each side of the indicator

A thermo-bulb on the outlet side of the engine oil tank provides the engine oil temperature input signal. A transducer on the forward right side of the engine firewall provides the engine oil pressure input signal to the gauge.

ABAA_3121_000006

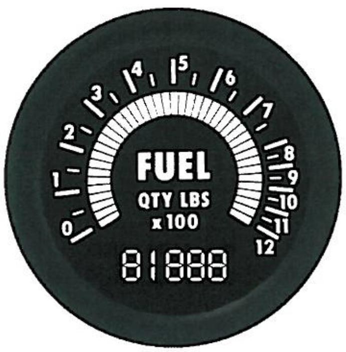
# CREW COMPARTMENT

## Propulsion Instruments (Cont)

**Transmission Oil Pressure/Temperature Indicator** – The transmission oil pressure/temperature indicator displays oil pressure in PSI on the left side of the instrument and oil temperature in degrees Celsius on the right side. This information displays on the LCD trend arcs. The 28 VDC bus, through its own circuit breaker, powers each side of the indicator.

A thermo-bulb installed on the transmission oil filter manifold provides the transmission oil temperature input signal. A transducer, mounted on the transmission oil manifold, provides the transmission oil pressure input signal to the gauge.


ABAA_3121_000005

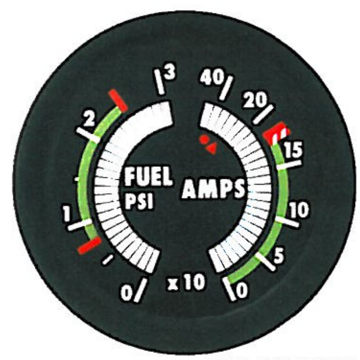



ABAA_3121_000007

**Fuel Quantity Indicator** – The fuel quantity indicator displays total fuel quantity in pounds from three quantity probes in the fuel cells. The fuel quantity displays on the LCD arcs and on the digital display. The 28 VDC bus powers the fuel quantity indicator through its own circuit breaker.

The fuel quantity gauge indicates the total usable fuel in both fuel tanks. If an auxiliary tank is installed, the auxiliary quantity is automatically computed into the aft tank quantity.

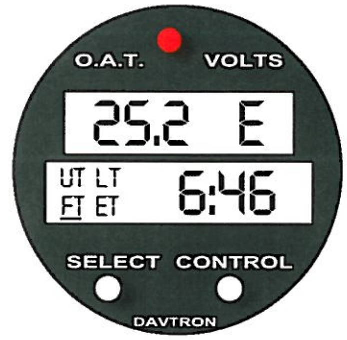
**Fuel Pressure/Ammeter Indicator** – The fuel pressure/DC ammeter indicator displays fuel pressure in PSI on the left side of the instrument and DC current load on the generator on the right side of the instrument, on the LCD trend arcs. A pressure transducer below the fuel valve measures the output of both boost pumps and provides this indication. The ammeter indicates the load in amperes, which the generator is supplying to the 28 VDC bus. The 28 VDC bus powers each side of the indicator through its own circuit breaker.




ABAA_3121_000008

# CREW COMPARTMENT

## Miscellaneous Instruments



ABAA_3160_000011

**Clock/OAT/Voltmeter** - The Clock/OAT/Voltmeter is a multifunction indicator mounted in the upper left area of the instrument panel. The 28 VDC bus powers the clock through its own circuit breaker. When the battery switch is turned on, the instrument defaults to the VOLTS mode and displays voltage available at the bus. A red button located on the top center of the instrument selects outside air temperature (OAT) in either degrees Celsius or Fahrenheit and VOLTS. The lower half of the instrument is a chronometer that displays:

- Universal Coordinated (ZULU) Time (UT) in 24-hour format
- Local Time (LT) in 12 or 24-hour format
- Elapsed Time (ET) count up timer to a maximum of 99 hours, 59 minutes; elapsed time count down timer from 59 hours, 59 minutes
- Flight Time (FT) count up or down a maximum of 99 hours, 59 minutes, activated when the aircraft is in flight, i.e. weight off gear

ABAA_3121_000009

**Engine Hourmeter** – The engine hourmeter mounts on the aft bulkhead of the battery compartment. The 28 VDC bus powers the hourmeter through its own circuit breaker. When the $N_G$ is greater than 55% and the aircraft is in flight, i.e. weight off gear, the system activates.

# CREW COMPARTMENT

## Exceedance Monitoring

Exceedances are limits of operation above which maintenance action may be required.

ABAA_3121_000003

TRQ, MGT, and $N_G$ gauges contain a microprocessor with a 10-year lithium backup battery with the ability to record in its Non-Volatile Memory (NVM) up to 50 events or exceedances. For each event, the gauge records the date, duration, and peak value of the exceedance.

The CHECK INSTR caution light illuminates and the gauge LCD trend arcs begin flashing to provide an advance warning that an exceedance is about to be recorded. If the pilot makes control inputs to reduce the instrument readings below the advisory values, the gauge trend arcs return to normal (i.e. stop flashing), and the CHECK INSTR caution light extinguishes. No exceedances are recorded.

If a limit is exceeded, the gauge trend arcs stop flashing and an "E" followed by the exceedance value, appears in the digital readout. The CHECK INSTR caution light remains on until the pilot acknowledges the exceedance by pushing the INSTR CHK button. Once the pilot pushes the INSTR CHK button, the numeric value of the last recorded exceedance appears for up to 11 seconds on the gauge digital readout. Releasing the INSTR CHK button causes the "E" and the exceedance value to disappear from the digital display and the CHECK INSTR light extinguishes. The instrument will then function normally. The "E" will re-appear only after power has been removed from and reapplied to the gauge.

An "E" and the value continues to display each time the gauge is powered up until the exceedance(s) is/are removed from the NVM of the gauge. This information will be stored in the instrument's NVM until cleared with a computer.

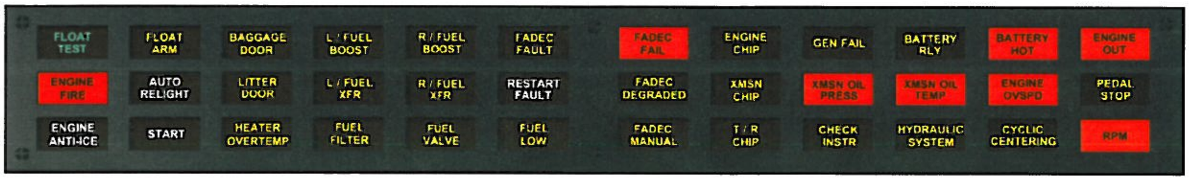
If the gauges have recorded an exceedance, prior to operating the helicopter the pilot shall review the aircraft maintenance logbook to verify the exceedance has been entered in the logbook and corrective maintenance has been performed. If corrective action has not been taken, the aircraft is considered not airworthy. Refer to the aircraft maintenance manual and maintenance personnel for corrective action. Corrective maintenance action shall be performed to return the aircraft to service.

If the affected gauge has not had the exceedance cleared with the LITTON furnished software, the "E" continues to appear each time the aircraft is powered up. The pilot can verify the exceedance value by pressing the INSTR CHK button. The exceedance value disappears from the display when the pilot releases the INSTR CHK button, and the instrument functions normally.

# CREW COMPARTMENT

## Caution and Warning System

The caution and warning system consists of the Caution Warning and Advisory Panel (CWAP), engine RPM sensor and warning horn, rotor RPM sensor and low rotor RPM warning horn, and the FADEC Warning horn.




ABAA_3141_000038

### Caution Warning and Advisory Panel (CWAP)

The 28 VDC bus powers the CWAP through its own circuit breaker. Each segmented indicator light is in series with its respective system. Warning lights are RED, Caution lights are AMBER, and Advisory lights are WHITE or GREEN. Pressing the C/W LT TEST button illuminates the CWAP and the FADEC MODE SWITCH.

### Definitions

### Land As Soon As Possible

Land without delay at the nearest suitable area (i.e. open field) at which a safe approach and landing is reasonably assured.

### Land As Soon As Practical

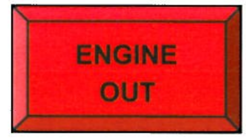
The landing site and duration of flight are at the discretion of the pilot. Extended flight beyond the nearest approved landing area is not recommended.

# CREW COMPARTMENT

### Caution and Warning System (Cont)

#### Engine Out

The ENGINE OUT warning system provides both visual and audible indications of an engine out condition. This includes an ENGINE OUT warning light, located on the CWAP, and the ENGINE OUT warning alarm (intermittent tone) located on the overhead console.



ABAA_3141_000001

The FADEC Electronic Control Unit (ECU) controls the engine out relay in AUTO mode. When gas producer RPM drops below 55 ± 1%, the ECU closes the ENGINE OUT relay, activating the warning light and alarm. In the FADEC manual mode, the ENGINE OUT relay is connected directly to the $N_G$ speed sensor, and when the gas producer RPM drops below 55 ± 1%, the engine out relay closes, activating the warning light and alarm.

#### RPM

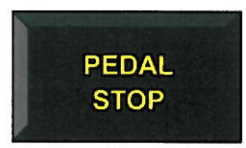
The rotor RPM warning system provides both visual and audible indications of a low or high rotor RPM condition. The rotor RPM light and alarm (continuous tone) activates when the $N_R$ sensor detects the rotor RPM decelerating below 95%.

ABAA_3141_000002

The rotor RPM warning light illuminates when rotor RPM exceeds 107%. The RPM warning audio does not activate for high RPM.

#### Pedal Stop

The PEDAL STOP amber caution light illuminates when the electro-magnetically operated pedal stop mechanism fails to engage or disengage. The PEDAL STOP caution light illuminates if the pedal stop fails to engage when the airspeed accelerates above 55 KIAS or if the pedal stop fails to disengage when airspeed decelerates below 50 KIAS. A hand operated mechanical release manually disengages the pedal stop mechanism. VNE 60 KIAS.



ABAA_3141_000003

#### Battery Hot

Thermal switches installed in the battery activate the BATTERY HOT warning light when the battery reaches preset temperature limits (approximately 63°C for the 17 amp/hour battery and 71°C for the 28 amp/hour battery). Battery switch OFF. **LAND AS SOON AS PRACTICAL.**

ABAA_3141_000004

#### Battery Relay

The BATTERY RELAY caution light illuminates when the battery switch is placed in the OFF position and the battery relay remains closed. If the BATTERY HOT light is also illuminated,
**LAND AS SOON AS POSSIBLE**.

ABAA_3141_000005

# CREW COMPARTMENT

### Caution and Warning System (Cont)

### Engine Overspeed

The light activates when $N_P$ versus TRQ is above the maximum continuous limit. (102.1% $N_P$ at 100% TRQ to 108.6% $N_P$ at 0% TRQ) (FADEC Version 5.202)

ABAA_3141_000006

$N_P$ speed is at or above 102.1% for 2.5 seconds (FADEC version 5.356/5.358). If an $N_P$ overspeed exceedance has occurred, this light will illuminate along with the FADEC DEGRADED light upon engine shut down. (FADEC 5.358)

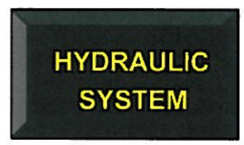
### Cyclic Centering

The CYCLIC CENTERING caution light illuminates when a sensor connected to the pilot's cyclic detects the cyclic is not centered when the aircraft is on the ground. To minimize mast stress at the main rotor hub, the pilot should keep the cyclic centered when the aircraft is on the ground. The cyclic centering system is enabled through the WEIGHT ON GEAR switch and is not functional when the aircraft is in flight.

ABAA_3141_000007

### Hydraulic System

The HYDRAULIC SYSTEM caution light illuminates when hydraulic pressure decreases below 650 PSI. The light extinguishes when the pressure increases above 750 PSI.



ABAA_3141_000008

If illuminated in flight, perform hydraulic system failure procedure.
**LAND AS SOON AS PRACTICAL.**

### Generator Failure

The GEN FAIL caution light illuminates when the generator is not on line or has failed.

ABAA_3141_000009

GEN switch, RESET, then ON. If light remains illuminated, GEN switch OFF.
**LAND AS SOON AS PRACTICAL.**

### Check Instrument

The CHECK INSTR caution light illuminates when TRQ, $N_G$ or MGT has had, or is about to have, an exceedance.

ABAA_3141_000010

## CREW COMPARTMENT

### Caution and Warning System (Cont)

### Transmission Oil Pressure

The transmission oil pressure warning system includes the XMSN OIL PRESS warning light, a pressure transducer, and an oil pressure switch mounted on the transmission oil manifold assembly. The light illuminates when pressure decreases to 30 PSI ± 2 and extinguishes when pressures increases to 38 PSI.
**LAND AS SOON AS POSSIBLE.**

ABAA_3141_000011

### Transmission Oil Temperature

A thermal switch is located on the transmission-mounted oil filter manifold. If the oil temperature rises above 110°C ± 5°C, the XMSN OIL TEMP warning light illuminates.
**LAND AS SOON AS PRACTICAL.**

ABAA_3141_000012

### Engine Chip Detector

The engine chip detector caution system includes the ENGINE CHIP caution light, and two magnetic chip detectors. If ferrous metal particles are in the engine oil, the magnets in the detectors attract them, completing the circuit and illuminating the ENGINE CHIP caution light.
**LAND AS SOON AS POSSIBLE.**

ABAA_3141_000013

### Transmission Chip Detector

The transmission chip detector caution system includes the XMSN CHIP caution light, two transmission chip detectors and a freewheeling unit chip detector. If ferrous metal particles are in the transmission oil, the magnets in the detectors attract them, completing the circuit and illuminating the XMSN CHIP caution light.
**LAND AS SOON AS POSSIBLE.**

ABAA_3141_000014

### Tail Rotor Gearbox Chip Detector

The tail rotor gearbox chip detector caution system includes the T/R CHIP caution light and the tail rotor gearbox chip detector. If ferrous metal particles are in the gearbox oil, the magnet in the detector attracts them, completing the circuit and illuminating the T/R CHIP caution light.
**LAND AS SOON AS POSSIBLE.**

ABAA_3141_000015

### FADEC Fail

The FADEC FAIL warning light indicates a failure of the FADEC system. A FADEC FAIL warning light illuminates and the FADEC FAIL horn sounds when the ECU detects a failure of the FADEC system.
**LAND AS SOON AS PRACTICAL.**

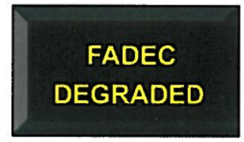
ABAA_3141_000016

# CREW COMPARTMENT

### Caution and Warning System (Cont)

### FADEC Degraded

The FADEC DEGRADED caution light receives its signal from the ECU and indicates a degraded condition.
**LAND AS SOON AS PRACTICAL.**

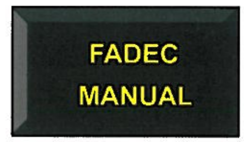



ABAA_3141_000017

The FADEC DEGRADED caution light may also illuminate during engine shutdown, indicating the ECU has records a FADEC related error code.

### FADEC Manual

The FADEC MANUAL caution light receives its signal from the ECU when the FADEC is operating in manual mode.
**LAND AS SOON AS PRACTICAL.**




ABAA_3141_000018

### FADEC Fault

The FADEC FAULT caution light receives its signal from the ECU. This light indicates $N_P$ or $N_G$ automatic limiting circuits may not be functional or the permanent magnet alternator (PMA) has failed.
**LAND AS SOON AS PRACTICAL.**


ABAA_3141_000019

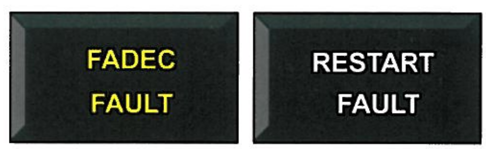
### Restart Fault

The RESTART FAULT light is a white advisory light. When illuminated, automatic engine start may not be possible.


ABAA_3141_000020

### FADEC Fault/Restart Fault

The FADEC FAULT and the RESTART FAULT lights receive a signal from the ECU. These lights indicate MGT automatic limiting may not be functional.
**LAND AS SOON AS PRACTICAL.**

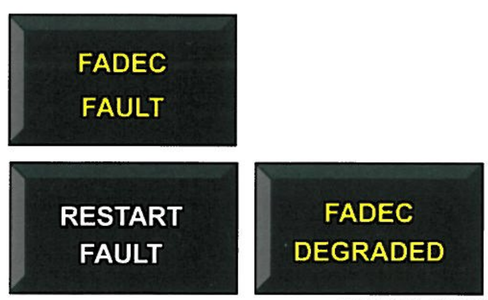



ABAA_3141_000021

### Primary Governor Failure (FADEC)

Failure of the FADEC primary channel. 5.356/5.358

Illumination of the FADEC FAULT, RESTART FAULT and FADEC DEGRADED LIGHTS indicates reversionary governor mode
**LAND AS SOON AS PRACTICAL.**




ABAA_3141_000022

### Caution and Warning System (Cont)

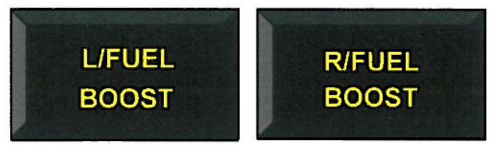
#### Fuel Low

The FUEL LOW caution light illuminates when approximately 100 ± 10 pounds of fuel remains in the aft (main) cell. The input from the low-level detector passes through the fuel signal conditioner, which provides a 13-second (± 3 seconds) time delay to reduce the possibility of intermittent annunciator flickering due to fuel sloshing.
**LAND AS SOON AS PRACTICAL.**

ABAA_3141_000023

#### Fuel Boost Pumps

A fuel pressure switch located at the base of each pump controls the L/FUEL BOOST and R/FUEL BOOST caution lights. The respective switch activates when fuel pump pressure drops to 1.5 PSI ± 0.5 or lower, indicating a pump failure. The lights extinguish when fuel pressure increases to 5 PSI and above.



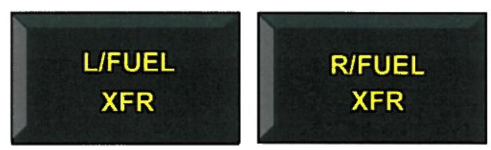
ABAA_3141_000024

If one pump fails, descend below 8000 ft HP, Jet A Fuel, 4000 ft HP, Jet B fuel.
**LAND AS SOON AS PRACTICAL.**

If both boost pumps fail, **LAND AS SOON AS POSSIBLE**.

#### Fuel Transfer Pumps

The fuel pressure switch at the base of each pump also controls THE L/FUEL XFR and R/FUEL XFR caution lights. The respective switch activates when fuel pump pressure drops to 1.5 PSI ± 0.5 or lower, indicating a pump failure. The lights extinguish when fuel pressure increases to 5 PSI and above. In the event of dual transfer pump failure, up to 135 pounds of Jet B fuel or 151 pounds of Jet A fuel may be trapped (unusable) in the forward tank.



ABAA_3141_000025

Aircraft SN 53001 to 53174 caution lights illuminate for approximately 2 1/2 minutes then extinguish when fuel transfer from the forward tank is completed. Aircraft SN 53175 and subsequent caution lights illuminate only in the event of a pressure drop indicating a pump failure.
**LAND AS SOON AS PRACTICAL.**

#### Fuel Valve

The FUEL VALVE caution light illuminates when the fuel valve position differs from the switch position, or when the valve moves between open and closed positions.
**LAND AS SOON AS PRACTICAL.**

ABAA_3141_000026

# CREW COMPARTMENT

## Caution and Warning System (Cont)

### Fuel Filter

The FUEL FILTER caution light illuminates when the airframe fuel filter is in an impending bypass condition (approximately 1 PSI differential). The airframe fuel filter bypasses at approximately 4 PSI differential.
**LAND AS SOON AS PRACTICAL.**

ABAA_3141_000027

### Baggage Door

A micro switch on the baggage door illuminates the BAGGAGE DOOR caution light when the door is open or not securely fastened.
**LAND AS SOON AS PRACTICAL.**

ABAA_3141_000028

### Litter Door

The litter door caution system includes the LITTER DOOR caution light, a micro switch for the upper door striker, and related wiring. If the litter door is not securely fastened or has been removed, the caution light illuminates.
**LAND AS SOON AS PRACTICAL.**

ABAA_3141_000029

### Heater Overtemp

The HEATER OVERTEMP caution light receives its signal from one of three temperature sensors, one each below the pilot and copilot seats and one located in the passenger compartment heater ducting. The HEATER OVERTEMP caution light illuminates when the temperature in any sensor reaches 104° C.
**TURN HEATER SWITCH OFF IMMEDIATELY.**

ABAA_3141_000031

### Auto Relight

The AUTO RELIGHT advisory light illuminates whenever the engine igniter is operating. During the start sequence, the AUTO RELIGHT advisory remains on until $N_G$ reaches 60%, when the FADEC is operating in the manual mode above 55% $N_G$ , or FADEC detects an engine out condition with $N_G$ above 50%.

ABAA_3141_000032

### Start

The START light is a white advisory light that illuminates when the start relay is energized. The START light extinguishes when $N_G$ reaches 50%.

ABAA_3141_000034

# CREW COMPARTMENT

## Caution and Warning System (Cont)

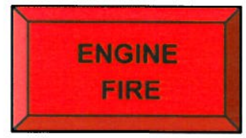
### Engine Anti-Ice

The ENGINE ANTI-ICE white advisory light illuminates when the engine anti-icing system is operating. This light is standard on ship serial numbers 53095 and subsequent.


ABAA_3141_000035

### Engine Fire (FMS-21)

The ENGINE FIRE warning light and fire test button are installed with the optional engine fire detector kit. The ENGINE FIRE caution light illuminates when a heat sensitive wire mounted on the inboard side of the engine compartment senses excessive heat. The ENGINE FIRE warning light also illuminates when the press-to-test FIRE DET TEST button is pressed.




ABAA_3141_000033

### Float Arm (FMS-1)

The FLOAT ARM caution light illuminates when the pilot selects the "protected" float arm switch located on the collective head to the "ARMED" position. The floats allow the aircraft to land on water if an emergency occurs during flight.


ABAA_3141_000036

This float kit has an airspeed restriction of 60 KIAS or less for inflation. Refer to FMS-1 for additional limitations and emergency procedures.

### Float Test (FMS-1)

The FLOAT TEST advisory light illuminates when the pilot simultaneously presses and holds the float test switch and the float inflation switch.


ABAA_3141_000037

# CREW COMPARTMENT

## Overhead Circuit Breaker Panel


ABAA_9670_000001

# CREW COMPARTMENT

## Electrical Fail Safe Systems

**Engine Anti-Ice System** - The hot air solenoid valve is powered from the 28 VDC bus through the ENGINE ANTI-ICE circuit breaker and the ENG ANTI-ICE switch. When the solenoid valve is de-energized (ENG ANTI-ICE switch is ON), bleed (hot) air passes from the compressor diffuser through the anti-ice valve to the engine inlet housing. This hot air helps prevent ice formation on the hollow inlet guide vanes. Electrical power is provided to the solenoid only when the switch is positioned to OFF.

**Particle Separator** - The particle separator kit is equipped with a PART SEP purge switch located on the overhead console. With the switch in the ON position, engine bleed air purges debris from the separator. Electrical power is provided to the solenoid only when the switch is positioned to OFF. Performance charts in the appropriate supplement provide data for use with this kit.

**Hydraulic System Switch** - The HYDRAULIC SYSTEM switch is located on the overhead console and controls the operation of the hydraulic bypass solenoid. The solenoid is electrically de-energized open when the hydraulic switch is in the ON position, allowing pressurized hydraulic fluid to flow into the hydraulic manifold. Electrical power is provided to the solenoid only when the switch is positioned to OFF.

**Airspeed Actuated Pedal Stop System** – The system includes a Pedal Restrictor Control Unit (PRCU), an actuating rotary solenoid, positioning sensing microswitch, and a press-to- test (PTT) switch. The PRCU calculates airspeed from the pitot and static inputs and, when greater than 55 ±5 KIAS, drives the solenoid to extend the pedal stop restrictor into the left pedal range of travel, at which time the ENGAGED light illuminates. Electrical power is required to keep the solenoid engaged. If aircraft electrical power is lost, the solenoid will disengage. Airspeed is now restricted to 60 KIAS.

# CREW COMPARTMENT

## Lighting Systems

The lighting system includes both interior and exterior lighting. The interior lighting system includes the cockpit map light, instrument and control panel lighting, and cabin lighting. The exterior lighting system includes position, anti-collision, and landing lights.

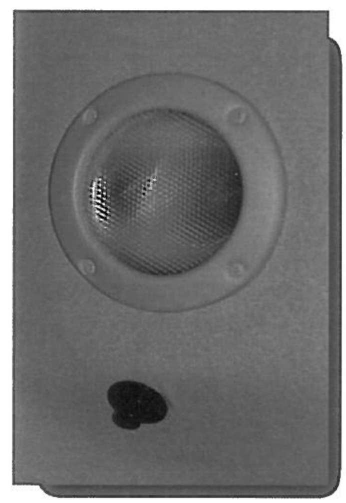
The cockpit map light features a narrow spotlight or wide floodlight beam. It is a multipurpose utility light designed to provide blue or white illumination. The cockpit light body provides controls for ON/OFF, BRIGHT/DIM, BLUE/WHITE, and SPOT/FLOOD. 28 VDC power is supplied through the CKPT LIGHTS circuit breaker located on the overhead console.

Instrumentation lighting 28 VDC power is supplied through the INSTR LIGHTS circuit breaker located on the overhead console. The INSTR LT rheostat knob, also located on the overhead console, adjusts light intensity. Rotation of this knob operates the power OFF/BRT switch that provides power to both the 28 and 5 VDC lighting systems. The INSTR LT rheostat also enables the CWAP BRT/DIM switch.

**Bright/Dim Switch** - With the INSTR LT rheostat adjusted out of the OFF position, the pilot can adjust the CWAP light intensity to bright or dim intensity by positioning the BRT/DIM switch to the desired position. The CWAP lights remain at the selected intensity until the pilot turns the INSTR LT rheostat to another position or power is removed from the DC bus. The ENGINE OUT, RPM, BATTERY HOT, ENGINE OVSPD, and FLOAT TEST warning and advisory lights do not dim.

**Interior Lighting System** - Two individual white cabin lights, protected by their own circuit breaker, provide passenger area illumination at the pilot's discretion. The CABIN LT switch and individual switches near each light regulate the light.

- In CABIN LT position, both cabin lights illuminate.
- In PASS position, the passenger can control each light.



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**Exterior Lighting System** - The exterior lighting system includes position lights, anti-collision light, landing lights, and related wiring and components. Position lights are located on the outboard side of the auxiliary fins, the lower cabin left and right side, and on the aft side of the tail rotor gearbox fairing.

The anti-collision light mounts on the top portion of the vertical fin.

Dual landing lights are in the lower portion of the helicopter nose section.

# CREW COMPARTMENT

## Flight Controls

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The cyclic controls the tilt of the main rotor disc by changing the pitch angle of the rotor blades in their cycle of rotation.

**Cyclic Switches** - A two position transmit switch is mounted on the cyclic grip. Depressing the switch to the first detent position activates the internal communication system (ICS) between the pilot and passengers. Depressing the switch to the second detent allows the pilot to transmit on the radio selected on the audio panel.

The collective, located to the left of the pilot's seat, controls the pitch angle of the main rotor blades.

**Collective Switches** - If the aircraft is equipped with floats, the collective control head will have a guarded FLOAT ARM switch on the lower left area to arm the floats. A FLOAT INFLATE push button is located to the right of the arming switch.

The upper surface of the collective head has a LDG LTS switch and a START/DISENG switch.

An indexed bezel helps match the Power Lever Angle (PLA) throttle position to engine Ng during transition from FADEC automatic mode to manual mode.


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**Tail Rotor Pedals And Adjusters** – The tail rotor pedals control the pitch of the tail rotor blades. The tail rotor control pedals contain a bellcrank pedal adjuster, which provides for manual adjustment of pedal position. The pedals also may be manually adjusted to re-position the pedal footrests.

# CREW COMPARTMENT

## Ventilation System

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Opening the sliding window in each of the crew and passenger entrance doors provides air for cabin ventilation.

Pulling out the VENT control knobs and positioning the DEFOG blower switch on the overhead console to the DEFOG (ON) position provides additional ventilation to the cockpit.

**Ram Air** - When the VENT control knobs below the instrument panel are pulled out; ram air is forced into the air intake grills and directed through the plenum and flapper valve. The VENT control knob opens and closes the flapper valve assembly.


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**Defog** - Two electrically driven axial flow blowers are installed in the inlet end of the defroster nozzles. The DEFOG blower circuit breaker switch on the overhead console controls the blowers.

The defog system provides airflow for ventilation and defogging when the helicopter is on the ground or hovering.

# CREW COMPARTMENT

## Seating

**Crew And Passenger Seats** - All seat back cushions are constructed of soft sponge with styrene backing. Seat cushions are constructed of soft urethane foam. Back and seat cushions are bonded to tubular frames and upholstered with fabric resistant to flame, fluids, and climatic conditions.

A small storage area is located behind each crew seat. Aft facing seats are equipped with an adjustable headrest. (Refer to the placard on the inside of the litter door).

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### Restraint Assemblies

Each crew and passenger seat is equipped with a restraint assembly that consists of an inertial reel, shoulder harness, and an adjustable seat belt. The inertia reel has an anti-rebound lock feature and can retract 22 inches of web belt.

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# CREW COMPARTMENT

## Flight Manual, Airworthiness Certificate, and Aircraft Registration Certificate Case

The Rotorcraft Flight Manual contains information needed for the safe operation of the aircraft, and shall be on-board for all operations.

Airworthiness and Registration Certificates are located in an aircraft document case mounted on the right forward side of the instrument panel console. These certificates are required to be carried in the aircraft at all times.


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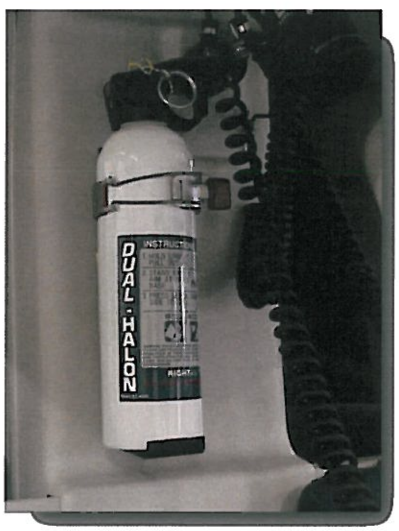
## Fire Extinguisher

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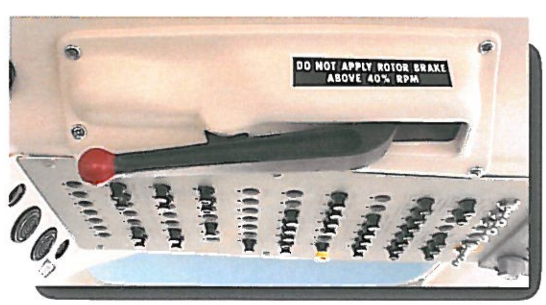
A manually operated fire extinguisher is furnished with each helicopter. The extinguisher is located in the crew compartment between the pilot and co-pilot seats. The mounting bracket is a quick-opening type for rapid removal of the extinguisher.

## Rotor Brake

If a rotor brake is installed, application is limited to ground operation after the engine has been shut down and $N_R$ has decreased to 40% or lower. Engine starts with the rotor brake engaged are prohibited.

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# DRIVETRAIN AND ROTORS

THE BELL TRAINING ACADEMY
Est. 1947

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## Drivetrain and Rotors

# DRIVETRAIN AND ROTORS

## Drivetrain

The drivetrain system provides a means of transmitting power from the engine to the main and tail rotor assemblies. The drivetrain includes the freewheeling unit assembly, main drive shaft, transmission, mast, tail rotor drive shaft, oil cooler blower, and tail rotor gearbox.

## Freewheeling Unit Assembly

The freewheeling unit is mounted on the engine gearbox and driven under power from the engine power takeoff gear shaft. Engine power is transmitted to the outer race of the freewheeling unit, through the engaged sprag clutch and into the freewheeling inner shaft. The freewheeling inner shaft couples the engine to the main driveshaft on the forward attachment plate. The tail rotor drive system is driven through a flexible coupling and a splined adapter mounted on the aft end of the freewheeling inner shaft.

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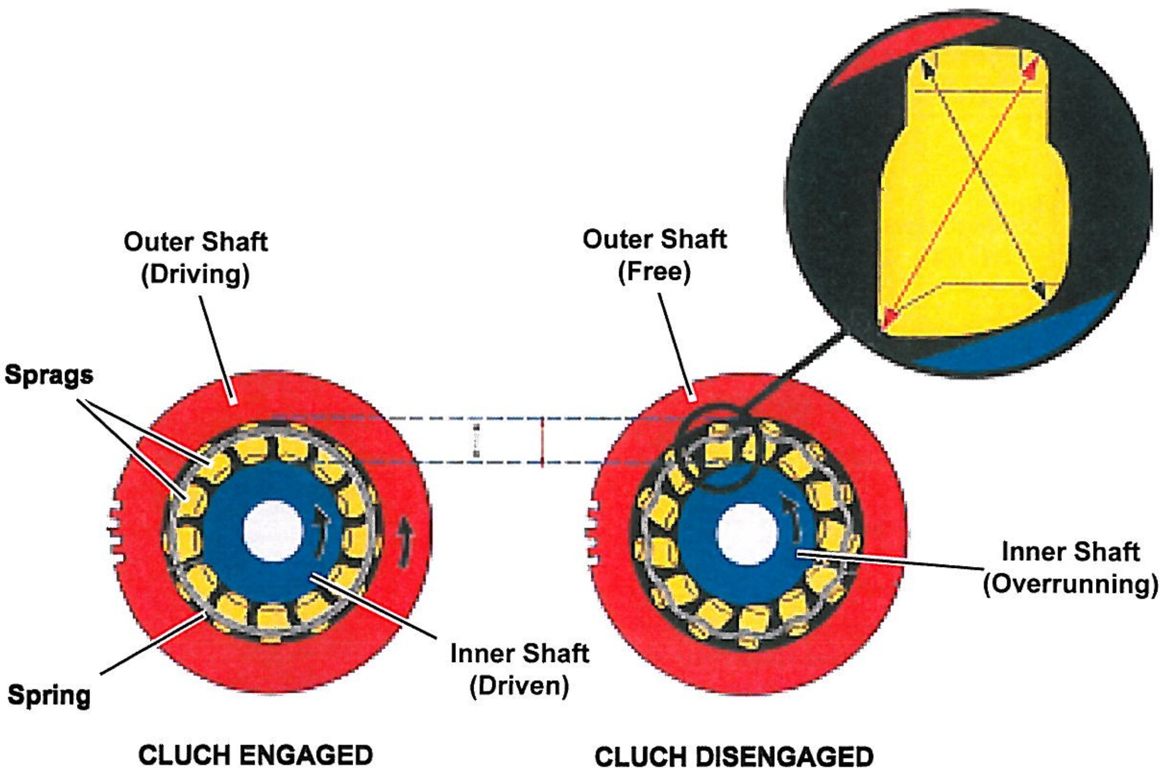
If the engine fails, the sprag clutch disengages and the rotational forces of the main rotor are utilized to drive the transmission accessories and tail rotor drive system.

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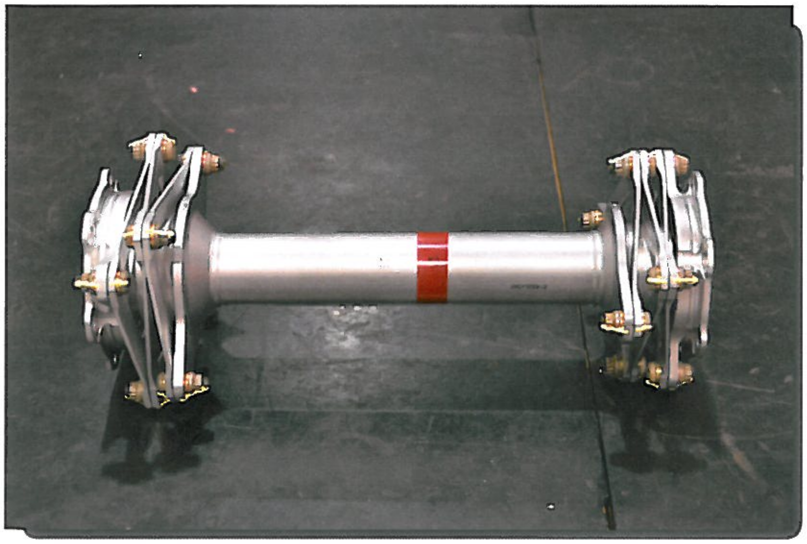
# DRIVETRAIN AND ROTORS

## Main Driveshaft

The engine to transmission Kaflex main driveshaft provides a flexible drive connection between the freewheeling unit and transmission. This flexible shaft allows for the smooth transfer of torque when the transmission and freewheeling unit are not aligned.



Flexibility is provided by three rectangular plate sets in each coupling. Each plate flexes, providing both angular misalignment and length changes to accommodate movement of the transmission on its mounts. Each coupling can be considered a truss work in which torque loads are carried as axial loads in straight members of each plate. The main driveshaft turns 6317 RPM at 100 % $N_P$.

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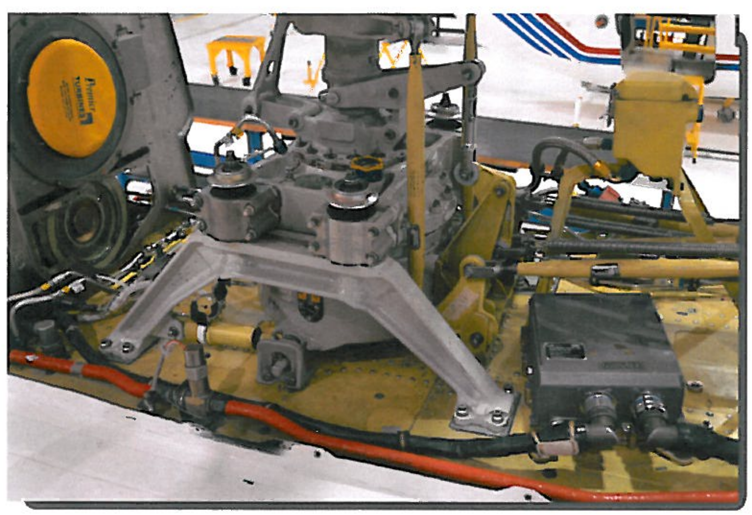
## Transmission

The transmission is utilized to transfer engine torque through the mast assembly to the main rotor system. The two stage transmission will reduce input from 6317 engine RPM to 413 rotor RPM, a reduction ratio of 15.29 to 1.

