FLIGHT LINE SUPPLEMENT

UH-60L BLACKHAWK

2-224th Aviation Regiment
Sandston, VA 23150

Revised December 2008

UH-60L Flight Line Supplement
Table of Contents

The purpose of the flight line supplement is to provide basic supplemental information about the UH-60L to Army Aviators. The following references were used for this supplement:

TM 1-1520-237-10 (Operators Manual)
TM 11-1520-237-23-1 (Maintenance / General)
TM 11-1520-237-23-2 (Maintenance/ Airframe)
TM 55-1500-342-23 (Weight and Balance)
GE Training Guide for the T-700 Engine
Sikorsky Training Guide for the UH-60
ASET-AT Training CD and Student Handout
UH-60 ATM and AR 95-1

HYDRAULIC SYSTEM

The #1 and #2 hydraulic pumps are driven by separate transmission accessory gearbox modules and supply 3000 PSI hydraulic pressure when the main rotors are turning. The backup hydraulic pump is powered by a 115 VAC electric motor. If AC power is available, the backup pump is able to provide hydraulic pressure should one or both hydraulic pumps fail.

Each hydraulic pump is a constant pressure, variable volume pump. They will maintain the required 3000 PSI over any rate of control movement for their own system. However, if the #1 and #2 hydraulic pumps are inoperative at the same time, there will be a slight restriction in the maximum rate of flight control movement. This is due to the backup pump's inability to provide the required volume because it exceeds the flow rate capability of the pumps.

Each pump has a pressure and return filter. The filter indicator buttons will extend when the differential pressure reaches 70±10 PSID (pounds per square inch differential). Additionally, the return filter has a bypass capability, should the differential pressure reach 100±10 PSID. This is to allow hydraulic fluid to return from the system, regardless of filter condition. If a return filter were to become completely clogged and have no bypass capability, fluid would be blocked in the return lines. This would result in a hydrostatic lock, and cause a flight control lock up.

The #1 hydraulic pump supplies hydraulic pressure to the #1 fore, aft, and lateral primary servos and the #1 tail rotor servo.

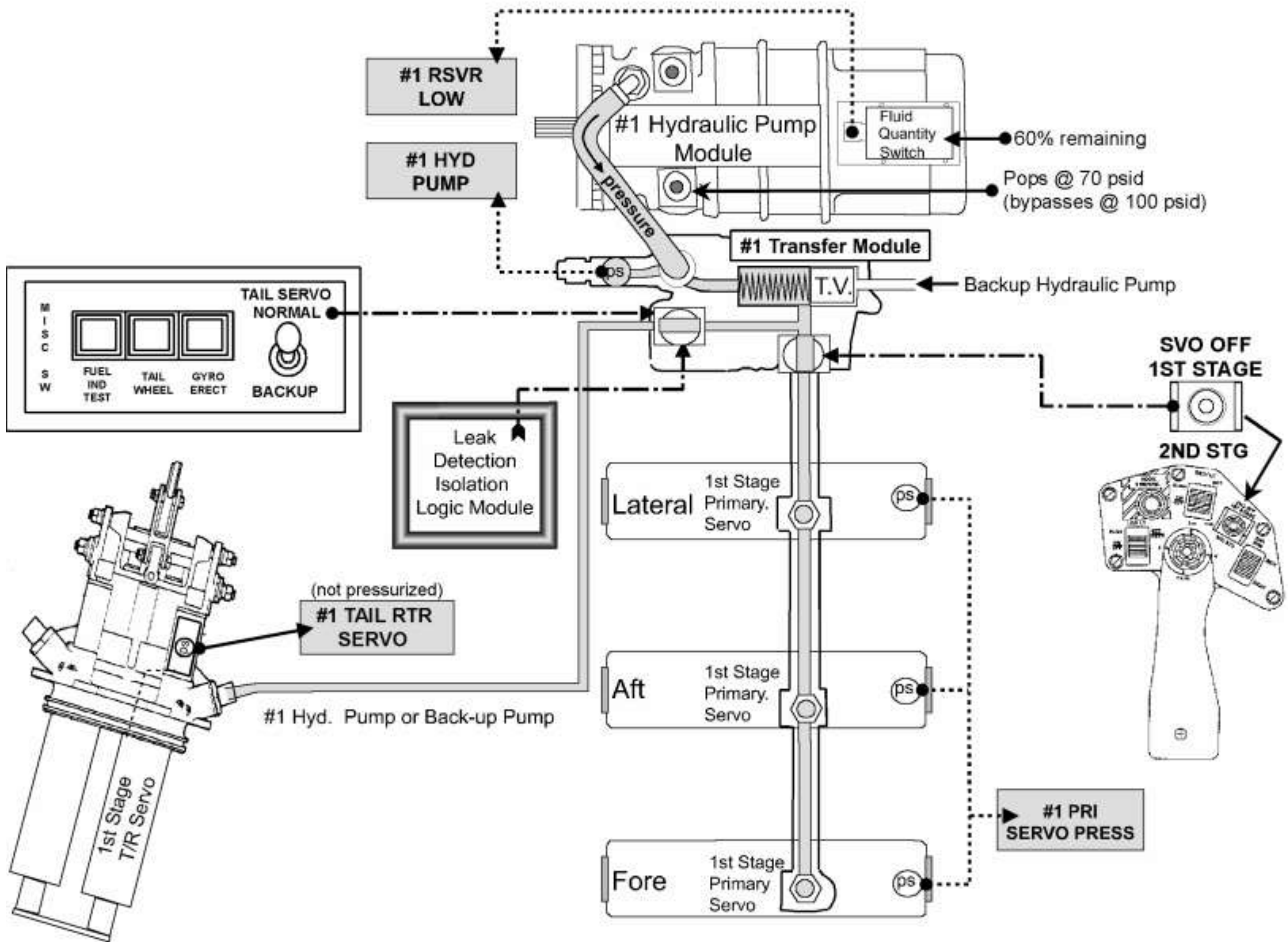
The #2 hydraulic pump supplies hydraulic pressure to the #2 fore, aft, and lateral primary servos and the components of the pilot assist area; pitch, roll, and yaw SAS actuators, pitch boost servo, collective and yaw boost servos, and the pitch trim assembly. A pressure reducer with relief valve in the pilot assist module reduces the 3000 PSI pump pressure to 1000 PSI for use by the pitch trim servo. The pressure reducer relief valve indicator button will extend when pressure goes above 1375 PSI indicating the relief valve opened allowing fluid to bypass the pitch trim servo.

The backup hydraulic pump can supply hydraulic pressure to the #1 and/or #2 hydraulic systems independently or simultaneously. It is also the only pump that can supply hydraulic pressure to the 2nd stage tail rotor servo and the APU accumulator. The backup pump supplies hydraulic pressure to flight control components during ground checks (main rotors not turning). There are four caution lights and one advisory light that turn on the backup pump when AC power is available:

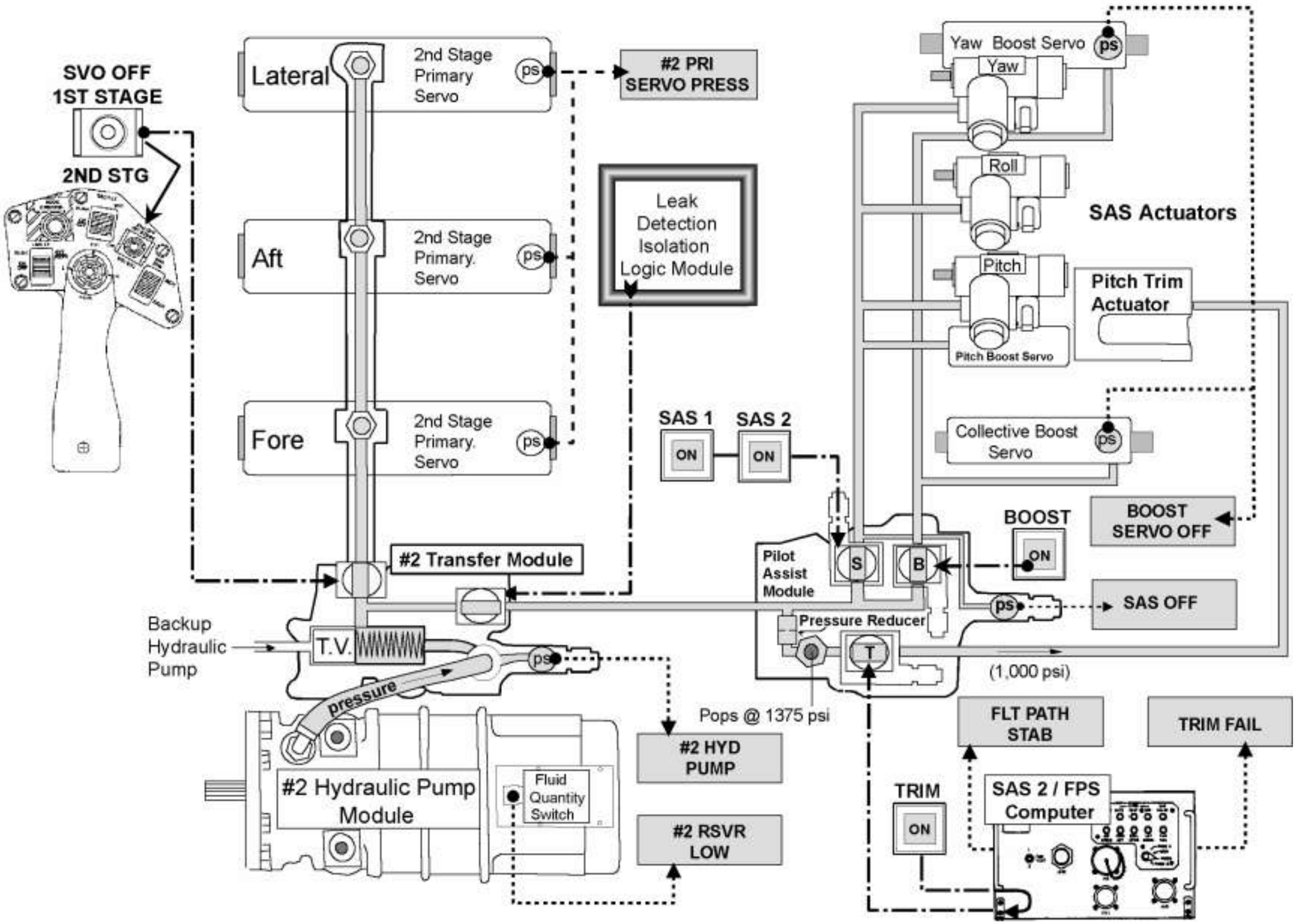
- #1 Reservoir Low (caution light)
- #1 Hydraulic Pump (caution light)
- #1 Tail Rotor Servo (caution light)
- #2 Hydraulic Pump (caution light)
- APU Accumulator Low (advisory light)

An electrical interlock is provided to prevent the #1 and #2 primary servos from being mechanically isolated simultaneously. The primary servo shutoff valves located on the transfer modules require electrical power to close. If one set of primary servos (#1 or #2) is to be isolated, the other primary servo set must be pressurized to at least 2350 psi. There is no priority between the pilot’s and copilot’s servo off switch. The first switch actuated takes precedence.