The transmission assembly consists of a top case, support and a lower case. The lower case assembly contains the input pinion, bevel gear and shaft arrangement sun gear, planetary gear train, accessory gear drive, oil pump, oil filter housing, rotor RPM monopole pickup and a magnetic chip detector. Components attached to the transmission are the mast assembly, swashplate, main driveshaft, and hydraulic pump.



The transmission assembly is attached to the helicopter roof by a pylon installation. The pylon installation uses two side beams, four elastomeric corner mounts, and two fore/aft restraint springs.

Elastomeric corner mounts are used to minimize vibrations being transferred into the airframe. The transmission is equipped with chip detectors and a temperature switch to provide an indication of a related system malfunction.

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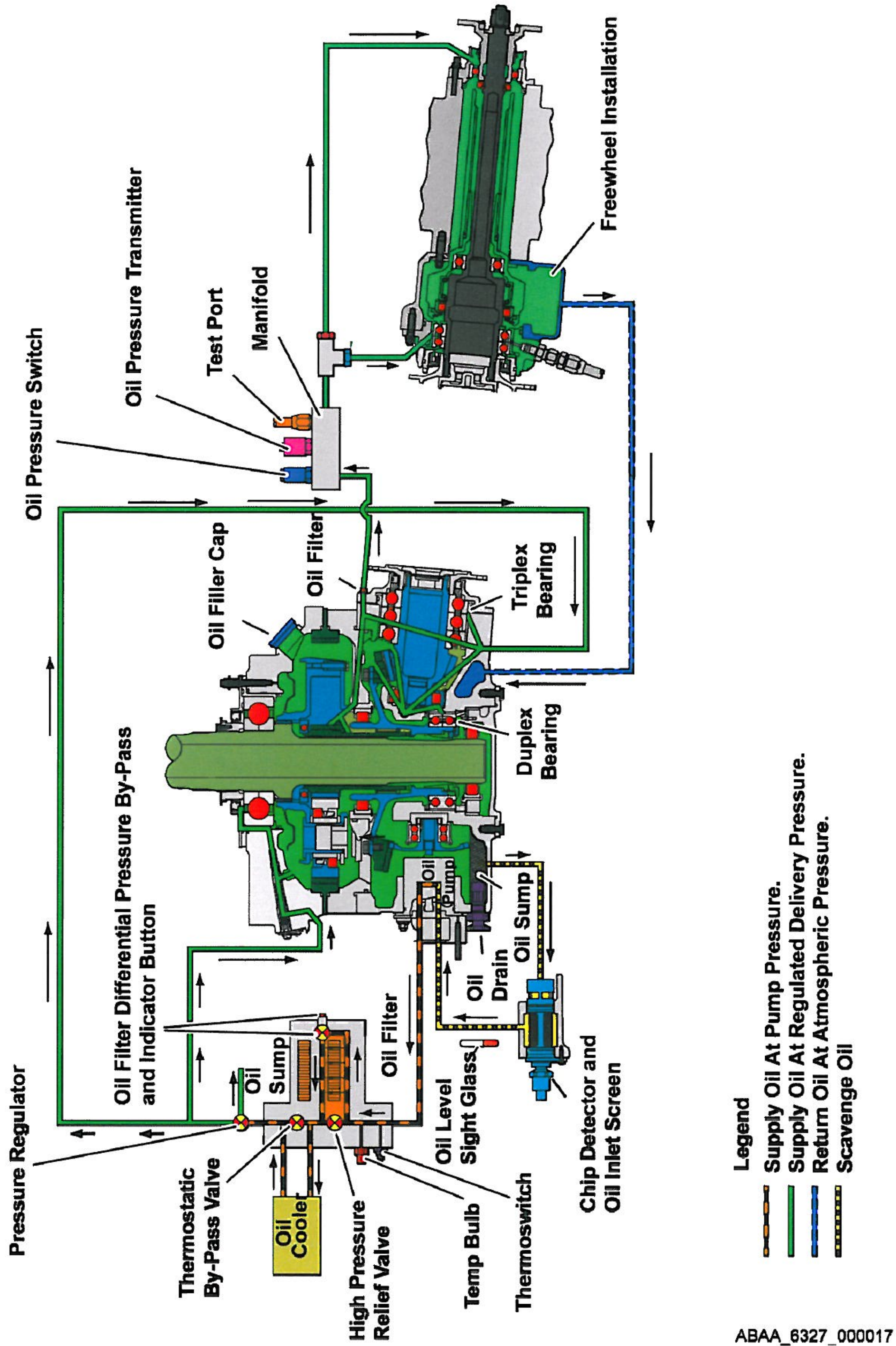
# DRIVETRAIN AND ROTORS

## Transmission Oil System

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# DRIVETRAIN AND ROTORS

**Transmission Oil System (Cont)**

The transmission oil system provides lubrication to the transmission, mast and freewheeling unit assembly.

An oil level sight glass is located on the right side of the transmission lower case and may be viewed through a cutout in the air induction cowling.

A non-vented filler cap is located on the right side of the top case to service the transmission oil system.

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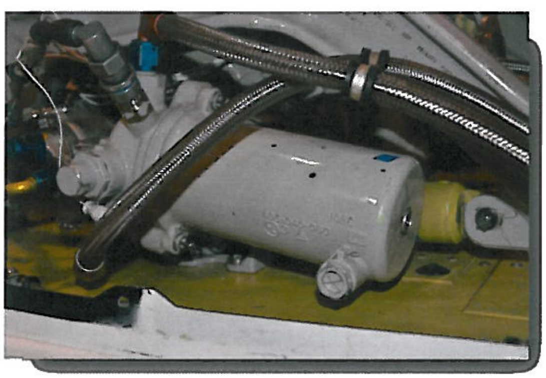
The transmission accessory gear drives the transmission oil pump. The pump output is between 6.0 to 6.7 gallons per minute. The pressure is regulated by the transmission oil pressure regulator valve and is set at approximately 52 PSI. The pump scavenges oil from the lower case sump through a wire screen and the lower chip detector. Oil is then directed to the transmission mounted oil manifold and the filter element.



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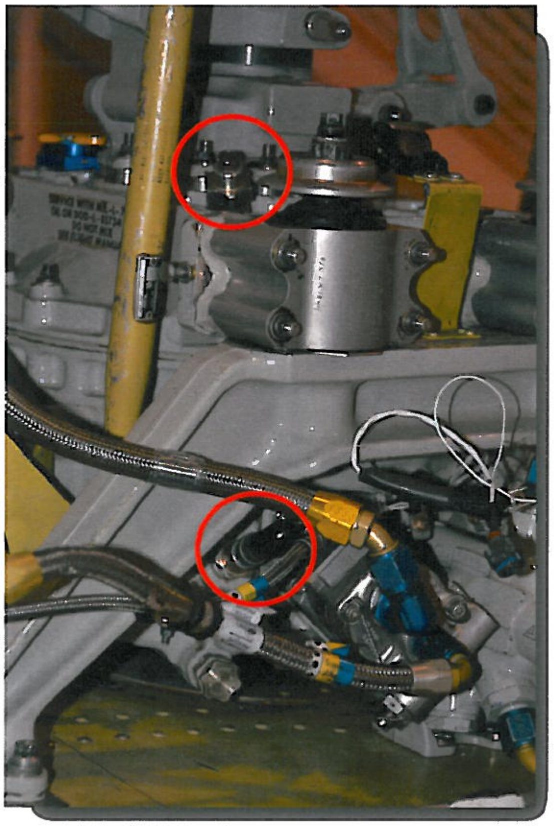
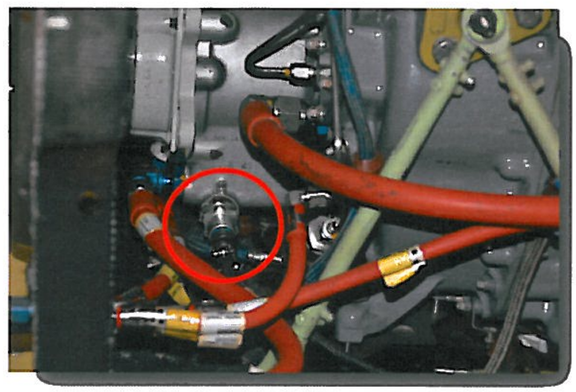
To ensure oil flow is not restricted, the filter incorporates an impending bypass indicator button on the end of the filter housing which extends at 14± 2 PSID. The filter bypass valve opens and the oil totally bypasses the filter element at approximately 17 PSID.

# DRIVETRAIN AND ROTORS

## Transmission Oil System (Cont)



Transmission oil is monitored for metallic contamination by three magnetic chip detectors. Two of these chip detectors are located on the left side of the transmission case, one upper and one lower. The third chip detector is found on the freewheeling unit. The chip detector consists of a self-locking bayonet probe with a permanent magnet at the end. Free ferrous metal particles in the oil are attracted to the magnet. When sufficient metal is attracted to complete the circuit between pole and ground, the XMSN CHIP detector segment on the caution panel illuminate.



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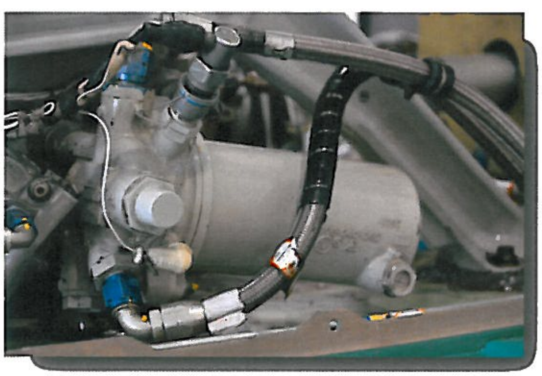

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The transmission oil filter manifold incorporates a thermostatic valve, which controls the flow of oil to the oil cooler. At an oil temperature below 150°F (66°C), the oil cooler is bypassed and oil is returned into the transmission. As the oil temperature increases above 150°F (66°C), oil is gradually directed to the oil cooler until at 178°F (81°C) all oil is directed through the cooler. A temperature bulb for the oil temperature indicator and a thermoswitch for the XMSN OIL TEMP light are also located on the transmission oil filter manifold.

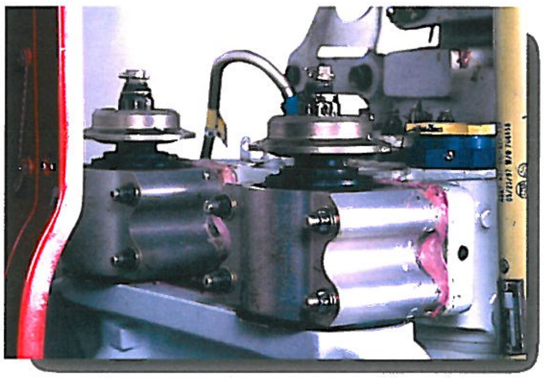
After the oil exits the cooler (or bypasses the cooler through the thermostatic by-pass valve), it is directed to the transmission to lubricate the various gears and bearings. The oil is then directed to the oil manifold located below the main driveshaft. A pressure transducer and oil pressure switch are mounted on the transmission oil manifold. The transducer provides signals to the oil pressure gage and the pressure switch controls the illumination of the XMSN OIL PRESS light.

## DRIVETRAIN AND ROTORS

### Transmission Oil System (Cont)

After leaving the oil manifold, oil flows through the forward firewall and into a "T" fitting. The "T" fitting is equipped with two restrictors. The restrictors decrease the oil pressure flow and direct oil into the freewheeling forward duplex bearing and the aft bearing. After lubricating the bearing in the aft housing, oil moves forward through the hollow engine output driveshaft to the freewheeling sprag clutch and bearing where it is collected in the forward freewheeling housing, where the electrical chip detector monitors for metal contamination. The oil is then returned to the main transmission lower case.

### Transmission Mounts



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The transmission is mounted to the helicopter roof by two pylon assemblies. The pylons are bolted to the aircraft roof in such a manner to give the transmission and mast a one degree left tilt to compensate for translating tendency and a five degree forward tilt to maintain level cabin attitude in forward flight. Four elastomeric corner mounts connect the transmission to the pylons. These mount assemblies are the primary vibration isolating components that provide for a vibration free ride throughout the helicopter's speed range.

At the base and rear of each pylon are two elastomeric transmission restraints spring assemblies which are used to maintain the proper alignment of the engine to transmission main drive shaft. On the forward end of each restraint are stop deck fittings that limit transmission movement.

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### Mast

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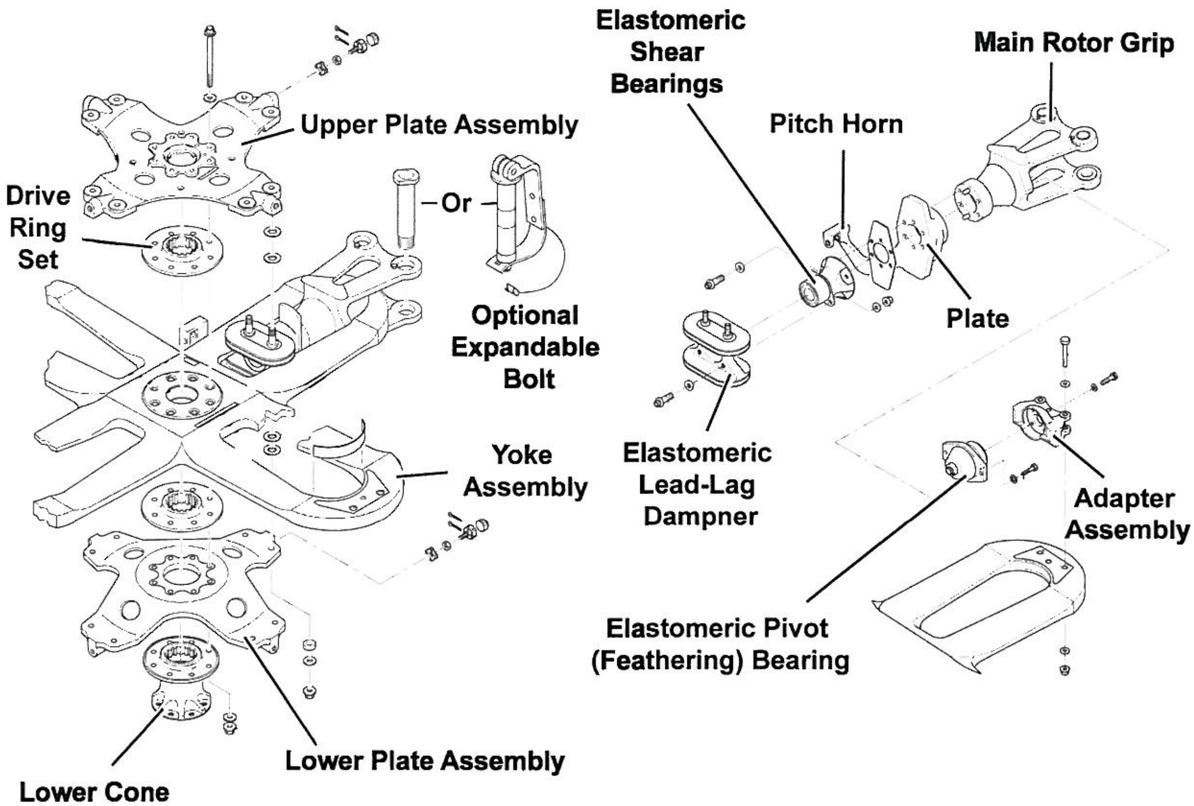
The mast is attached to the transmission by means of a mast locking plate and studs on the top of the transmission case. The mast is a hollow steel shaft that transmits rotational energy from the transmission to the rotor system. It incorporates three sets of splines and two threaded areas into its design. The upper spline is utilized to position and attach the main rotor. The swashplate drive splines are designed to receive a collar set that connects and provides drive to the swashplate drive links. The lower splines are the drive splines where the rotational energy from the transmission is transmitted to the mast from the planetary section. The upper portion of the mast is threaded to receive the main rotor retaining nut while the lower threads are used to secure the mast bearing retaining nut.

# DRIVETRAIN AND ROTORS

## Main Rotor

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ABAA_6220_000011

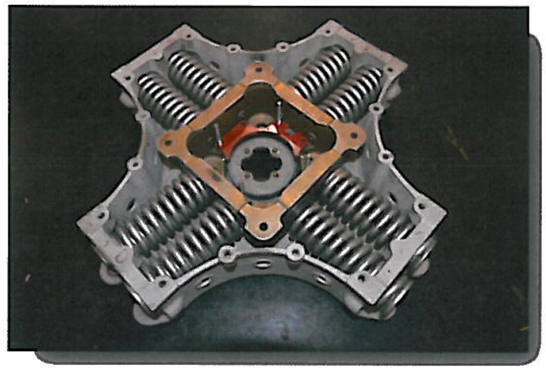
The main rotor is a 35 foot diameter, soft - in - plane flex beam (flapping flexure) type yoke/hub with four interchangeable blades. Elastomeric technology is incorporated and allows for blade movement. The blades and yoke are all composite. The rotor is designed to rotate at 413 RPM at 100% $N_R$.

# DRIVETRAIN AND ROTORS

## Main Rotor Hub

The main rotor hub contains a glass/epoxy composite yoke that acts as a flapping flexure. A flapping stop and a droop stop located at the inboard portion of each spindle protect and limit the composite yoke from excessive flexing. Cyclic centering is incorporated into this aircraft to reduce stress on the yoke and mast when the aircraft is on the ground (weight on gear). The feathering bearings and lead lag dampers are elastomeric elements that require no scheduled maintenance and have benign failure modes. Blade flapping is accomplished by a virtual hinge in the yoke (flapping flexure). The hub plate, pitch horns, and spindles are made of aluminum forgings for strength and reduced weight.

## Frahm Damper



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A Frahm damper is installed atop the main rotor head to dampen four per rev vibrations from the main rotor and hub.

A formed steel block is held in position (floats independent of the upper and lower housing) by 8 heavy gage springs. As the main rotor blade encounters in-flight convective turbulence (frequency), the Frahm damper steel block begins to vibrate at a similar frequency, canceling the vibrations prior to their descent into the main rotor shaft and the aircraft fuselage.

## Main Rotor System Cover

A convex fiberglass cover is aerodynamically shaped for drag and noise reduction. The cover is installed on top of the Frahm assembly to protect the Frahm assembly mechanism from any environmental factors.

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# DRIVETRAIN AND ROTORS

## Main Rotor Blades

The main rotor blades are asymmetrical for optimum rotor blade efficiency. The blades incorporate a -13° twist from hub to tip to provide a more equal distribution of lift along the length of the blade and to reduce blade stress. The main rotor blades are composite design consisting of three structural members; a fiberglass/epoxy spar, a Nomex honeycomb core, and a fiberglass/epoxy skin. The leading edge is a nickel plated stainless steel strip and is coated with conductive paint to protect the main rotor blade against lightning strikes and static charges. The blades are dynamically balanced and interchangeable.

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Product balance weights reduce inherent rotor vibrations during pitch changes.

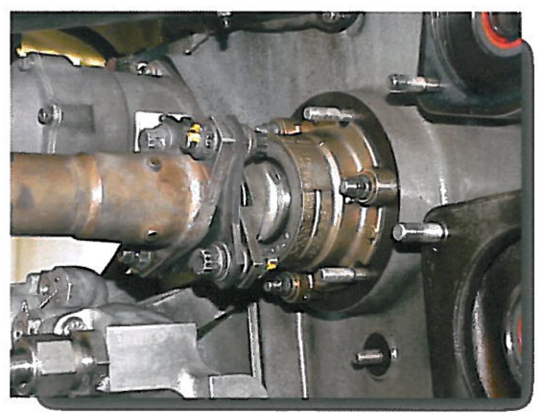
Tip weights increase rotor inertia for better autorotative characteristics.

Each main rotor blade weighs approximately 58 pounds.

## Tail Rotor Drivetrain

### Tail Rotor Drive Shaft

The tail rotor drive shaft is a strong, flexible segmented assembly that provides power to drive the tail rotor gearbox. The drive shaft consists of several segments. The most forward segment is a steel shaft mounted between the accessory gearbox and the aft firewall. The second shaft segment is a short steel shaft that drives the oil cooler blower.



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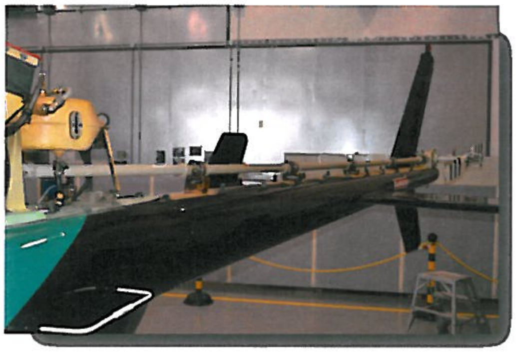
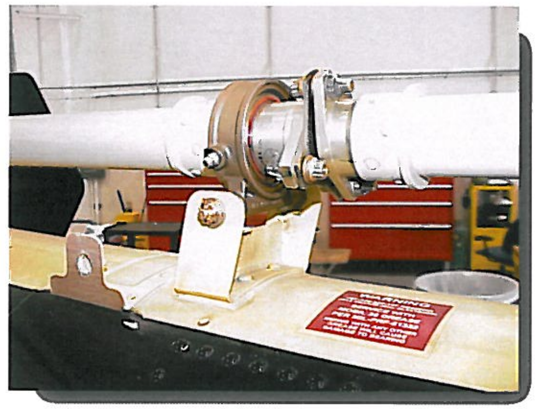
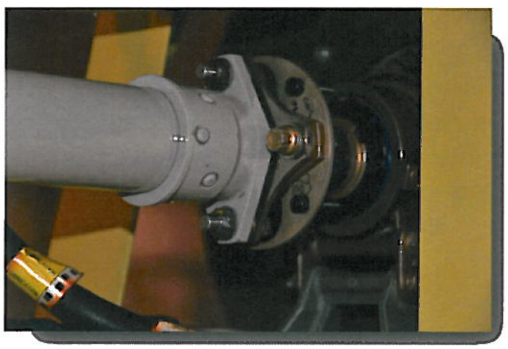
# DRIVETRAIN AND ROTORS

## Tail Rotor Drivetrain (Cont)







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Aft of the oil cooler blower are five equal length aluminum segmented shafts that extend along the top of the tailboom.

### Hanger Bearings

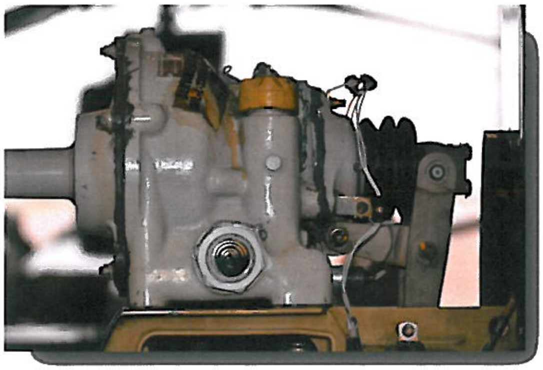
Seven hanger bearing assemblies are utilized to support the tail rotor driveshaft segments. These hanger bearings are grease lubricated and assist in maintaining the alignment of the driveshaft along the top of the tailboom.

### Thomas Couplings

Eight coupling disc packs (Thomas couplings) are used to connect the tail rotor drive shaft segments together. Each disc pack is made up of 9 to 12 steel plates. Each plate's circumference is cut in two places, parallel to the grain of the steel. When assembled each plate is turned 90° from the previous plate to alternate the grain of the steel plates. This increases the strength and flexibility of the disk pack. Verify only the grip portion (non-threaded section or shoulder) of the bolts are positioned to contact the disc pack.

# DRIVETRAIN AND ROTORS

## Tail Rotor Gearbox



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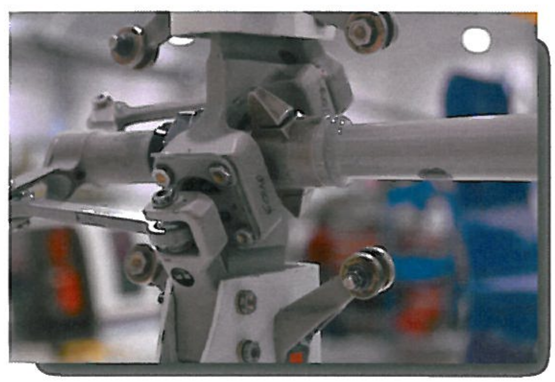
The tail rotor gearbox is located on the aft end of the tailboom. The gearbox drives the tail rotor. The direction of drive is changed 90° by means of two spiral bevel gears positioned at a 90° angle to each other. There is a speed reduction of 2.53:1 at the gearbox. The magnesium housing is attached to the tailboom by means of four studs and two alignment pins. The assembly includes a non-vented filler cap, vent line, oil level sight glass, and a combination magnetic chip detector/self-closing drain valve. The chip detector consists of a self-locking bayonet probe with a permanent magnet at the end. Free ferrous metal particles in the oil are attracted to the magnet. When sufficient metal is attracted to complete the circuit between pole and ground, the T/R CHIP detector segment on the caution panel illuminates. The valve automatically closes and prevents loss of oil when the chip detector is removed for inspection. The valve also serves as a drain plug when oil needs to be drained from the gearbox.

## Tail Rotor

### Static Stop and Yield Indicator

A static stop and yield indicator is installed on the tail rotor output shaft inboard of the yoke. This static stop and yield indicator provide the ability to visually determine if the tail rotor yoke has been stressed beyond design limits. This will be evident by deformation of the static stop yield indicators due to excessive contact with the yoke. If any deformation of either yield indicator is evident, maintenance action may be required.



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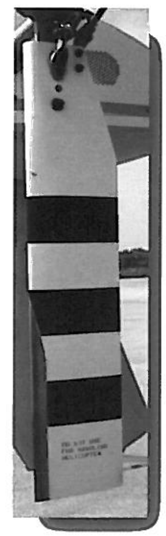
# DRIVETRAIN AND ROTORS

### Tail Rotor Hub and Blades

The tail rotor is a two bladed teetering rotor with a 5.42 feet diameter. A stainless steel yoke is attached to the output shaft by means of a splined trunnion. The trunnion provides a flapping axis through the use of elastomeric bearings and compensates for dissymmetry of lift. There are two pitch change bearings per blade on the pitch axis which permits the pitch angle of the blades to change. Span-wise balance is accomplished by using washers on the blade bolts at the yoke. Chord-wise balance is accomplished by using weights and washers on the trunnion bearing housing restraining bolts.

The tail rotor yoke fits into the root of the tail rotor blade. Two bolts attach the blade and yoke together. A pitch horn is attached from each blade root to the crosshead to provide angle of attack changes.

The all composite tail rotor blade is made of a fiberglass/epoxy spar and skin assemblies, Nomex honeycomb core, and a nickel plated stainless steel leading edge abrasion strip. The blades are painted with conductive paint for lightning protection and conductivity.



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## Servicing

TRANSMISSION and TAIL ROTOR GEARBOX

Oils conforming to the following specifications are approved for use in transmission and tail rotor gearbox.

DOD-PRF-85734 is limited to ambient temperatures above –40°C (-40°F).

MIL-PRF-7808 is limited to ambient temperatures below –18°C (-0°F).

NOTE

It is recommended that DOD-PRF-85734 oil be used in the transmission and tail rotor gearbox to maximum extent allowed by temperature limitations. Refer to BHT–407–FM-1 for transmission and tail rotor gearbox oil limitations.

Sight glasses are provides to determine quantity of oil in transmission and tail rotor gearbox.

Transmission capacity 5.0 U.S. quarts (4.7 liters)

Tail Rotor Gearbox capacity 0.33 U.S. quarts (0.31 liters)

# FLIGHT CONTROLS AND HYDRAULICS

THE BELL TRAINING ACADEMY
Est. 1947

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## Flight Controls and Hydraulics

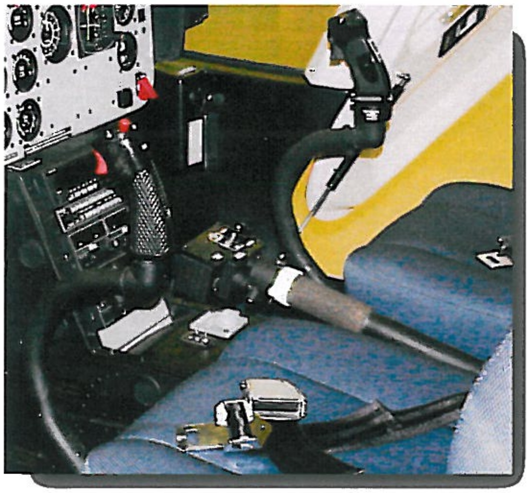


Bell Helicopter
A Textron Company

# FLIGHT CONTROLS AND HYDRAULICS

## Main Rotor Controls

The main rotor and tail rotor flight control systems consist of cyclic, collective, anti-torque pedals, and associated mechanical linkage. These controls regulate the helicopter attitude, altitude, and direction of flight. All controls are hydraulically boosted to reduce pilot effort and to counteract control feedback forces.



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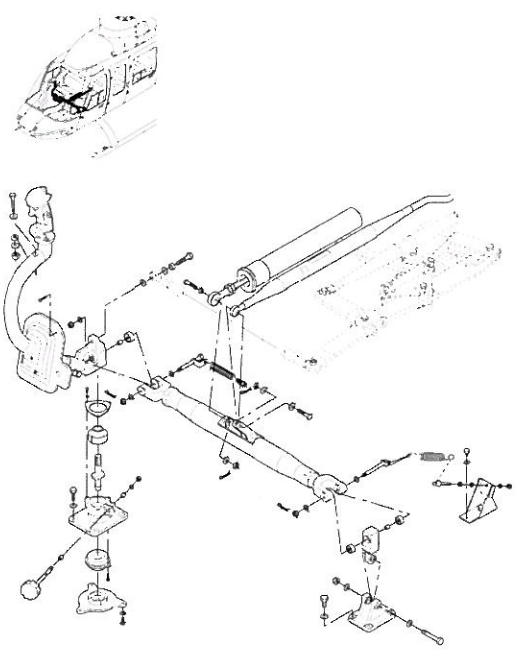
The main rotor cyclic and collective controls regulate pitch/roll attitude and thrust. Push-pull tubes transmit control inputs from the cyclic and collective to their respective hydraulic servo actuator. The actuators connect to the cyclic and collective levers and control the elevation and tilt of the swashplate.

The swashplate converts linear control inputs from the cockpit to the rotating controls and allows cyclic and collective pitch input to the rotor systems.

Springs installed in parallel to the cyclic and collective push-pull tubes on the cabin roof and under the copilot's seat assist the pilot with the increased control feedback forces if hydraulic pressure to the servo actuators is lost.

## Cyclic



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The cyclic is mounted under the pilot's seat and protrudes from the crew seat support of the seat. Fore and aft cyclic input connects through push-pull tubes to the fore and aft hydraulic servo actuator. This input also feeds through a cam assembly that automatically adds lateral cyclic input proportional to the fore and aft cyclic movement. A force gradient cartridge in line with the cam input permits cyclic movement if the cam assembly becomes jammed.

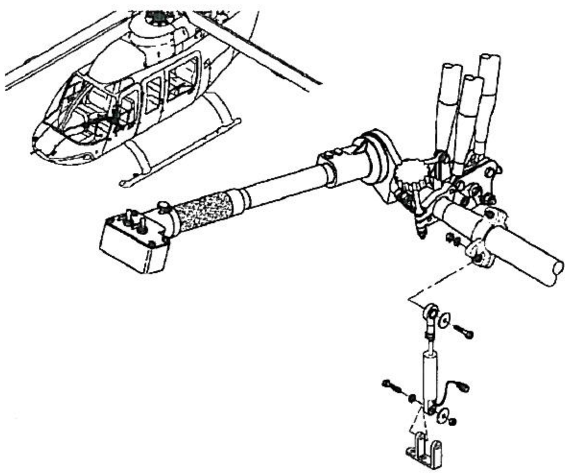
The lateral cyclic input connects through push-pull tubes to the lateral hydraulic servo actuator. The hydraulic servo actuators operate bellcranks and push-pull tubes that tilt the swashplate non-rotating ring. The swashplate rotating ring will tilt in unison and actuate the pitch links that control the plane of rotation of the main rotor.

The cyclic grip contains a two-position intercommunication/radio transmit switch and a cargo hook release switch. The grip also has additional locations for optional kits.

An adjustable friction control knob at the base of the cyclic stick allows the pilot to set the desired control friction for flight.

# FLIGHT CONTROLS AND HYDRAULICS

## Collective



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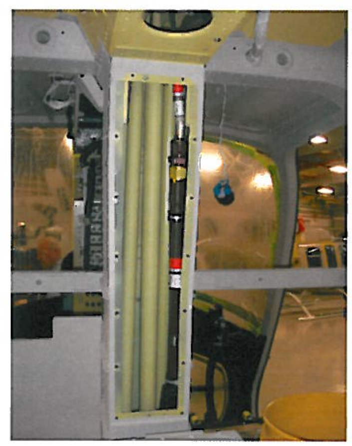
The collective is mounted between the pilot and co-pilot seats. The collective connects to the collective hydraulic servo actuator through push-pull tubes. This servo actuator connects to the collective lever. The collective lever raises and lowers the swashplate ball-sleeve assembly and cyclic levers to change main rotor blade angle without affecting the cyclic path.

An adjustable collective friction knob near the base of the collective between the pilot and co-pilot's seats allows the pilot to set the desired collective friction for flight.

A twist grip throttle for the engine is mounted on the collective stick. A mechanical idle release push button is in front of the twist grip throttle. A switchbox located on the forward end of the collective provides a mount for the engine start switch, landing light switch and accessory switches.

## Control Column

The flight controls route beneath the pilot and copilot's seats aft to the vertical control column then up to the cabin roof. This control column also serves as a primary cabin support structure. Access panels on the aft side of the column, the bottom of the aircraft, and seat panels allow inspection of control components and maintenance accessibility. The cyclic controls are mixed with collective control through the mixing levers located at the base of the control column.



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# FLIGHT CONTROLS AND HYDRAULICS

## Swashplate Assembly


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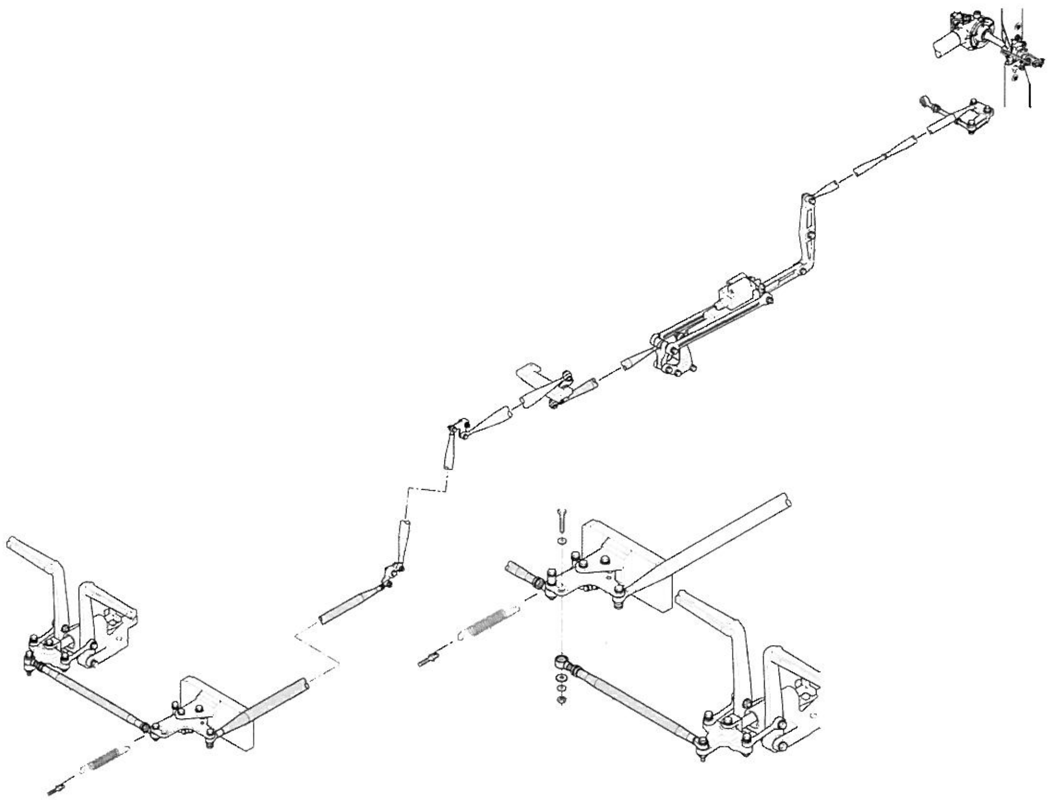
The main rotor controls consist of the swashplate and support assembly, sleeve drive link, and pitch links. The swashplate and support assembly transfers cyclic control motions from the non-rotating to the rotating control system. The swashplate and support encircle the mast directly above the transmission. The swashplate mounts on a universal support (pivot sleeve or uniball) that permits it to move in any direction. Movement of the cyclic results in a corresponding tilt of the swashplate and the main rotor.

The collective lever and link assembly are mounted to the swashplate support assembly and transfers collective inputs to the lower swashplate. Movement of the collective pitch lever actuates the sleeve assembly that in turn raises or lowers the swashplate and transmits collective input to the main rotor. The swashplate drive assembly consists of a collar set, two idler links, and idler levers. The collar set attaches to the mast and the idler links attach to the outer ring of the swashplate. This connects the upper swashplate to the mast, causing it to rotate with the mast.

The pitch link assembly connects the pitch horn on the blade grips to the swashplate that transmits control input from both the collective and cyclic controls.

# FLIGHT CONTROLS AND HYDRAULICS

## Tail Rotor Controls



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The tail rotor controls include the control pedal assembly, pedal adjuster, control tubes, bellcranks, hydraulic actuator, and a pitch control mechanism mounted through the tail rotor gearbox shaft. Moving the pedals causes pitch change in the tail rotor blades to offset main rotor torque and control the directional heading of the helicopter.

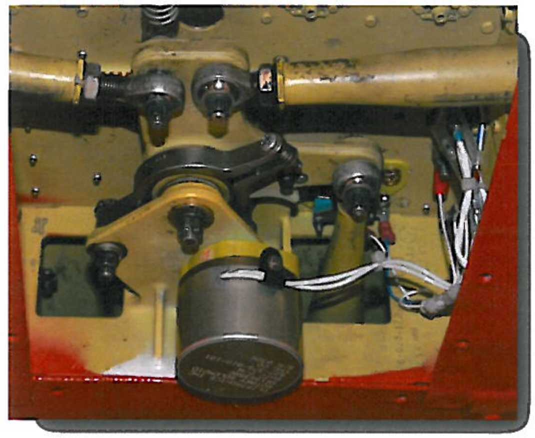
The tail rotor pedals connect to a pedal adjuster that provides for manual adjustment of the pedal position according to pilot needs.

Tail rotor pitch control is accomplished by means of a bellcrank, rod, and lever assembly mounted on the tail rotor gearbox that actuates a control tube through the hollow output drive shaft to the crosshead and pitch links.

# FLIGHT CONTROLS AND HYDRAULICS

## Pedal Stop Solenoid

This solenoid extends the pedal stop restrictor into the left pedal range of travel. The restrictor allows full left pedal at less than 50 knots and limits left pedal travel when airspeed increases to 55 knots and above.



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## Dual Controls

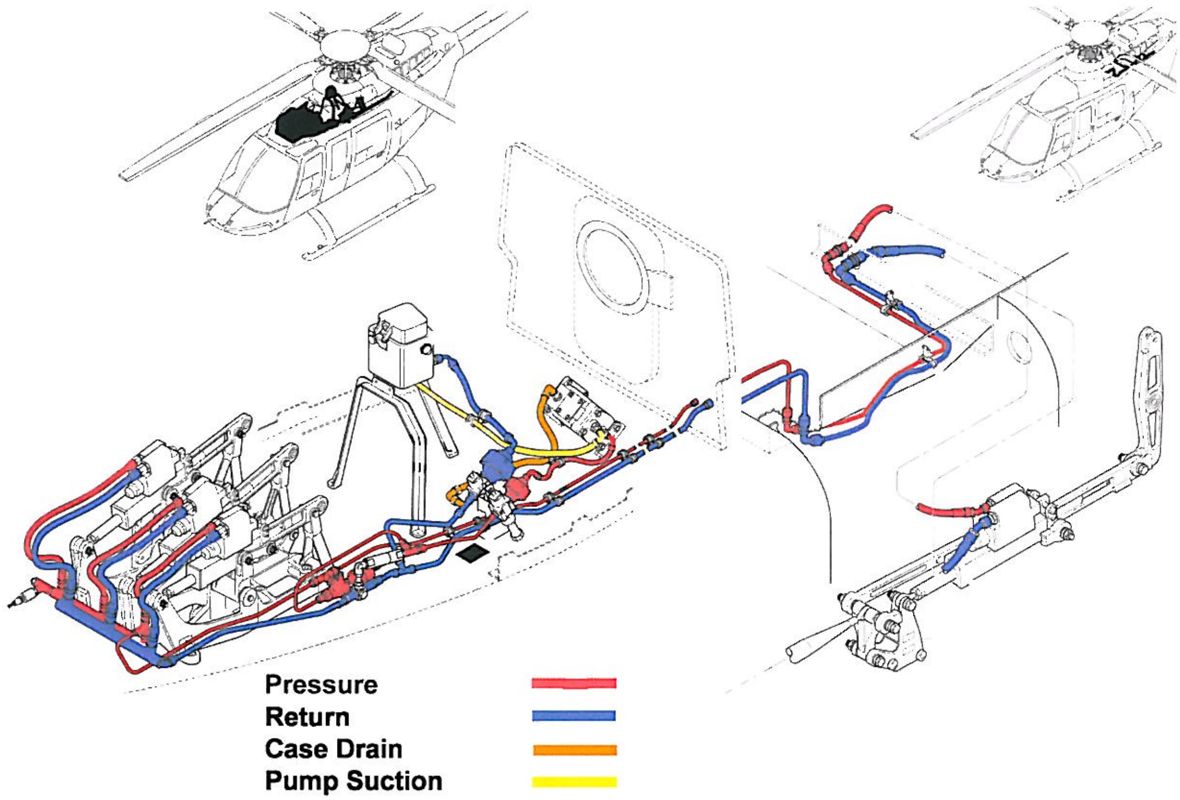
Installation of dual controls provides a collective, cyclic, and tail rotor control pedal assembly for the copilot. The copilot's controls connect to the pilot's by means of an elbow fitting for the collective, lever assembly for the cyclic and a control tube for the tail rotor pedals and electrical wiring.

Quick disconnects are provided for the collective and cyclic.

The copilot's controls do not provide electrical cargo release, flight idle stop, throttle bezel, markings, starter switch, or landing light controls.

# FLIGHT CONTROLS AND HYDRAULICS

## Hydraulic System General

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### Hydraulic System

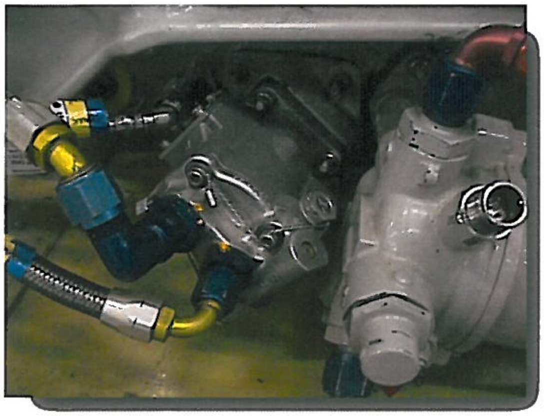
The hydraulic system provides pressurized fluid to operate the cyclic, collective, and tail rotor flight control servo actuators. The hydraulic system switch electrically controls the system. When the hydraulic switch is ON, (the solenoid valve de-energized (open)), pressurized hydraulic fluid flows to the four servo actuators. When the switch is OFF (solenoid valve energized closed), pressurized hydraulic fluid flows to the reservoir and bypasses the four servo actuators. In case of a total electrical failure, the system is fail-safe ON.

The hydraulic reservoir and cover are constructed from magnesium alloy. The reservoir is mounted on a brace and support forward of the main transmission.

The hydraulic system has no specific cooling system. Heat dissipates from the hoses and lines as air circulates beneath the forward cowling.

# FLIGHT CONTROLS AND HYDRAULICS

## Hydraulic System Components



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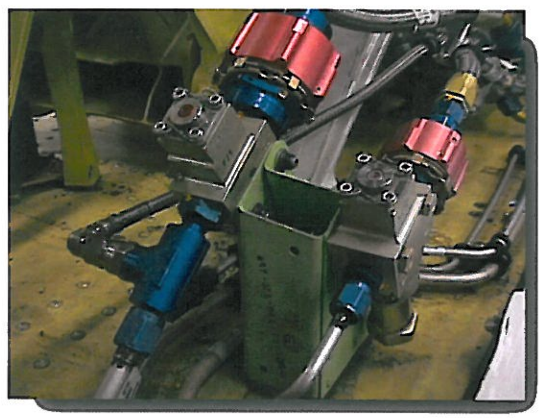
The hydraulic pump is mounted on the transmission and driven by the transmission accessory drive.

The pump is a constant pressure, variable delivery, self-lubricated type designed to operate continuously and provide a rated discharge pressure of 1,000 ± 25 PSI.

Two hydraulic filters are on a bracket on the forward left side of the cabin roof near the main transmission. An indicator on each filter assembly indicates an impending clogged filter. The indicator is a red button on the filter assembly housing that pops up when differential pressure across the filter is 70±10 PSID. To prevent inaccurate indications of bypass, the indicator will not work when the hydraulic fluid temperature is less than 35° F (2°C). If the hydraulic fluid temperature is more than 35° F (2°C), the indicator gives the correct indication of clogging, even if the ambient (OAT) temperature is below 35° F.



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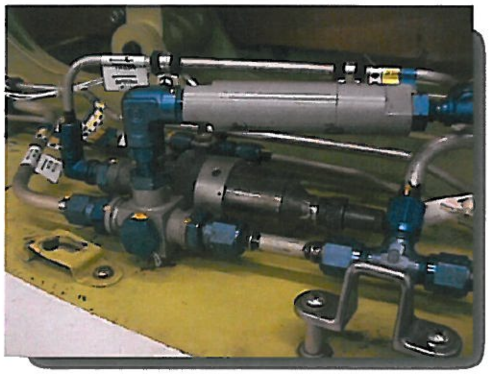
One filter is in the pressure line and the other in the return line. The pressure filter assembly does not have bypass capability. The return line filter does incorporate a bypass valve to prevent overpressurization of the return system. If the filter element in the return line becomes clogged, the bypass valve opens to allow fluid to bypass the filter and return to the reservoir.

ABAA_2910_000013

The hydraulic solenoid valve is mounted on the cabin roof on the left side of the hydraulic servos. The valve is electrically closed and is controlled by the hydraulic system switch on the overhead panel. The electrical circuit is protected by a circuit breaker on the overhead console. When the switch is turned off, hydraulic fluid bypasses the servos and returns to the hydraulic reservoir.

## Hydraulic System Components (Cont)



ABAA_2910_000013

The relief valve is installed forward of the transmission and on the left side of the cyclic and collective servo actuator support. During normal operations, the relief valve is closed.

If system pressure increases to 1,075 to 1,375 PSI, the valve will open to protect the system from damage by returning excess hydraulic fluid to the pump.

When system pressure returns to normal, the relief valve will close.

The cyclic and collective servo actuator support is on the cabin roof forward of the transmission. It serves as a mount for the three servo actuators and associated bell-cranks. The collective servo actuator is in the center position, and the two cyclic servo actuators are on the outboard positions.

Springs are installed in parallel to the cyclic and collective push pull tubes on the cabin roof. If hydraulic pressure is lost to the servo actuators, these springs assist the pilot with the increased control feedback forces.

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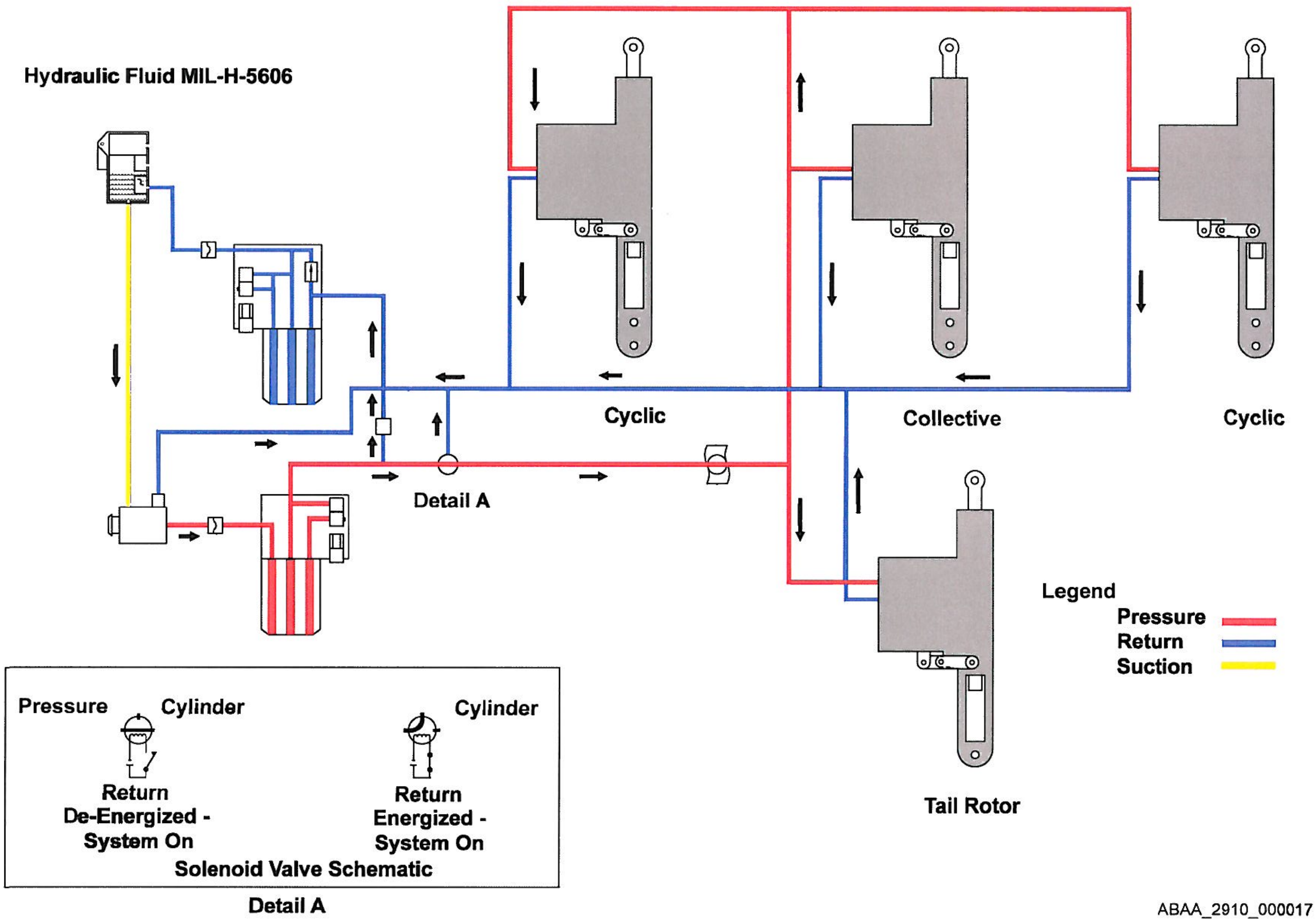
ABAA_2910_000015

The tail rotor servo actuator is in the intermediate section of the fuselage above the baggage compartment.


Hydraulic Fluid MIL-H-5606
Detail A
Cyclic
Collective
Cyclic
Tail Rotor
Legend
Pressure
Return
Suction
Pressure
Cylinder
Return
De-Energized -
System On
Cylinder
Return
Energized -
System On
Solenoid Valve Schematic
Detail A
ABAA_2910_000017

# FLIGHT CONTROLS AND HYDRAULICS

## Hydraulic Systems Check

(Modified Excerpt from M407 Flight Manual)

PRELIMINARY HYDRAULIC SYSTEMS CHECK

**NOTE**

Uncommanded control movement or motoring (with hydraulic system off) may indicate hydraulic system malfunction.

HYDRAULIC SYSTEM switch — OFF, caution light illuminated, switch — ON.

HYDRAULIC SYSTEMS CHECK

**NOTE**

HYDRAULIC SYSTEMS CHECK is to determine proper operation of hydraulic actuators for each flight control system. If abnormal forces, unequal forces, control binding, or motoring are encountered, it may be an indication of a malfunctioning flight control actuator.

Collective — Full down

ROTOR — 100% RPM

HYDRAULIC SYSTEM switch — OFF, caution light illuminated

Cyclic — Centered

Cyclic control — Check normal operation by moving cyclic forward and aft, then left and right (approximately 1 inch), center cyclic.

Collective — Check for normal operation by increasing collective slightly (1 to 2 inches).

Repeat 2 to 3 times as required. Return to full down position.

Pedals — Check normal operation by displacing pedals slightly (1 inch).

HYDRAULIC SYSTEM switch — ON, caution light extinguished

# FLIGHT CONTROLS AND HYDRAULICS

## Servicing

Hydraulic fluids conforming to MIL-PRF-5606 (NATO H-515) are approved for use in hydraulic flight control system.

| | |
|---|---|
| Hydraulics reservoir capacity: | 1.3 U.S. pints (0.5 liter). |
| Total hydraulic system capacity | 1.68 Quarts |

The hydraulic reservoir is located on top of the fuselage, forward of the transmission under the transmission faring. A sight glass is provided to determine the quantity of hydraulic fluid in the reservoir.

Service the hydraulic system as follows:

1. Open and support the forward fairing.
2. Remove the transmission fairing.
3. Remove the cap and fill the reservoir until the sight glass is full of hydraulic fluid.
4. Secure the cap and fairings.

# AIRFRAME

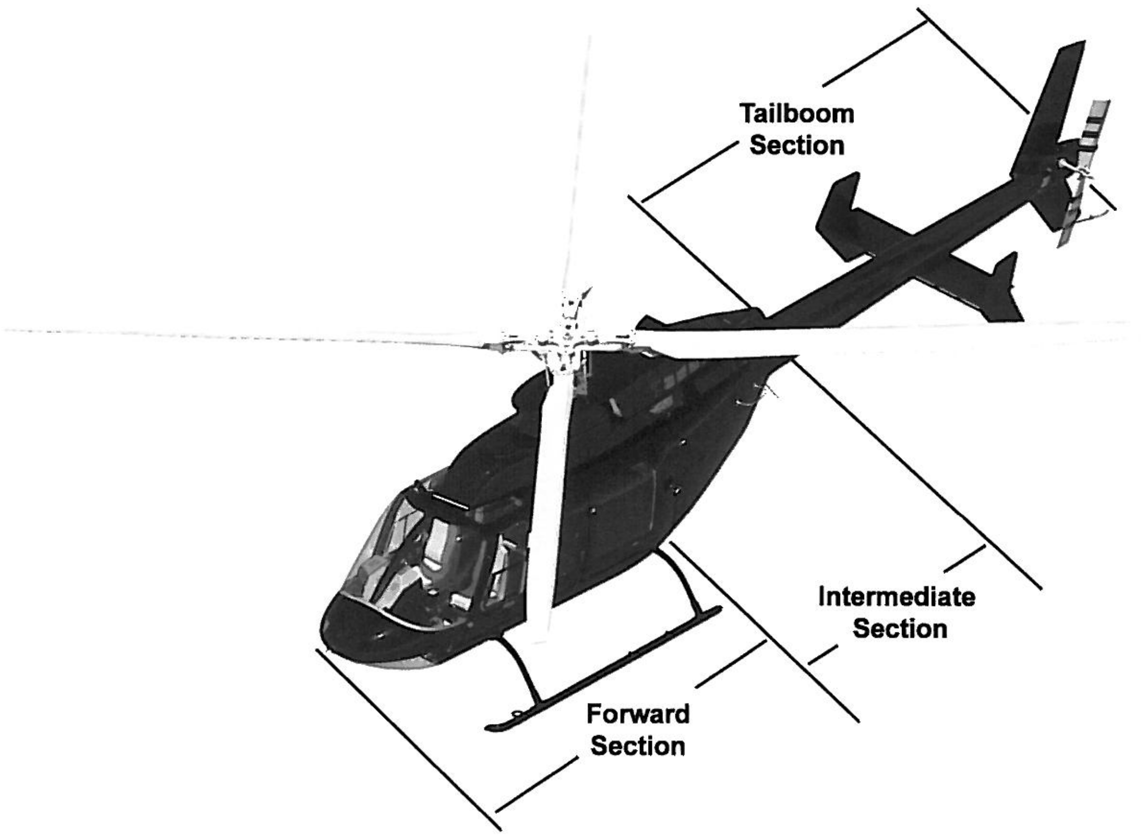
THE BELL TRAINING ACADEMY
Est. 1947

# Table of Contents

## Airframe

# Airframe

## General

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The fuselage consists of three main sections: the Forward Section, which extends from the cabin nose to the bulkhead aft of the passenger compartment, the Intermediate Section, which extends from the bulkhead aft of the passenger compartment to the tailboom, and the Tailboom Section.

# Airframe

## Aircraft Materials


ABAA_5300_000009

# Airframe

## Forward Section

The forward section is constructed of aluminum honeycomb and carbon graphite structures and provides the major load carrying elements of the forward cabin. This section provides for pilot and passenger seating, up to seven people, instrumentation, electrical controls, hydraulics, fuel cell locations, and pylon and transmission support.

Each side of the aircraft is assembled with a tapered fairing to reduce aerodynamic drag. The fairing begins at the forward door hinge (no additional width) and continues back to the rear bulkhead (3½ inches per side).

The 407 is equipped with five entrance/exit doors. Crew and passenger doors are located on both sides of the fuselage. Crew and passenger doors are equipped with an interior and exterior door handles. Door locks are installed in the crew, litter, passenger, and baggage compartment doors to provide security to the cockpit, cabin, and baggage areas. Locks are keyed alike, except for the baggage compartment door lock.

ABAA_5310_000004

Aft of the crew doors, two passenger doors are installed on each side of the fuselage to provide access to the cabin area. Each is equipped with the same latch assembly, which may be operated from either side of the door, and a lock installed in the exterior door handle. The left passenger door is hinged on the litter door, so the two may be opened together and allow convenient access for passengers seated in the aft facing seats to enter or exit, patients on a litter (stretcher) to be loaded for ambulance transport, and large cargo to be loaded in the main cabin area. The aircraft may be flown without the litter door installed; however, the aft left door must also be removed.

The 407 seats are covered with flame resistant fabric, leather, or other customer requested coverings. The seats are easily removed, which then allows for 85 cubic feet of cargo space. Designed into the forward section of the aircraft are the two fuel cells, which make up the seat supports and mounting for passenger seating. A parcel shelf is located behind the forward facing passenger seats.

Two cabin roof windows, made of dark tinted acrylic plastic, are located over the forward crew seats and allow overhead viewing. The crew and passenger doors contain windows of light gray tinted acrylic plastic. The crew and passenger doors have a sliding window for ventilation. This window opening is adjustable by using the handle to slide the window in the window track. The handle also prevents the window from sliding out of the track.

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# Airframe

### Forward Section (cont)

The windshields are fabricated of tinted acrylic plastic and are supported by formed aluminum alloy sections. The two lower windows are made of tinted acrylic plastic, located in the lower cabin nose section.

A variety of hinged doors and panels provide access for aircraft inspection, servicing, and miscellaneous storage. The battery access door, located on the nose of the helicopter, provides access to the battery and hourmeter. Two fixed landing lights are mounted below the battery and are controlled by a switch located on the pilot collective. Two position lights are located below the crew seats, forward of the forward crosstube. The main rotor flight controls, hydraulics, transmission, landing gear, and optional cargo hook are also mounted to the forward section.

## Intermediate Section

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The intermediate section consists of an aluminum/composite semi-monocoque construction with bulkheads for strength. This section provides a deck for engine installation, compartment under the engine deck for heater and electrical equipment, baggage compartment located below the electrical equipment, and attachment point for the tailboom.

The engine pan and the forward and aft firewalls are constructed of titanium and fitted to form the engine compartment. The engine pan, located below the engine, acts as both a drip pan and firewall. The pan is designed to provide sufficient clearance to allow for the removal of accessories without removing the engine.

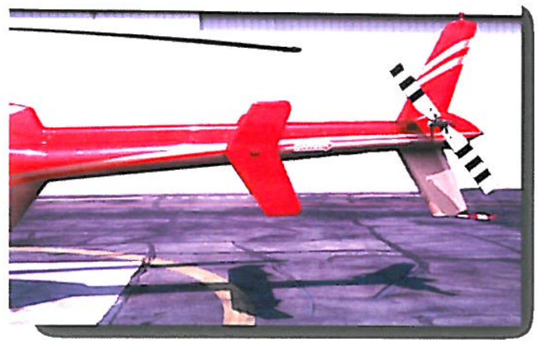
The equipment compartment is located below the engine, behind the parcel shelf, and above the baggage compartment. This area contains aircraft electrical equipment, fuel signal conditioner and has space for optional equipment.

The baggage compartment is located below the equipment compartment, and has a volume capacity of 16 cubic feet with a maximum floor weight of 250 pounds. An optional auxiliary (aux) fuel tank may be installed in the baggage compartment. Since the aux tank is mounted to the forward bulkhead and adds no weight to the baggage compartment floor, all 250 pounds may be carried provided aircraft weight and balance and limitations are maintained. The baggage door, located on the left side of the fuselage, is hinged at the forward end, opens the full width and height of this storage area, and is secured by means of two push-button latches and a keyed lock. In-flight door security is monitored in the cockpit by a Baggage Door caution light. The compartment has ten tiedown loops to secure cargo.

# Airframe

## Tailboom Section

As a full monocoque design, the tailboom obtains its strength through the skin and internal bracing. Four bolts attach the tailboom to the intermediate fuselage. A tail rotor driveshaft, tail rotor gearbox, horizontal stabilizer, two auxiliary fins, and vertical fin are mounted to the tailboom.



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Attached to the top along the length of the tailboom are brackets that provide a mount for the hanger bearings that support the tail rotor driveshaft. Mounted to the aft end of the tailboom is the 90° tail rotor gearbox.

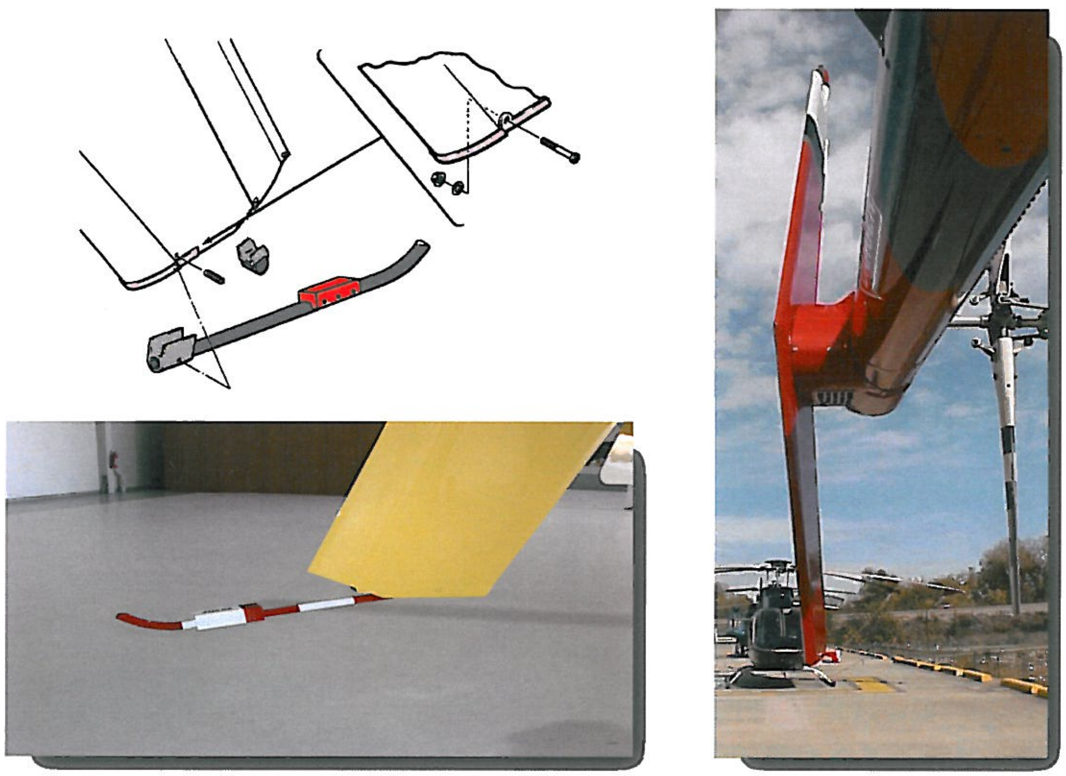
Mounted through the tailboom is the horizontal stabilizer. This stabilizer is a one-piece aluminum honeycomb inverted airfoil. This device provides a downward resultant lift on the tailboom to maintain the cabin in a nearly level attitude throughout all cruise airspeeds. A Gurney flap on the right and left trailing edge reduces mast stress at high speeds. A leading edge slat is installed to improve pitch stability during climbs

Mounted to each end of the horizontal stabilizer are end plates/auxiliary fins. The leading edges of the fins are both offset 5 degrees from the helicopter centerline. This improves the dihedral (roll) stability of the aircraft in forward flight. A position light is mounted on the outboard end of each end plates/auxiliary fins.

# Airframe

## Vertical Fin

The vertical fin, composed primarily of aluminum and honeycomb construction, provides directional (yaw) stability. The leading edge is canted outboard 9 degrees to reduce the required amount of tail rotor thrust during forward flight at cruise speed. A flicker vertigo tab and anti-collision light are mounted on top of the fin. A tubular tailskid is installed on the lower end to prevent the tail rotor from making contact with the ground in the event of a tail low landing.



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# Airframe

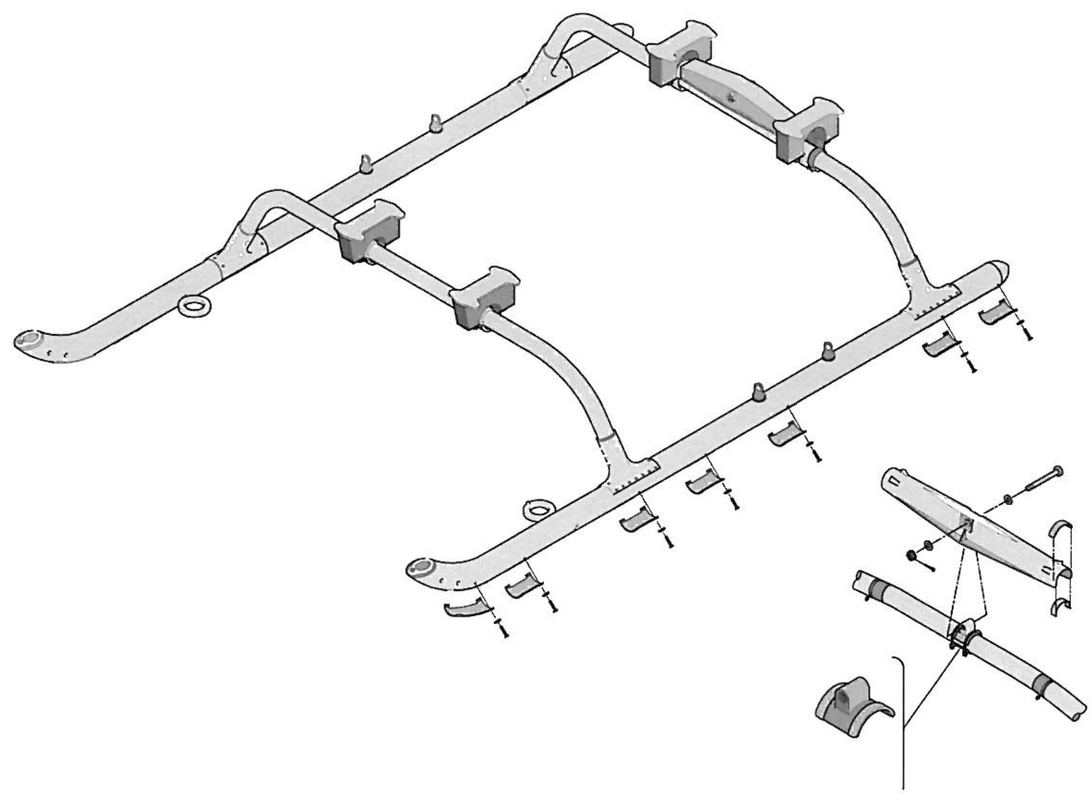
## Landing Gear

The skid type landing gear consists of two skids attached to arched cross tubes that are secured to the fuselage by means of two strap assemblies and a pivot beam. Each skid tube is fitted with a forward end step, a tow fitting, two saddles with sockets for the crosstube, seven separate shoes along the bottom, a rear cap, and two eyebolt fittings for mounting of ground handling wheels.

A WEIGHT ON GEAR switch is mounted to the lower fuselage and is activated by the flexing of the forward cross tube, through a bracket mounted on the forward crosstube. Through this switch, three systems are energized; cyclic centering, hour meter, and flight time. When the aircraft is on the ground, the cyclic centering system and related caution light are energized. The cyclic is positioned to extinguish the cyclic centering caution light. This cyclic position minimizes mast stress while the aircraft is on the ground. The hourmeter and flight time are disabled when the aircraft is on the ground. When the aircraft is airborne, the cyclic centering system is disabled and the hourmeter and flight time are enabled.

The aft crosstube uses a pivot beam with the attaching hardware that allows limited lateral movement of the airframe while the helicopter is on the ground. This minimizes the effect of ground resonance. During preflight, the fuselage must rock freely on the beam. The landing gear can be equipped with fairings to give the crosstube less air resistance.

It is recommended that no components be attached to the landing gear assembly except as designated by the manufacturer. Doing so could lead to failure of the crosstube.



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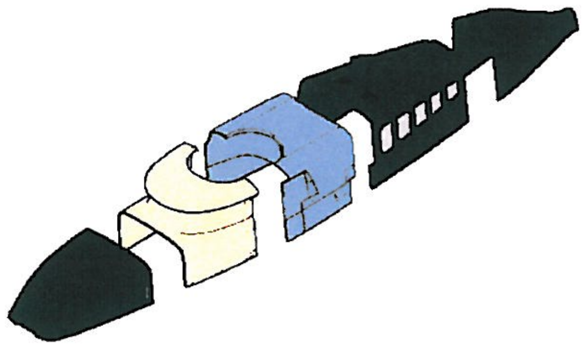
# Airframe

## Cowlings

Cowlings and fairings enclose the various roof and tailboom mounted assemblies. Cowlings provide for inspection of interior areas through the use of hinge mounts, access doors, and inspection windows or cutouts. They are manufactured from composite or aluminum materials and are readily removable for maintenance access. Fairings are constructed of composite materials and reduce drag.

The forward cowling is composite and incorporates a hinge located at the forward end of the cowling. Support rods are attached internally on each side of the cowl and fit into roof mounted clips to support the cowl in a raised position. Two toggle hook latch fittings secure it in the closed position.

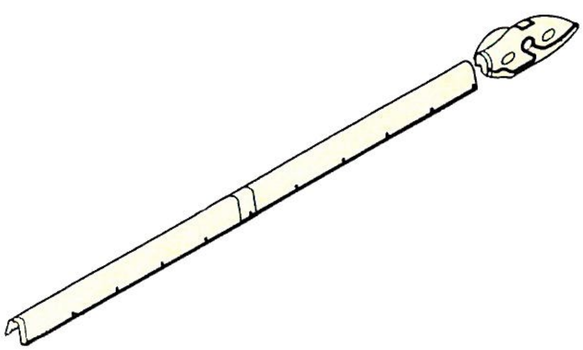


The transmission cowling is composite and encloses the forward half of the main transmission and controls. Two cutouts are provided for visual inspection of aircraft fluids. The transmission oil level cut out is on the right side and the hydraulic fluid level cut out is on the left side.

The air induction cowling is aluminum and encloses the aft half of the main transmission. Inlet ducts on the left and right side of the air induction cowling direct airflow into the induction screen and into the engine. Two hinged access doors, one on each side, provide for inspection of the main transmission, transmission mounting, and main driveshaft.

The engine cowling has composite hinged doors and a composite upper structure. The doors are held in the open position with a mechanical folding brace. The doors and upper structure of the engine cowling incorporate screened vents to allow air movement through the engine compartment. A titanium engine exhaust stack protrudes through the upper structure. A smaller opening, facing aft and located just forward of the engine exhaust, is the engine bleed air induction port vent.

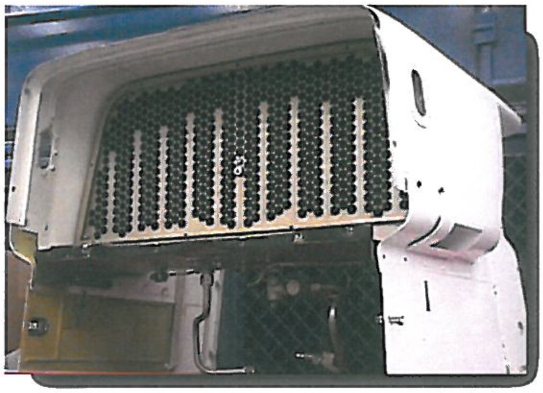
The aft cowling is composite and encloses the oil cooler, blower assembly, and engine oil tank. The cowling has a cutout on the left-hand side to view the engine oil level. Two access doors allow viewing of the tail rotor driveshaft and engine oil servicing. Aluminum screens allow for air circulation.



A two piece composite cowling encloses the tail rotor driveshaft. The driveshaft cover is mounted in a manner that prevents the cover from making contact (chafing) with the top of the tailboom.

The composite tail rotor gearbox fairing encloses the tail rotor gearbox and is attached to the tailboom and vertical fin. It incorporates a white position light mounted on the aft end of the fairing. There are two hinged inspection doors; one on the upper fairing used for tail rotor gearbox oil servicing, and a lower door providing access to the tail rotor gearbox chip detector. Screened openings provide for air movement and gearbox oil level viewing.

## Particle Separator



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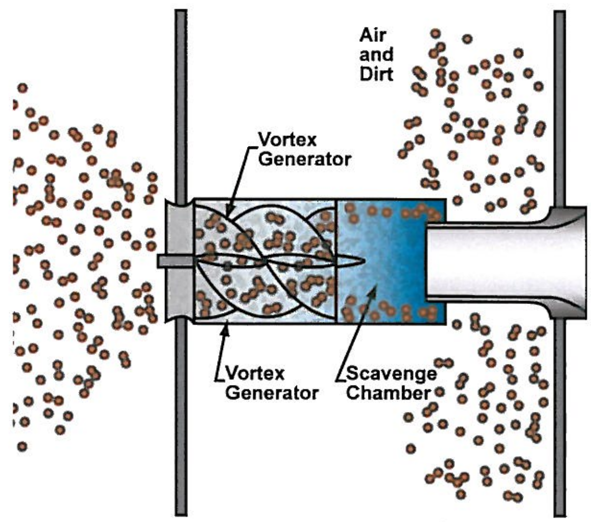
An optional particle separator kit may be installed forward of the engine air inlet just aft of the transmission fairing instead of the induction screen. The particle separator provides continuous protection of the engine against damage from the ingestion of sand, dust, and other foreign material. The unit consists of the separator, bleed air tubing and hoses, compressor wash fittings, and other associated hardware.

The separator section has 533 filter elements and is positioned so all inlet air must pass through the filter elements before entering the engine. Each of the filter elements (vortex tube assemblies) in the separator consists of a vortex generator bonded into an inlet tube and a second, smaller tube to form a scavenge chamber. Materials such as hay, leaves, etc., which are too large to enter the filter elements, are stopped at the face of the separator. Particles such as dust, dirt, sand, grass, etc., enter the filter elements and are spun in the vortex generator. The particles are centrifugally routed into the scavenge chamber where they are ejected overboard through ejector tubes by engine bleed air. The separator has a tested efficiency of 85% by weight for coarse particles (27 micron nominal). Due to its inertial (centrifugal) action, the separator is more efficient for particles larger than 27 microns and less efficient for particles smaller than 27 microns.

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Turning on the purge feature (crew compartment overhead panel switch marked (PART SEP) takes bleed air from the engine diffuser scroll and reroutes this air to the base of the particle separator, through the venturi tubes, and out the ejector tubes.

A plexiglass window is installed on each side of the cowling to permit visual inspection of the separator plenum chamber between the aft side of the particle separator and the engine inlet. The ejector tubes are mounted on each side just below these windows.