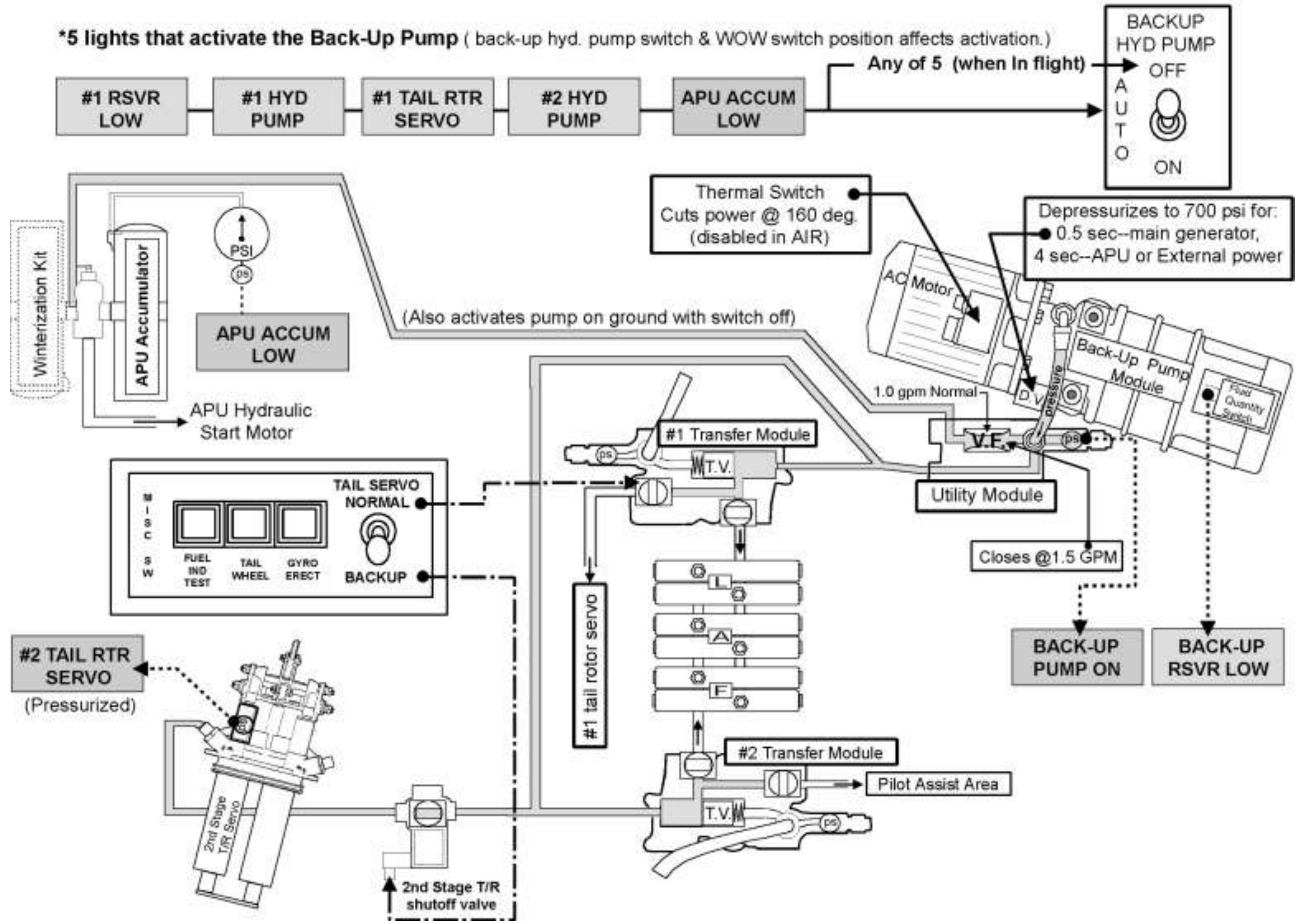
UH-60 #1 HYDRAULIC SYSTEM

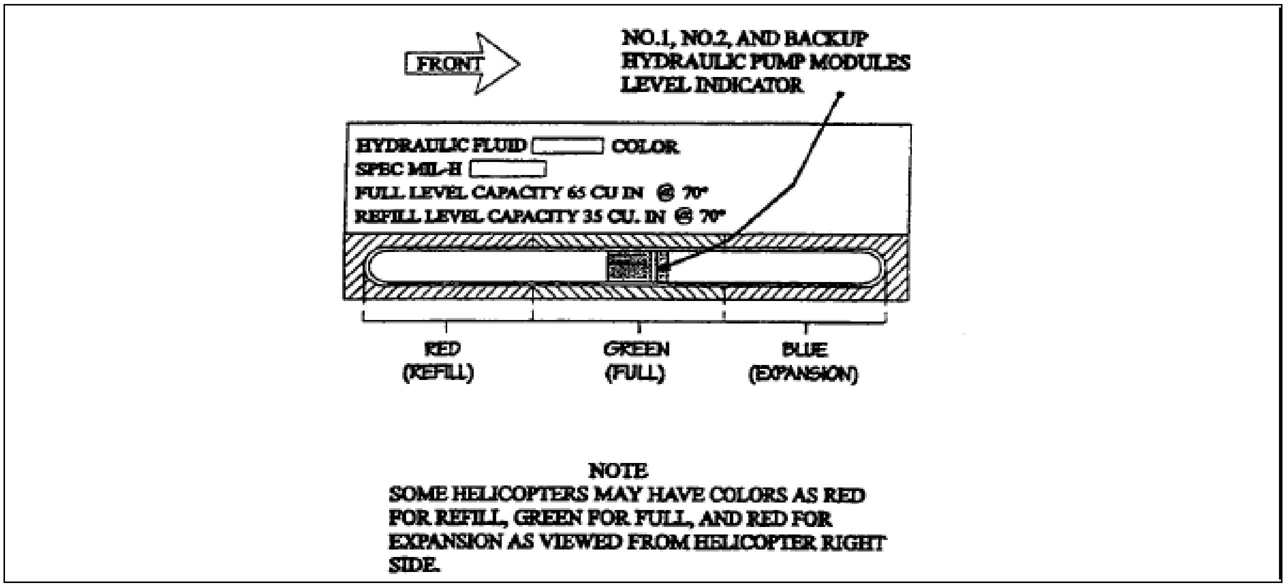


UH-60 #2 HYDRAULIC SYSTEM



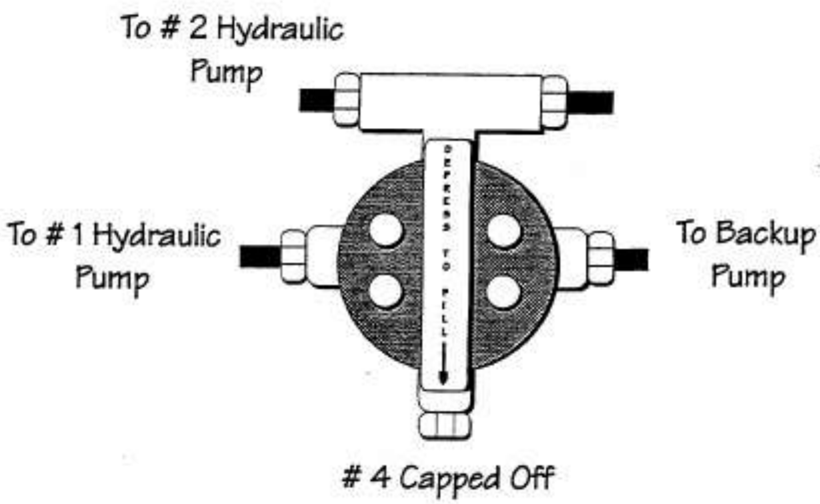
UH-60 BACKUP HYDRAULIC SYSTEM



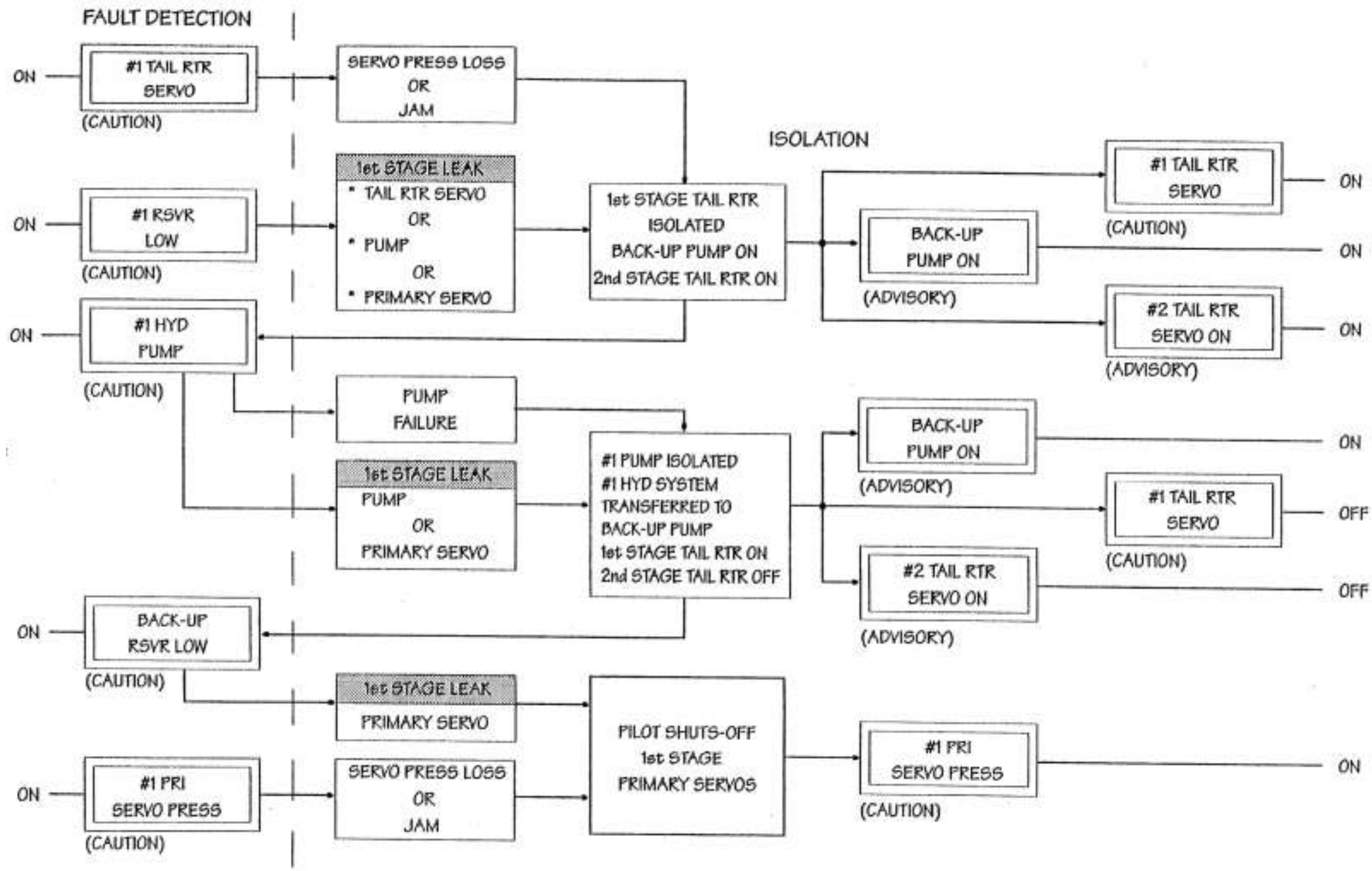


Hydraulic fluid levels may be up to 3/8” into the overfill zone (black or blue depending on type of reservoir) when fluid is hot.

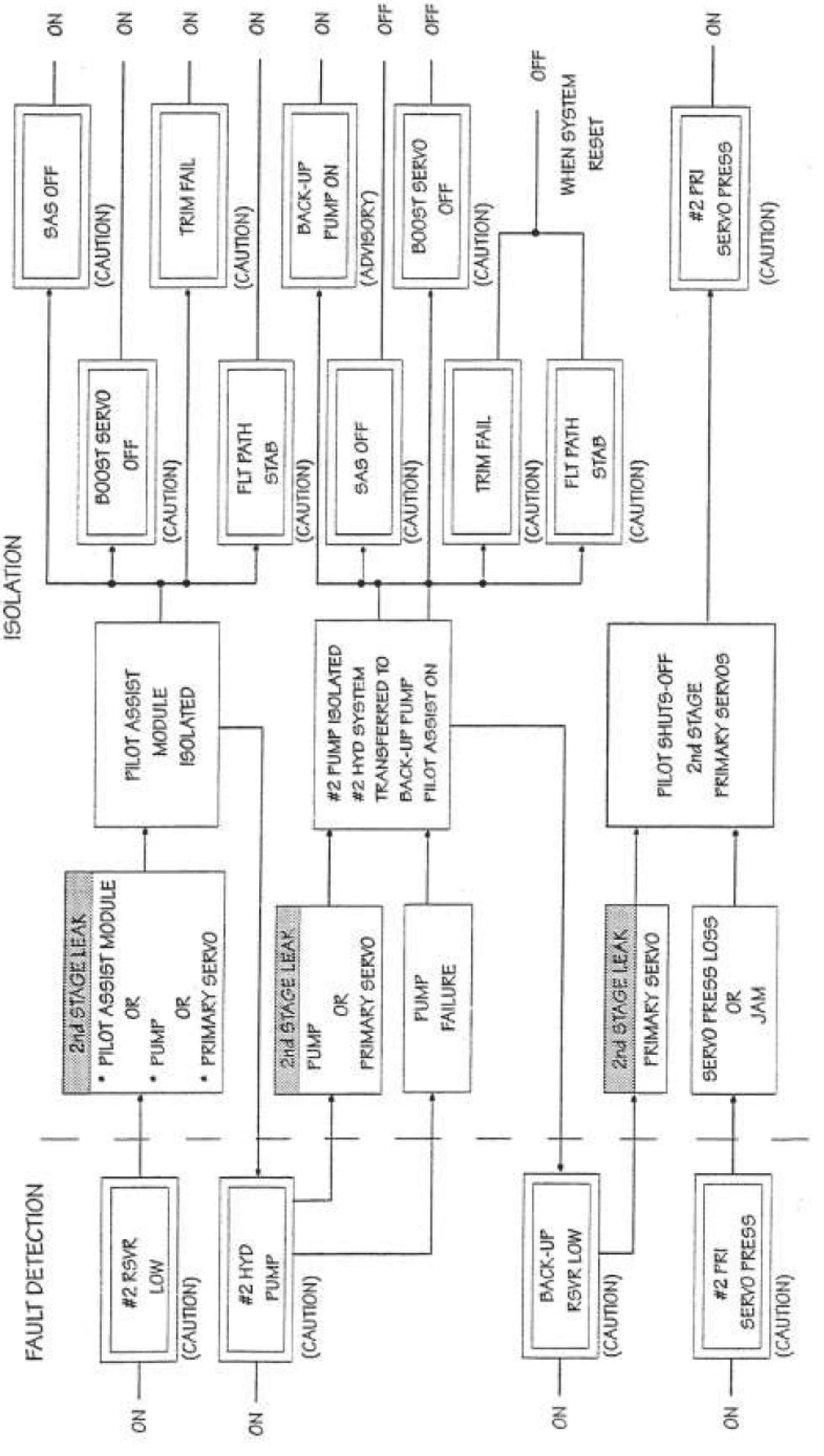
There are temperature sensitive labels used on all hydraulic pumps. There should be one label on the pressure side and one label on the return side. When the temperature exceeds 132°C (270°F) on the label an entry shall be made in the on DA Form 2408-13-1. The aircraft should not be flown until appropriate maintenance action has been taken.



The hydraulic pump modules on the aircraft consist of a combination hydraulic pump and reservoir that is sealed from the atmosphere. A spring in the reservoir maintains a positive pressure in the system at all times. Because the system is sealed, a servicing hand pump and reservoir are incorporated to allow for replenishment of hydraulic fluid. A selector valve allows for the servicing hand pump to be connected to the #1 hydraulic pump, the #2 hydraulic pump, or the backup hydraulic pump (labeled #3 on the selector valve). When not in use the selector valve should be rotated to the #4 (capped) position. To fill a hydraulic reservoir, select the desired position, and while depressing the selector valve, rotate the servicing hand pump handle in the direction indicated on the reservoir (clockwise). Use caution not to overfill the reservoir being serviced. Upon completion of servicing, rotate the selector valve to the #4 (capped-off) position.



1st STAGE HYDRAULIC SYSTEM FAULT DETECTION AND ISOLATION CHART