The use of bleed air to purge the system causes a slight increase in MGT indication when the purge is turned on. When the unit is installed, 13.2 pounds of weight is added to the empty weight of the aircraft.

A compressor wash fitting is installed inside the right side transmission access panel. This fitting allows aircraft operators to rinse the aircraft engine.

# Airframe

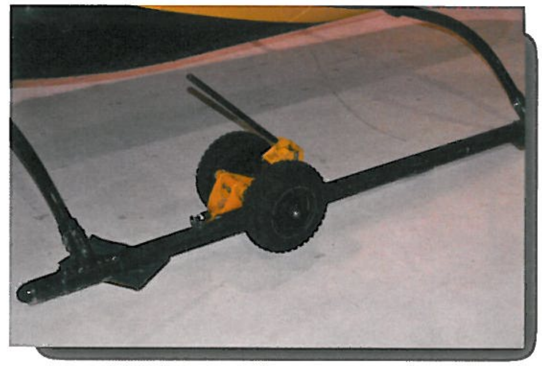
## Parking, Mooring, and Storage

### Ground Handling

Ground handling of the helicopter consists of towing, parking, securing, and mooring. Model 205 or 206L ground handling wheels are used for towing. Refer to BHT-407-MM-2 for more detailed ground handling information.

**CAUTION**

DO NOT TOW THE HELICOPTER IF THE GROSS WEIGHT IS MORE THAN 5000 LBS (2270 kg) FOR MODEL 205 GROUND HANDLING WHEELS OR 4450 LBS (2020 kg) FOR MODEL 206L GROUND HANDLING WHEELS.



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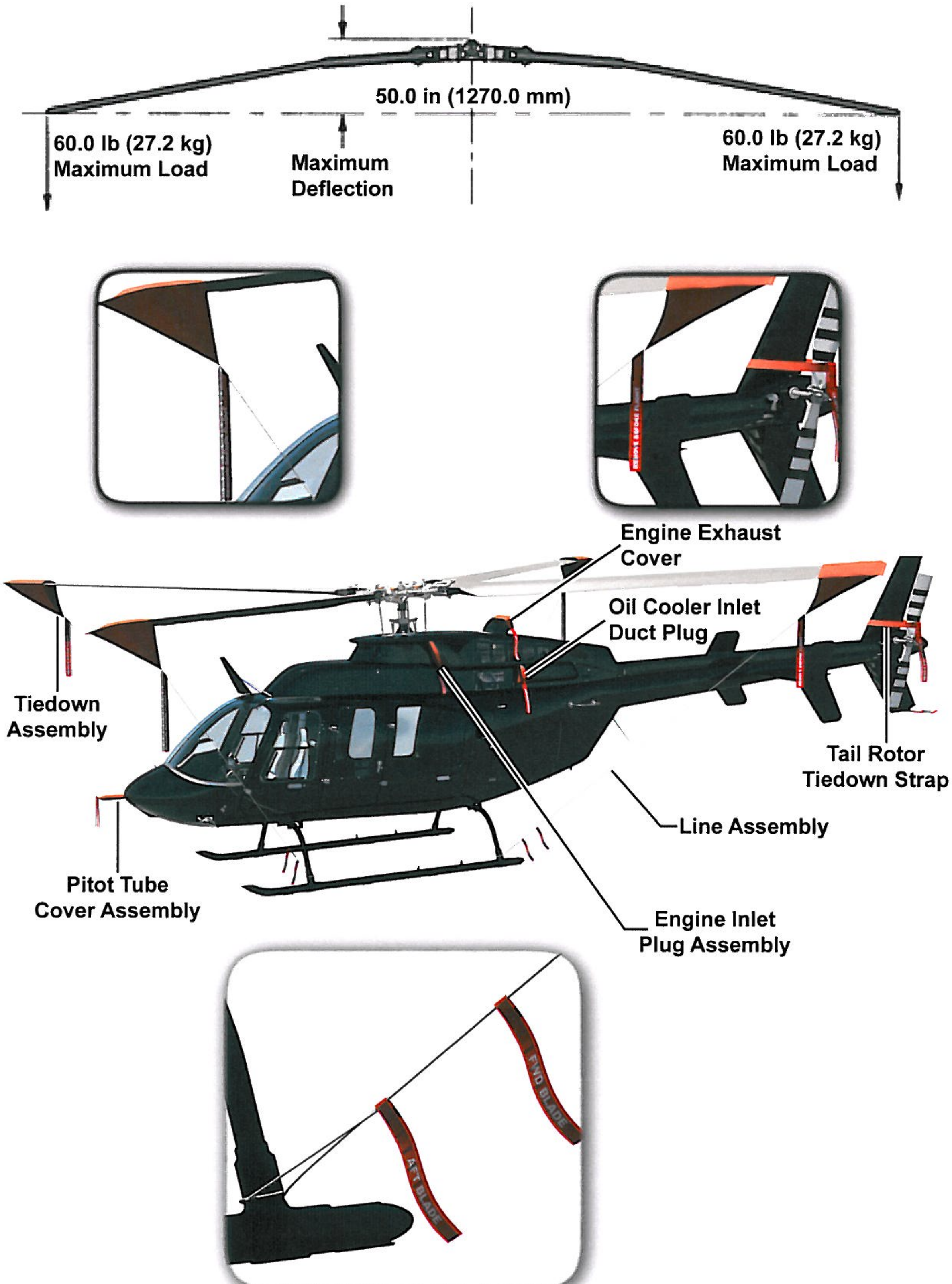
# Airframe

## Parking, Mooring, and Storage (Cont)

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# Airframe

### Covers and Tiedowns

Protective covers and tiedowns are furnished as loose equipment and are used for parking and mooring of helicopter. Additional equipment such as ropes, cables, devises, ramp tiedowns, or dead man tiedowns are required during mooring.

### Cover — Engine Inlet

Engine inlet plug assemblies are red, flame resistant, and each cover is attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To install the engine inlet plugs, make sure that the side marked TOP is up. Push the engine inlet plug into the engine air inlet.

### Plug Assembly – Oil Cooler Blower Inlet Duct

Oil cooler blower inlet duct plug assemblies are red, flame resistant, and each plug is attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To install the oil cooler blower inlet duct plug, push it into oil cooler blower inlet duct.

### Cover — Pitot Tube

The pitot tube cover assembly is red, flame resistant, and attached with a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. To install the pitot tube cover, push it over the pitot tube. Attach tie cord.

**WARNING**

THE PITOT TUBE CAN BE HOT.

### Cover — Engine Exhaust

The engine exhaust cover is red, flame resistant, and includes a red streamer stenciled in white letters, REMOVE BEFORE FLIGHT. A 1/4-inch diameter elastic tie cord is attached to the cover for securing to engine exhaust. To install the engine exhaust cover, push it over the exhaust tailpipe. Attach the cord.

### Tiedown — Main Rotor

For each main rotor blade, there is a main rotor tiedown assembly. Each tiedown assembly has a sock assembly and a line assembly. Use these to attach the blades to the landing gear crosstube. The sock assembly is red and has a red streamer, stenciled with white letters, REMOVE BEFORE FLIGHT, attached to it. The line assembly is made of 0.19 inch (4.83 mm) diameter nylon and has a ring and an attached flag. The flag is stenciled with the letters FWD BLADES or AFT BLADES.

Install the main rotor tiedown assemblies as follows:

**CAUTION**

DO NOT CAUSE THE MAIN ROTOR BLADES TO BEND MORE THAN THE LIMITS SHOWN.

# Airframe

### Tiedown — Main Rotor (Cont)

Align the main rotor blades at the same time the tail rotor blades are aligned with the vertical fin. This makes it possible to install the tail rotor tiedown.

1. Turn the main rotor blades until there are two blades aft of the fuselage station of the main rotor hub. When looking down on the helicopter, the four blades make an X over the vertical center line of the fuselage.

2. Install the two FWD BLADES sock assemblies on the ends of the main rotor blades that are forward of the fuselage station of the main rotor hub.

3. Put a line assembly around each outboard end of the forward crosstube of the landing gear.

**NOTE**

Rings are pre-set to apply the necessary tension to the forward and aft main rotor blades.

4. Attach the snaps of the two line assemblies to the rings of the two FWD BLADES sock assemblies.

5. Install the two AFT BLADES sock assemblies on the ends of the two aft main rotor blades.

6. Put a line assembly around each outboard end of the aft crosstube of the landing gear.

7. Attach the snaps of the two line assemblies to the rings of the two AFT BLADES sock assemblies.

### TIEDOWN — TAIL ROTOR

The tail rotor tiedown strap is made of 0.025 x 1.0 x 92.0 inch (0.635 x 25 x 2340 mm) nylon webbing. It is red and stenciled with white letters, REMOVE BEFORE FLIGHT.

Install the tail rotor tiedown as follows:

**CAUTION**

DO NOT TIE DOWN TAIL ROTOR TO EXTENT THAT TAIL ROTOR BLADE FLEXES.

1. Turn the main rotor blades until there are two blades aft of the fuselage station of the main rotor hub. When looking down on the helicopter, the four blades should make an X over the vertical center line of the fuselage. Align the tail rotor with the vertical fin.

2. Install the tail rotor tiedown strap in the loop on the lower left side of the vertical fin.

3. Wind the tail rotor tiedown strap around the tail rotor blade.

4. Attach the tail rotor tiedown strap to the loop on the lower left side of the vertical fin.

# POWERPLANT

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## Powerplant

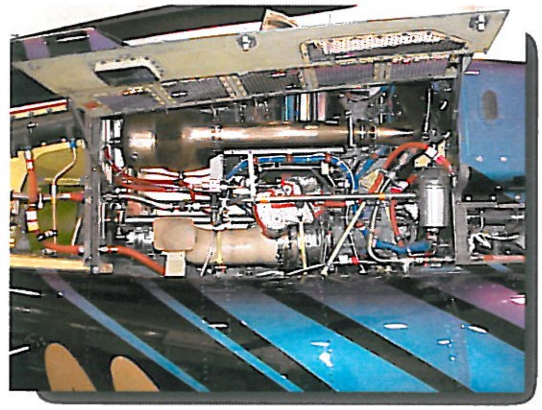


Bell Helicopter
A Textron Company

# Powerplant

## General

The Rolls Royce 250-C47B engine is an internal combustion turboshaft engine featuring a free turbine. The gas generator is a single stage, single entry centrifugal flow compressor directly coupled to a two-stage gas generator turbine. The power turbine is a two-stage free turbine gas coupled to the gas generator turbine.



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The integral reduction gearbox has front and rear drive splines to mate with aircraft drives. The engine has a single combustion chamber with single ignition. The output shaft centerline is below the centerline of the engine rotor and the single exhaust outlet is directed upward. The engine incorporates a Full Authority Digital Electronic Control (FADEC) system.

The 250-C47B is a thermodynamically rated 813 shaft horsepower (SHP) engine. To provide reserve power for high altitude and hot day performance, the engine torque has been derated through the Hydro-Mechanical Unit (HMU) or fuel control to 674 SHP for takeoff and 630 SHP for continuous operation.

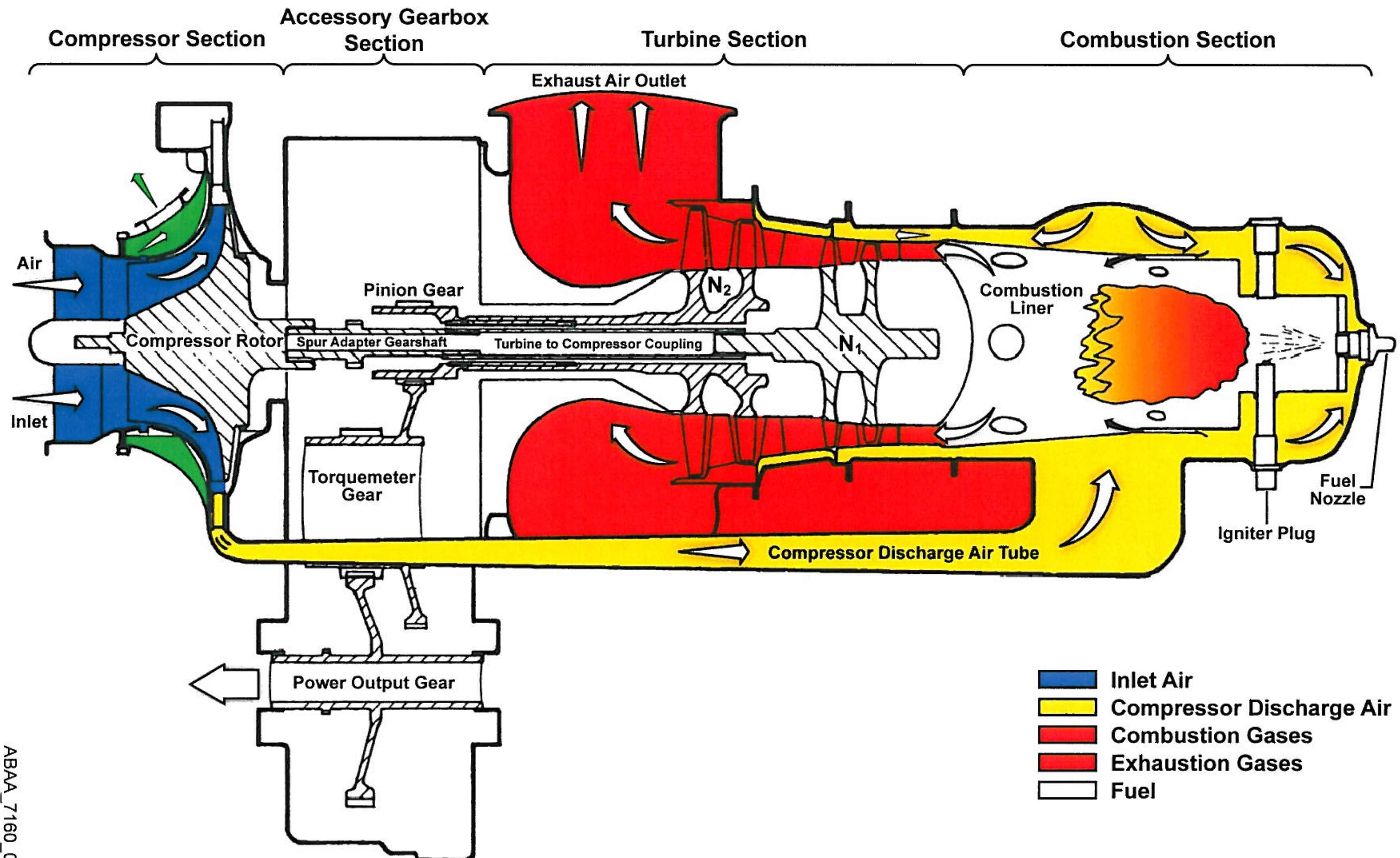
To clarify the 250-C47B engine horsepower further, the engine data plate identifies the engine's rated horsepower as 650. Rolls-Royce guarantees the engine will provide this rated horsepower at sea level at a specific fuel flow and MGT. This ensures all 250-C47B engines sold to Bell Helicopter meet or exceed the rated horsepower specification. Therefore, the rated horsepower shown on the engine data plate is not the maximum horsepower the engine will deliver when installed in the 407 helicopter.

The engine mounts horizontally aft of the transmission and above the fuselage to simplify the drive system, improve the inlet/exhaust arrangement, and reduce cabin noise. To provide better structural integrity, three bipod and one horizontal mount support the engine on the engine deck.

The major engine components are:

- Compressor Section
- Combustion Section
- Turbine Section
- Accessory Gearbox

## Air Flow

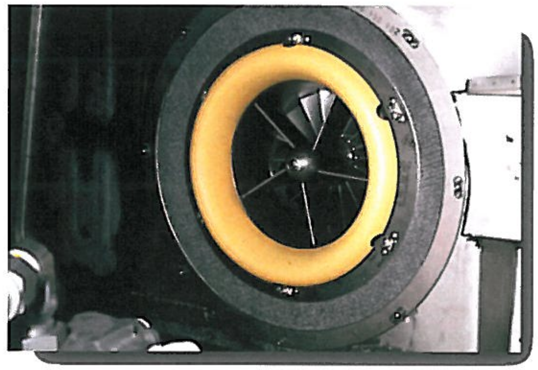



ABAA_7160_000005

# Powerplant

## Compressor

The compressor assembly consists of a compressor front support, shroud housing, diffuser, rear support assembly, centrifugal impeller, scroll assembly, mount assembly, and bearings. The front support encloses the front bearing and supports it through five hollow inlet struts. The struts are hollow to provide anti-icing using compressor discharge air and to permit introduction of oil to and from the compressor front bearing. The compressor rear bearing is mounted in the rear support assembly and is lubricated from the gearbox.



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The compressor impeller is a single piece of forged titanium. The vanes transition from axial to centrifugal, eliminating the need for stators. At 100% $N_G$ the compressor rotates at 51,000 RPM. The compression ratio is approximately 9.2:1 (9.2 Bars or 133 PSI). This rapid compression increases the temperature of the compressed air approximately 291°C (555°F).

The compressor shroud housing has a narrow slot, known as the inducer bleed port, machined into the perimeter near the front. At low $N_G$ RPM, a portion of the airflow vents into the atmosphere, unloading the compressor to allow for more rapid acceleration.

At higher $N_G$ RPM when maximum engine efficiency is required, atmospheric air enters the compressor, increasing the volume of air entering the engine.

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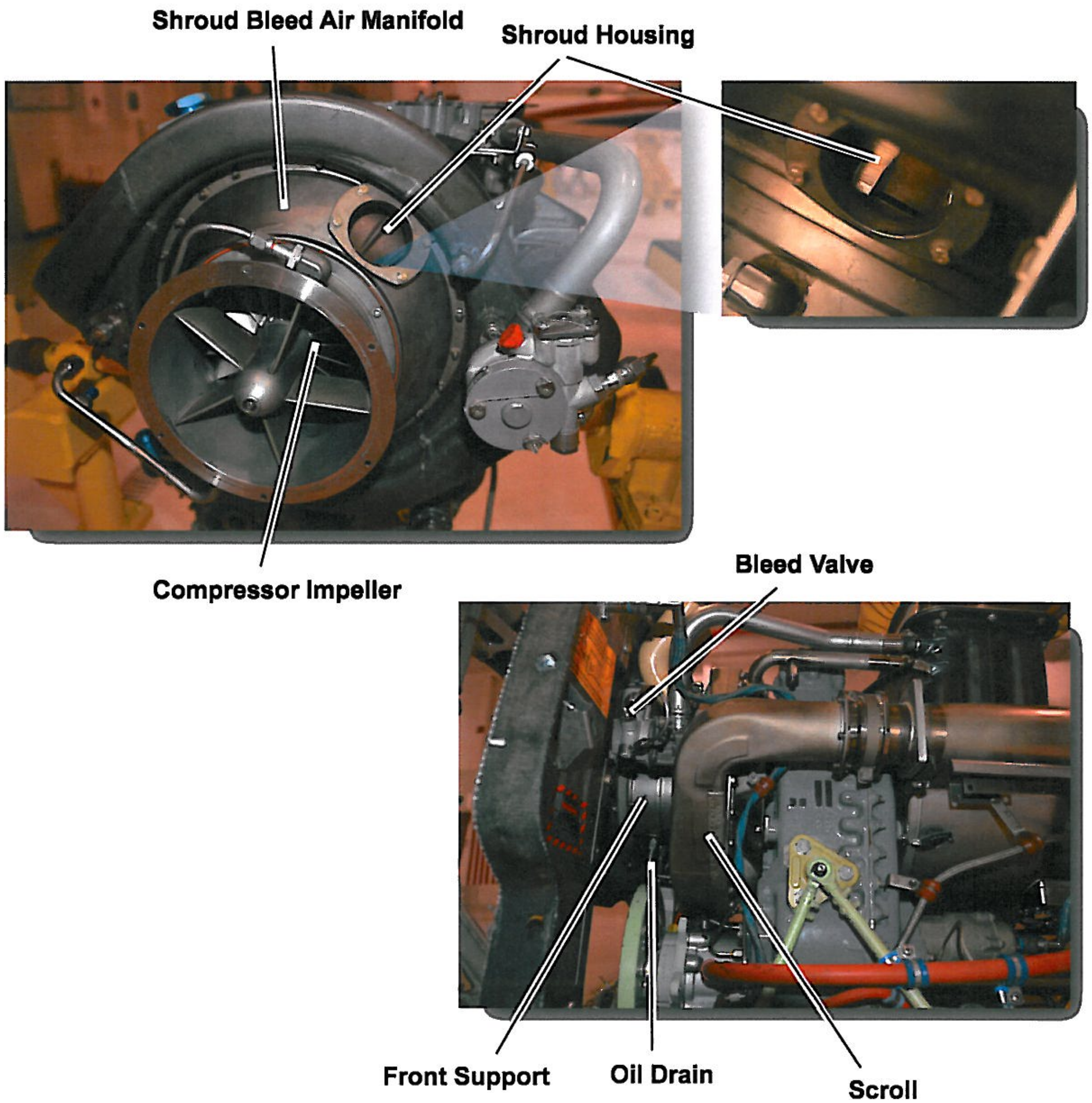
# Powerplant

## Compressor (Cont)

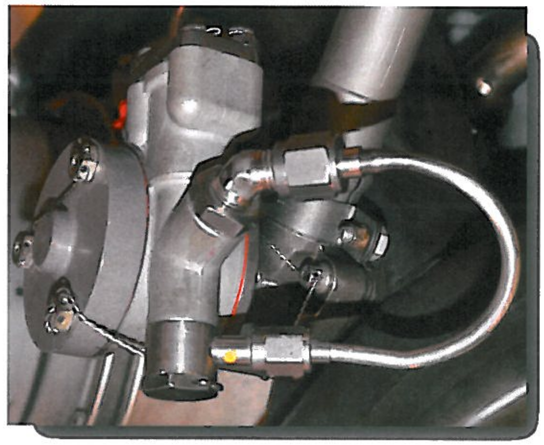



ABAA_7230_000003

# Powerplant

## Bleed Valve

The bleed control valve is open during starting and idle operation and remains open until a predetermined pressure ratio is obtained. At this pressure ratio, the valve begins to modulate from open to close. It is normally open during the start cycle and ground idle, modulates during acceleration to full operational speed, and remains closed during flight operation speeds. Pressure sensing for bleed control valve operation is within the valve.



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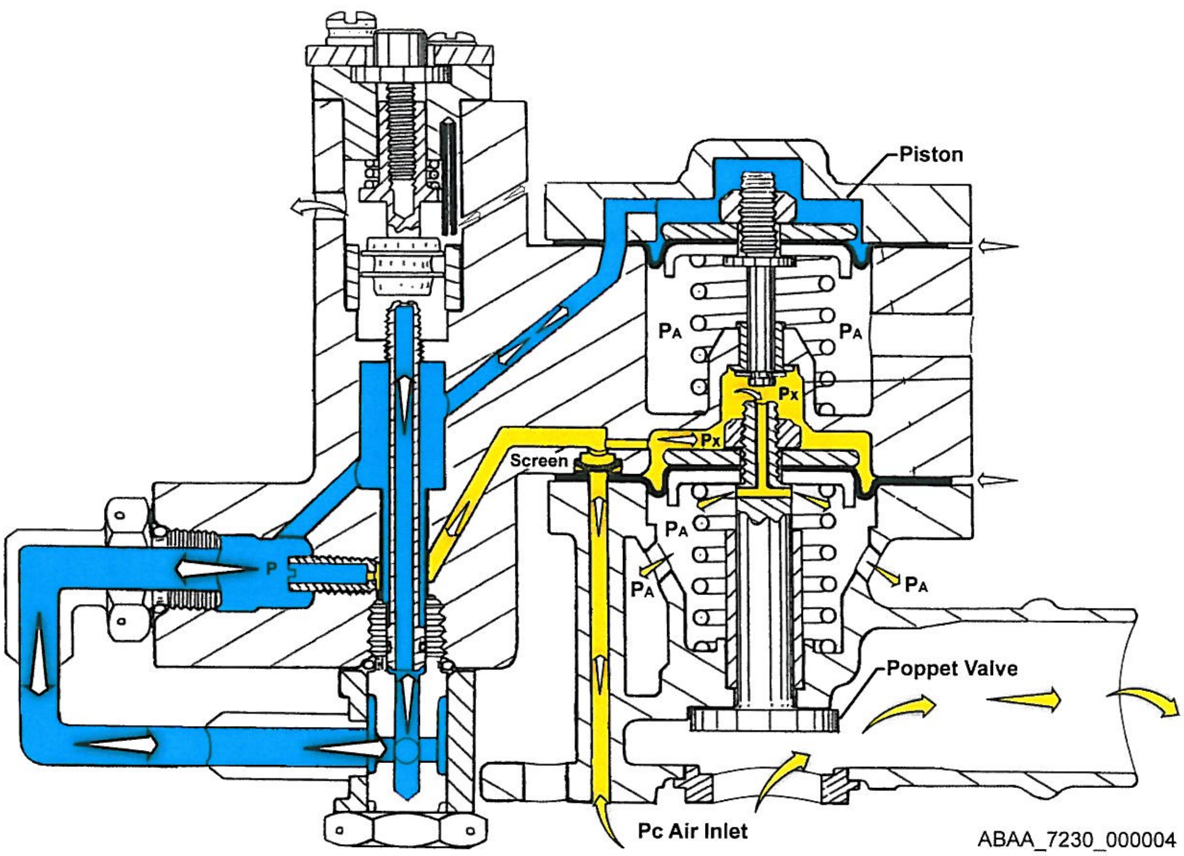


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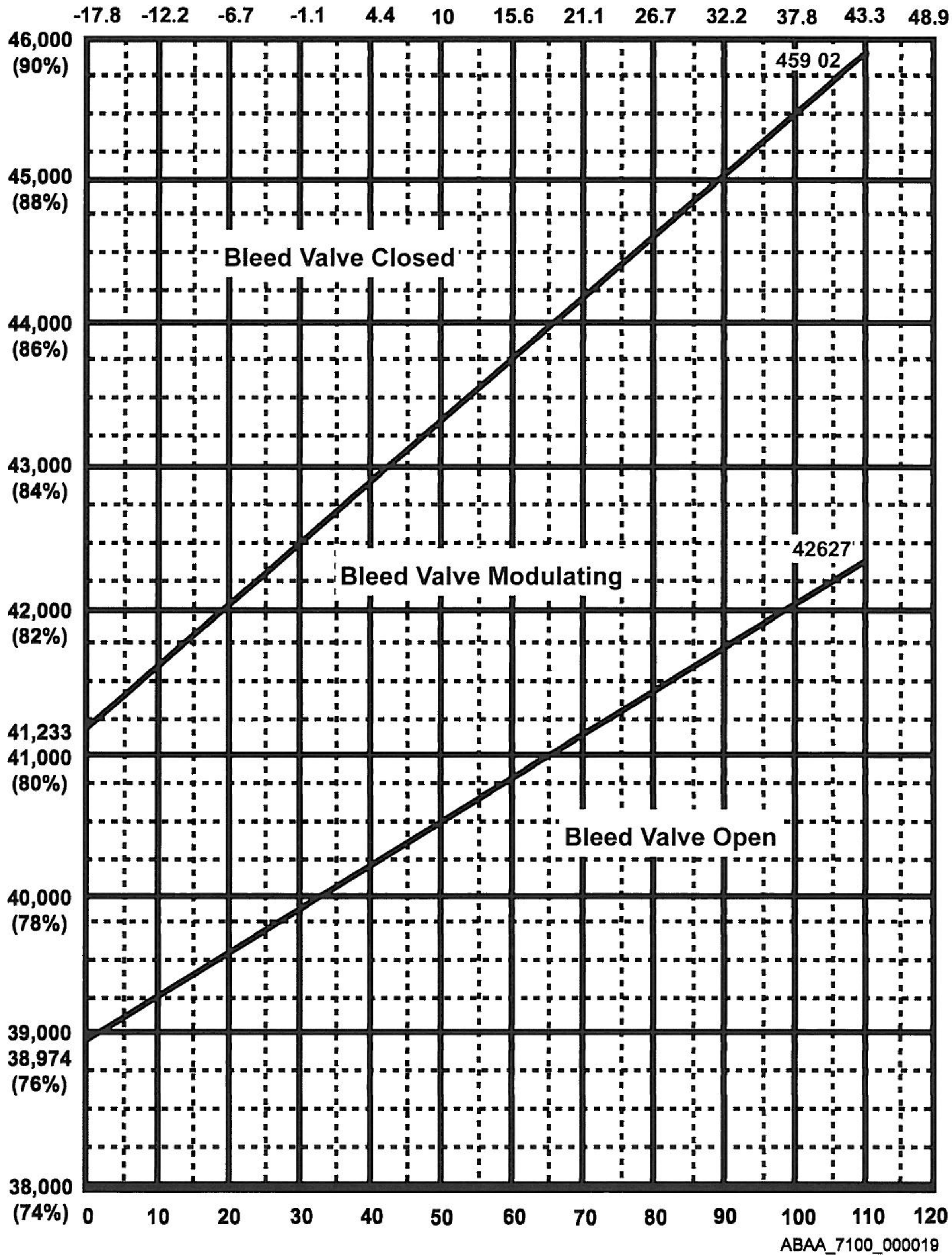
## Powerplant

### Bleed Air Valve Chart

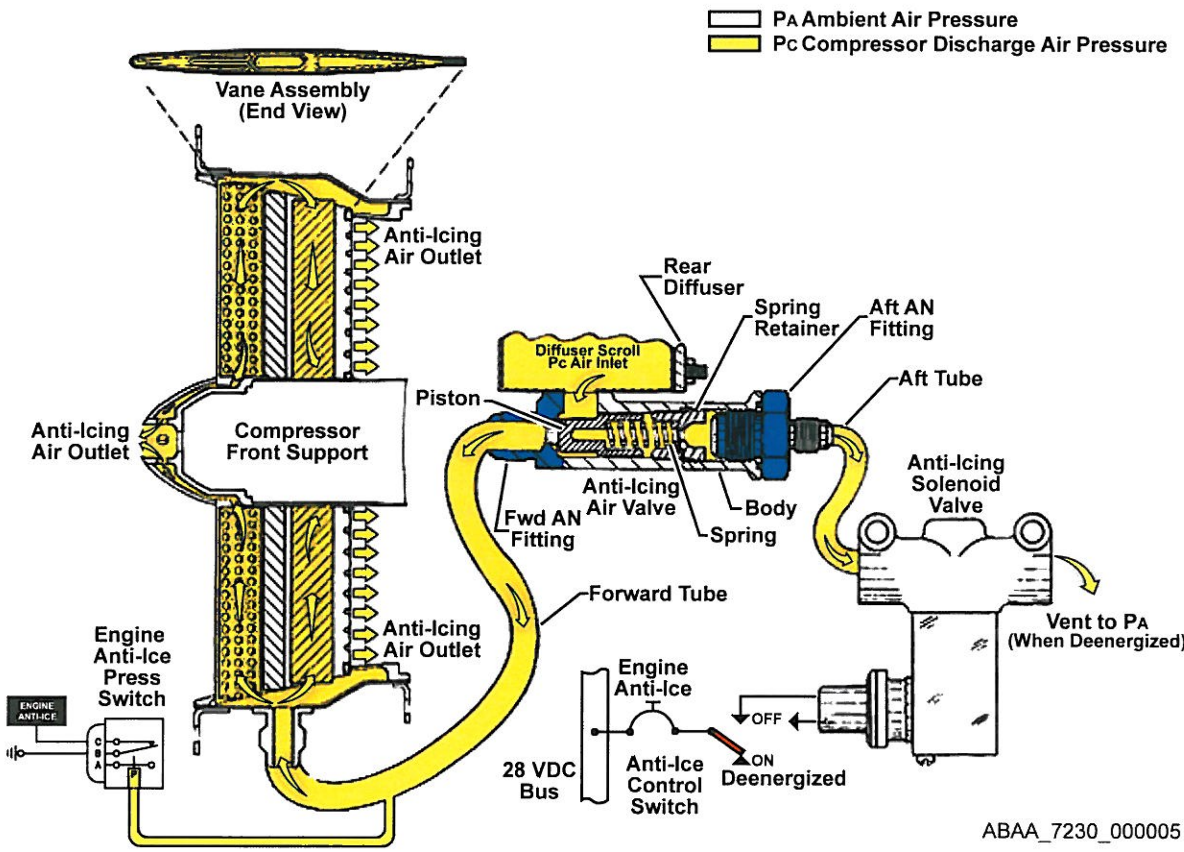
This chart shows the relationship between ambient temperature and $N_G$ speed, as it applies to the engine bleed valve's modulation from an open to closed position.



# Powerplant

## Anti-Icing System

Operation of the engine during icing conditions could result in ice formations on the compressor front support. If ice built up, airflow to the engine could be affected and performance decreased. The engine anti-icing system is designed to prevent ice formation on the compressor front support. The pilot activates the system by placing the engine anti-ice switch to the ENG ANTI ICE (on) position. The ENG ANTI ICE white advisory light illuminates when the system is activated. The ENG ANTI ICE white advisory light is optional on aircraft prior to Serial Number (S/N) 53095 and a standard installation on subsequent aircraft.

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When the system is in operation, compressor discharge air, which has been heated due to compression, flows through the anti-icing valve and tube to the compressor front support passages. Hot air flows between the double wall outer shell and into the five hollow radial struts. The hot air flowing through the radial struts exhausts either from small slots in the trailing edge of the struts or from the double wall bullet nose hub of the compressor front support.

The compressor inlet guide vanes and front bearing support hub are the only engine components with anti-icing provisions. The bleed air shutoff valve is solenoid controlled. In the event of a total electrical failure, the system is fail-safe to on.

# Powerplant

## Combustion Section

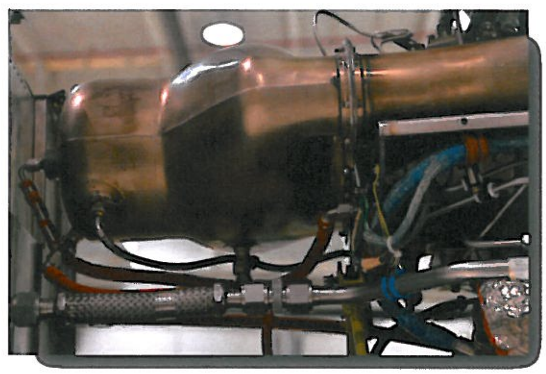
The combustion section consists of an outer combustion case and a combustion liner. The combustion liner is inside the outer case. The fuel nozzle mounts in the aft end of the outer combustion case. The igniter plug mounts near the aft end of the case.

Air enters the single combustion liner at the aft end through holes in the liner dome and skin. The air mixes with fuel sprayed from the fuel nozzle and combustion takes place. Combustion gases move forward out of the combustion liner to the first stage $N_G$ gas producer turbine nozzle. Approximately 20% to 25% of the air delivered to this section is required to burn the fuel. The other 75% to 80% is used to cool the internal parts. Approximately 2% of the air is used to seal oil passages.

Most of the cooling air enters the combustion liner in such a manner that the flame pattern is prevented from impinging on the wall of the combustion liner. In addition, because the gases of combustion and the cooling air mix before passing through the turbine sections, the resultant temperature stays within acceptable limits.

## Turbine Section

As the gas stream leaves the combustion chamber, it passes to the turbine section. This high-energy gas stream powers the two turbine sections to sustain the airflow through the engine and provide output power. The turbine section consists of a gas producer turbine support, power turbine support, a two-stage $N_G$ turbine rotor, a two-stage power turbine rotor, and an exhaust collector support. The gas producer turbine drives the compressor and certain engine accessories through the $N_G$ drive train. The power turbine drives the power output shaft, and certain engine accessories.



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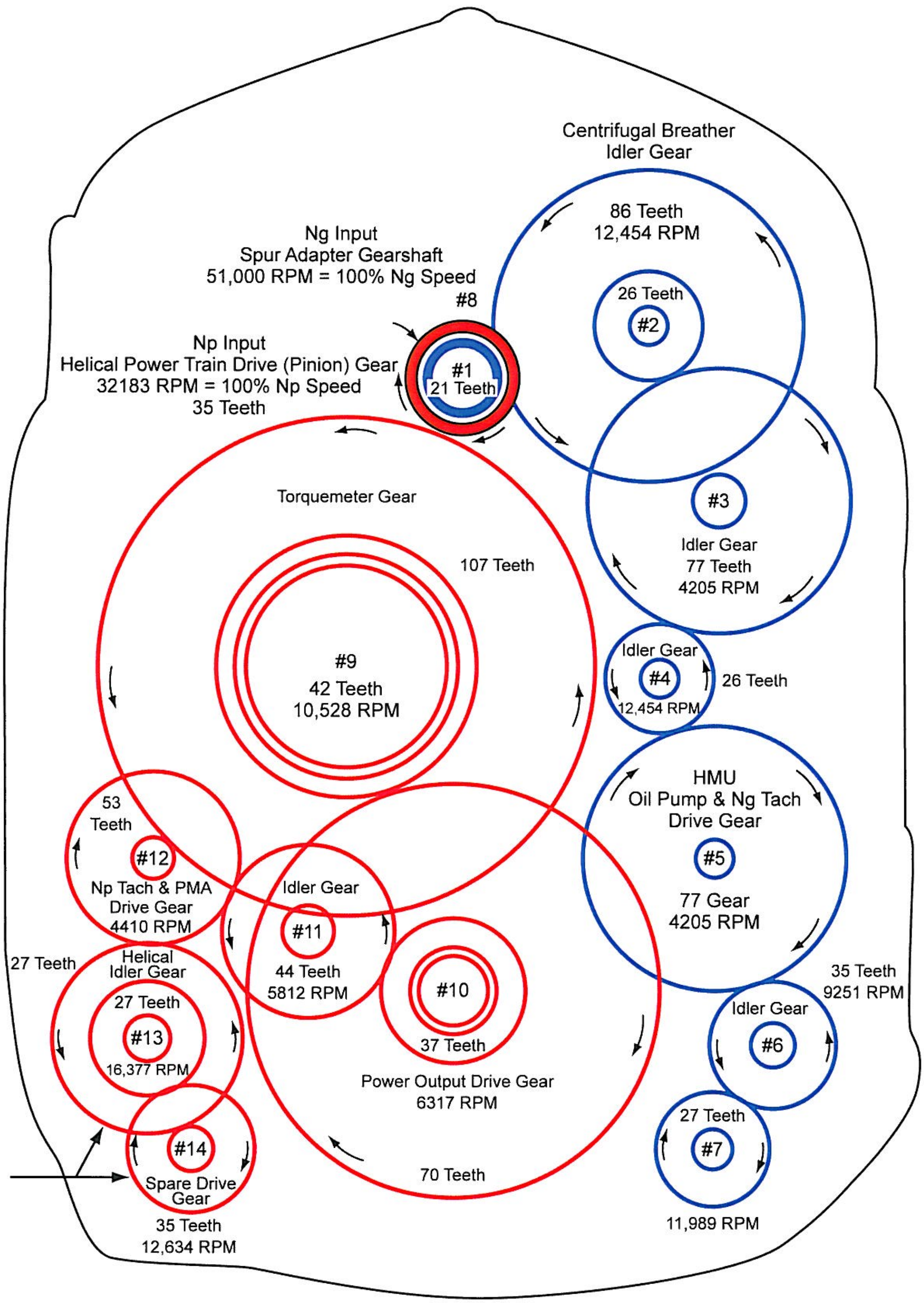
## Measured Gas Temperature (MGT)

Four thermocouples located between the $N_G$ and $N_P$ turbine wheels sense the temperature of the gases passing through the turbine. Each thermocouple senses a voltage proportional to the gas temperature. An average of the four voltages is displayed on the cockpit MGT gauge.