2nd STAGE HYDRAULIC SYSTEM FAULT DETECTION AND ISOLATION CHART

ELECTRICAL SYSTEM

Alternating current (AC) is the primary source of electrical power for the UH-60. The AC power priority allows the #1 and/or #2 main generator(s) to automatically supersede the APU generator, which in turn will supersede external power (on the ground), or the battery (in flight).

Two generators must be operating for the blade de-ice system to function. If one main generator should fail and cannot be reset, the APU must be started to provide power for blade de-ice. The APU GEN ON advisory light will not appear during this time.

Primary DC power is obtained by two 115 VAC to 28 VDC converters. The battery is the secondary source providing power to the BATT UTIL BUS, BATT BUS, and DC ESS BUS only when AC power is not available.

The distribution of electrical power is as follows for a normal start sequence:

1. With the battery connected, power is supplied to the Battery Utility Bus.

2. When the battery switch is placed on, power is supplied to the Battery Bus and the DC Essential bus.

3. After the APU is started and the APU generator switch is turned on, AC power from the APU generator is supplied to the #2 AC Primary Bus. From there it goes through the 60 amp current limiters to the #1 AC Primary Bus. The #1 AC Primary Bus then supplies power to the AC Essential Bus.

4. AC power from each AC Primary Bus supplies power to each converter, which supply DC power to their respective DC Primary Busses. The #1 DC Primary Bus supplies power to the DC Essential Bus and the Battery Bus.

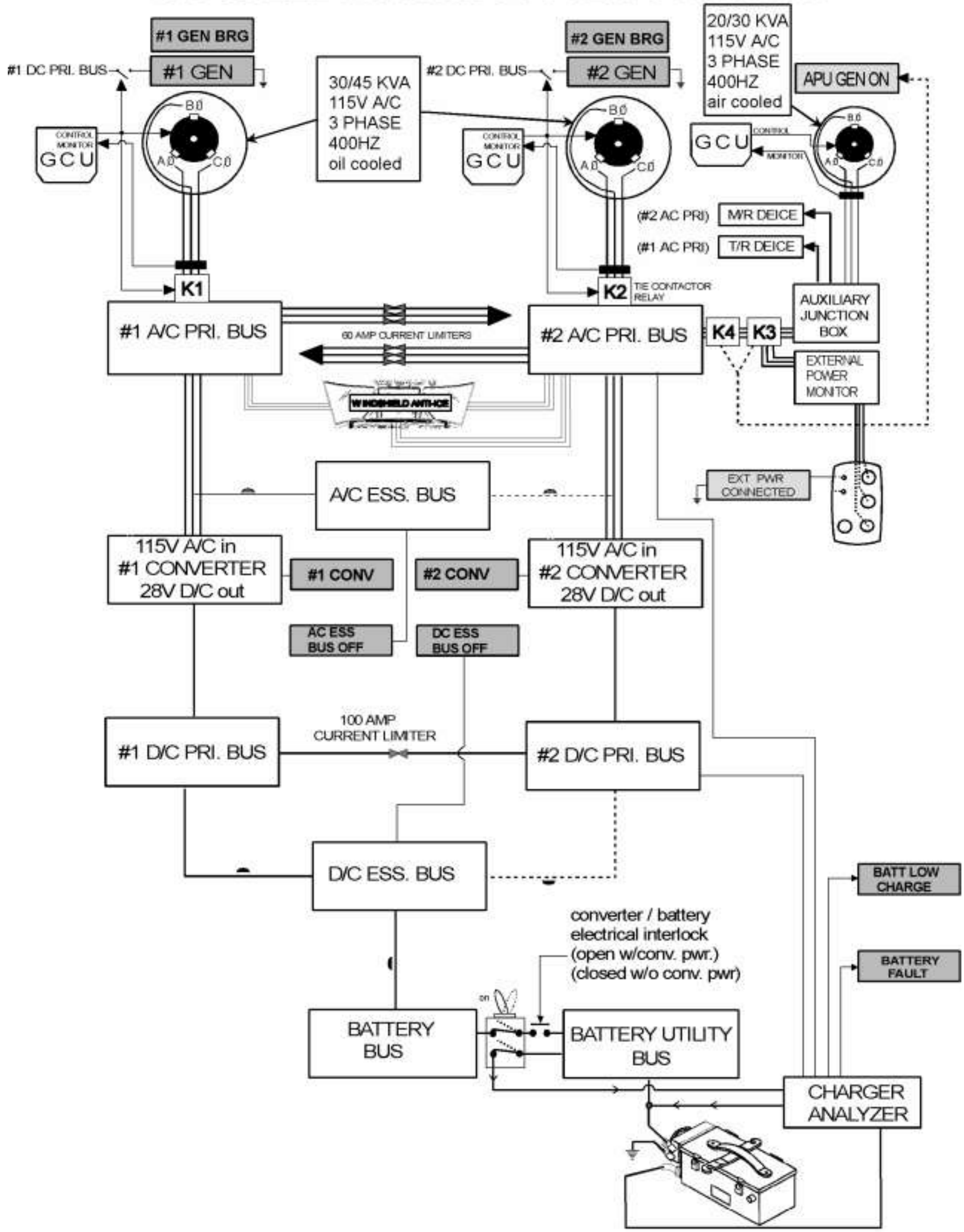
5. When the main generators come on line, they supply AC power to the AC Primary Busses. The AC power priority will supersede the APU generator, and the APU GEN ON advisory light will go off.

Should a generator be disconnected by its GCU, 60 amp current limiters will connect the AC Primary Bus of the disconnected generator with the other AC Primary Bus. Current limiters are designed to protect the good primary bus from a possible short circuit in the faulty primary bus. The only way for the AC Essential Bus to receive power from the #2 AC Primary Bus is for the #1 AC Primary Bus not to be powered.

Should a converter fail, a 100 amp current limiter will connect the DC Primary Bus of the failed converter with the other DC Primary Bus. The current limiter is designed to protect the good primary bus from a possible short circuit in the faulty primary bus. The only way for the DC Essential Bus to receive power from the #2 DC Primary Bus is for the #1 DC Primary Bus not to be powered.

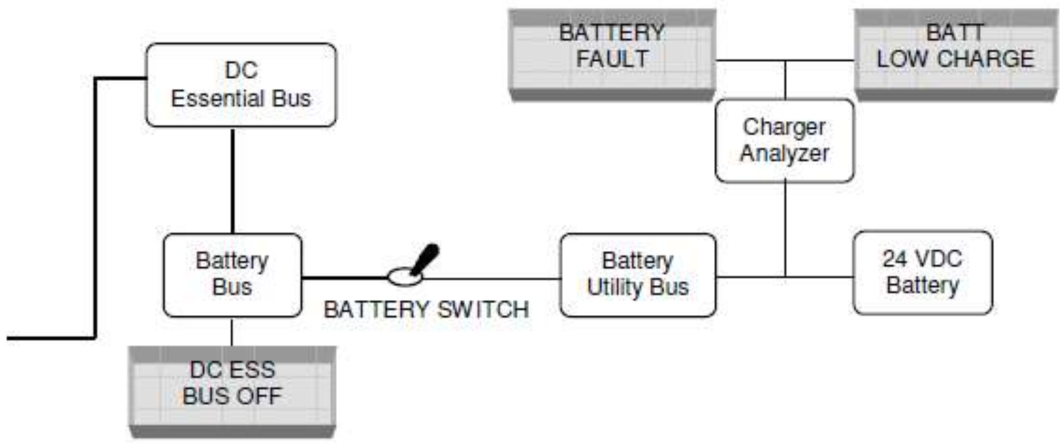
The APU generator cannot supply power to windshield anti-ice and the backup pump concurrently. Priority is given to the backup pump.

UH-60 ELECTRICAL SYSTEM OVERVIEW

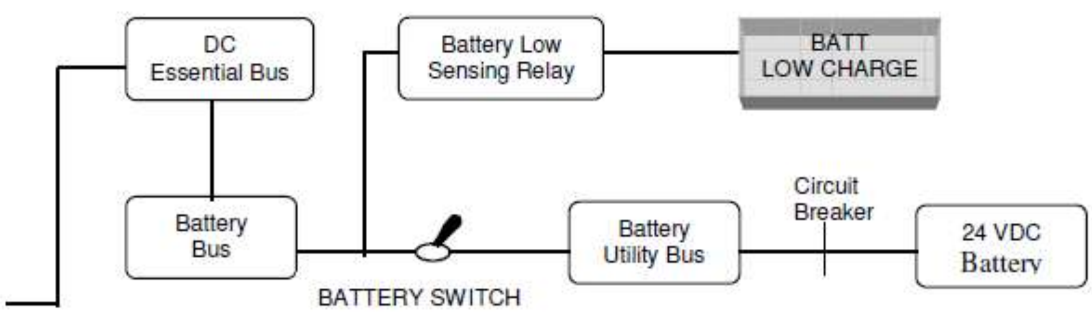


Battery power is supplied from one of two different types of batteries. A Nickel Cadmium wet cell, open vent type battery or a flooded wet cell sealed lead acid battery (SLAB). Both batteries supply 24Vdc power for secondary or emergency power.

The NiCad battery is a 24 vdc 5.5 amp hour 20-cell battery supplies dc power to the battery and battery utility buses for operating dc essential equipment during primary dc malfunction. Battery condition is monitored and maintained by a charger analyzer. The charger analyzer charges the battery through a converter, whenever AC power is applied. One of the cells monitors internal temperature and cell dissimilarity. An internal battery temperature of 70°C or dissimilarity between cells will cause the BATT FAULT caution light to illuminate, and the battery will be disconnected from the charging circuit by the charger analyzer. When only battery power is available, and the battery is charged to 80%, battery life will be approximately 22 minutes day and 14 minutes night. At 35 to 45% charge the BATT LOW caution light will illuminate. At 30 to 40% charge the DC essential Bus will be disconnected from the battery. At 35% capacity the battery can provide two APU starts. If the battery is left ON the battery will completely discharge in less than 3.5 hours.



The sealed lead acid battery (SLAB) is also a 24 vdc 9.5 amp hour, sealed battery. The battery is charged by the battery-charging relay. A battery low sensing relay will cause the BATT LOW CHARGE caution light to illuminate when the battery charge reaches 23 volts. The battery will not be disconnected from the dc essential bus. No internal monitoring of the battery is possible, and the BATT FAULT caution light cannot illuminate. For a battery at 80% charged, battery life will be approximately 38 minutes day, and 24 minutes night. If the battery is left ON the battery will completely discharge in less than 6 hours.



FUEL SYSTEM

The aircraft fuel system consists of lines from two interchangeable main fuel tanks, firewall-mounted fuel selector valves, prime/boost pump, and engine driven boost pumps.

Fuel from the main tanks is, by design, provided to the engine by the low-pressure suction engine driven boost pumps, resulting in a negative pressure in the fuel lines. Two electrically operated submerged boost pumps provide positive pressure fuel (25-27 psi) from the main tanks.

An automatic engine fuel prime feature is activated during engine start and stops when the engine starter drops out. During single engine starts with the engine fuel system selector in cross-feed the automatic prime feature is unable to prime the engine being started if the other engine is already operating.

FUEL LOW caution light(s) flash and the MASTER CAUTION light on the Master Warning Panel flashes when the fuel in the respective cell reaches approximately 172 pounds.

Fuel tanks are crashworthy and ballistic-resistant. The fuel line network includes self-sealing, breakaway valves, and all main engine fuel lines are self-sealing at the fittings. The APU fuel lines are not self-sealing.

The Fuel System Selector Levers allow:

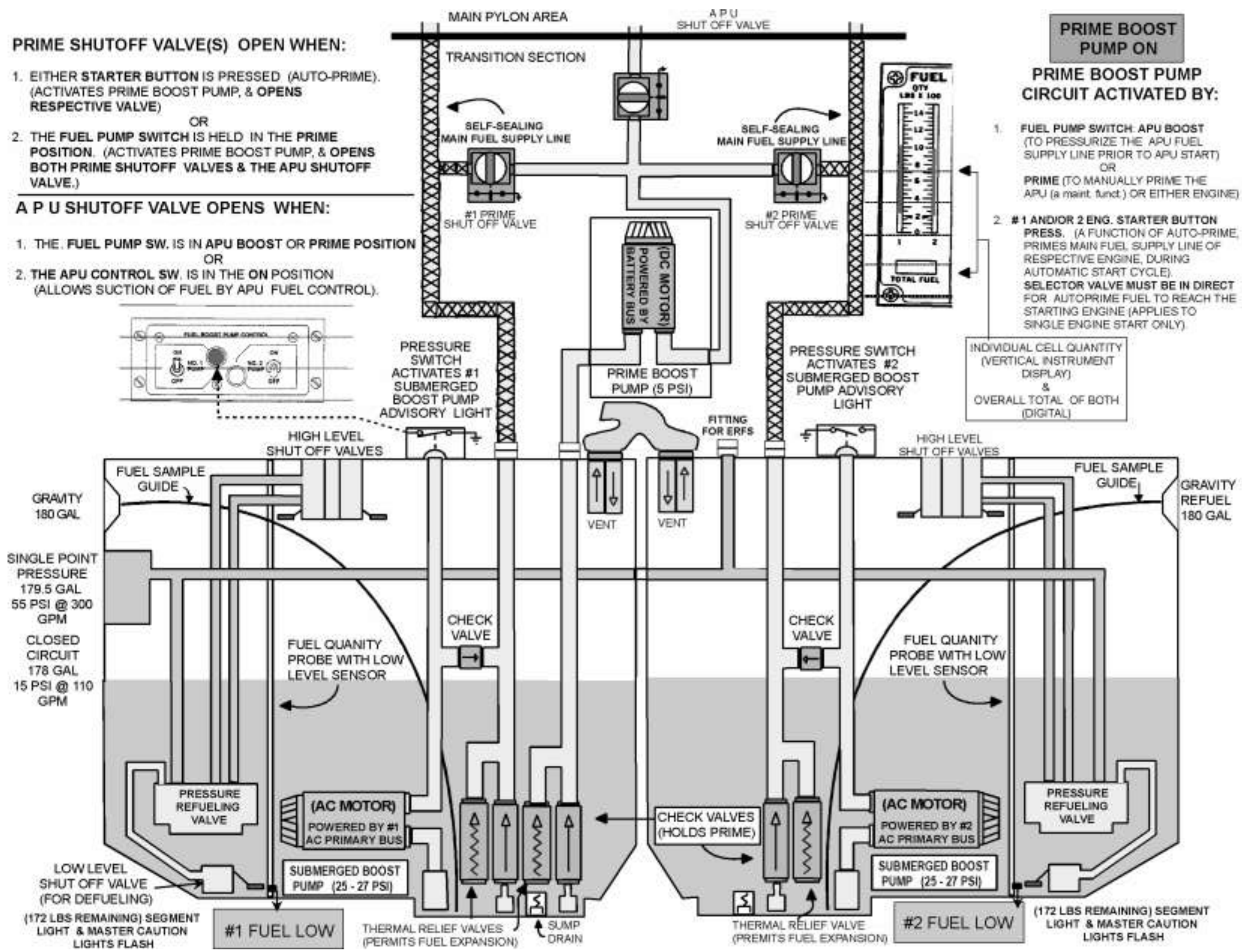
► DIR - Fuel is drawn to the respective engine (#1 fuel tank to #1 engine, and #2 fuel tank to #2 engine).