# Powerplant

## Power and Accessory Gearbox

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# Powerplant

## Power and Accessory Gearbox (Cont)

The main power and accessory drive gear trains are enclosed in a single gear case. The gear case serves as the structural support of the engine. All engine components are attached to the case. There are two independent drive trains in the gearbox, gas producer ($N_G$), and power turbine ($N_P$).

The $N_G$ gear train drives the hydro-mechanical unit (HMU), the starter generator, and the oil pump (inside the gearbox).

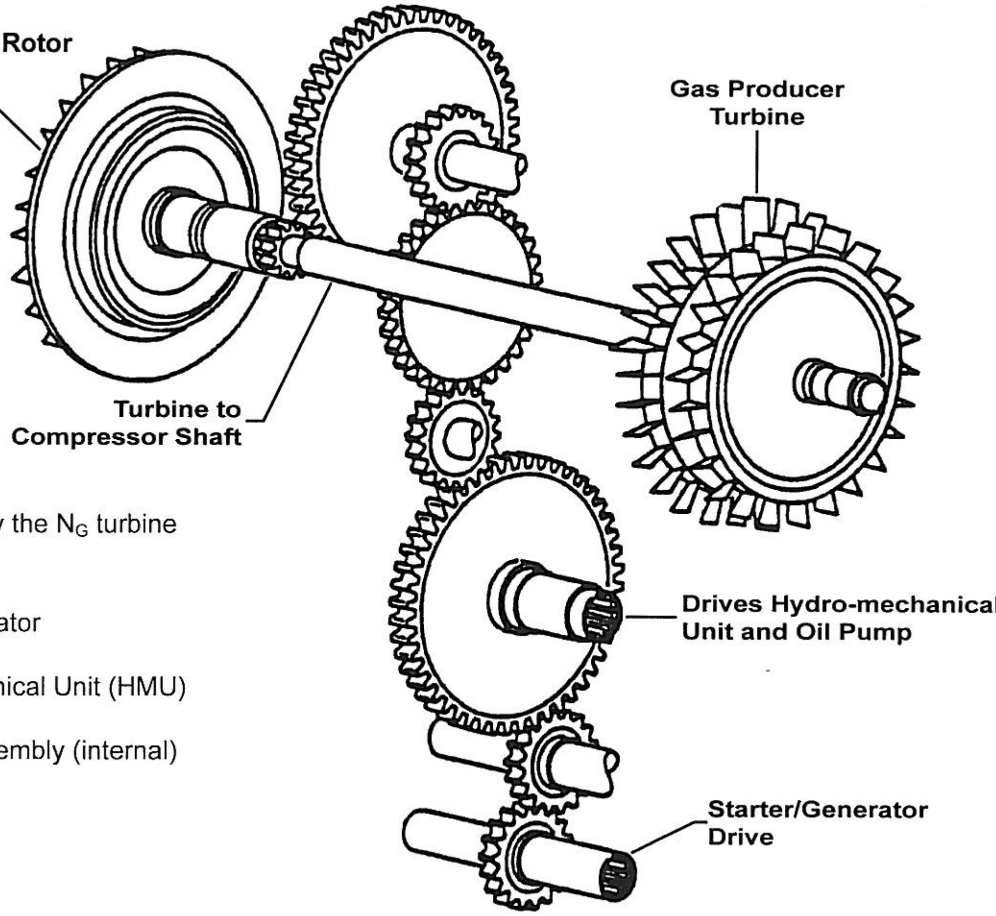
A two-stage helical and spur gear set is used to reduce $N_P$ rotational speed from 32,183 RPM at the power turbine to 6317 RPM at the output drive spline.

The $N_P$ gear train drives the permanent magnet alternator (PMA), the torquemeter, and the freewheeling unit.

The gearbox also contains two dual coil monopole sensors. These sensors are magnetic coils that produce an electrical signal. One sensor measures $N_G$ speed, and the other measures $N_P$ speed. On each sensor, both coils send input signals to the Engine Control Unit (ECU) and to the appropriate indicator: either the $N_G$ gauge or the $N_P$ needle on the dual tachometer.

# Powerplant

## Power and Accessory Gearbox (Cont)

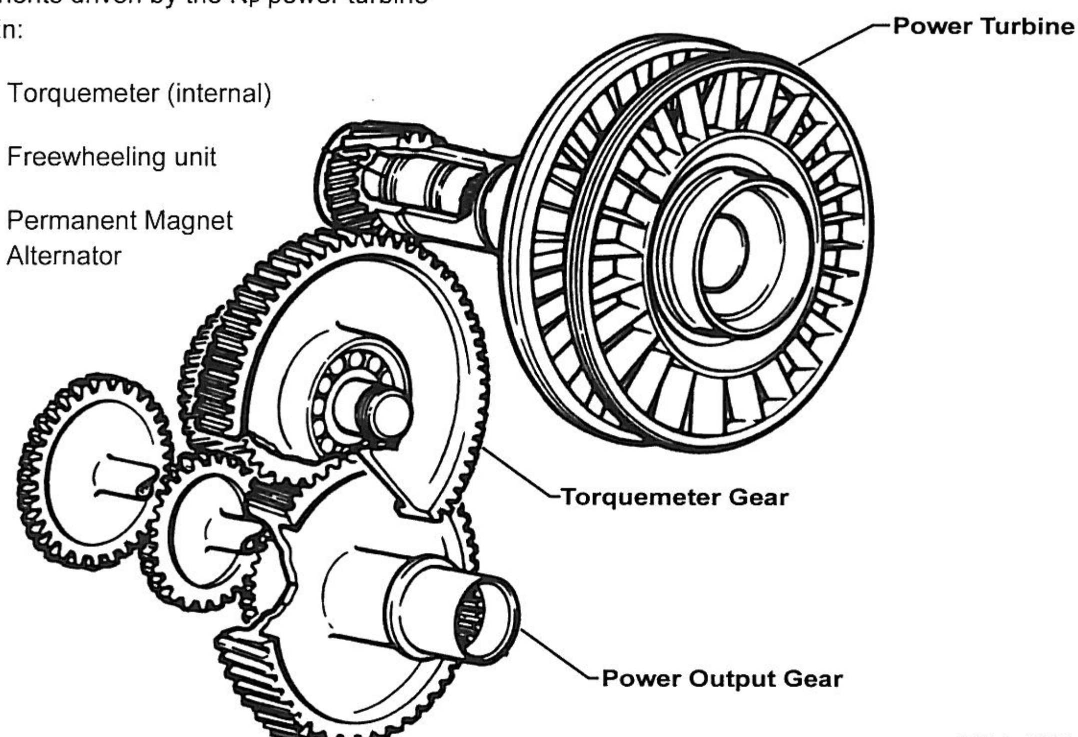



Components driven by the $N_G$ turbine geartrain:

- Starter Generator
- Hydro Mechanical Unit (HMU)
- Oil Pump Assembly (internal)

Components driven by the $N_P$ power turbine geartrain:

- Torquemeter (internal)
- Freewheeling unit
- Permanent Magnet Alternator

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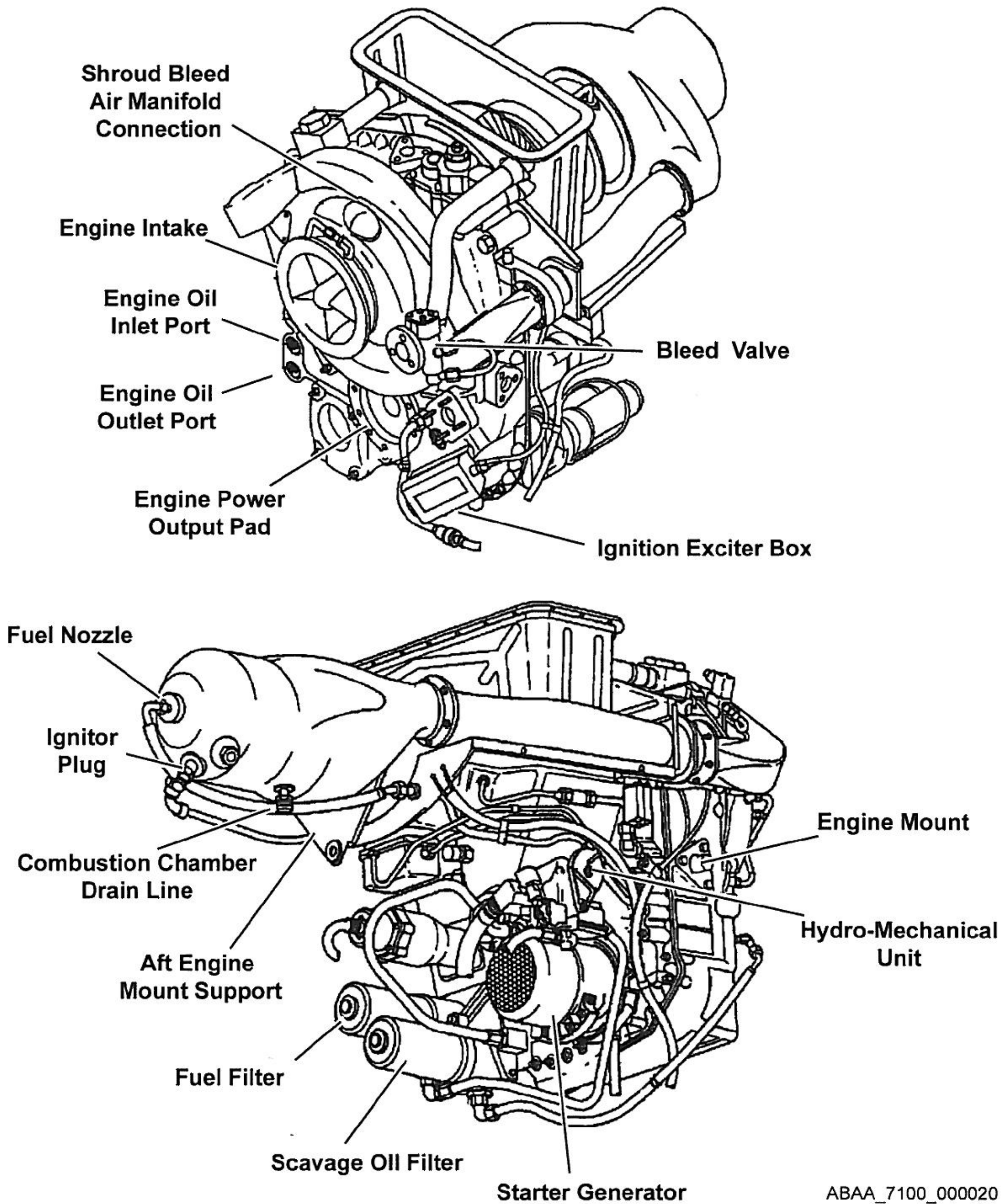
# Powerplant

## Powerplant Components

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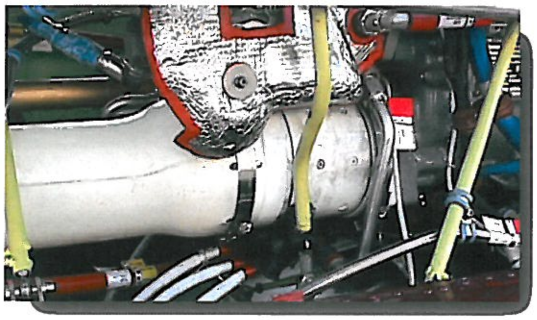
# Powerplant

## Starter/Generator



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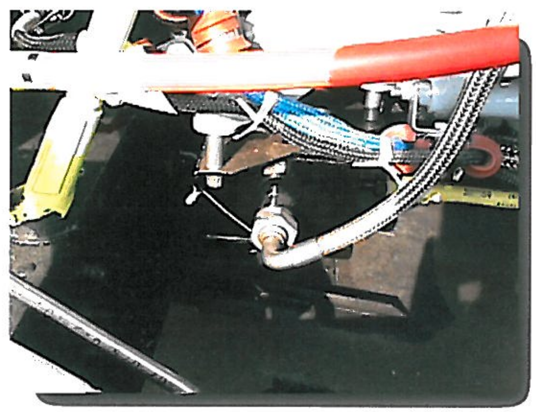
The starter/generator is used as a DC motor to drive the engine during the starting cycle. Once the engine starts and the generator switch is turned on, the starter/generator functions as a DC generator to supply main electrical power for the helicopter and to charge the battery.

## Ignition System

The exciter box amplifies 28 VDC into high voltage arcs at the igniter plug that is required during the starting cycle. The igniter plug is threaded into the combustion outer case and extends into the combustion liner, providing ignition sparks that ignite the fuel/air mixture during start. Once the engine starts, combustion is continuous and the ignition source is no longer required. Electrical power is supplied from the electrical bus and is protected by a 5 amp circuit breaker (CB) marked IGNTR.



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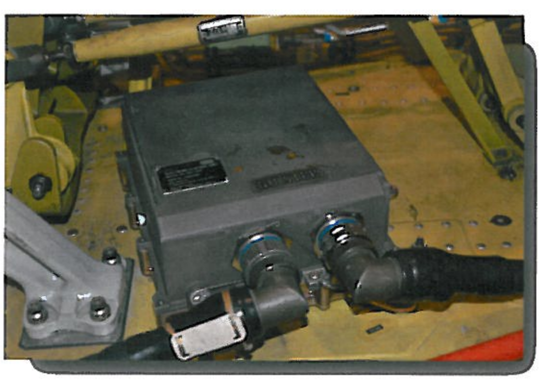
## Fuel System



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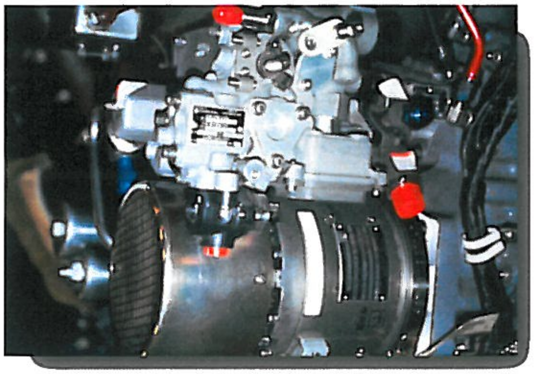
The engine fuel system consists of the following components: the ECU, Combined Engine Filter Assembly (CEFA), HMU, and fuel nozzle.

The ECU, part of the FADEC system, monitors engine power demand and signals the metering valve in the HMU to provide the proper amount of fuel to the combustion chamber.

The CEFA filters both fuel and oil supplied to the engine. The fuel portion of the assembly filters the fuel prior to entering the HMU. A bypass valve allows the fuel to bypass the filter in the event it becomes blocked.

# Powerplant

## Fuel System (Cont)



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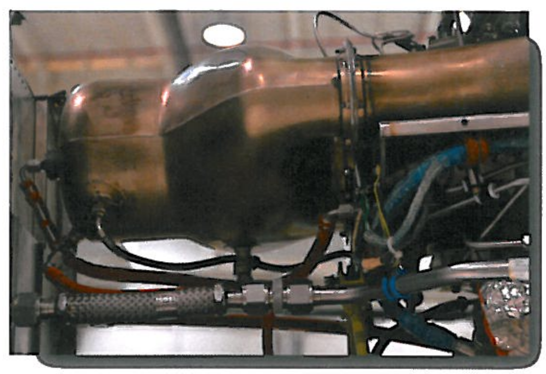
The HMU consists of a fuel pump and a fuel control assembly. The fuel pump provides high-pressure fuel to the fuel control. The fuel control sends the proper fuel flow to the combustion chamber as directed by the ECU

The fuel nozzle atomizes and injects fuel into the combustion liner at the proper spray angle to support combustion.

## Burner Drain Valves

The burner drain valves prevent accumulation of water or fuel in the combustion section. During start, the drain valves close when the air pressure within the combustion section exceeds the air pressure on the outside of the combustion section by a predetermined value. The valves open on shutdown by means of spring action.



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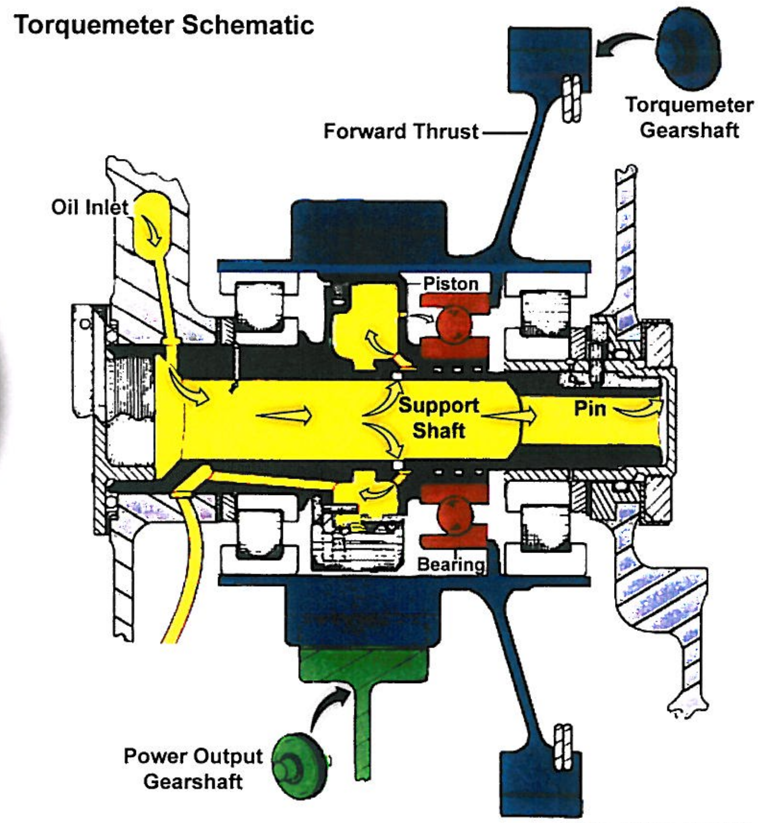
# Powerplant

## Torquemeter


The torquemeter in the 250 series engine gearbox is a hydraulic type that uses the engine lubrication system as its oil (hydraulic) pressure source. In order to minimize friction effects and provide accurate measurement of torque, the axial gear thrust on the helical torquemeter gearshaft is high. Engine oil system pressure must always be greater than the torquemeter oil pressure and is regulated to a value of 115-130 psi.

The power turbine gear train has two stages of helical gearing. The helical powertrain drive gear driving the larger diameter gear on the helical torquemeter gear shaft accomplishes the first stage reduction. The smaller diameter gear on the helical torquemeter gearshaft driving the helical power takeoff gear-shaft accomplishes the second stage reduction. The helix angles on the gears are engineered to insure that both stages of reduction produce a forward axial thrust on the helical torquemeter gear shaft. If friction is neglected, this axial thrust is directly proportional to the torque transmitted through the gears.

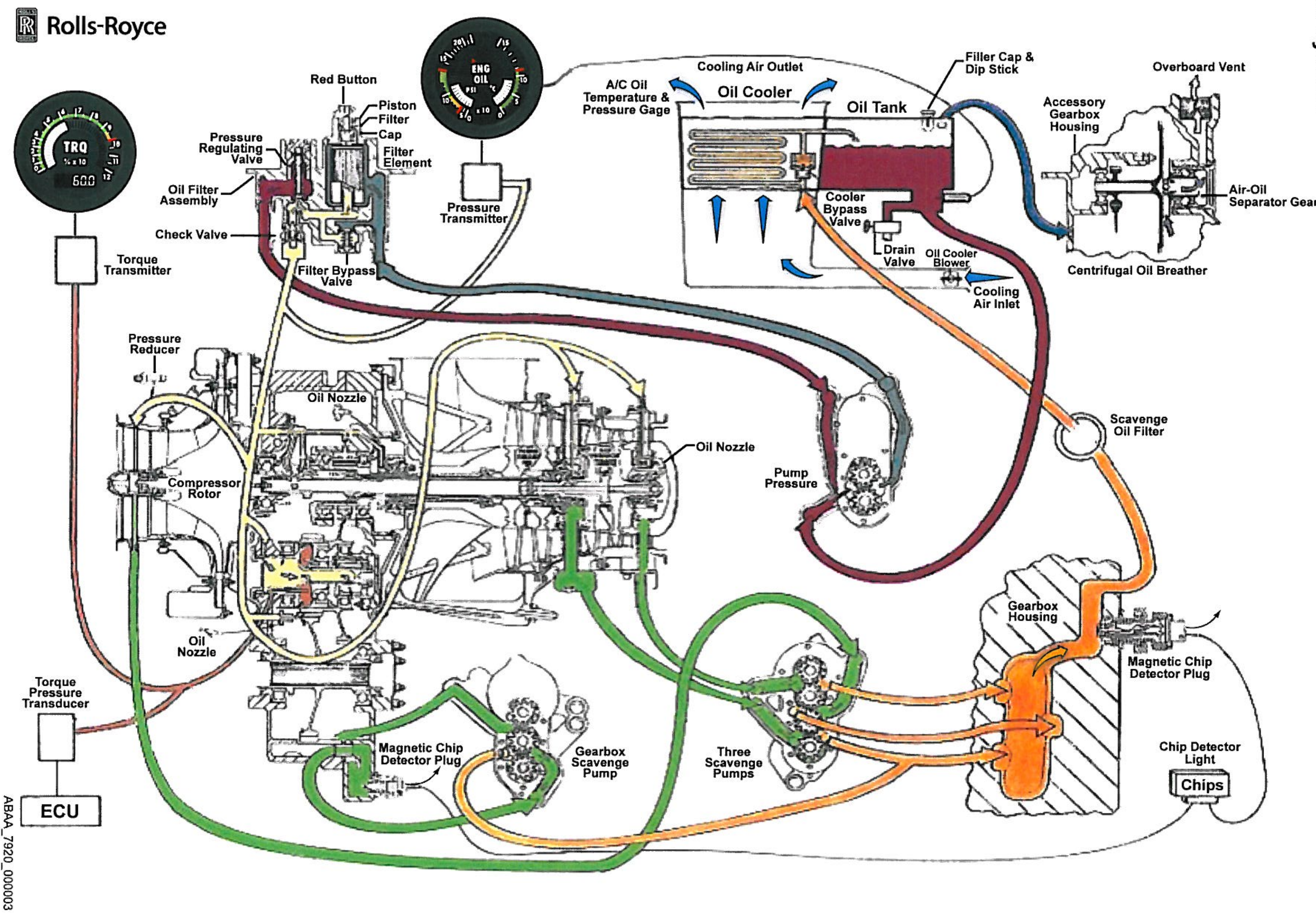
A ball bearing transmits helical gear thrust from the rotating helical torquemeter gear shaft to the non-rotating torquemeter piston. This piston has an anti-rotation pin that contacts a second anti-rotation pin on the flange of the support shaft. The piston is free to slide axially on the non-rotating and retained support shaft. This sliding piston serves as a variable position valve that will admit regulated engine oil pressure from the support shaft to the oil chamber formed by the piston and the flange on the support shaft. The flange also has an external groove that houses a piston ring and expander.

# Powerplant

## Torquemeter (Cont)

The support shaft and piston are arranged in such a manner that an increase in gear thrust increases the piston inlet port opening. Because the outlet openings of the piston remain constant, the pressure inside the oil chamber is directly proportional to torque. Pressure in the oil chamber is directed to the torquemeter pressure sensing port on the front side of the accessory gearbox. The chamber oil is transferred through the support shaft anti-rotation pin and filter screen that is located in this passage.

During a stabilized power condition, the axial thrust force acting on the piston is counter balanced by the torquemeter pressure in the oil chamber. When the torque output of the engine is increased, the result is increased axial thrust acting on the piston and an unbalanced condition on the piston. The piston moves forward slightly to increase the piston inlet port opening. With a larger opening, there is less restriction to the flow of oil into the oil chamber. As a result, the pressure in the oil chamber increases. The pressure in the oil chamber continues to increase until its force acting on the piston counter balances the axial thrust acting on the piston. When the two forces are equal, the piston stops moving and the pressure in the oil chamber is higher than it was prior to the torque increase. The torquemeter indicating system, sensing a higher pressure, registers the increased torque.

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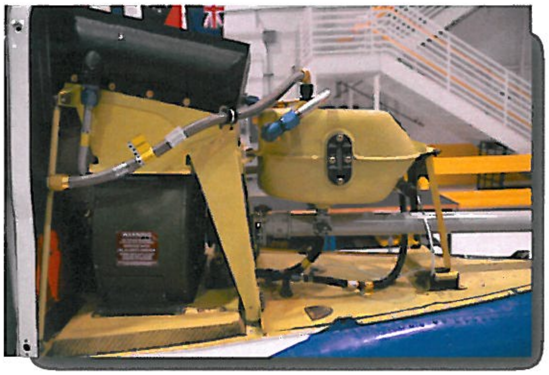
# Powerplant

## Oil System (Cont)



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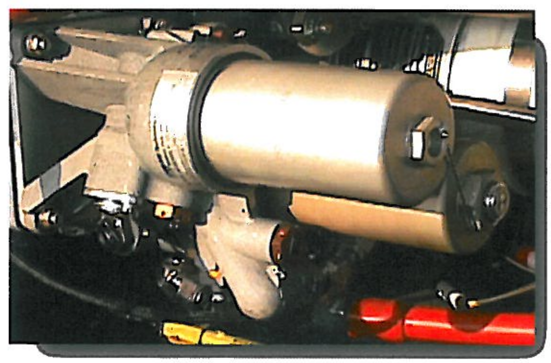
The engine incorporates a dry sump oil system with an externally mounted supply tank and an oil cooler located on the top aft deck of the intermediate section. The oil system supplies adequate lubrication and cooling as needed to the bearings, splines, and gears, regardless of the helicopter attitude or altitude. Oil is supplied from the tank to a pressure pump mounted within the engine accessory gearbox. The oil filter assembly, consisting of an oil filter, filter bypass valve, and pressure regulating valve, is located externally on the upper left-hand side of the gearbox. Magnetic chip detectors are installed in the bottom and right hand forward side of the gearbox. The engine is lubricated internally through oil jets and nozzles. External pressure and scavenge lines are used for the number 1 compressor bearing and the number 6/7 and 8 bearings in the gas producer and power turbine supports. The scavenge pumps remove oil from the various sump cavities and return this oil to the supply tank.

The CEFA, located on the lower left hand portion of the gearbox, provides both fuel and scavenge oil filtration within a single filter assembly. The oil filter portion of the assembly filters the oil prior to entering the oil cooler. A bypass valve allows the oil to bypass the filter in the event the filter becomes blocked.



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The assembly consists of an oil filter bowl, oil bypass valve, oil differential pressure indicator, and a disposable oil filter element.

The bypass valve and scavenge oil filter allow flow to go into a bypass condition if differential pressure across the filter is 8.8 to 10.8 PSI. The differential pressure indicator on the back of the filter bowl provides a visual indication of a filter bypass. The indicator is reset manually following replacement.

# Powerplant

## Oil System (Cont)


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A temperature bulb in the outlet side of the engine oil reservoir senses the temperature of oil from the tank. The temperature signal is sent to the engine oil temperature gauge through the aircraft wiring. 28 VDC powers the oil temperature indicating system and the circuit is protected by a 3-amp circuit breaker labeled ENGINE OIL TEMP.

Return oil is routed from the engine oil outlet port to the oil cooler. An impeller assembly is mounted on the tail rotor drive shaft and provides cooling air to the oil cooler. The oil passes through the cooler element and then returns to the tank.

The normal capacity of the engine oil tank is 6.0 U.S. quarts. The oil level is checked through a sight glass on the left side of the reservoir. The oil tank provides port openings for the supply tube, vent tube, scavenge tube, temperature bulb, and drain valve.

Oil pressure is monitored by routing regulated oil pressure through a transducer to the engine oil pressure gauge in the cockpit. The pressure signal is sent to the pressure input of the engine oil gauge through the aircraft wiring. 28 VDC powers the oil pressure indicating system and the circuit is protected by a 3-amp circuit breaker labeled ENGINE OIL PRESS.

## Throttle

ABAA_7610_000034

The twist grip throttle on the collective controls the mechanical movement to the HMU and consists of a flexible control cable that extends from the throttle arm on the rear of the collective stick to a bellcrank assembly mounted on the engine deck. A control tube is connected between the bellcrank and a lever mounted on the HMU.

# Powerplant

## Servicing

Certain oils conforming to following specifications are approved for use in engine:

| Specification | OAT Range |
|---|---|
| M1L-PRF-7808 (NATO 0-148) | Any OAT |
| MIL-PRF-23699 (NATO 0-156) | OAT above -4O°C (-40°F) |
| DOD-PRF-85734 | OAT above -40°C (-40°F) |

**NOTE**

Per Rolls-Royce, the preferred engine oils for MIL-PRF-23699 are Mobile Jet Oil 254 and Aeroshell 560.

Because of availability, reduced coking, and better lubricating qualities at higher temperatures, Rolls-Royce prefers qualified MIL-PRF-23699 oils.

Long-term use of DOD-PRF-85734 oil may increase probability of seal leakage in accessory gearbox.

Refer to BHT-407-FM-1 for engine oil limitations.

Capacity: 6.0 U.S. quarts (5.7 liters).

Engine oil tank is located under aft fairing, and access doors are provided for filling and draining oil tank. A sight glass and filler cap dipstick is provided to determine quantity of oil in tank.

# Powerplant

## FADEC

The FADEC system is designed to enhance flight safety and reduce pilot workload as well as provide other important benefits, such as engine automatic starts and precise control of main rotor speed, and features, such as redundant signal sensing, continuous monitoring, and self-diagnostics. Because much of the redundant design is transparent to the pilot, the caution/warning/advisory system has been expanded to alert to conditions resulting from the increased monitoring.

Although a FADEC system failure is unlikely, pilots and maintenance personnel must have an operational understanding of the FADEC system, along with sound knowledge of emergency and troubleshooting procedures. Bell Helicopter recommends personnel involved with the 407 familiarize themselves with the procedure for FADEC FAILURE, as prescribed in the Emergency Procedures Section of the Rotorcraft Flight Manual.

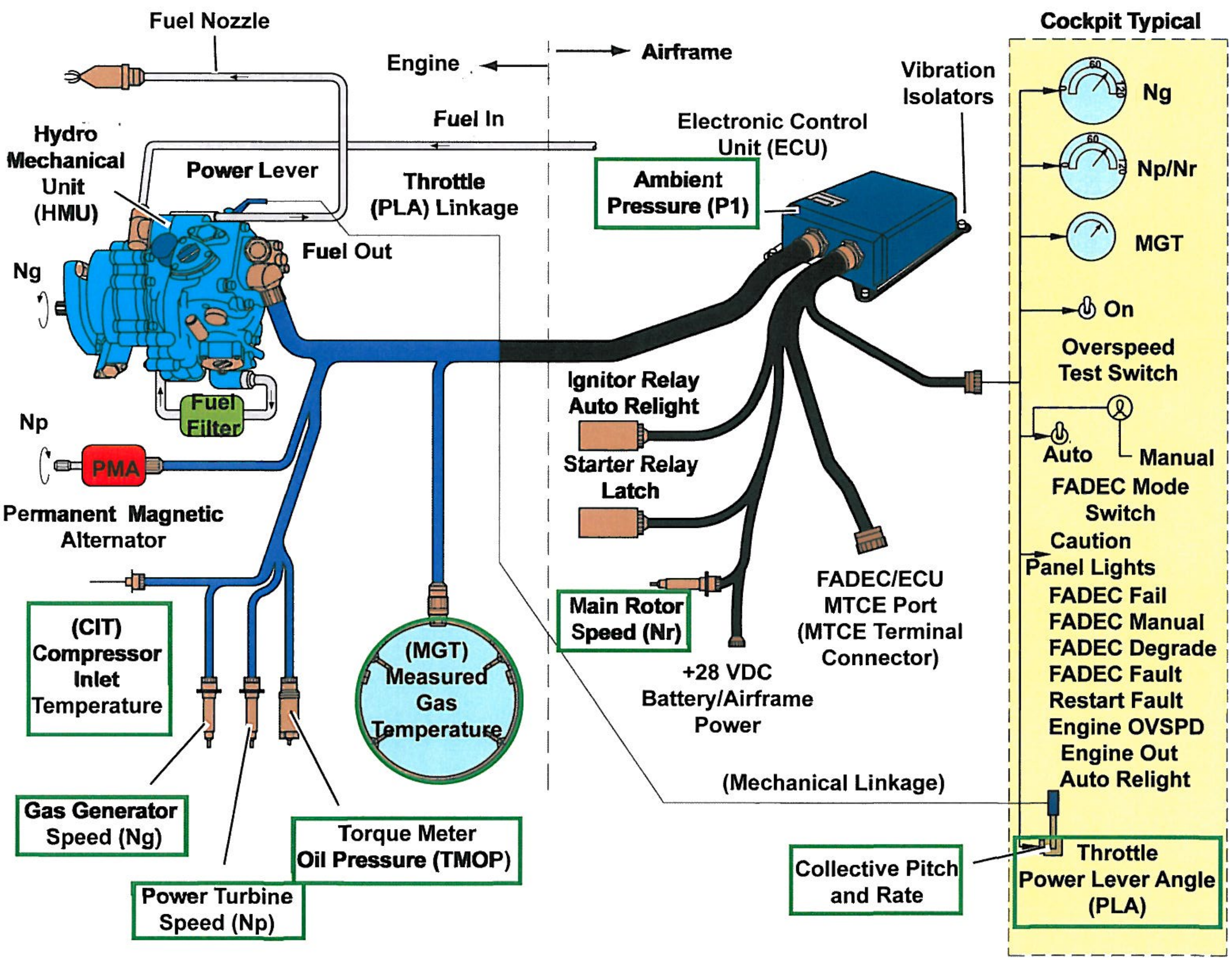
The 407 uses either a single channel control (5.202) with one microprocessor or the dual channel (5.356/5.358) FADEC system. There is also a MANUAL mode hydro mechanical backup. The FADEC system has two main components:

- Airframe-mounted ECU
- Engine-mounted HMU

The ECU monitors numerous internal and external inputs to modulate fuel flow and control engine speed, acceleration rate, temperature, and other engine parameters. The ECU provides inputs to the HMU to modulate fuel flow based on the continuous monitoring of the following: Measured Gas Temperature (MGT), Gas Producer speed ($N_G$), Power Turbine speed ($N_P$), Main Rotor speed ($N_R$), Engine Torque Meter Oil Pressure (TMOP), Collective Pitch (CP) and rate, Compressor Inlet Temperature (CIT), Ambient Pressure (P1), and Power Lever Angle (PLA)/throttle position. During flight in AUTO mode with the throttle in FLY detent position (PLA 70°), the FADEC has complete control over engine operation to maintain $N_R$ within limits.

The HMU consists of a two-stage suction fuel pump, fuel metering assembly, Auto/Manual changeover solenoid valve, electric overspeed valve, mechanical fuel shut off valve, hot start fuel solenoid valve, and altitude compensated bellows. The HMU provides fuel modulation via a stepper motor in AUTO mode and a Hydro Mechanical actuator in MANUAL mode.

The 28 VDC bus supplies electrical power to the FADEC ECU. When the engine achieves 85% $N_P$ and above, the engine power turbine driven PMA provides a dedicated power source for the FADEC ECU. The FADEC ECU selects the higher voltage source between 28 VDC bus and the PMA.