► XFD - Allows the engines to draw fuel from the opposite tank (#1 engine draws fuel from the #2 fuel tank, and #2 engine draws fuel from the #1 fuel tank).

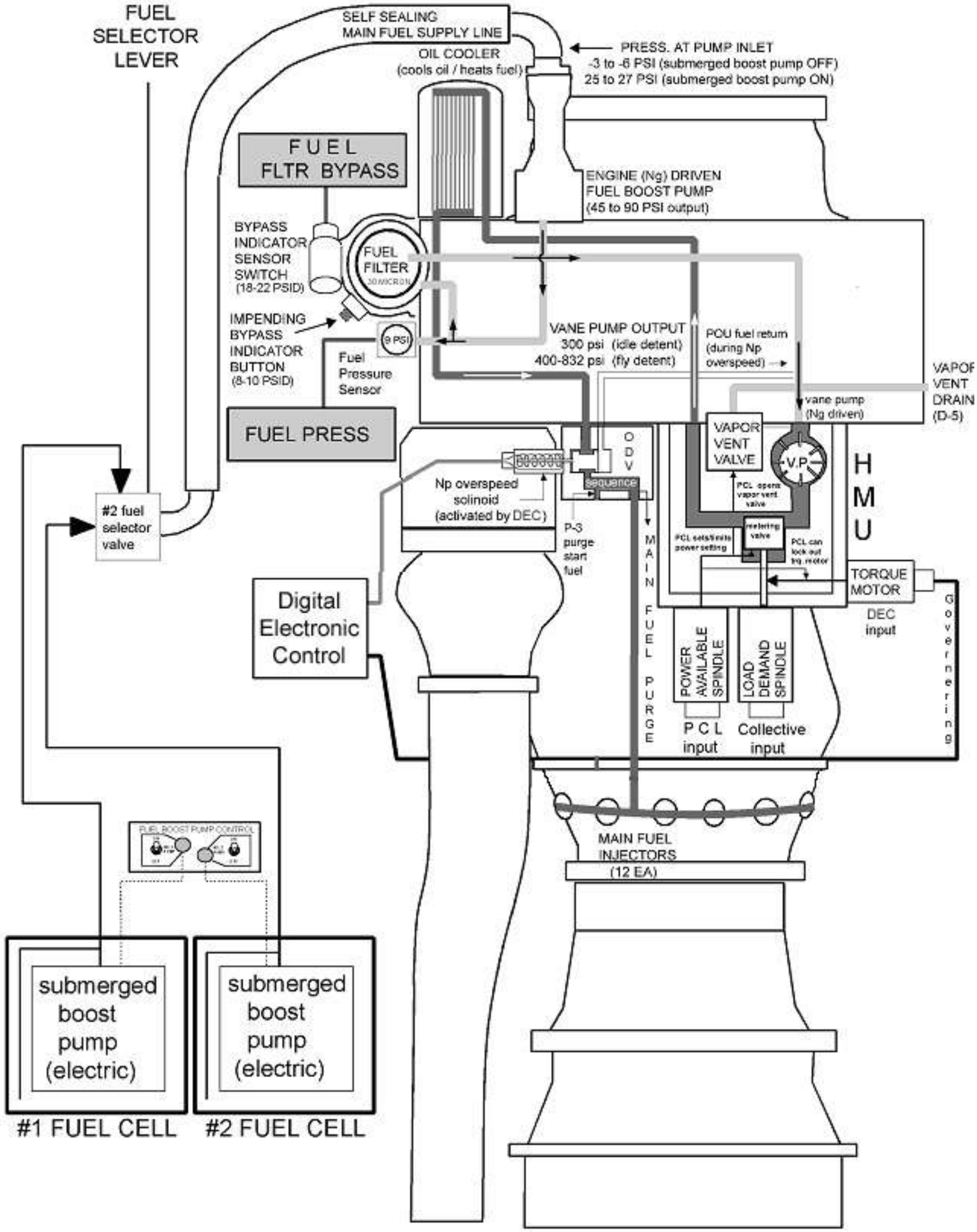
► OFF - Fuel to the main engines is shut off (valves closed). Fuel for the APU is drawn from the #1 fuel cell ONLY.

The engine fuel supply system consists primarily of the low pressure suction engine driven boost pump, fuel filter, fuel filter bypass valve, fuel pressure sensor, hydro mechanical unit (HMU), and overspeed drain valve (ODV).

Fuel is drawn from a main tank up through the fuel selector valve and proceeds through the low pressure suction engine driven boost pump, fuel filter, HMU, liquid-to-liquid cooler, then to the ODV, out to the main fuel nozzles for starting and engine operation.


PRIME SHUTOFF VALVE(S) OPEN WHEN:
1. EITHER STARTER BUTTON IS PRESSED (AUTO-PRIME). (ACTIVATES PRIME BOOST PUMP, & OPENS RESPECTIVE VALVE)
OR
2. THE FUEL PUMP SWITCH IS HELD IN THE PRIME POSITION. (ACTIVATES PRIME BOOST PUMP, & OPENS BOTH PRIME SHUTOFF VALVES & THE APU SHUTOFF VALVE.)
A P U SHUTOFF VALVE OPENS WHEN:
1. THE. FUEL PUMP SW. IS IN APU BOOST OR PRIME POSITION
OR
2. THE APU CONTROL SW. IS IN THE ON POSITION (ALLOWS SUCTION OF FUEL BY APU FUEL CONTROL).
MAIN PYLON AREA
A P U SHUT OFF VALVE
TRANSITION SECTION
SELF-SEALING MAIN FUEL SUPPLY LINE
SELF-SEALING MAIN FUEL SUPPLY LINE
#1 PRIME SHUT OFF VALVE
#2 PRIME SHUT OFF VALVE
(DC MOTOR) POWERED BY BATTERY BUS
PRIME BOOST PUMP (5 PSI)
PRESSURE SWITCH ACTIVATES #1 SUBMERGED BOOST PUMP ADVISORY LIGHT
PRESSURE SWITCH ACTIVATES #2 SUBMERGED BOOST PUMP ADVISORY LIGHT
FITTING FOR ERFS
HIGH LEVEL SHUT OFF VALVES
HIGH LEVEL SHUT OFF VALVES
FUEL SAMPLE GUIDE
FUEL SAMPLE GUIDE
VENT
VENT
GRAVITY 180 GAL
GRAVITY REFUEL 180 GAL
SINGLE POINT PRESSURE 179.5 GAL 55 PSI @ 300 GPM
CLOSED CIRCUIT 178 GAL 15 PSI @ 110 GPM
CHECK VALVE
CHECK VALVE
FUEL QUANITY PROBE WITH LOW LEVEL SENSOR
FUEL QUANITY PROBE WITH LOW LEVEL SENSOR
PRESSURE REFUELING VALVE
PRESSURE REFUELING VALVE
(AC MOTOR) POWERED BY #1 AC PRIMARY BUS
(AC MOTOR) POWERED BY #2 AC PRIMARY BUS
SUBMERGED BOOST PUMP (25 - 27 PSI)
SUBMERGED BOOST PUMP (25 - 27 PSI)
CHECK VALVES (HOLDS PRIME)
LOW LEVEL SHUT OFF VALVE (FOR DEFUELING)
(172 LBS REMAINING) SEGMENT LIGHT & MASTER CAUTION LIGHTS FLASH
#1 FUEL LOW
THERMAL RELIEF VALVES (PERMITS FUEL EXPANSION)
SUMP DRAIN
THERMAL RELIEF VALVE (PREMITS FUEL EXPANSION)
#2 FUEL LOW
(172 LBS REMAINING) SEGMENT LIGHT & MASTER CAUTION LIGHTS FLASH
PRIME BOOST PUMP ON
PRIME BOOST PUMP CIRCUIT ACTIVATED BY:
1. FUEL PUMP SWITCH: APU BOOST (TO PRESSURIZE THE APU FUEL SUPPLY LINE PRIOR TO APU START)
OR
PRIME (TO MANUALLY PRIME THE APU (a maint. funct.) OR EITHER ENGINE)
2. #1 AND/OR 2 ENG. STARTER BUTTON PRESS. (A FUNCTION OF AUTO-PRIME, PRIMES MAIN FUEL SUPPLY LINE OF RESPECTIVE ENGINE, DURING AUTOMATIC START CYCLE). SELECTOR VALVE MUST BE IN DIRECT FOR AUTOPRIME FUEL TO REACH THE STARTING ENGINE (APPLIES TO SINGLE ENGINE START ONLY).
INDIVIDUAL CELL QUANTITY (VERTICAL INSTRUMENT DISPLAY)
&
OVERALL TOTAL OF BOTH (DIGITAL)
FUEL
TOTAL FUEL

701C FUEL SYSTEM SCHEMATIC



HYDROMECHANICAL UNIT (HMU)
T-700-GE-701C

The HMU is a basic fuel control that includes a high-pressure pump and a variable geometry servo-actuator. It meters required fuel to the engine as a function of Ng (compressor speed), T2 (compressor inlet temperature), P3 (compressor discharge pressure), and trimming signals from the DEC, plus inputs from the Power Available Spindle (PAS) and Load Demand Spindle (LDS).

The functions of the HMU are:

1. High Pressure Fuel **P**umping - Fuel needs to be at a high pressure to provide a precise spray pattern and good atomization in the combustion chamber.

2. Fuel Flow **M**etering - The HMU meters the proper amount of fuel to the POU in response to the PAS position, LDS position, torque motor servo inputs from the DEC, and sensed engine variables.

3. **C**ollective Pitch Compensation through the LDS - When the collective is moved, the desired Ng is reset to provide immediate and accurate gas producer response. The new setting is trimmed by the DEC via the torque motor servo. This helps prevent rotor droop as collective is increased and helps prevent rotor overspeed as collective is reduced

4. **A**cceleration/Deceleration Flow Limiting – Prevents the Ng from accelerating or decelerating faster than the engine can safely handle. Uncontrolled acceleration could cause compressor stalls or engine damage and uncontrolled deceleration could cause an engine flame out.

5. **N**g Limiting - Limits maximum torque available under low temperature conditions.

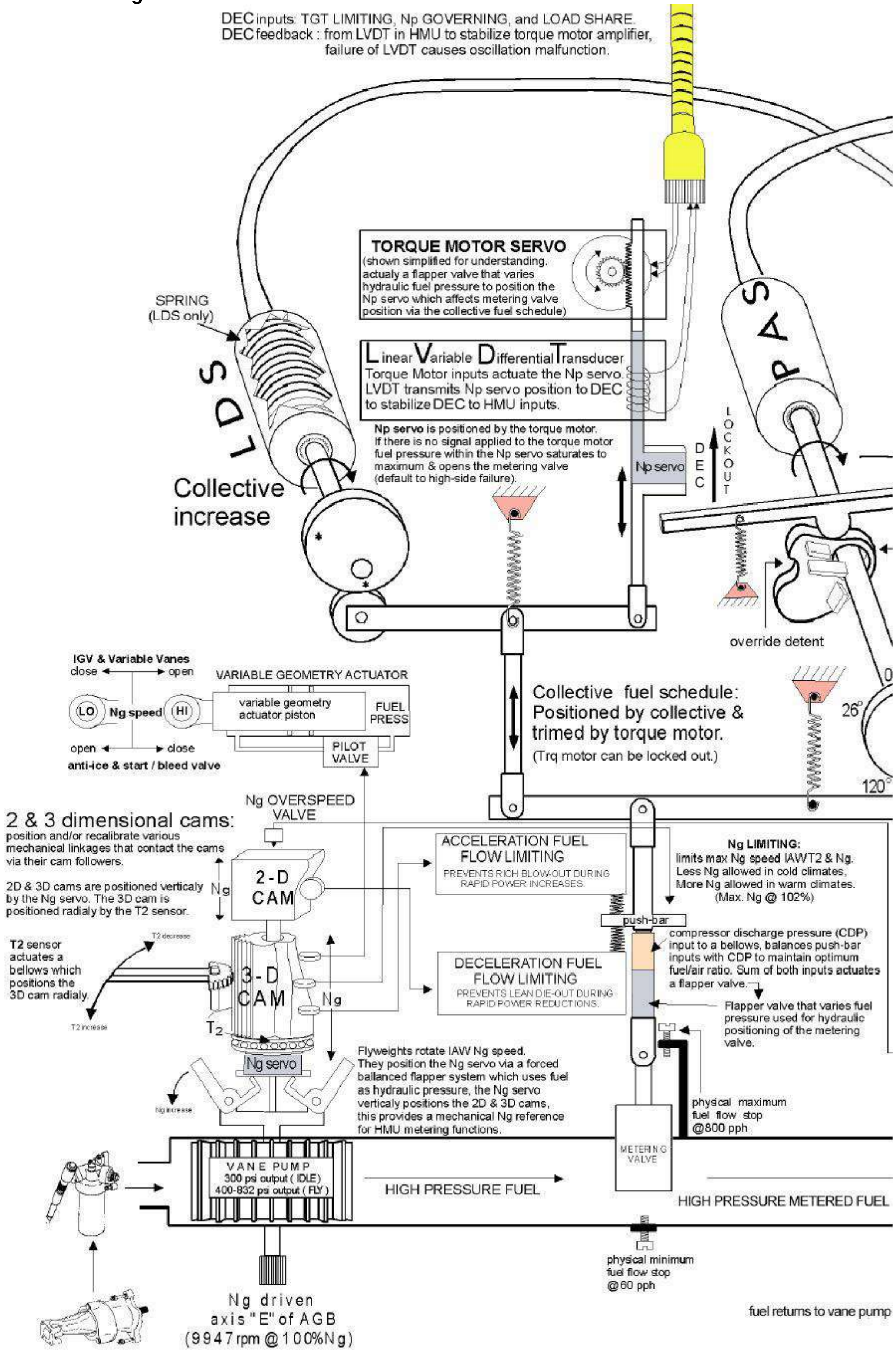
6. **V**ariable Geometry Positioning - The HMU positions the inlet guide vanes and the first two stages of variable guide vanes. The use of a variable geometry system permits optimum performance over a wide range of operating conditions, facilitates rapid stall-free accelerations, and optimizes fuel flow under partial power conditions.

7. D**E**C lockout via the PAS (Power Available Spindle) Override detent - Allows the pilot to mechanically bypass the torque motor inputs and control the engine manually by advancing the PAS past the fly detent, to the lockout position and then retarding the PAS to trim the engine speed manually. This is known as DEC lockout. TGT limiting, adjusting the Np reference from the cockpit with the engine speed trim switch (INCR/DECR), Load sharing/Tq matching, and Np governing are disabled during DEC lockout. The pilot must keep TGT within limits and Np within the operating range. Only the torque motor servo is being overridden, all other HMU functions will still function normally. (Note: DEC lockout will have no effect on the Np over speed protection system for that engine.)

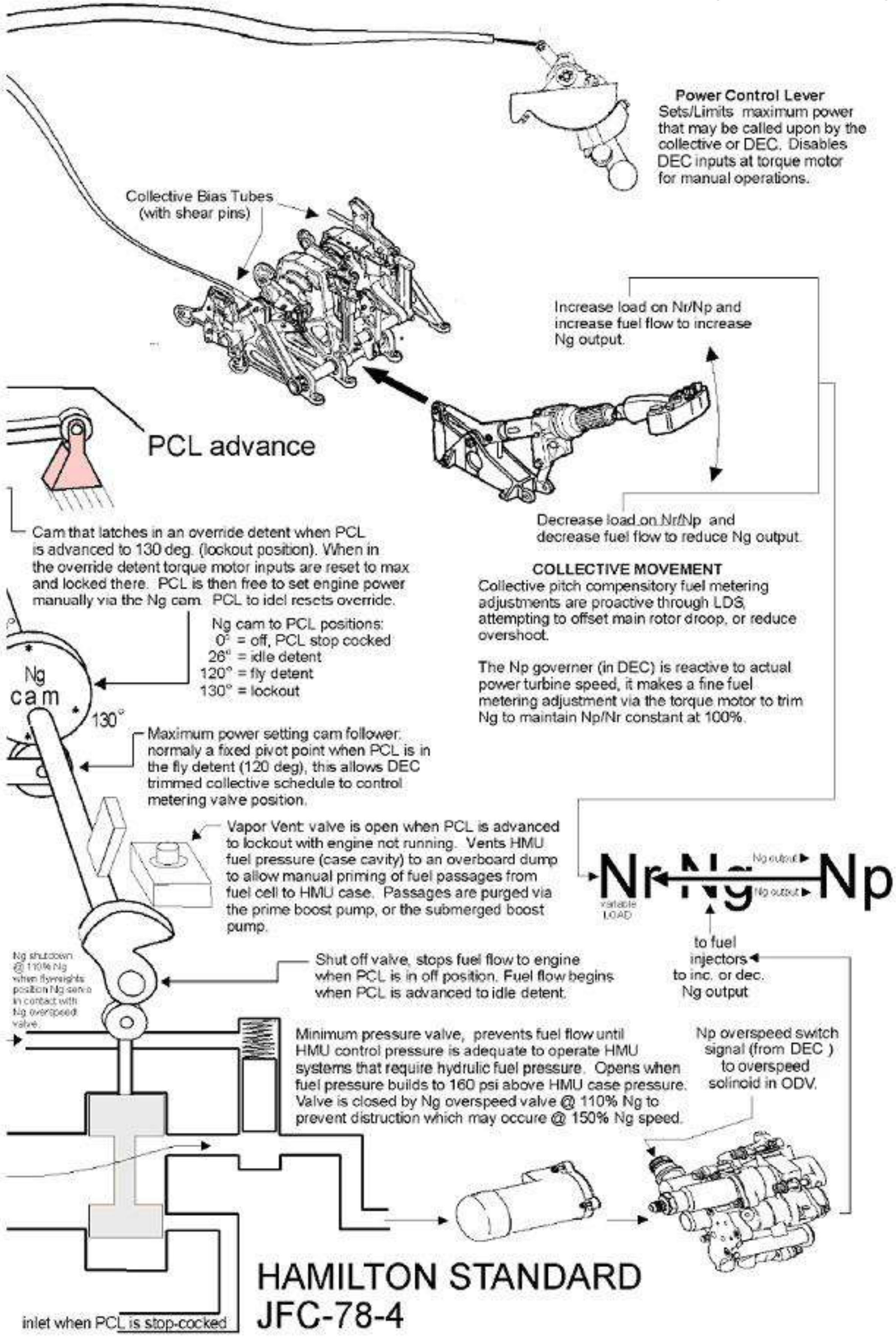
8. **T**orque Motor Servo to Trim Ng Output - The DEC sends signals to the torque motor servo in the HMU where the electrical input is changed to a mechanical input utilizing fuel pressure in the HMU case. The torque motor can retard Ng no further than ground idle (63%), and can allow power to increase no further than that set by the PAS setting. The DEC uses the torque motor to fine-tune the output of the engine to meet its requirements. The torque motor servo may be overridden by advancing the engine power control lever(s) to DEC lockout.

9. **O**pens Vapor Vent for HMU Priming - Certain situations require that the HMU be primed (air removed from within the HMU). By advancing the power control lever past the fly detent to the lockout position, the vapor vent is opened. When the PCL is retarded to the idle position, the vapor vent closes.