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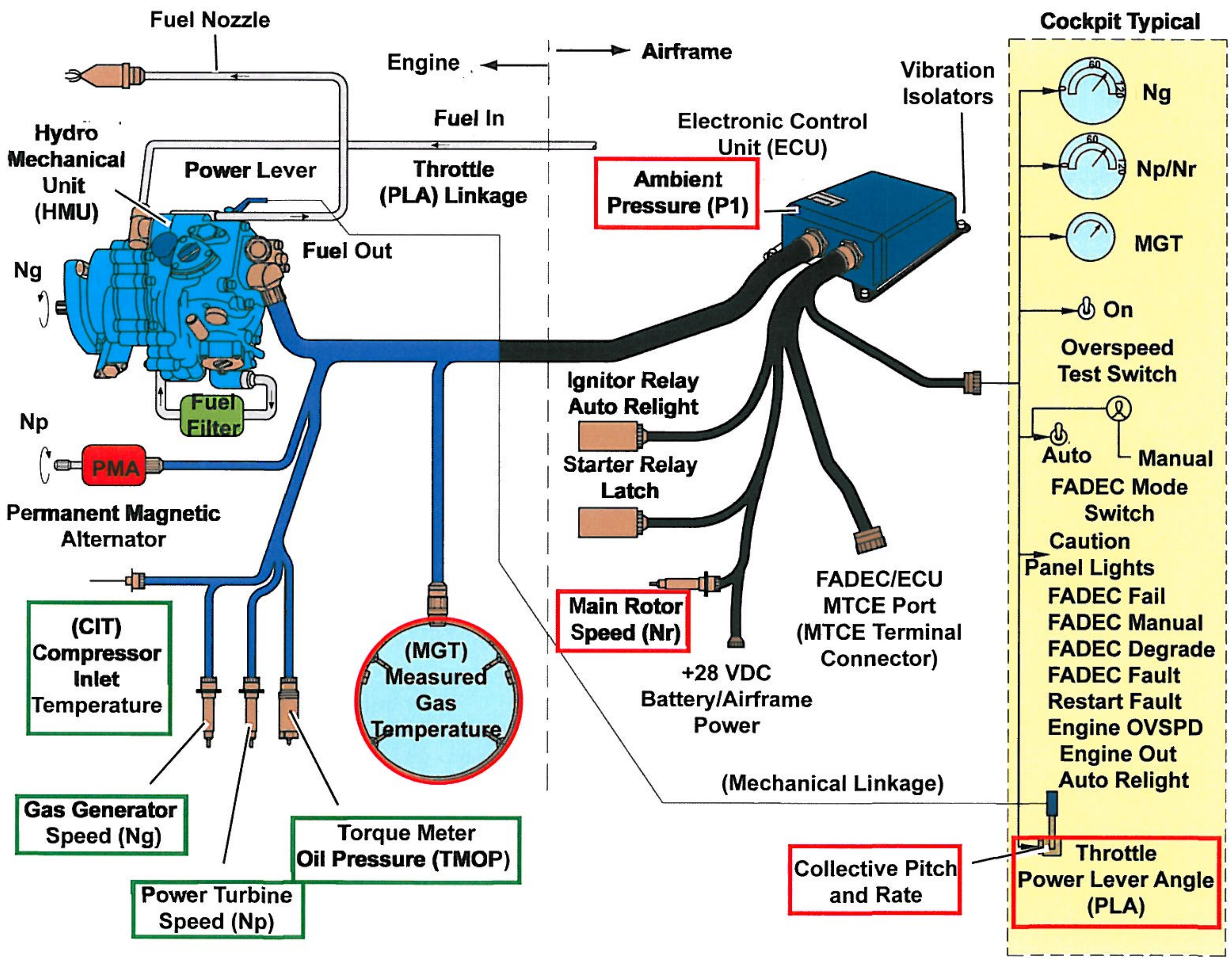


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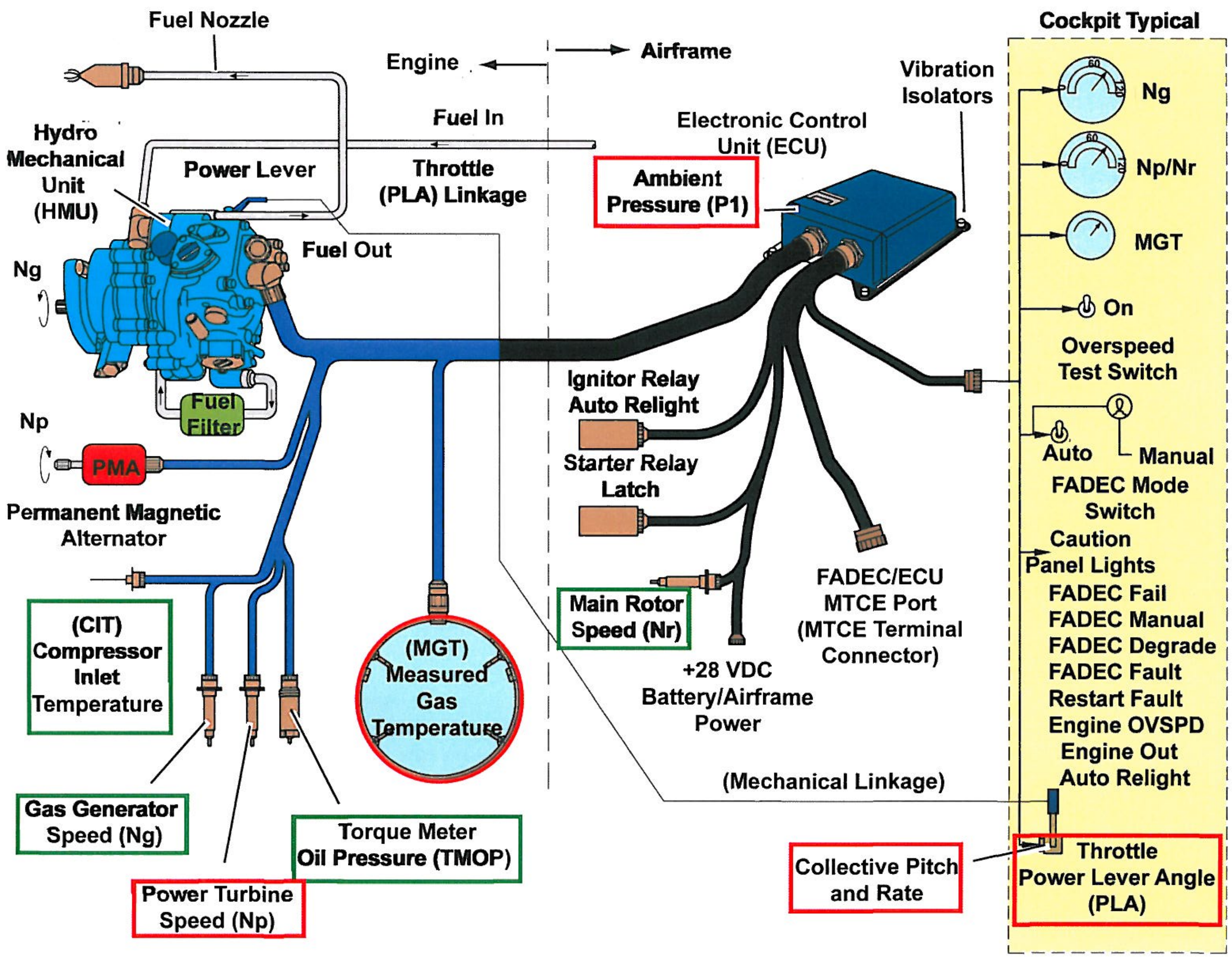


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# Powerplant

## Power Up Mode and Built In Test (BIT)

The FADEC system incorporates logic and circuitry to perform self-diagnostics. In general, the system checks sensors for continuity, rate, and proper range and discrete inputs for continuity. The system also monitors output drivers for current demand to sense failed actuators and open or shorted circuits. A FADEC power up check exercises output drivers and actuators to ensure system functionality and readiness. If the self-test detects any faults, the appropriate FADEC caution panel light illuminates to alert the pilot.

## Start in Auto Mode

An automatic start function is designed into the 407. Throttle modulation of fuel flow by the pilot is not required because fuel flow is controlled automatically by the FADEC. The system incorporates "Hot Start Abort Logic" to cut off fuel flow and abort a start if:

- Start MGT reaches 843°C (FADEC version 5.202/5.356) 885°C (FADEC version 5.358),
- At pressure altitudes less than 10,000 feet and if residual MGT is less than 82.2°C at initiation (5% $N_G$) of start
- Start MGT reaches 912°C (FADEC Version 5.202/5.356/5.358) at pressure altitudes greater than 10,000 feet
- And/or if residual $MG_T$ is greater than 82.2°C at initiation (5% $N_G$) of start,
- Voltage to the FADEC ECU drops below 10.3 VDC.

Although the start sequence is automatic, the pilot must monitor the start process and take appropriate action, if required. Therefore, both the throttle and start switch should be guarded until the start is completed. Do not initiate a start if FADEC-related caution panel lights are illuminated.

## Alternate Start Mode

This procedure may be used in hot and/or high altitude environments where aborted hot starts have been experienced and when prior troubleshooting has not revealed any engine maintenance issues. Refer to the BHT-407-FM-1 Section 2 Normal procedures, paragraph 2-5-B for the proper procedure.

## Start In Manual Mode

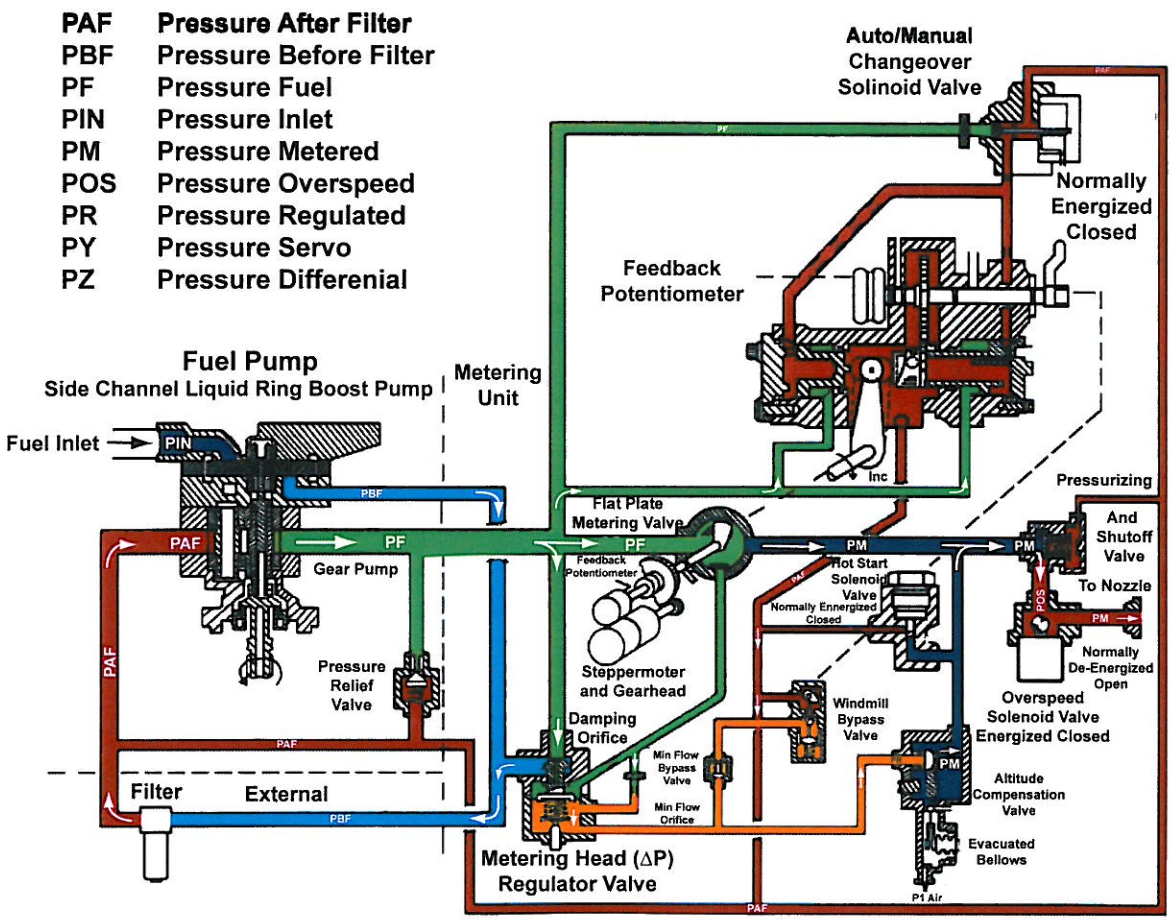
In accordance with the Rolls-Royce 250-C47B Operation and Maintenance Manual, Manual Mode starting on the ground is not authorized except for use under emergency conditions or under special permit from the local aviation authority. Refer to the Rolls-Royce 250-C47B Operation and Maintenance Manual to ensure adherence to all Manual Mode Operational Procedures. Automatic hot start abort features are not available in MANUAL mode.

# Powerplant

## In-flight Operation in Auto Mode

During flight in AUTO mode, with the throttle in FLY detent position (PLA 70°), the FADEC has complete control over engine operation to maintain $N_R$ within limits.

**Fuel System Hydromechanical (HMU) Schematic**
**250 Engine FADEC System**
**Automatic Electronic Operation**

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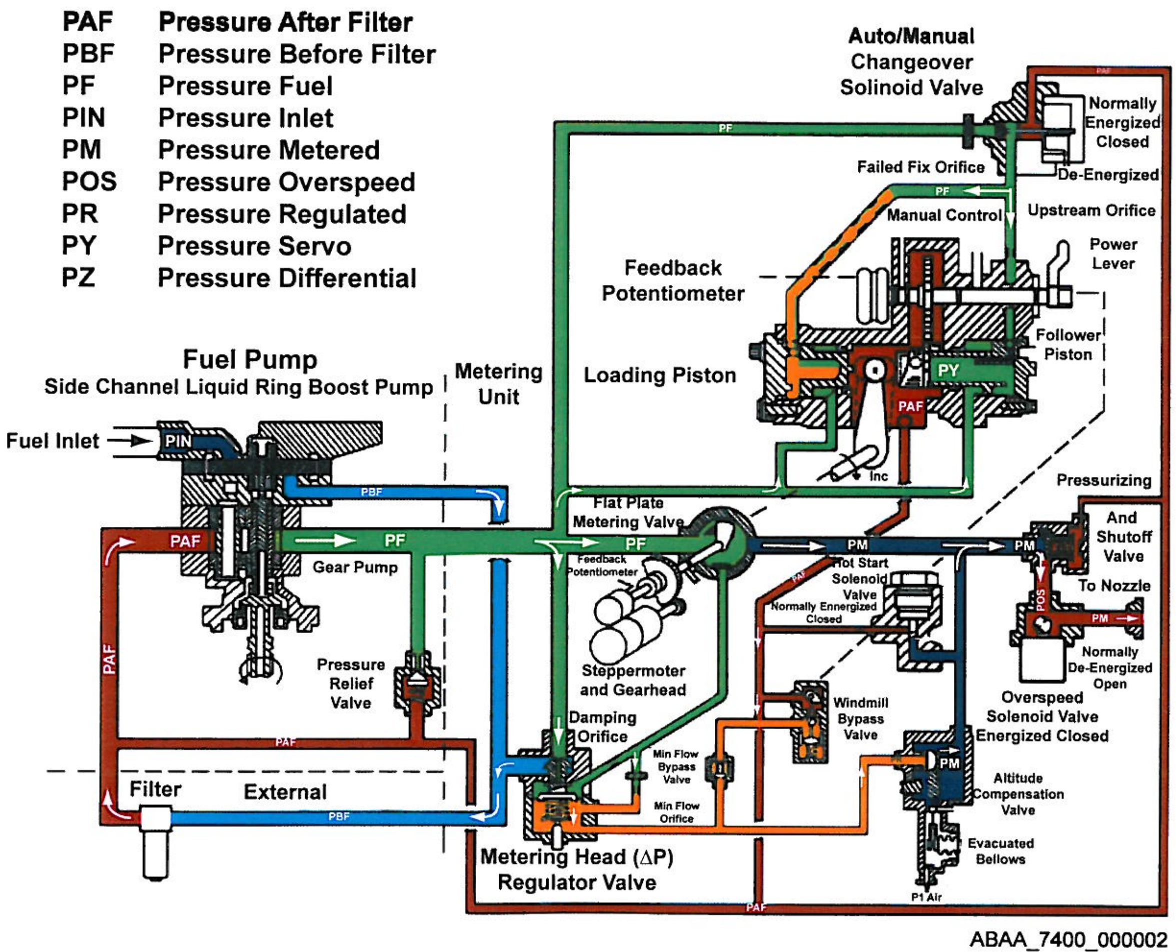
The ECU receives engine and airframe parameter inputs and cockpit command signals, processes them, and modulates the HMU stepper motor to provide adjustment to the Fuel Metering Valve and achieve desired engine performance. Precise electronic control provides automatic maintenance of rotor RPM throughout the normal range of operation and lessens pilot workload.

In AUTO mode, the FADEC can detect an engine flameout by sensing $N_G$ deceleration. Without any pilot action, the auto-relight sequence initiates by establishing a controlled fuel flow and activating the ignition system. The FADEC controls the MGT and accelerates the engine to the proper state.

## In-flight Operation in Manual Mode

In AUTO mode, the stepper motor receives electronic signals from the FADEC ECU and adjusts the fuel metering valve to maintain proper engine performance. In MANUAL mode, the fuel metering valve responds to the position of the throttle (PLA). In MANUAL mode, high-pressure fuel enters both sides of the HMU interior and moves the pistons from parked positions into contact with the fuel metering valve lever. Throttle manipulation changes PLA position to achieve variable fuel pressure in the follower piston side of the HMU. Varying fuel pressure in the HMU moves the fuel metering valve lever and valve to enable the pilot to effectively control engine RPM.

**Fuel System Hydromechanical (HMU) Schematic**
**250 Engine FADEC System**
**Manual Electronic Operation**

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# Powerplant

## FADEC System Faults

The FADEC ECU continuously monitors the FADEC system for faults and makes appropriate accommodations to continue operations. The FADEC controls eight lights in the caution/warning/advisory panel. A FADEC FAIL warning horn and FADEC FAIL/FADEC MANUAL warning lights alert the crew to any failure that requires pilot action in the ECU/HMU or in one of the input/output signals that significantly impacts the ECU or control of the HMU. If the detected failure does not significantly impair the function of the ECU, a FADEC DEGRADED caution light, FADEC FAULT caution light, RESTART FAULT advisory light, or combination thereof alerts the pilot.

FADEC does not communicate minor faults to the pilot with the engine running but identifies them as maintenance advisory faults, which display during shutdown via a FADEC DEGRADED light when the pilot places the throttle in the cutoff position and $N_G$ speed decays below 9.5%.

Additional detailed information regarding FADEC operation is in the Manufacturer's Data Section of the Rotorcraft Flight Manual. Pilots and Maintenance Personnel should research this section for a more thorough understanding of FADEC operation and characteristics.

## FADEC Software Version 5.202

FADEC faults fall into one of five types:

In-flight Faults:

1. FADEC fail to Manual (DRTM)
2. FADEC DEGRADED
3. FADEC FAULT
4. RESTART FAULT

A combination of these lights may occur, depending on the type of fault.

Maintenance Advisory Faults

FADEC DEGRADED light illuminates when $N_G$ decays to 9.5%.

## FADEC Software Version 5.356/5.358

In-flight Faults are:

1. Primary Governor Failure (reversion to the reversionary governor).
2. FADEC fail to Manual (DRTM)
3. FADEC DEGRADED
4. FADEC FAULT
5. RESTART FAULT

A combination of these lights may occur depending on the type of fault.

Maintenance Advisory Faults

FADEC DEGRADED light illuminates when $N_G$ decays to 9.5%.

# Powerplant

## FADEC Fail to Manual (DRTM) (5.202, 5.356, 5.358)

If the FADEC fails to the Manual Mode (DRTM) the pilot receives the following indications in the cockpit.

1. The FADEC Chime tone (audio) activates.
2. FADEC FAIL light illuminates.
3. FADEC MANUAL light illuminates.
4. RESTART FAULT light illuminates.
5. FADEC Mode Switch changes from AUTO to MAN

   To silence the FADEC Chime tone (audio), push the FADEC Mode switch one time. At this time, the AUTO RELIGHT light illuminates to indicate the igniter is operating.

### FADEC Degraded

Illumination of the FADEC DEGRADED light indicates loss of some feature of the FADEC system that may cause degraded performance. This may result in $N_R$ droop, $N_R$ lag, or reduced power capability. The ECU displays these faults, but operation continues in the AUTO Mode. The pilot should fly the helicopter smoothly and non-aggressively under these conditions.

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Illumination of the FADEC DEGRADED light after engine shut down and $N_G$ speed reaches 9.5% indicates a minor fault occurred during the previous engine run.

### FADEC Degraded and Restart Fault

In conjunction with the FADEC DEGRADED light, the RESTART FAULT light may also illuminate under certain fault conditions.

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### FADEC Fault

Illumination of the FADEC FAULT light indicates the PMA has failed. This light could also be an indication the automatic limiting circuits for overspeed protection may not be functional.

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### FADEC Fault and Restart Fault

Illumination of the FADEC FAULT and RESTART FAULT lights indicates the MGT automatic limiting circuit is not functioning.


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### Restart Fault

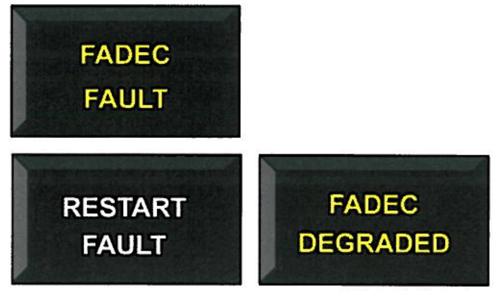
Illumination of the RESTART FAULT light indicates a subsequent automatic engine start may not be possible. The fault does not require immediate pilot action and should not affect performance of the helicopter.

ABAA_3141_000020

# Powerplant

## FADEC Degraded, FADEC Fault and Restart Fault (5.356, 5.358)

Illumination of these lights indicates failure of the FADEC primary governor and activation of the reversionary (backup) governor.



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Failure of the primary governor can cause $N_R$ droop, $N_R$ lag or reduced maximum power capabilities.

When the reversionary governor activates due to failure of the primary channel, the reversionary governor does not support the following:

1. Automatic start with starter and ignition operation
2. $N_G$ power modulation based on PLA)
3. Temperature limiting
4. Automatic relight
5. Surge detection, recovery and avoidance
6. Quiet mode

## Engine Ovspd

$N_P$ overspeed limiting is available in both the AUTO and MANUAL modes by independent analog circuits integral to the ECU.

ABAA_3141_000006

**FADEC software version 5.202**

Activation of the ENGINE OVSPD light occurs when $N_P$ or $N_P$ versus torque is above the continuous limit of 102.12 $N_P$ @ 100% torque to 108.6 $N_P$ at 0% torque.

**FADEC software version 5.356/5.358**

Activation of the ENGINE OVSPD light occurs when $N_P$ reaches or exceeds 102.1 for 2.5 seconds or exceeds 107.3% $N_P$.

**FADEC software version 5.202 or 5.356**

If the ECU has recorded an $N_P$ exceedances, the FADEC DEGRADED light illuminates after engine shutdown when $N_G$ speed reaches 9.5%.

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**FADEC software version 5.358**

If the ECU has recorded an $N_P$ exceedance, the FADEC FAULT light illuminates in flight and remains illuminated until engine shutdown.

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After engine shutdown, the FADEC DEGRADED and ENGINE OVSPD lights illuminate.

ABAA_3141_000017

ABAA_3141_000006

# Powerplant

## Overspeed System Test

The operator should accomplish a functional test of the Engine Overspeed on the first shutdown of the day. The following are required to perform the test:

1. Collective pitch is below 10% travel.
2. Throttle position is at IDLE.
3. $N_G$ is between 60% and 66%.
4. $N_P$ is less than 75%.

Press and hold the OVSPD TEST button switch for a minimum of one second but less than 10 seconds. Release the test button switch; the ENGINE OVSPD light illuminates briefly and the engine shuts down. Continue with shutdown procedures.

## FADEC Fault Codes

When the ECU detects a FADEC fault, either in flight or after engine shutdown, maintenance action is required prior to further flight. The preferred method of determining FADEC faults and/or exceedances is with the ECM-35A maintenance terminal. If the maintenance terminal is not available, use the FADEC Maintenance Advisory Mode feature (Fault code checking) to identify faults and/or exceedances. This feature allows operators to identify faults through a sequence of flashing light displays on the cockpit Caution, Warning, and Advisory panel (CWAP).

The different types of FADEC faults are:

1. Current Faults (faults that occurred after engine shutdown)
2. Last Engine Run Faults (faults that occurred while the engine was running)
3. Accumulated Faults (Fault History)
4. $N_P$ overspeed exceedance

The pilot can perform FADEC Fault Code checking by using the flashing light sequence of the CWAP. If checking for Last Engine Run Faults, the throttle must be in the CLOSED position. If checking for Current Run Faults, the throttle must be in the IDLE position

The Manufactures Data section of the flight manual (BHT-407-MD-1) provides the procedure for performing the Maintenance Advisory Mode feature (FADEC Fault Code checking).

# Powerplant

## Piston Parking

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If the pilot does not follow engine shutdown procedures correctly, the manual mode pistons will begin to engage if the battery switch is turned OFF before the $N_G$ stops turning. After reapplication of electrical power to the aircraft and completion of the Built In Test (BIT), the FADEC DEGRADED and RESTART FAULT lights illuminate. If the pilot positions the throttle from CUTOFF to IDLE, the FADEC DEGRADED and RESTART FAULT lights remain illuminated. The pilot shall accomplish the piston parking procedure prior to engine starting.

The Piston Parking procedure is in the Manufactures Data section of the flight manual (BHT-407-MD-1).

1. Position throttle to CUTOFF
2. Igniter circuit breaker – OUT
3. BATT switch – ON
4. Power up check – Complete
5. FADEC mode switch – MAN
6. Motor engine (throttle in CUTOFF) – 10 seconds
7. Wait for $N_G$ to decay to 0% $N_G$.
8. FADEC mode switch – AUTO
9. Motor engine (throttle in CUTOFF) – 10 seconds
10. Wait for $N_G$ to decay to 0% $N_G$.
11. BATT switch – OFF
12. Igniter Circuit breaker – IN

# FUEL SYSTEM

THE BELL TRAINING ACADEMY

Est. 1947

# Table of Contents

## Fuel System

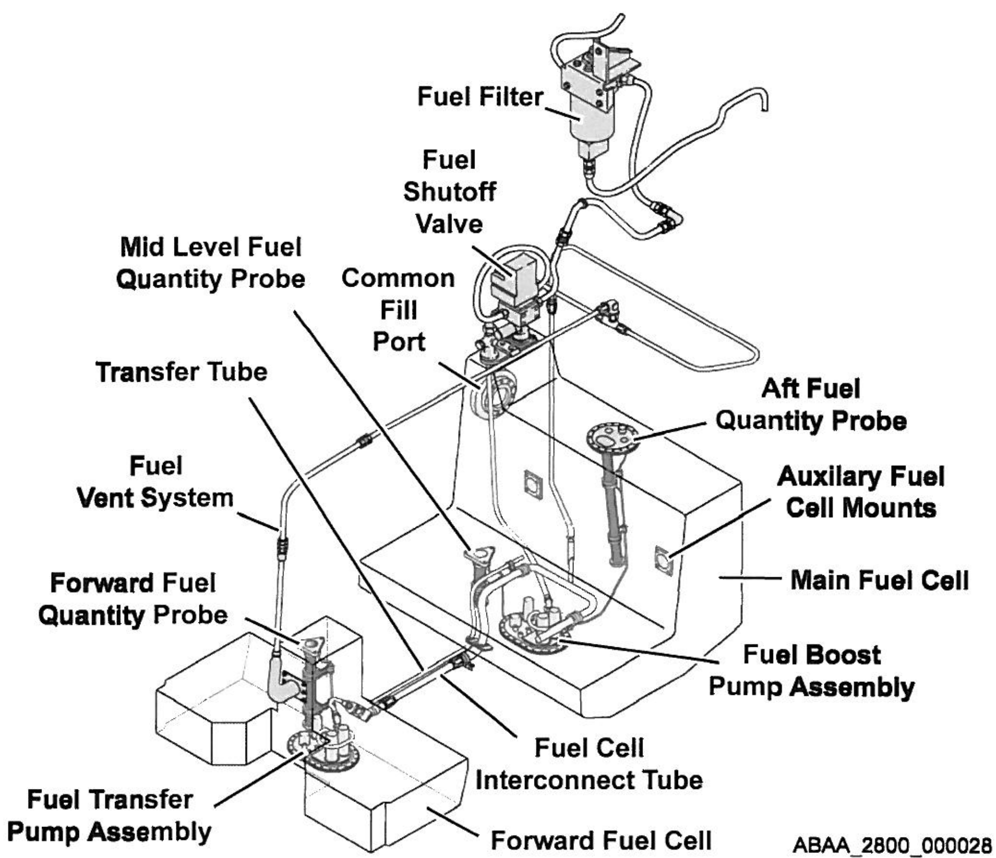
# FUEL SYSTEM

## System Description

The basic 407 fuel system consists of two fuel cells, pumps for transferring fuel between the two cells, pumps for supplying fuel to the engine, a fuel cell venting system, a fuel cell quantity indicating and low level detection system. Each fuel cell has an electrically-operated fuel drain solenoid valve. An electrically-operated fuel shut off valve is located above the aft fuel cell. Several caution lights indicate abnormal operation of their related components. Fuel is transferred between the aft and forward cells by two means:

- A gravity transfer line permits fuel to transfer from the aft to forward cell when the level of fuel in the aft cell is above the height of the standpipe.
- A pressure transfer line transfers fuel from the forward transfer pumps to the aft (Main) cell.

The pressure system has two electrical pumps mounted to the sump plate in the bottom of the aft cell. The transfer system has two electrical pumps mounted to the sump plate in the bottom of the forward cell.



# FUEL SYSTEM

## System Components

### Fuel Cells

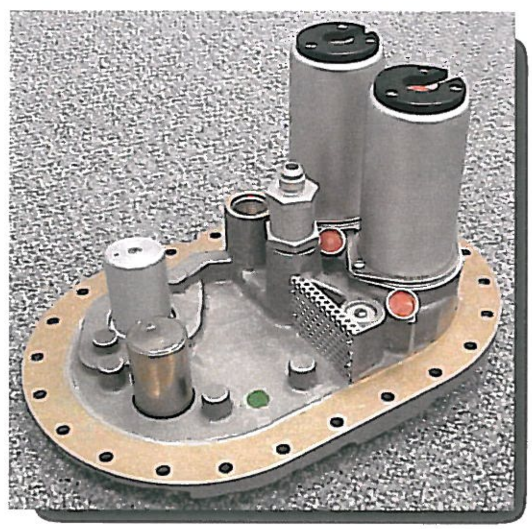
The basic aircraft has two crash-resistant bladder type fuel cells, which are serviced through a filler port located on the right side of the aircraft. These cells have a combined capacity of 130 US gallons (492.05 Liters), usable 127.8 US gallons (483.7 Liters), 869 LBS. The aft cell located underneath and behind the aft passenger seats, has a total capacity of 92 US gallons (348 Liters), 626 LBS. The forward cell located underneath and between the middle passenger seats, has a capacity of 37.6 gallons (142 Liters), 256 LBS. Both cells are connected by gravity and return lines on the floor of the main cabin covered by a protective fiberglass cover.

### Auxiliary Fuel Cell

Every aircraft has provisions for an optional 19 US gallons (72 Liters) 129 LBS. The auxiliary fuel cell can be installed in the baggage compartment. The auxiliary cell has no fuel capacitance probe or fuel pump. The auxiliary cell is mounted in such a way that the cell itself and the fuel do not add any additional weight to the baggage floor. Installation of the AUX TANK does not reduce the baggage compartment maximum load of 250 pounds.

### Forward Fuel Transfer Pumps

Two transfer pumps mounted on a forward sump plate, located on the floor of the forward cell, transfer all usable fuel to the aft cell. Each pump has a pressure switch that controls a transfer pump caution light on the CWAP. With the pump circuit breaker switch on and the transfer pump producing normal pressure, the related caution light extinguishes to indicate proper operation.



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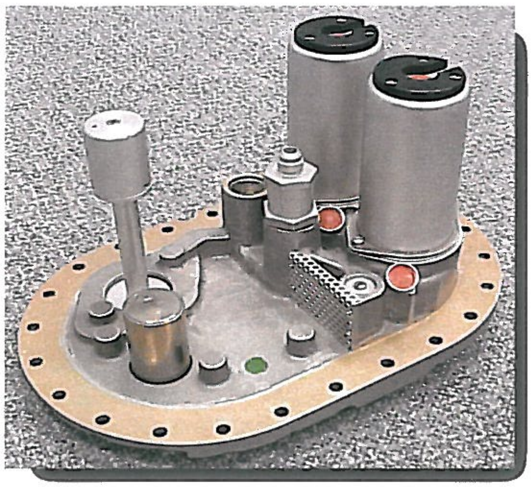
If a transfer pump fails, the pressure switch senses low pressure and causes the respective FUEL XFR caution light to illuminate on the CWAP.

Each transfer pump sends fuel through a one way check valve located at the outlet of each pump. This check valve ensures that if either transfer pump becomes inoperative, fuel is pumped to the aft cell and not through the inoperative pump back into the forward fuel cell.

Individual fuel lines are connected to a 'T' fitting that connects the supply from each pump to a single line.

### System Components (Cont)

#### Fuel Boost Pumps



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Two boost pumps mounted on an aft sump plate are located on the floor of the aft cell. Each individual pump can lift all usable fuel past the fuel pressure transducer, through the fuel valve, then through two filters, and on to the HMU high pressure pump. A pressure switch in each boost pump controls a caution light on the CWAP. With the pump circuit breaker switch on and boost pump operating normally, the related caution light should be extinguished.

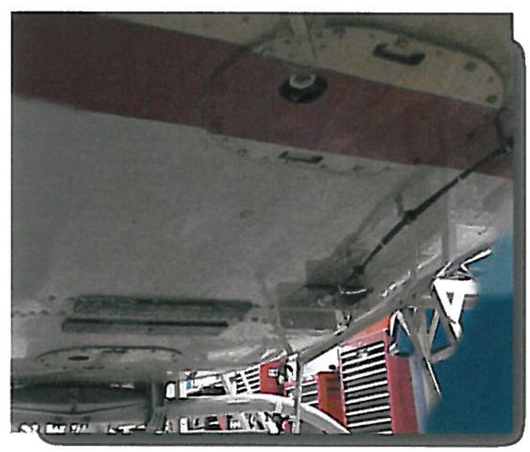
Each boost pump sends fuel through a one-way check valve located at the outlet of each pump. This check valve ensures that if either boost pump becomes inoperative, fuel is pumped to the engine and not through the inoperative pump back into the aft fuel cell.

Individual fuel lines are connected to a 'T' fitting that connects the supply from each pump to a single line.

#### Fuel Sump Drain Valves

Fuel drain switches provide a means of draining fuel from both the forward and aft cells. Sump drain valves are located on the forward and aft sump plates and are electrically operated by means of the drain switches located aft of the right-side passenger door. Drain switches are wired through the fuel valve switch on the instrument panel.

The fuel valve switch must be in the OFF position to operate either fuel cell drain valve.



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#### Fuel Pressure Transducer

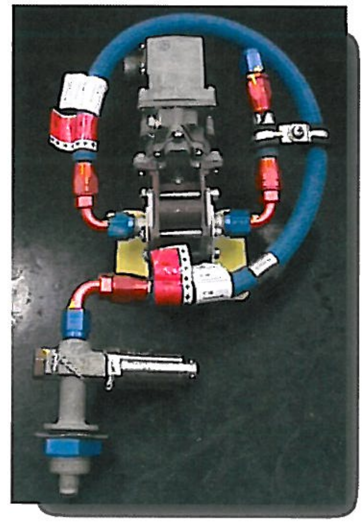
Transducers convert a wet line (fluid passing thru tubing) signal into an electrical signal. Aft cell boost pump fuel pressure is displayed in PSI (pounds per square inch) on the Fuel Pressure indicator by means of a fuel pressure transducer.

# FUEL SYSTEM

## System Components (Cont)

### Fuel Valve

The fuel valve is located on the right hand side above the refueling port and is electrically controlled by a switch located on the lower right corner of the instrument panel. A fuel valve caution light illuminates when the fuel valve is in transit or has stopped somewhere between the full OPEN or full CLOSED position.



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### Fuel Filter

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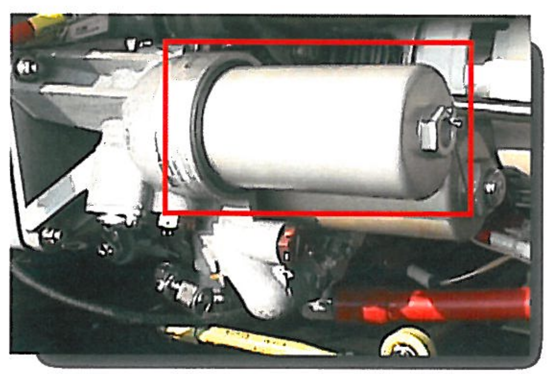
The airframe fuel filter is located on the forward firewall on the right side of the powerplant. It is equipped with a built-in filter bypass valve and impending bypass switch that is electrically connected to a caution light on the CWAP. At .875 ± .125 PSID (differential fuel pressure), the airframe filter caution light illuminates. The filter bypasses at 3.75 ± 25 PSID, however there are no further cockpit indications. A red press-to-test button is located on the top of the filter to check the electrical connection to the fuel filter caution light.

### CEFA Fuel Filter

The CEFA is located on the lower left hand portion of the engine gearbox. The fuel filter portion consists of a fuel filter bowl, fuel bypass valve, fuel differential pressure indicator, manifold assembly and a disposable fuel filter element. The impending bypass indicator extends when differential pressure is 2.5 PSI (±.5 PSI) and bypasses at 3.4 PSI. The impending bypass indicator is located on the lower outboard side of the manifold assembly.



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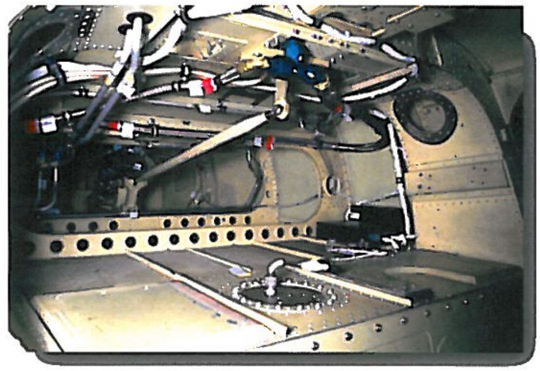
# FUEL SYSTEM

## Fuel Quantity System

A signal conditioner is located on the electrical shelf behind the rear forward-facing seat. Three separate fuel capacitance probes (one located in the forward cell and two located in the aft cell) send signals indicating fuel quantity (based on fuel levels and density) to the signal conditioner. A microprocessor in the signal conditioner uses the information provided by the three fuel capacitance probes to compute the weight of the fuel, and displays the fuel quantity in pounds on the fuel quantity gage.



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The fuel signal conditioner senses the auxiliary cell's presence by a micro switch in the baggage compartment. With the AUX TANK installed, the signal conditioner recalculates fuel quantity in the main (aft) cell, including the additional 19 gallons in the auxiliary tank using the aft fuel capacitance probe. The combined total quantity of fuel displays on the fuel gauge.

With the AUX TANK installed, fuel capacity increases to 149 US gallons (563 Liters), 1013 LBS.

The signal conditioner carries out a four-second power Built-In-Test (BIT) () when the unit is first provided power. If a failure is detected during the BIT, the signal conditioner blanks the fuel quantity Indicator. The signal conditioner also performs a continuous BIT whenever the unit is powered. If an error is found during the continuous BIT, the signal conditioner blanks the fuel quantity Indicator display.

### Fuel Low Light

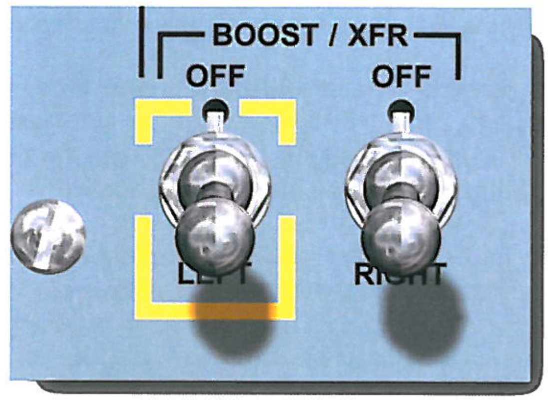
A float type low-level switch is mounted on the aft fuel cell sump plate to detect the fuel low condition. The input from the low level switch is sent to the fuel signal conditioner, which provides a 13 ± 3 second delay to reduce the possibilities of intermittent annunciator flickering due to fuel sloshing. The FUEL LOW light illuminates when approximately 100 ± 10 pounds of useable fuel remains in the AFT tank.

# FUEL SYSTEM

## Fuel Quantity System (Cont)

### LEFT and RIGHT BOOST / XFR Switch

The left and right fuel BOOST / XFR (transfer) switches are a circuit breaker toggle design located on the overhead console. Each switch powers a set of pumps. The left BOOST / XFR pump switch controls the left boost pump in the aft fuel cell and the left transfer pump in the forward cell. The right BOOST / XFR pump switch controls the right boost pump in the aft cell and the right transfer pump in the forward cell. In normal ground run and in-flight conditions, the aircraft battery switch is in the ON position. DC power (battery and generator) is connected to the single aircraft bus powering all items wired to the bus, including the left boost and transfer pumps. In the event the battery switch is positioned to OFF during aircraft operations (fuel valve on), an alternate circuit provides electrical power directly from the battery to the left boost and left transfer pumps.

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The LEFT FUEL BOOST / XFR switch is outlined by a yellow border to identify that it has an alternate circuit. In the event a short circuit or battery hot condition occurs in the helicopter and all DC bus power (battery and generator) is shut off, it is desirable to maintain the operation of one transfer and one boost pump, allowing access to all fuel in the fuel cells. To enable this feature, the fuel valve switch must be positioned to ON, the left fuel boost / XFR circuit breaker switch must be ON, and the battery switch must be turned OFF.

## Fuel Transfer System

When the forward fuel cell nears depletion, the signal conditioner uses programmed time delays to control the transfer pumps and lights. The time delay allows continued operation of the transfer pumps to ensure all of the fuel in the forward cell is transferred to the aft main cell. When the forward fuel cell is empty, approximately 193 pounds of fuel remain in the aft fuel cell.

### S/N 53000 Thru 53174

When the forward fuel cell nears depletion, the signal conditioner uses time delays of 2½ minutes to control the transfer pumps and caution lights. The time delay allows continued operation of the transfer pumps to ensure all of the fuel in the forward cell is transferred to the aft cell.

The caution lights remain operational during the time delay and illuminate if pump pressure drops. Once the time delay periods end, the transfer pumps and the L/FUEL XFR and R/FUEL XFR light circuits are deactivated.

The transfer pump and light circuits stay inoperative as long as the forward fuel cell is empty.

# FUEL SYSTEM

## Fuel Transfer System (Cont)

**S/N 53175 and Subsequent**

Activation of either annunciator circuit is dependent on the relationship between the fuel quantity in the forward fuel cell and the total quantity of the fuel system.

(Refer to the table on the following page). When the fuel signal conditioner detects a forward fuel quantity of less that 25 pounds and a total quantity of less than 250 pounds for 10 consecutive seconds, L/FUEL XFR or R/FUEL XFR lights are deactivated to prevent intermittent light flickering while remaining fuel is transferred from the forward cell to the aft cell.

When the forward cell capacitance probe and the low level detector sense a low fuel condition, input signals to the fuel conditioner are removed. With input signals removed, the fuel signal conditioner utilizes a 360 second time delay prior to disabling the caution light. The pumps continue running for 360 seconds to ensure all forward cell fuel is transferred to the main fuel cell. The annunciator and transfer pump circuits stay inoperative until the fuel system is refueled.

The L/FUEL or R/FUEL XFR light illuminates when its respective fuel pressure switch senses a decreasing boost pump pressure of 1.5 PSI ± 0.5 PSI providing the proper conditions are met.

# FUEL SYSTEM

## Fuel Drain Procedures

(Excerpt from the Model 407 Flight Manual)

FUEL SUMP

Drain fuel sample as follows:

a. Left and Right FUEL BOOST/XFR circuit breaker switches - OFF

b. BATT switch — BATT (ON).

c. FUEL VALVE switch — OFF.

d. PUSH the FUEL CELL DRAIN button(s), (on the right aft side of the aircraft fuselage) — press, drain sample, then release.

Airframe Fuel filter

Drain and check before first flight of the day as follows:

a. FUEL VALVE switch — ON.

b. Left and Right FUEL BOOST/XFR circuit breaker switches — ON.

c. Fuel Filter Drain Valve — Open, drain sample, then close.

**NOTE**

Filter test button is located on top of fuel filter.

Fuel filter test button — Press and check FUEL FILTER caution light illuminated. Release switch and check light extinguished.

a. FUEL VALVE switch — OFF.

b. BAT switch — OFF

# FUEL SYSTEM

## Fuel Flow and Charts

Fuel flow charts present aircraft fuel consumption during level flight as a function of altitude, OAT, airspeed, and gross weight. These charts, which are based on estimates and limited flight test data, are applicable to the basic helicopter with all doors installed and without any optional equipment that appreciably affects lift, drag, or power available. This data does not include effects of bleed air heater, ECS (air-conditioning), particle separator purge, or anti-ice operation on fuel consumption. Also, fuel consumption may vary between engines under the same operating conditions. Bell recommends the operator conduct fuel consumption checks to adjust the presented data as necessary.

The expanded performance section of the Manufacturer's Data provides Fuel Flow vs. Airspeed charts ranging from sea level to pressure altitude 17,000 feet.

EXAMPLE:

**Problem**: What is the fuel burn for 4600 pounds, at 4000 feet pressure altitude, and 7° C, at 120 knots indicated airspeed?

**Solution**: Find the appropriate chart, and begin by locating the indicated airspeed of 120 knots at the bottom of the chart. Locate the aircraft weight of 4600 pounds (lower left of chart center), then follow the curved line to the right and upwards till you intersect the appropriate indicated airspeed line of 120 knots. Proceed horizontally to the LEFT until you intersect the FUEL FLOW LB/HR (fuel flow pounds per hour) of approximately 315 lbs. per hr. You may continue further LEFT and read the approximate aircraft torque for these conditions.

### Long Range Cruise (LRC)

The solid line labeled LRC indicates the optimum long-range cruise airspeed for best fuel economy. LRC is not depicted where it would occur above the continuous transmission limit line.

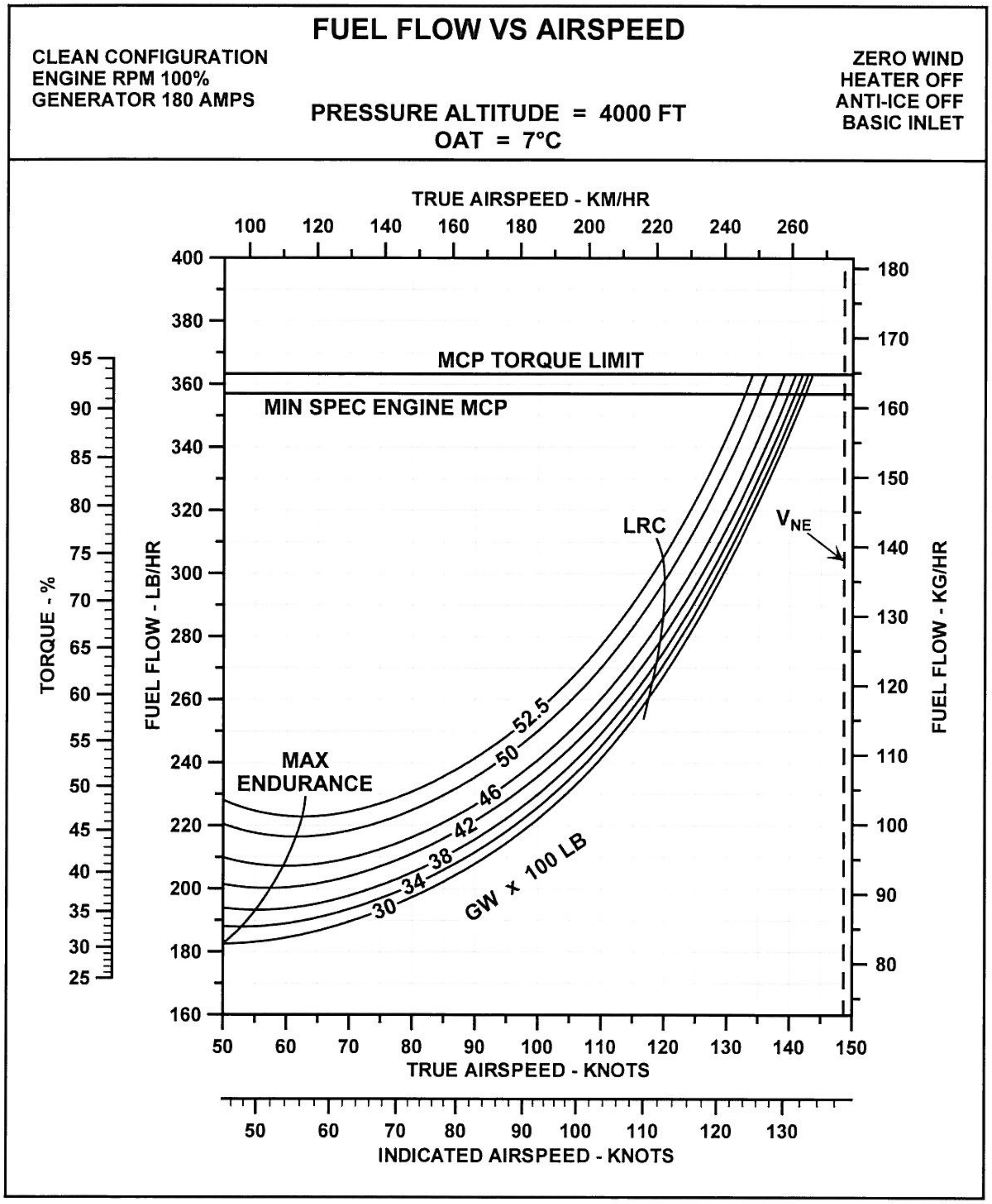
The long range cruise airspeed for 4600 pounds is found the same way as above, however enter the appropriate chart at the aircraft gross weight of 4600 pounds and then proceed along the curved line to the right and upwards, till you intersect the LRC line. Proceed DOWN vertically to the bottom of the chart and read the aircraft indicated (112 knots) or true airspeed (118 knots) for these conditions.

### Maximum Endurance (Max End)

The solid line labeled Max End indicates the aircraft gross weight and airspeed relationship that would yield the minimum fuel burn and give the aircraft the most time aloft.

# FUEL SYSTEM

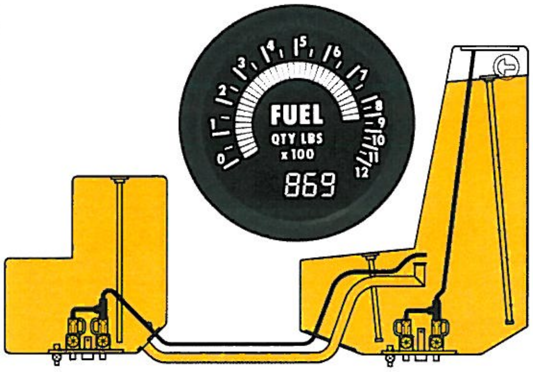
## Fuel Flow vs. Airspeed Chart



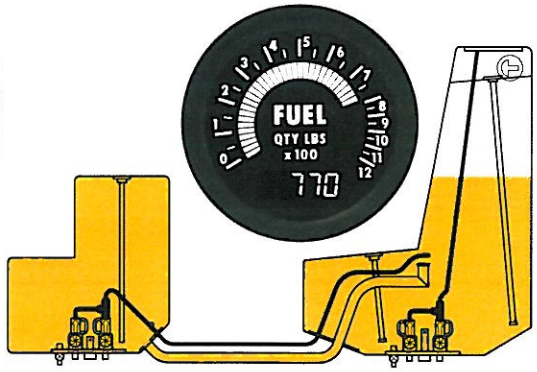

ABAA_2800_000026

# FUEL SYSTEM

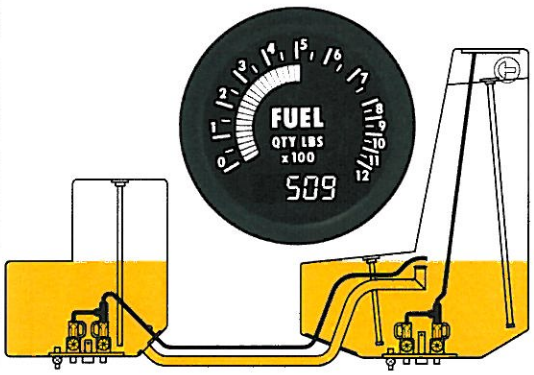
## Fill and Burn



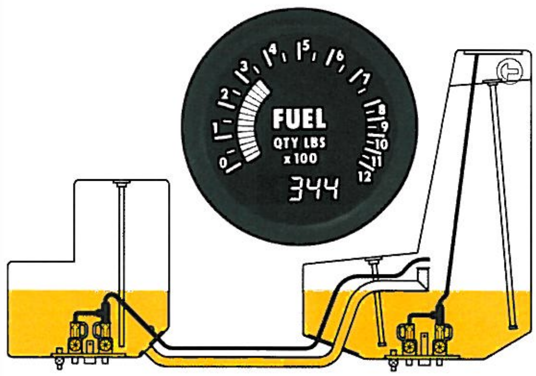

Full Fuel (869 LBS) With Transfer pumps operating, fuel is circulating from the forward cell through the transfer line to the aft cell then returns to the forward cell.



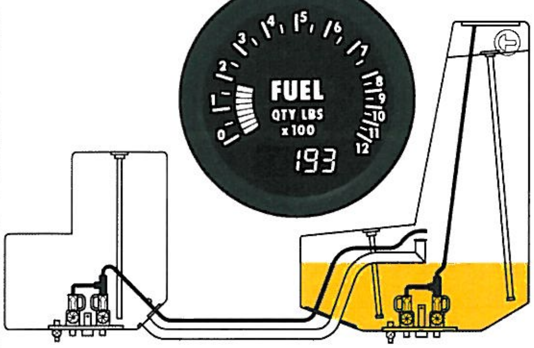

Partial Fuel (770 LBS) Fuel is consumed from the aft tank. At this point the fuel transfer from the forward cell starts. CG is beginning to move forward.



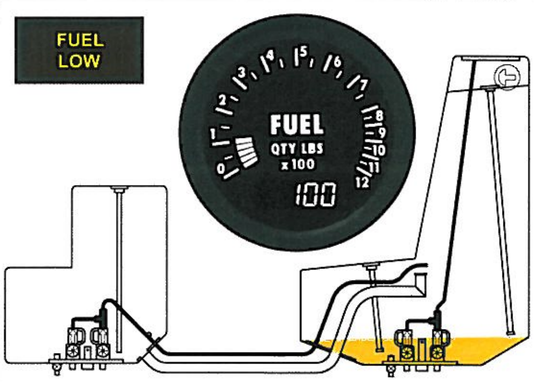

Partial Fuel (508.6 LBS) Fuel is consumed equally from the aft and forward cells. This point is the most forward CG during fuel burn.




Partial Fuel (344.1 LBS) Fuel is consumed from the forward cell until depleted. CG moves aft during this portion of the fuel burn.




Partial Fuel (193.1 LBS) The forward cell is empty, the remainder of the fuel in the aft cell is consumed. This point is the most aft CG during fuel burn.

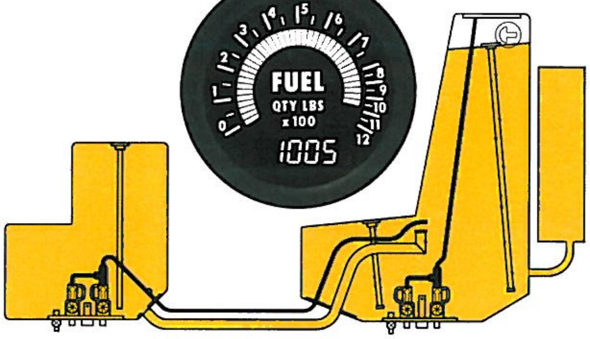



Partial Fuel (100 ± 10 LBS) Fuel low light will illuminate. At this point the CG will shift slightly forward.

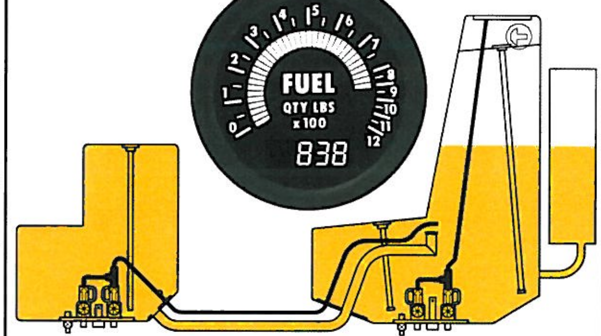
ABAA_2820_000030

# FUEL SYSTEM

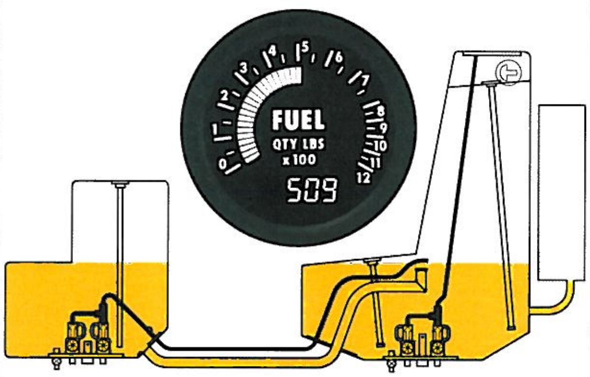
## Fill and Burn (Cont)



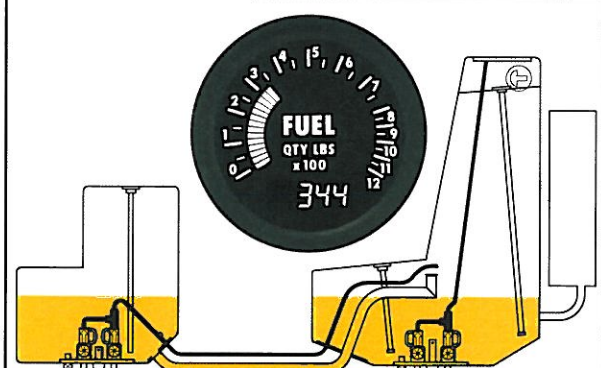

Full Fuel (1005 LBS) With Transfer pumps operating, fuel is circulating from the forward cell through the transfer line to the aft cell then returns to the forward cell. Most Aft C.G.



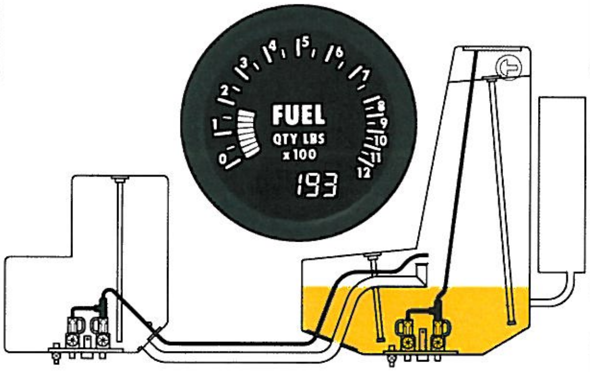

Partial Fuel (838 LBS) Fuel is being consumed from the aft and aux cells. At this point the fuel transfer from the forward cell starts.



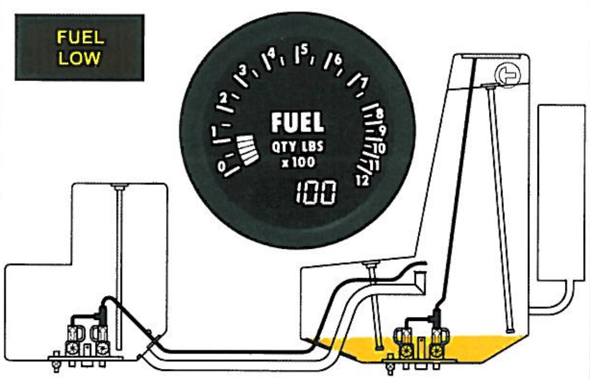

Partial Fuel (508.6 LBS) Fuel is consumed equally from the aux aft and forward cells. This point is the most forward CG during fuel burn. Aux Tank is empty.




Partial Fuel (344.1 LBS) Fuel is consumed from the forward cell until depleted. CG moves aft during this portion of the fuel burn.




Partial Fuel (193.1 LBS) The forward cell is empty, the remainder of the fuel in the aft cell is consumed.




Partial Fuel (100 ± 10 LBS) Fuel low light will illuminate. At this point the CG will shift slightly forward.

ABAA_2820_000030

# ELECTRICAL SYSTEM

THE BELL TRAINING ACADEMY

Est. 1947

# Table of Contents

## Electrical System



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# ELECTRICAL SYSTEM

## General

The helicopter is equipped with a 28VDC electrical system. A nickel-cadmium (Ni-Cad) battery, a starter/generator and/or an external power source provide aircraft electrical power.

Controls for the electrical system are located on the overhead console and instrument panel. Control relays, power relays, generator control unit, and circuit breakers control and regulate the voltage transfer. Malfunction monitoring circuits are located in individual compartments.

The major components of the DC power system include the battery, starter/generator, Generator Control Unit (GCU)/voltage regulator, relays, 28VDC bus, and circuit breakers. All circuits in the electrical system are single wire with fuselage common ground return.

## Battery System


ABAA_2400_000001

The battery system includes the battery, a battery relay, and a battery switch with related wiring. The battery may be a Ni-Cad 24 volt 17 ampere-hour battery or an optional 24 volt 28 ampere-hour battery. The battery is located in the nose of the aircraft.

The battery relay is located below the battery in the landing light compartment and connects battery current to the main bus bar. The battery switch located in overhead console opens and closes the circuit to the relay energizing coil.

## External Power System

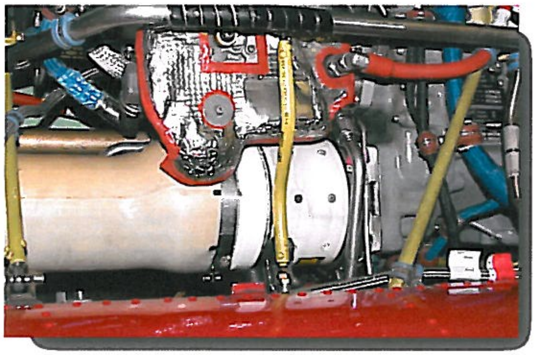
The external power system includes the external power receptacle, external power relay, and related wiring. The external power receptacle is located in the lower front center section of the nose and is a polarized receptacle used as a contact point for external power plug-in.

The external power relay is located in the nose below the battery in the landing light compartment. The relay is an electrically-operated switch between the external power receptacle and the main bus bar. It is controlled through the small positive pin from the external power source that energizes the circuit to the activating coil of the relay.

The 28VDC Ground Power Unit (GPU) shall be limited to 500 amperes or less to reduce the risk of starter damage due to excessive heat.

# ELECTRICAL SYSTEM

## Generator System



ABAA_2430_000001

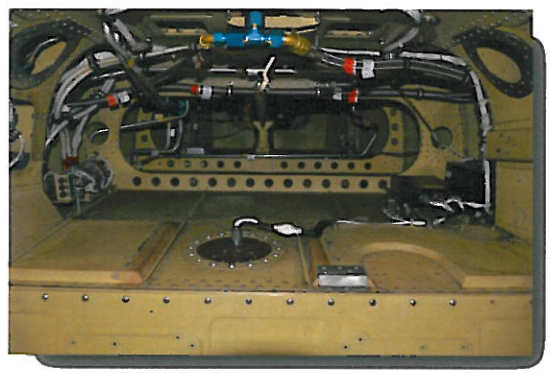
The generator system includes the generator portion of the starter/generator, generator control unit (GCU), line control relay, generator reset switch, and generator shunt.

The starter/generator is located on the lower right aft side of the accessory gearbox. The generator portion furnishes power at a regulated voltage for all DC electrical components on the helicopter.

During normal operations, with the generator switch in the ON position, the voltage regulator will monitor and control the output voltage of the generator, and connect the generator to the 28 VDC bus through the generator relay.

## Electrical Equipment Shelf

The GCU, which controls operation of the electrical system, operates the line control relay, regulates generator voltage, and provides protection against overvoltage and reverse current. The GCU contains an electronic voltage regulator to control voltage output of the DC generator and a circuit to energize the line contactor when correct conditions exist.



ABAA_9600_000011

The line control relay is installed on the electrical panel assembly located on the equipment shelf above the baggage compartment. It opens or closes the power circuit between the DC bus and the generator, controlled by the proper output voltage from the generator control unit.

The generator switch closes the generator field circuit in the GEN position, supplies voltage to reset the generator field circuit in the RESET position, and opens the generator field circuit in the center OFF position.

The generator shunt, installed on the electrical panel assembly on the equipment shelf above the baggage compartment, provides a voltage drop proportional to the generator current to operate the ammeter.

# ELECTRICAL SYSTEM

## Starter/Igniter System

The starter/igniter system includes the starter portion of the starter/generator, starter relay, FADEC/Start relay, igniter relay, igniter, and starter switch.

The starter relay, installed on the electrical panel assembly located on the electrical shelf above the baggage compartment, energizes the starter to start the engine. DC power is supplied to the starter through the starter relay when the start switch is engaged.

The igniter relay is installed on the electrical panel assembly located on the electrical shelf above the baggage compartment. The FADEC/Start relay is installed in the forward left side of the pedestal. DC power is supplied to the igniter and start field suppress section of the GCU through the FADEC/Start relay and igniter relay when the starter is engaged.

The tension capacitor discharge ignition exciter (igniter exciter) is located on the lower left section of the engine accessory case. This exciter provides increased voltage to the igniter plug to ensure fuel ignition during the engine start cycle.

## Switches, Circuit Breakers and Circuit Breakers Switches


ABAA_9870_000002

Switches circuit breakers, and circuit breaker switches are mounted in the overhead console, instrument panel, and the collective switch box. Operating these push/pull circuit breakers and/or the two-position circuit breaker switches opens and closes the circuits.

### Battery Switch

The battery switch installed on the overhead console controls the battery relay that connects the battery to the DC bus. The switch positions are OFF and BATT.

### Generator Switch

The generator switch on the overhead console controls generator output by opening and closing the generator field circuit. The switch is a double-pole, double throw, spring-loaded design with only momentary contact in the RESET position. The switch positions are GEN, OFF, and RESET.

### Start Switch

The start switch on the collective switch box contains two sets of spring-loaded contacts that provide momentary contact in either the START or DISENG positions. When the switch is positioned to start or disengage, only one set of contacts moves.

# ELECTRICAL SYSTEM

## Switches, Circuit Breakers and Circuit Breakers Switches (Cont)

### Hydraulic System Switch

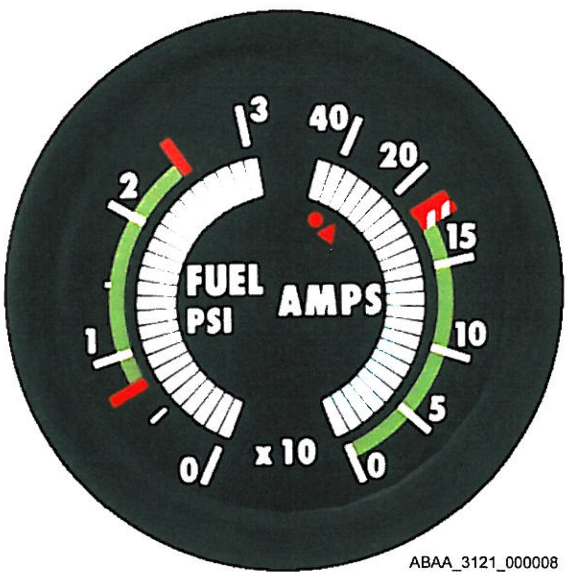
The HYDRAULIC SYSTEM switch on the overhead console controls operation of the hydraulic bypass solenoid. The solenoid is electrically de-energized open when the hydraulic switch is in the ON position, allowing pressurized hydraulic fluid to flow into the hydraulic manifold. Electrical power flows to the solenoid only when the switch is OFF.

### Avionics Master Switch

The avionics master switch allows activation and deactivation of all avionics simultaneously (except the VHF1 NAV/COM and audio panel ICS). Triangles next to circuit breakers on the overhead console identify the circuit breakers connected to the avionics master switch.

## Electrical System Indicators

Electrical system indicators include a DC ammeter, voltmeter, BATTERY HOT light, BATTERY RELAY light, START light and a GEN FAIL light.




ABAA_3121_000008

### Fuel Pressure / DC Ammeter

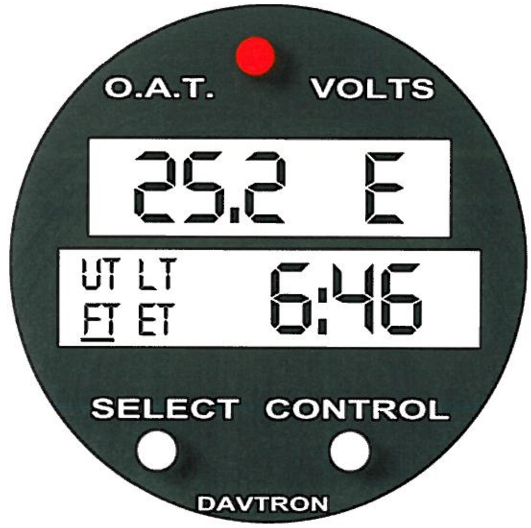
The fuel pressure/ammeter indicator is a dual display instrument. The ammeter indicates the load in amperes being supplied to the 28VDC bus by the generator. The ammeter side of the indicator is powered through its own circuit breaker.

### Ammeter 400 Amp Max Scale

The ammeter indications are continuous, regardless of load. The indicator does not drop to zero if limits are exceeded. To insure limits are not exceeded, the pilot may switch the generator OFF prior to a generator exceedance and after a brief time, the generator switch can be positioned to GEN (on).

### Voltmeter

A multifunction indicator mounted in the upper left area of the instrument panel includes the voltmeter. The red button on the top of the instrument changes the top display between Volts (e.g. 28E) and OAT (Celsius and Fahrenheit). When power is applied to the instrument, the display defaults to the voltmeter reading. The Voltmeter display receives power from the 28VDC bus through the OAT/V INSTR circuit breaker. The 28VDC bus, through this circuit breaker, is also the source of the voltage displayed by the Voltmeter. When all electrical power is turned off, the Voltmeter display disappears.




ABAA_3160_000011

# ELECTRICAL SYSTEM

## Electrical System Indicators (Cont)

### Battery Hot Annunciator

The BATTERY HOT light applies to either the basic ship 17 amp/hour or the optional 28 amp/hour battery. The 17 amp/hour battery incorporates two thermal switches whereas the 28 amp/hour battery incorporates three thermal switches. Although both 17 amp/hour battery thermal switches are used in the BATTERY HOT light circuit, only two of the three thermal switches on the 28 amp/hour battery are used.

The 17 amp/hour battery thermal switches close at a temperature of 145 ± 5°F (62.7 ± 2.8°C). The 28 amp/hour battery switches close at a temperature of 160 ± 5°F (71.1± 2.8°C). If any one of the thermal switches closes, the BATTERY HOT light illuminates.

### Battery Relay Annunciator

The BATTERY RELAY light illuminates if the battery relay remains in the closed (energized) position after the battery switch has been set to OFF. If the battery relay remains closed after the battery switch has been set to OFF, battery power remains on the 28VDC bus. This powers the Caution, Warning, and Advisory Panel (CWAP) to illuminate the BATTERY RELAY light even if the generator is OFF.

### Start Annunciator

The START light illuminates when the starter relay is energized. The starter relay energizes when the start switch is positioned as follows:

> With the FADEC MODE switch positioned to AUTO, the starter relay stays engaged until the gas producer ($N_G$) speed reaches 50 ±1%.
>
> With the FADEC MODE switch positioned to MAN, the starter relay stays engaged until the start switch is released from the START position.

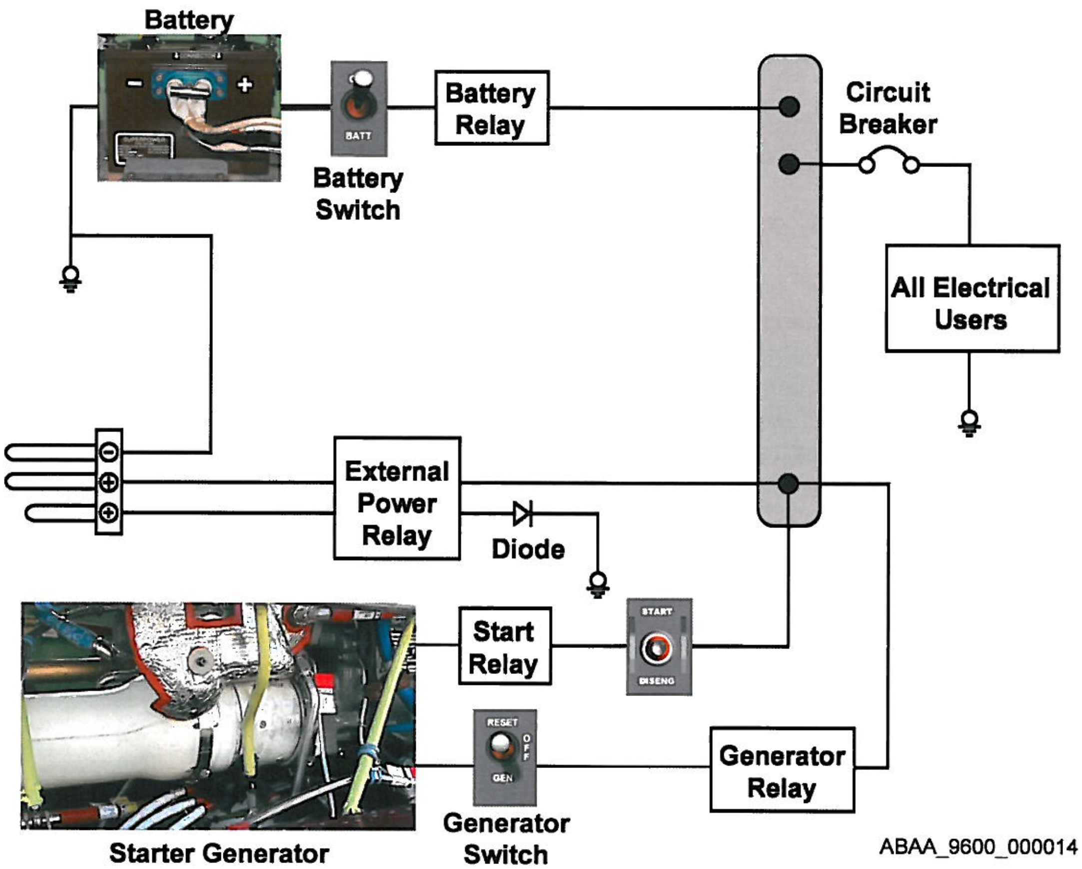
### Generator Fail Annunciator

The generator relay controls the GEN FAIL light, which illuminates when the generator relay is de-energized and is not connecting the generator output to the DC bus. The GCU energizes the generator relay when generator output climbs through a threshold of 24 ± 2.4 VDC.

Before the generator relay is energized, the GEN FAIL light is on. Once the generator relay is energized, the GEN FAIL light is extinguished.

# ELECTRICAL SYSTEM

## Starting Circuit




The engine start procedures are in the Normal Procedures section 2 of the 407 Flight Manual. (BHT–407–FM–1).

# ELECTRICAL SYSTEM

## Electrical System Failures

The 407 can function (fly) without electrical bus power. If circumstances require the pilot to turn off both the generator and battery in flight, the electrical system operates the left boost/transfer pumps directly from the battery. In the event of a complete electrical failure, a fully charged battery continues to operate the left boost/transfer pumps for approximately 1.7 hours (2.8 hours with optional 28 amp hour battery). This provides boosted fuel pressure to the engine and transfers fuel from the forward cell, making all fuel available and maintaining longitudinal CG on the aircraft.

Procedures for electrical system malfunctions are in the Emergency/Malfunction Section 3 of the 407 Flight Manual. (BHT—407—FM—1).

During a total electrical failure, **ALL** electrical power to the **BUS** is lost so the following systems fail to operate:

- Propulsion Instruments
- Navigation Instruments (except compass)
- Miscellaneous Instruments
- Flight Instruments (except Pitot/Static)
- Pedal Restrictor Control Unit (PRCU)

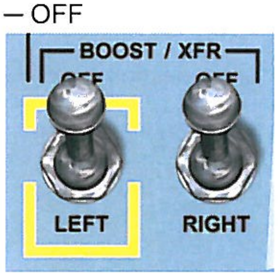
The following systems are ON regardless of switch position (fail safe on).

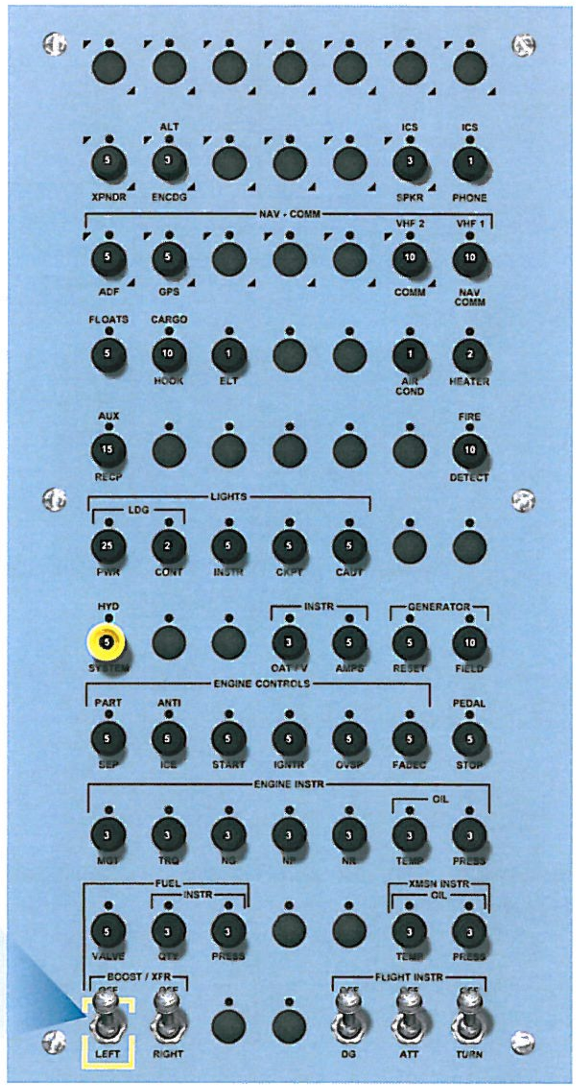
- Engine Anti Ice
- Hydraulics
- Particle Separator Purge (if installed)

## Alternate Circuit

The alternate electrical circuit for the LEFTBOOST/XFR pumps provides a means for the pumps to operate if no electrical power is provided to the BUS.