Left Side HMU Diagram



Right Side HMU Diagram



OVERSPEED AND DRAIN VALVE (ODV)
T700-GE-701 Engine

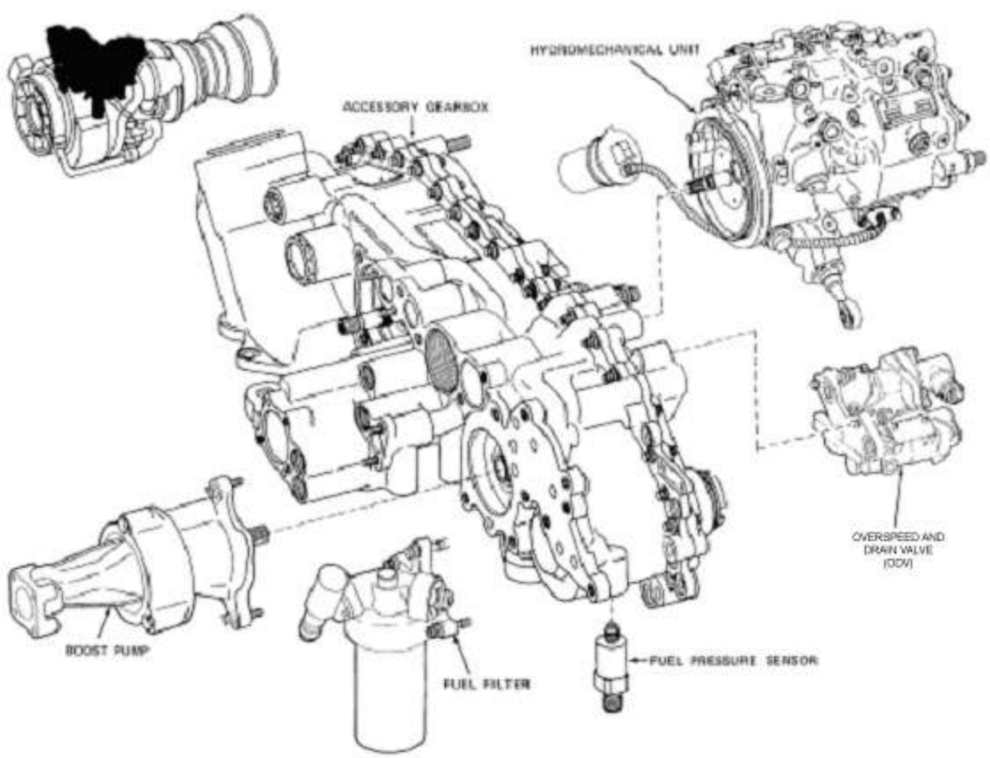
The ODV sends fuel through the main fuel manifold to the injectors for starting acceleration and engine operations. It also provides shutoff drain, start fuel & main fuel purge functions, and a reduced flow schedule for overspeed protection.

The ODV performs the following functions:

1. Sequences main fuel for engine starting and for engine operation.

2. Purges main fuel manifold and nozzles during engine start (after 3 seconds) and main fuel manifold and nozzles on shutdown to prevent coking.

3. Shuts off fuel to provide Np overspeed protection when activated from the overspeed circuit in the DEC at 120%+/-1%.



FUEL SYSTEM COMPONENTS

DIGITAL ELECTRONIC CONTROL (DEC)
T700-GE-701 ENGINE

The DEC controls the electrical functions of the engines and transmits operational information to the cockpit.

1. Trims the HMU output through the torque motor servo as determined by:

► Isochronous Np Governing - Simply stated, the DEC will maintain Np at the reference speed set by the pilot (Normally 100%). This is the DEC’s primary function, and is overridden only by TGT limiting.

► TGT Limiting - When the TGT reaches approximately 866°C (10-minute limiting value) or approximately 891°C (2 ½-minute limiting value), the TGT limit amplifier overrides the Np governing, load sharing channels, and limits fuel flow to hold a constant TGT. It is possible to see a transient increase above 903°C when the pilot demands maximum power. TGT limiting does not prevent overtemperature during engine starts, compressor stalls, or during DEC lockout operations.

► Np Reference from the cockpit - The pilot sets the reference %RPM (Np 1 and 2) in the DEC by adjusting the ENG RPM speed trim (INCR/DECR) switch on either collective. This is normally set at 100% but is adjustable from 96-100%.

► Load Sharing - The torque matching and load sharing system increases power on the lower-torque engine to keep engine torques approximately equal. The system does not allow an engine to reduce power to match a lower power engine. If an engine fails to the high side, the good engine will only try to increase torque upward until its Np is 3% above the reference Np.

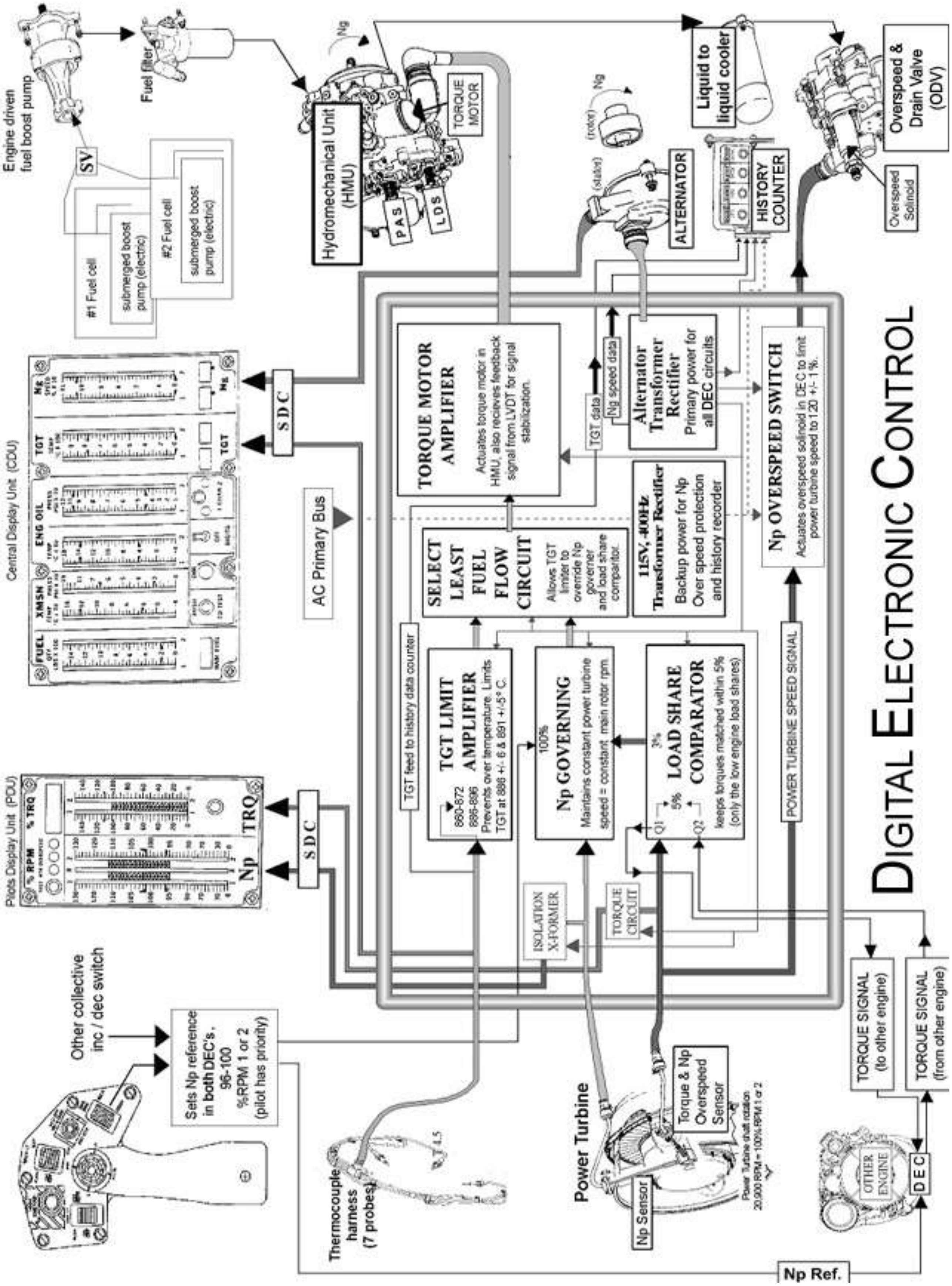
2. Np Overspeed System - The overspeed system includes two speed sensing circuits (A and B). Each circuit is calibrated to close a switch when Np reaches 120±1%. Both switches must close before the overspeed solenoid in the ODV will be energized. The switches will open when Np is reduced below 120±l%. This will cause a cycling of the over speed protection system as long as the reason for the overspeed condition exists. The overspeed circuits receive power from the engine alternator. As a backup, in the event of an alternator failure, the overspeed circuits receive power from the aircraft electrical system.

3. Sends Np, and torque signals to the cockpit. A TGT signal (pass through only) is also sent via the DEC to the cockpit.

4. Sends signals to the history counter. (Maintenance function)

CAUTION

When an engine is controlled with the engine power control lever in lockout or with the DEC inoperative, TGT limiting is inoperative and engine response is much faster. The pilot must ensure that TGT limits are not exceeded and that % RPM R and % RPM 1 and/or 2 are kept within operating limits.


Engine driven fuel boost pump
Fuel filter
SV
#1 Fuel cell
submerged boost pump (electric)
#2 Fuel cell
submerged boost pump (electric)
Hydromechanical Unit (HMU)
PAS
LDS
TORQUE MOTOR
Ng
Liquid to liquid cooler
(stator)
(rotor)
ALTERNATOR
HISTORY COUNTER
Overspeed Solinoid
Overspeed & Drain Valve (ODV)
Central Display Unit (CDU)
SDC
AC Primary Bus
TORQUE MOTOR AMPLIFIER
Actuates torque motor in HMU, also recieves feedback signal from LVDT for signal stabilization.
TGT data
Ng speed data
Alternator Transformer Rectifier
Primary power for all DEC circuits
SELECT LEAST FUEL FLOW CIRCUIT
Allows TGT limiter to override Np governer and load share comparitor.
115V, 400Hz Transformer Rectifier
Backup power for Np Over speed protection and history recorder
Np OVERSPEED SWITCH
Actuates overspeed solinoid in DEC to limit power turbine speed to 120 +/- 1%.
Pilots Display Unit (PDU)
Np
TRQ
TGT feed to history data counter
TGT LIMIT AMPLIFIER
860-872
888-896
Prevents over temperature. Limits TGT at 888 +/- 6 & 891 +/-5° C.
100%
Np GOVERNING
Maintains constant power turbine speed = constant main rotor rpm.
3%
Q1
5%
Q2
LOAD SHARE COMPARATOR
keeps torques matched within 5% (only the low engine load shares)
POWER TURBINE SPEED SIGNAL
DIGITAL ELECTRONIC CONTROL
Other collective inc / dec switch
Sets Np reference in both DEC's, 96-100 %RPM 1 or 2 (pilot has priority)
ISOLATION X-FORMER
TORQUE CIRCUIT
Thermocouple harness (7 probes)
T4.5
Power Turbine
Torque & Np Overspeed Sensor
Np Sensor
Power Turbine shaft rotation 20,900 RPM = 100% RPM 1 or 2
TORQUE SIGNAL (to other engine)
TORQUE SIGNAL (from other engine)
OTHER ENGINE
DEC
Np Ref.