1. LEFT BOOST/XFR switch – ON
2. FUEL VALVE switch – ON
3. CIRCUIT BREAKER – IN
4. BATT SWITCH – OFF

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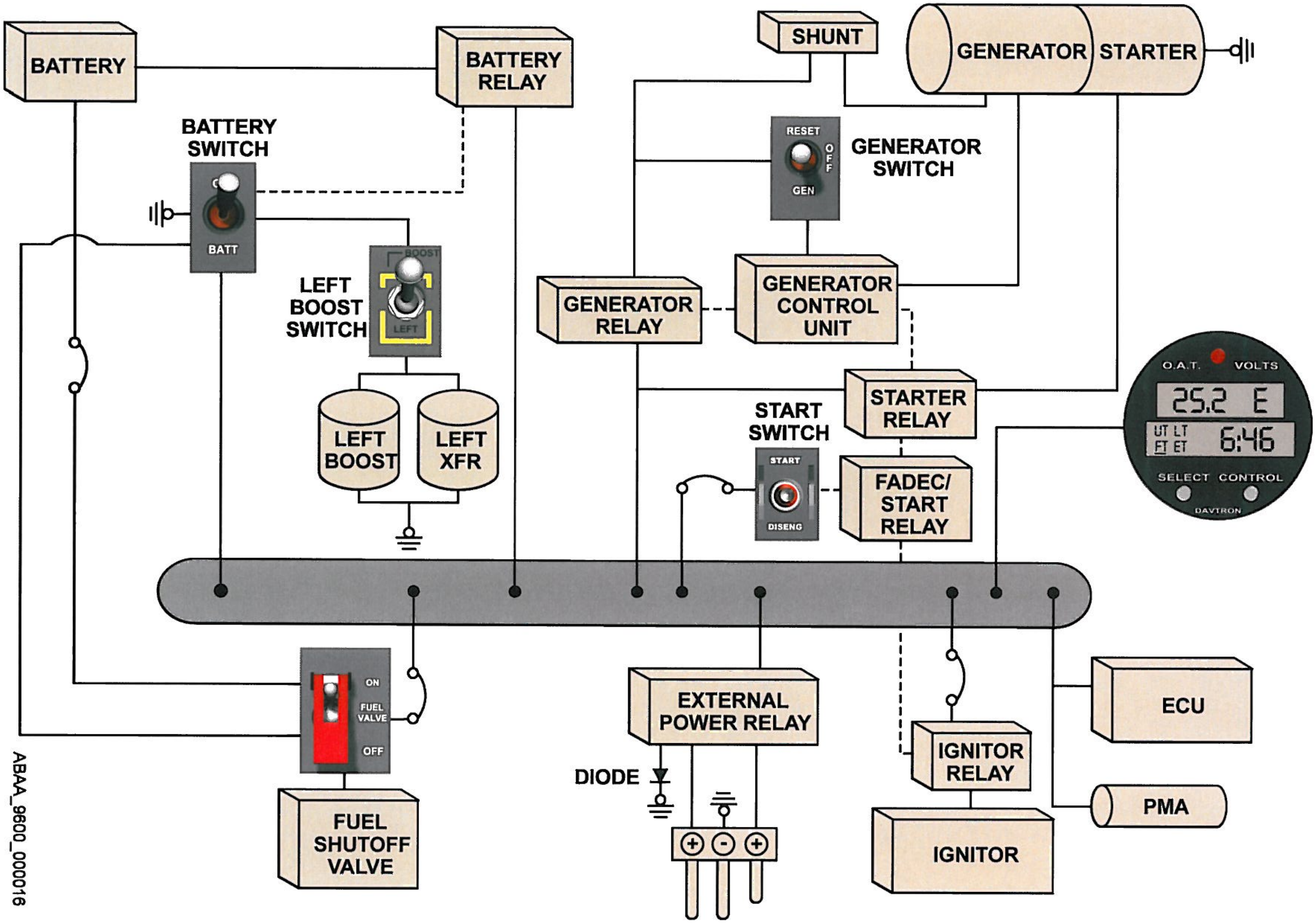
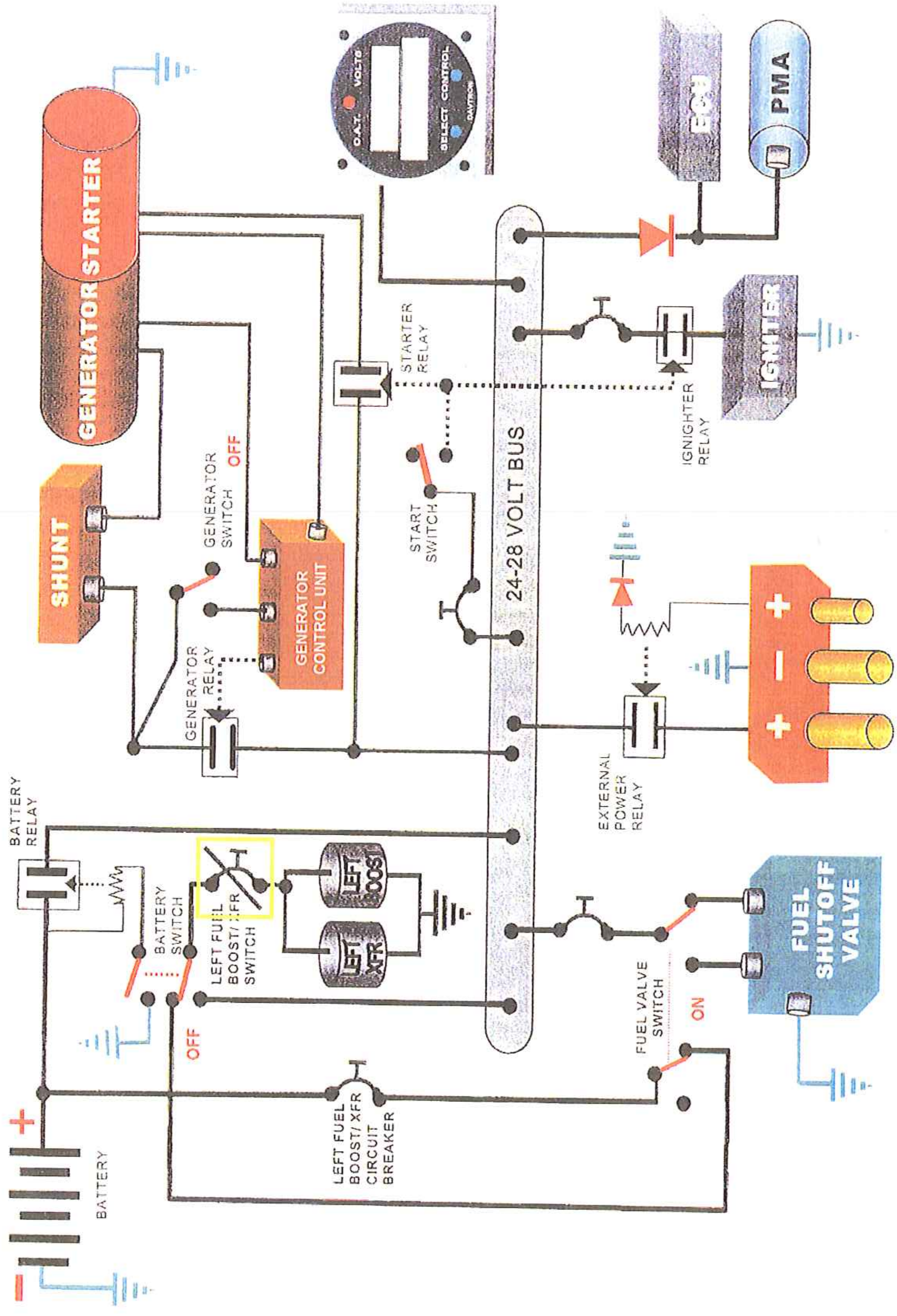
**NOTE**

A shut down aircraft in which the fuel valve and left fuel boost/transfer switch are left on will duplicate a total electrical failure situation. The left boost/transfer pumps will operate and deplete the battery.

## Electrical Schematic




ABAA_9600_00016


BATTERY
BATTERY RELAY
BATTERY SWITCH
OFF
LEFT FUEL BOOST/ XFR SWITCH
LEFT XFR
LEFT BOOST
LEFT FUEL BOOST/ XFR CIRCUIT BREAKER
FUEL VALVE SWITCH
ON
FUEL SHUTOFF VALVE
SHUNT
GENERATOR
STARTER
GENERATOR RELAY
GENERATOR SWITCH
OFF
GENERATOR CONTROL UNIT
STARTER RELAY
START SWITCH
24-28 VOLT BUS
O.A.T.
VOLTS
SELECT
CONTROL
EXTERNAL POWER RELAY
IGNIGHTER RELAY
IGNITER
ECU
PMA

# WEIGHT & BALANCE AND PERFORMANCE

THE BELL TRAINING ACADEMY

Est. 1947

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## Weight, Balance, and Performance



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Est. 1947

# WEIGHT, BALANCE, AND PERFORMANCE

## General Weight and Balance

Section 5 of the approved Rotorcraft Flight Manual (RFM) for the 407 presents the data necessary for the pilot to compute gross weight and center of gravity for various load configurations. It is the pilot's responsibility to ensure the maximum gross weight and center of gravity limitations are observed throughout each planned flight. Operations outside of the limitations are prohibited and may result in a reduction of aircraft performance, handling qualities, stability, and structural integrity.

Changes in aircraft configuration (example: doors on or off), loading, seating of passengers, positioning of cargo, and fuel consumption are all factors that must be considered in weight and balance calculations.

## Terms

The following terms are used in the calculation of weight and balance information:

**Weight:** The actual aircraft weight, weight of crewmembers and passengers, fuel, etc. is required to compute weight and balance accurately. The flight manual provides data based on either U.S. or metric measurements, expressed in pounds or kilograms.

**Arm:** When calculating weight and balance information, this term refers to the distance from a known point to where new weight is added. For a given quantity of weight added, the greater the arm, the greater the effect upon the balance point. In the 407 flight manual, this data is referred to as fuselage stations or buttock lines. It is provided in both U.S. and metric formats (inches or millimeters) and is provided for both longitudinal and lateral axes of balance.

**Moment:** A mathematical expression of the affect that weight (at a certain position) will have on the balance point. This number is obtained by multiplying the weight times the arm. The result is referred to as moment, and can be expressed as inch-pounds (U.S.) or kilograms-millimeters (metric).

**Center of Gravity:** Used to identify the position at which the aircraft is in balance. Maintenance personnel compute the Empty Weight Center of Gravity. Pilots use the empty weight CG to determine the Gross Weight Center of Gravity. Total Moment divided by total weight equals the center of gravity. The result is expressed in either inches (U.S.) or millimeters (metric).

**Datum:** An arbitrary point from which to measure fuselage station (arm) for longitudinal CG calculations. Fuselage Stations aft of the datum line are expressed as positive numbers, while points forward would be expressed as negative numbers.

**Center line:** A reference point for measuring distances from the lateral axis of the aircraft. Buttock lines are measured from the center line, either to the left or to the right. Positions to the right of the centerline are considered positive numbers, while positions to the left are considered negative numbers.

**Empty Weight and Center of Gravity:** computed by maintenance personnel. The empty weight configuration consists of the basic helicopter with required equipment, operational and special equipment, fixed ballast, hydraulic fluid, transmission and gearbox oil, unusable fuel, and undrainable engine oil. Weight empty and center of gravity is recorded on the Actual Weight Record, a copy of which should be carried in the helicopter for use in weight and balance calculations. Each time equipment is installed or removed, the actual weight record should be updated.

**Gross Weight Center of Gravity:** computed by the pilot. Gross weight includes the total weight of the helicopter, with contents. Contents include crew, passengers, engine oil, fuel, baggage, and cargo.

## Terms (Cont)

**Zero fuel weight (ZFW):** A calculated value used as a starting point to calculate changes in CG caused by fuel consumption during flight. ZFW includes helicopter empty weight, crew, passengers, engine oil, baggage and cargo.

## Effects of Fuel Consumption on Center of Gravity

Under normal circumstances, a center of gravity calculation completed prior to takeoff will be affected only by fuel consumption during the flight. Other loading considerations (crew, passengers, and cargo) will not change.

The automatic fuel transfer sequence will affect the aircraft center of gravity as fuel is consumed. Generally speaking, if the aircraft begins a flight with full fuel and lands with minimum fuel, the center of gravity will move forward, aft, then finally forward. It is the pilot's responsibility to ensure the helicopter remains within CG as fuel is consumed. To accomplish this, fuel shifts must be calculated. Data is provided in fuel loading tables to compute moment values for various quantities of fuel.

## Baggage Compartment Loading

The baggage compartment is accessible from the left side of the fuselage and contains approximately 16 cubic feet (0.45 cubic meters) of space. It has a load limit of 250 pounds (113.4 kg), not to exceed 86 pounds per square foot (4.2 kilograms/100 square centimeters). These are structural limitations only, and do not infer the CG will remain within approved limits. When weight is loaded into the baggage compartment, the pilot must compute gross weight and CG to assure loading within approved limits. The load shall be secured to tie-down fittings if shifting of the load in flight could result in structural damage to the baggage compartment or in gross weight center of gravity limits being exceeded. Tables and examples assume items in the baggage compartment have a longitudinal CG at the midpoint of the door opening

## Weight and Balance Calculation

As previously stated, it is the pilot's responsibility to ensure the maximum gross weight and center of gravity limitations are observed throughout each planned flight. In the case of the 407, improper cabin loading and fuel consumption may create a situation where the center of gravity could travel outside CG limits during the flight. For this reason, it is important to load the helicopter as evenly as possible, both longitudinally and laterally.

## Weight and Balance Worksheet

| | | Longitudinal | | Lateral | |
|---|---|---|---|---|---|
| | Weight (lb) | FS (in) | Moment (lb-in) | BL (in) | Moment (lb-in) |
| Weight Empty | | | | | |
| Oil | 13 | 205.0 | 2665 | 0 | |
| +Pilot | | 65.0 | | +14 | |
| +Forward passenger | | 65.0 | | -11.1 | |
| + Mid passenger(s) Left | | 91.0 | | -13.0 | |
| Right | | 91.0 | | +15.5 | |
| +Aft passenger(s) Left | | 129.0 | | -16.8 | |
| Center | | 129.0 | | 0.0 | |
| Right | | 129.0 | | +16.8 | |
| +Baggage | | 174.0 | | 0.0 | |
| Zero Fuel Weight | | | | | |
| +Fuel | | | | 0.0 | 0 |
| Gross takeoff weight (GTOW) | | | | | |
| Zero Fuel Weight | | | | | |
| +Critical fuel (Most FWD) | 508.6 | | 58998 | 0.0 | 0 |
| Most forward CG condition | | | | | |
| Zero Fuel Weight | | | | | |
| +Critical Fuel (Most AFT) | 193.1 | | 26455 | 0.0 | 0 |
| Most AFT CG condition | | | | | |

# WEIGHT, BALANCE, AND PERFORMANCE

## Weight and Balance Worksheet (Cont)

| | | Longitudinal | | Lateral | |
|---|---|---|---|---|---|
| | Weight (lb) | FS (in) | Moment (lb-in) | BL (in) | Moment (lb-in) |
| Weight Empty | | | | | |
| Oil | 13 | 205.0 | 2665 | 0 | |
| +Pilot | | 65.0 | | +14 | |
| +Forward passenger | | 65.0 | | -11.1 | |
| + Mid passenger(s) Left | | 91.0 | | -13.0 | |
| Right | | 91.0 | | +15.5 | |
| +Aft passenger(s) Left | | 129.0 | | -16.8 | |
| Center | | 129.0 | | 0.0 | |
| Right | | 129.0 | | +16.8 | |
| +Baggage | | 174.0 | | 0.0 | |
| Zero Fuel Weight | | | | | |
| +Fuel | | | | 0.0 | 0 |
| Gross takeoff weight (GTOW) | | | | | |
| Zero Fuel Weight | | | | | |
| +Critical fuel (Most FWD) | 508.6 | | 58998 | 0.0 | 0 |
| Most forward CG condition | | | | | |
| Zero Fuel Weight | | | | | |
| +Critical Fuel (Most AFT) | 193.1 | | 26455 | 0.0 | 0 |
| Most AFT CG condition | | | | | |

## Cabin and Baggage Loading (FM Table 5-2)

Cabin and Baggage Compartment
Table of Moments
Inch -Pounds

| Weight (LB) | Front Seat FS 65 | Mid -Pass. (Facing Aft) FS 91 | Aft -Pass. (Facing FWD) FS 129 | Litter Patient(S) FS 108 | Baggage FS 174 |
|---|---|---|---|---|---|
| 10 | 650 | 910 | 1290 | 1080 | 1740 |
| 20 | 1300 | 1820 | 2580 | 2160 | 3480 |
| 30 | 1950 | 2730 | 3870 | 3240 | 5220 |
| 40 | 2600 | 3640 | 5160 | 4320 | 6960 |
| 50 | 3250 | 4550 | 6450 | 5400 | 8700 |
| 60 | 3900 | 5460 | 7740 | 6480 | 10440 |
| 70 | 4550 | 6370 | 9030 | 7560 | 12180 |
| 80 | 5200 | 7280 | 10320 | 8640 | 13920 |
| 90 | 5850 | 8190 | 11610 | 9720 | 15660 |
| 100 | 6500 | 9100 | 12900 | 10800 | 17400 |
| 110 | 7150 | 10010 | 14190 | 11880 | 19140 |
| 120 | 7800 | 10920 | 15480 | 12960 | 20880 |
| 130 | 8450 | 11830 | 16770 | 14040 | 22620 |
| 140 | 9100 | 12740 | 18060 | 15120 | 24360 |
| 150 | 9750 | 13650 | 19350 | 16200 | 26100 |
| 160 | 10400 | 14560 | 20640 | 17280 | 27840 |
| 170 | 11050 | 15470 | 21930 | 18360 | 29580 |
| 180 | 11700 | 16380 | 23220 | 19440 | 31320 |
| 190 | 12350 | 17290 | 24510 | 20520 | 33060 |
| 200 | 13000 | 18200 | 25800 | 21600 | 34800 |
| 210 | 13650 | 19110 | 27090 | 22680 | 36540 |
| 220 | 14300 | 20020 | 28380 | 23760 | 38280 |
| 230 | 14950 | 20930 | 29670 | 24840 | 40020 |
| 240 | 15600 | 21840 | 30960 | 25920 | 41760 |
| 250 | 16250 | 22750 | 32250 | 27000 | 43500 |
| 260 | 16900 | 23660 | 33540 | 28080 | |
| 270 | 17550 | 24570 | 34830 | 29160 | |
| 280 | 18200 | 25480 | 36120 | 30240 | |
| 290 | 18850 | 26390 | 37410 | 31320 | |
| 300 | 19500 | 27300 | 38700 | 32400 | |
| 310 | 20150 | 28210 | 39990 | 33480 | |
| 320 | 20800 | 29120 | 41280 | 34560 | |
| 330 | 21450 | 30030 | 42570 | 35640 | |
| 340 | 22100 | 30940 | 43860 | 36720 | |
| 350 | 22750 | 31850 | 45150 | 37800 | |

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# WEIGHT, BALANCE, AND PERFORMANCE

## Fuel Loading (FM Table 5-4)

| QUANTITY (U.S. GAL) | JP-5 WEIGHT (LBS) | LONGITUDINAL CG (IN) | LONGITUDINAL MOMENT (IN-LBS) |
|---|---|---|---|
| 5 | 34.0 | 133.7 | 4,546 |
| 10 | 68.0 | 135.0 | 9,180 |
| 15 | 102.0 | 135.9 | 13,862 |
| 20 | 136.0 | 136.4 | 18,550 |
| 25 | 170.0 | 136.7 | 23,239 |
| 28.4 Δ | 193.1 | 137.0 | 26,455 |
| 30 | 204.0 | 134.3 | 27,397 |
| 35 | 238.0 | 127.8 | 30,416 |
| 40 | 272.0 | 122.9 | 33,429 |
| 45 | 306.0 | 119.1 | 36,445 |
| 50 | 340.0 | 116.0 | 39,440 |
| 50.6 ** | 344.1 | 115.7 | 39,812 |
| 55 | 374.0 | 116.1 | 43,421 |
| 60 | 408.0 | 116.2 | 47,410 |
| 65 | 442.0 | 116.2 | 51,360 |
| 70 | 476.0 | 116.1 | 55,264 |
| 74.8 □ | 508.6 | 116.0 | 58,998 |
| 75 | 510.0 | 116.1 | 59,211 |
| 80 | 544.0 | 117.7 | 64,029 |
| 85 | 578.0 | 119.0 | 68,782 |
| 90 | 612.0 | 120.3 | 73,624 |
| 95 | 646.0 | 121.4 | 78,424 |
| 100 | 680.0 | 122.3 | 83,164 |
| 105 | 714.0 | 123.4 | 88,108 |
| 110 | 748.0 | 124.6 | 93,201 |
| 115 | 782.0 | 125.6 | 98,219 |
| 120 | 816.0 | 126.6 | 103,306 |
| 125 | 850.0 | 127.5 | 108,375 |
| 127.8 ★ | 869.0 | 127.9 | 111,145 |

Δ Critical Fuel for Most Aft C.G. Condition

** Most Forward Fuel C.G.

□ Critical Fuel for Most Forward C.G. Condition

★ Full Fuel

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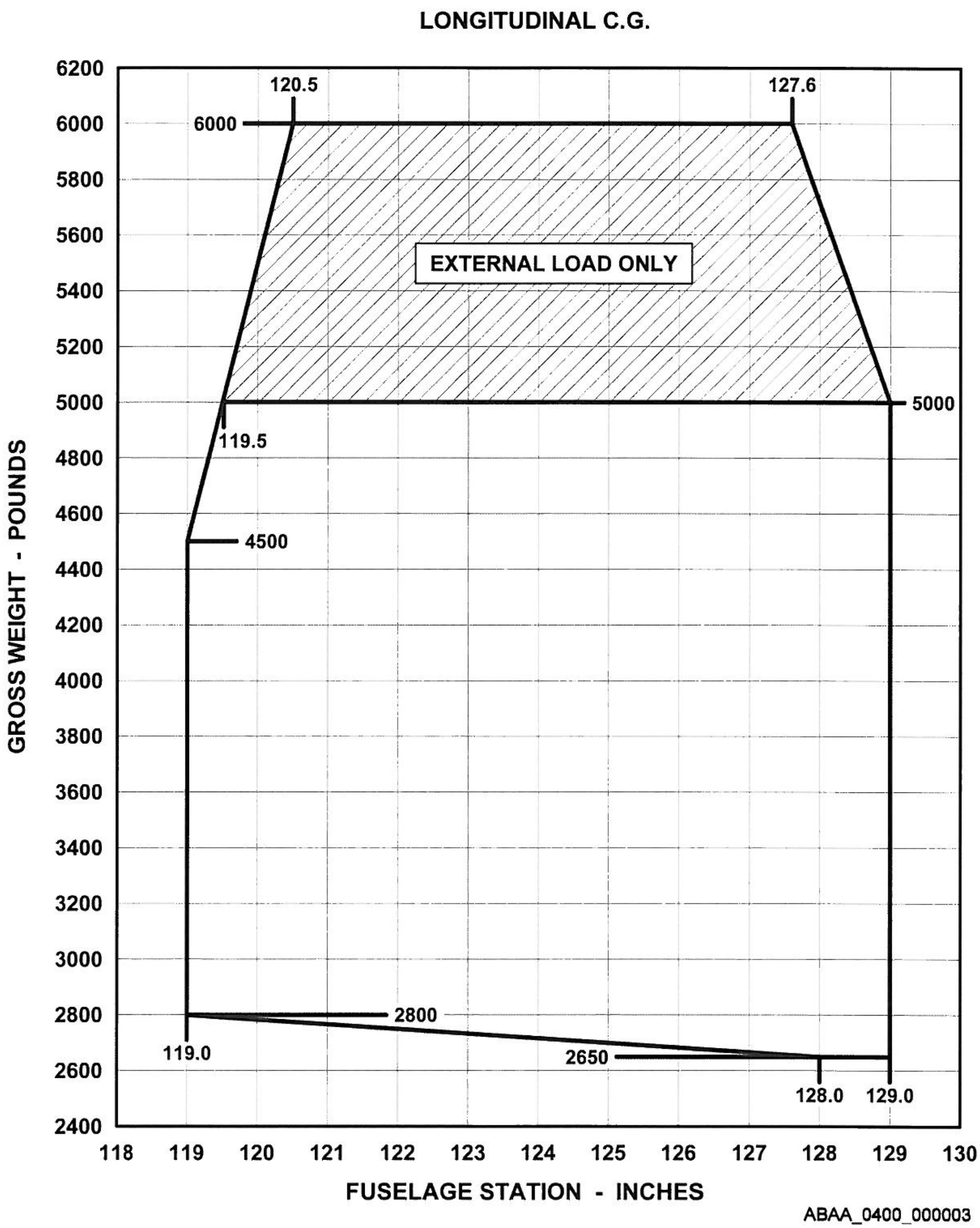
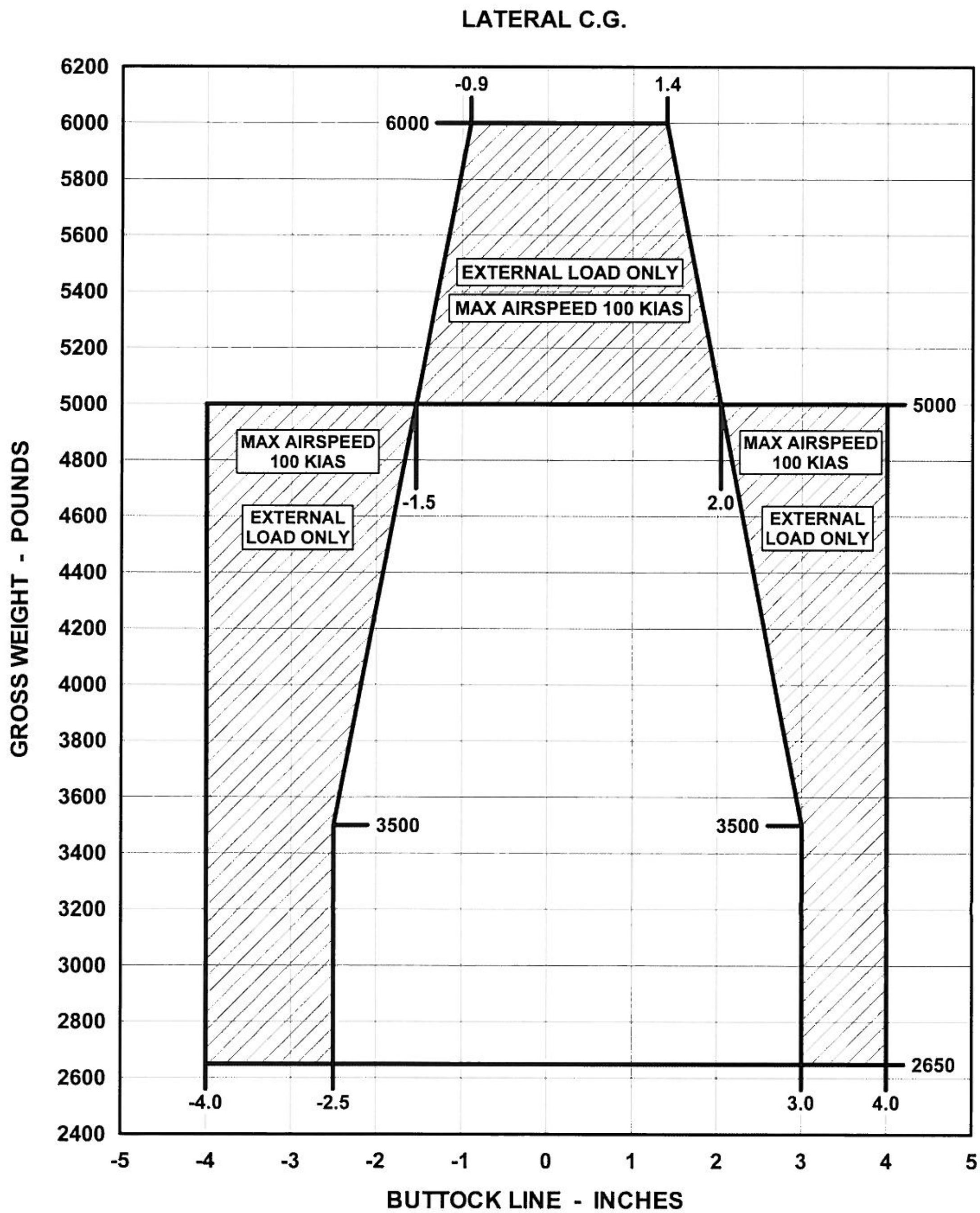
## WEIGHT, BALANCE, AND PERFORMANCE

### Gross Weight Longitudinal Center of Gravity Limits



# WEIGHT, BALANCE, AND PERFORMANCE

## Gross Weight Lateral Center of Gravity Limits

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# WEIGHT, BALANCE, AND PERFORMANCE

## Performance Planning - General

The performance section of the Rotorcraft Flight Manual (Section 4) contains the Bell 407 performance information and related charts. All performance charts are based on an engine meeting minimum Rolls-Royce specifications. The data shown is derived from actual flight tests and are intended to provide information to be used in conducting flight operations. This performance data is applicable to the 250C47B engine.

NOTE: The 407 basic configuration does not include a particle separator. If the aircraft has a particle separator installed, the correct performance charts are located in FMS-3. This supplement is not included in the training manual.

FMS-28 is the supplement that allows increased gross weight operation. When operating at takeoff gross weights over 5000 lbs, performance data from FMS-28 should be used. This supplement is not included in the training manual.

## Power Assurance Check

A power assurance check chart is provided for the Rolls-Royce 250C-47B engine. This chart indicates the maximum allowable MGT for an engine that meets minimum specifications. The engine must develop the required torque without exceeding chart MGT in order to meet performance data contained in this section.

To perform the power assurance check, turn off all sources of bleed air (anti-ice, particle separator purge, and heater). Establish level flight at an airspeed of 85 to 105 KIAS or Vne, whichever is lower. The check may also be conducted in a hover, depending on ambient conditions and gross weight.

With airspeed established, at sufficient altitude, record the following:

$H_P$

OAT

MGT

Torque

If the actual MGT value recorded is less than or equal to the chart MGT value, the engine meets minimum power specifications.

## Density Altitude

A density altitude chart is provided to aid in the calculation of performance and limitations. Density altitude ($H_D$) is defined as pressure altitude $H_P$) corrected for non-standard temperature. Pressure, temperature, and humidity determine air density. $H_D$ is an expression of the density of air in terms of height above sea level; hence, the less dense the air, the higher the $H_D$. For standard conditions of temperature and pressure, $H_D$ is same as $H_P$. As temperature increases above standard for any altitude, $H_D$ will also increase to values higher than $H_P$. The chart can also be used to compute a multiplication factor to determine true airspeed.

To use the chart, enter from the known temperature at the bottom of the chart. Proceed vertically until intercepting the $H_P$ line. Proceed left to determine $H_D$, and read right to determine the true airspeed conversion factor.

# WEIGHT, BALANCE, AND PERFORMANCE

## Height Velocity Envelope

The height velocity envelope charts define the conditions from which a safe landing can be made on a smooth, level, firm surface following an engine failure. For purposes of discussion, it is also important to note the chart is based on zero wind conditions.

Two other important factors that affect autorotational performance are gross weight and density altitude. In the model 407, an Altitude vs. Gross Weight chart is used to select one of four possible H-V diagrams. For a given outside ambient air temperature, pressure altitude, and gross weight the appropriate limiting envelope can be determined (Region A, B, C, or D).

To use the envelopes, begin on the Altitude vs. Gross Weight chart. Enter the chart at the appropriate ambient temperature, move up vertically until intercepting the pressure altitude, then move right until intercepting the appropriate gross weight line. The intercept point determines which envelope the aircraft is operating in. No interpolation is allowed.

Next, select the appropriate H-V envelope diagram (Region A, B, C, or D) and enter the chart either at a known skid height above the ground or a known airspeed. Entering the chart at a known skid height, then proceeding horizontally to intercept the avoid area envelope will calculate minimum airspeed. Entering at a known airspeed, then proceeding vertically to intercept the avoid area envelope will calculate minimum skid height.

## Hover Ceiling

Hover Ceiling In Ground Effect (IGE) and Out of Ground Effect (OGE) charts present hover performance expressed as allowable gross weight for varying conditions of Hp and OAT. These hovering weights are obtainable in zero wind conditions. The hover performance charts are based on 100% Rotor RPM.

Each hover ceiling chart is divided into two areas. Area A is non-shaded and presents hover performance for conditions where adequate control margins exists for all relative wind conditions up to 35 knots (lateral CG not exceeding ±2.5 inches) or up to 17 knots (lateral CG not exceeding ± 4.0 inches). Area B is shaded and presents hover performance for relative winds within ± 45 degrees of the nose of the helicopter up to 35 knots (lateral CG not exceeding ± 2.5 inches) or up to 17 knots (lateral CG not exceeding ± 4.0 inches).

Satisfactory stability and aircraft control have been demonstrated in each area of the hover ceiling charts with winds as depicted on the hover ceiling wind accountability chart.

To use the chart, enter at the appropriate OAT, and proceed vertically to the applicable $H_P$ line. Move horizontally to the right until intercepting the appropriate OAT line. Move vertically down and read maximum gross weight for hover.

# WEIGHT, BALANCE, AND PERFORMANCE

## Rate of Climb

Rate of climb charts are presented for various combinations of power settings and can be adjusted for doors off and anti-ice ON or OFF. Rate of climb data shown are "tapeline" rates, which means actual rates of climb. Rate of climb as measured with an altimeter will equal this rate of climb only on a standard day with a standard temperature lapse rate.

The rate of climb will be reduced with ANTI-ICE ON. Reduce rate of climb 100 feet per minute when operating with any combination of door(s) off.

Rate of climb charts are based on a recommended airspeed for best rate of climb 60 KIAS.

To use the charts, select the appropriate chart based on gross weight and enter at the Hp value. Proceed horizontally to the right until intercepting the OAT. Proceed vertically down and read rate of climb.

## Autorotation

An autorotation chart is provided in the Performance section of the flight manual (BHT—407—FM—1) for autorotation glide distance as a function of altitude.

## Airspeed Calibration

The airspeed calibration table presents calibrated airspeed for various indicated airspeeds. Calibrated airspeed is defined as indicated airspeed corrected for installation errors. Installation error changes based on flight condition, therefore calibrated airspeed values are presented for climb conditions as well as level flight.

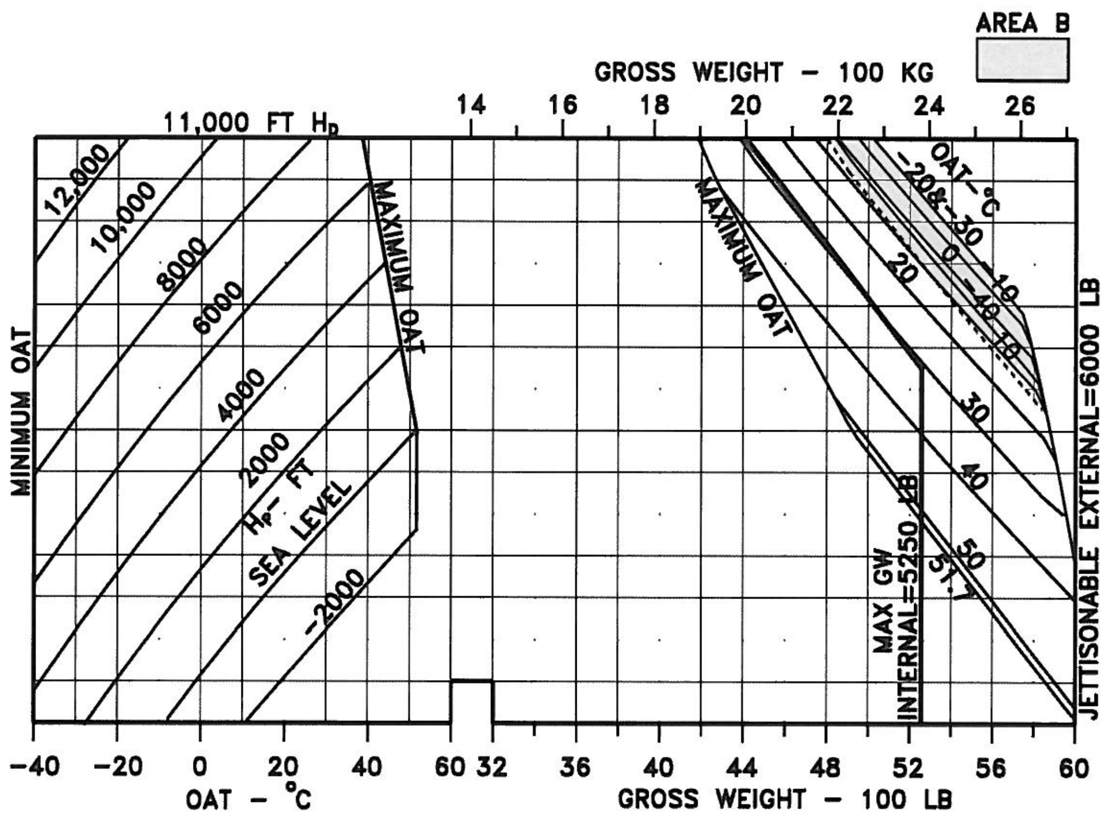
To use the table, enter at the appropriate KIAS value, and then read KCAS from either column.

## FMS-28 OGE Chart

**Hover Ceiling**
**Out of Ground Effect**

**Takeoff Power**
**Engine RPM 100%**
**Generator 180 AMPS**

**Skid Height 40 FT (12.2 Meters)**
**Heater and Anti Ice Off**
**Basic Inlet**

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