ENGINE ALTERNATOR
T700-GE-701 Engine

The engine alternator is mounted on the accessory gearbox of the engine. It contains three separate internal windings for its three functions that include:

1. Power to the ignition exciters during start (black engine wire).

2. Power to the DEC (yellow engine wire).

3. Ng signal which is sent to the cockpit CDU via the SDC's (green engine wire).

When the alternator Ng signal is interrupted, a loss of the associated engine Ng indication and **ENG OUT** warning and audio will occur. Because the DEC can utilize aircraft power, there will be no loss of associated **% RPM 1** or **2** and **% TRQ** indications.

If the alternator power supply to the DEC is interrupted, aircraft power is used to prevent engine (high side) failure. There will be a loss of the associated cockpit Ng indication and activation of **ENG OUT** warning and audio. Overspeed protection is still available.

MECHANICAL MIXING UNIT (MMU)

The mechanical mixing unit is designed to reduce inherent control coupling. (Inherent control coupling might be referred to as a side effect of a control input. You increase collective to bring the aircraft up to a hover. However, a side effect of this is the increased torque causing the nose to turn to the right). This is an average mixing for all flight conditions and is not based on any particular weight. The pilot, with assistance from AFCS, will make the adjustments needed if these mixings are too much or too little. The mechanical mixing unit helps reduce pilot workload through the following inputs and outputs:

1. **Collective** to **Pitch** mixing Compensates for downwash on the aft fuselage and stabilator. As collective is increased, the main rotor disk is tilted forward slightly, and as collective is decreased the main rotor disk is tilted aft slightly.

2. **Collective** to **Roll** mixing Compensates for translating tendency. As the collective is increased the main rotor disk is tilted slightly left and as the collective is decreased the main rotor disk is tilted slightly back toward the right.

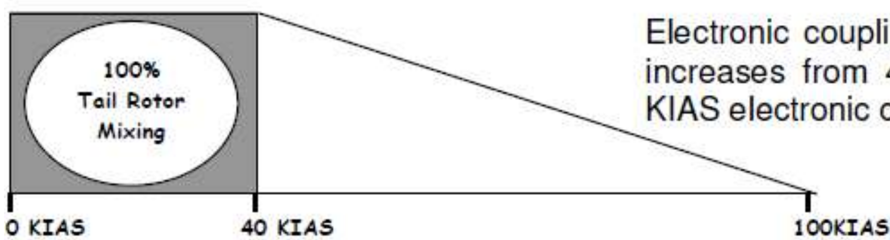
3. **Collective** to **Yaw** mixing Compensates for the torque effect of the main rotor. As collective is increased, tail rotor pitch is increased and as collective is decreased, tail rotor pitch is decreased.

4. **Yaw** to **Pitch** mixing Compensates for tail rotor lift vectors. As tail rotor pitch is increased the main rotor disk is tilted slightly aft, and as tail rotor pitch is decreased the main rotor disk is tilted slightly forward. The tail-rotor on the UH-60 is mounted to provide a vertical lift component to help offset the aft CG component associated with airframe design.

In the above, the first word associated with that particular mix, i.e. Collective to Yaw, is the pilot input to the flight controls. With a collective input from the pilot, the mixing unit provides a pitch, roll and yaw output. With a yaw input from the pilot or from AFCS (FPS coupled with trim), the mixing unit provides a pitch output.

ELECTRONIC COUPLING

COLLECTIVE to AIRSPEED to YAW This is usually discussed with the mechanical mixing unit, although it is not a function of the MMU. This is a separate function of the SAS/FPS computer. Collective to airspeed to yaw mixing, or electronic coupling, helps compensate for the torque effect in addition to the collective to yaw mechanical mixing by increasing or decreasing tail rotor pitch by use of the yaw trim actuator. Electronic coupling uses the yaw trim actuator to vary its input to the tail rotor based on both the collective position and airspeed signals. From 0 to 40 knots, electronic coupling is at its maximum input to the yaw trim actuator. As airspeeds increase above 40 knots, the tail rotor and cambered vertical fairing become more efficient. With increasing airspeed less tail rotor pitch is required for a given collective position. Electronic coupling continually decreases the amount of tail rotor pitch it programs in until 100 knots, where no electronic coupling is required. Note that the SAS/FPS computer may override this function as needed to maintain heading hold or turn coordination.



Electronic coupling is reduced as airspeed increases from 40-100 KIAS. Above 100 KIAS electronic coupling is reduced to zero.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

The AFCS enhances the static stability and handling qualities of the helicopter.
It is comprised of four subsystems:

1. SAS
2. Trim
3. FPS
4. Stabilator

STABILITY AUGMENTATION SYSTEM (SAS)

NOTE

Control authority is defined as the amount of input a system can make to the flight controls compared to how much the pilot can move those controls.

The UH-60 incorporates two SAS systems to help maintain a stable platform in flight. SAS 1 is an analog system and SAS 2 is digital system. Both provide short term rate dampening in the pitch, roll, and yaw axes. Operation of the two SAS is essentially the same. SAS 2 has self-diagnostic capabilities where SAS 1 does not.

The design of the SAS results in control inputs to the flight control surfaces without moving cockpit controls. Each SAS has 5% control authority for a combined total of 10%. Both SAS 1 and SAS 2 utilize hydraulic pressure from the #2 hydraulic system applied to a single SAS actuator for each of the three flight axes. The SAS actuators are located in the pilot-assist area beneath the over-head cowling.

SAS 1 is controlled by the SAS 1 amplifier located in the avionics compartment. SAS 1 receives a pitch signal from the #1 stabilator amplifier, a roll signal from the pilot's vertical gyro, and a yaw signal from an internal rate gyro.

SAS 2 is controlled by the SAS/FPS computer located aft of the lower console. SAS 2 receives a pitch signal from the #2 stabilator amplifier, a roll signal from a roll rate gyro and a yaw signal from a yaw rate gyro, co-located in an EMI hardened enclosure in the avionics compartment.

Both SAS utilize additional signal inputs, which are beyond the scope of this document.

Malfunctions of SAS may be intermittent or of continuous nature. A malfunction of SAS 1 may be accompanied by "pounding" in the flight controls or erratic tip path plane movements. There are no caution lights to indicate a failure of SAS 1. If a SAS 1 failure is suspected, SAS 1 should be turned off. SAS 2 utilizes a fault monitoring system to alert the pilot of a failure. There are no caution lights associated with SAS 2, however, there are failure advisory capsule lights located on the AFCS panel. A SAS off caution light indicates that the SAS actuators are not pressurized.

In the event of a SAS failure, the pilot must turn off the affected SAS. When the pilot turns off the malfunctioning SAS utilizing the **SAS 1** or **SAS 2** switches on the AFCS panel, the remaining SAS will double its gain. The remaining SAS will have up to 5% control authority, and will work at the same speed. Doubling the gain simply means that the sensitivity of the remaining SAS has been doubled. The effect of double the gain is normally sufficient so that handling qualities are not significantly degraded.

FLIGHT PATH STABILIZATION (FPS)

The FPS system provides long term rate dampening in the pitch, roll, and yaw axes. FPS provides basic autopilot functions using the trim actuators to maintain attitude in the pitch and roll axes, and heading hold/turn coordination in the yaw axis. When FPS is coupled with trim, it has 100% control authority. Because the FPS utilizes the trim actuators, all FPS inputs will be observed by corresponding cyclic and pedal movements.

In the pitch axis, the FPS computer uses the pitch trim assembly to make required pitch attitude adjustments. The pitch trim assembly is an electro-hydro-mechanical actuator meaning that it is electrically controlled, and converts hydraulic pressure into mechanical movement. Below 60 KIAS FPS will provide attitude hold. Above 60 KIAS FPS will provide attitude hold/airspeed hold.

In the roll axis, the FPS uses the roll trim actuator to make required roll attitude adjustments. The roll trim actuator is an electro-mechanical actuator meaning that it utilizes an electric motor to drive its output linkage. Attitude hold is provided regardless of airspeed.

In the yaw axis, the FPS uses the yaw trim actuator to make required yaw adjustments. The yaw trim actuator is also electro-mechanical. Below 60 KIAS, FPS provides heading hold. Above 60 KIAS FPS provides heading hold/turn coordination. To enter turn coordination the cyclic must be displaced about ½ inch and a roll attitude of 1.5 degrees or greater.

In order for FPS to function properly SAS 1 and/or SAS 2, Boost, and Trim must be operational. The stabilator operating in automatic mode enhances FPS operation but is not required.

The SAS 2 FPS computer constantly monitors the trim actuators when FPS is turned on. If a failure is detected, both Master Warning Lights, FPS, and Trim caution lights will illuminate. One or more of the failure advisory capsule lights on the AFCS panel will also illuminate. If these lights illuminate, the pilot should note the failure advisory indications. If the malfunction is of an intermittent nature, a power on reset should clear the malfunction. If the malfunction is of a continuous nature the affected axis can be controlled manually with no FPS or force gradient in that axis.

BELOW	**60 KIAS**	***ABOVE***
Attitude hold	Pitch Axis	Attitude Hold / Air Speed Hold
Attitude Hold	Roll Axis	Attitude Hold
Heading Hold	Yaw Axis	Heading Hold / Turn Coordination

TRIM

The trim system is comprised of three trim actuators. The roll and yaw trim actuators are electro-mechanical, and the pitch trim assembly is electro-hydromechanical. The trim system by itself provides a force gradient in the pitch, roll, and yaw axes.

Proper operation of the yaw trim actuator requires that the yaw boost servo be pressurized and operational.

Both electro-mechanical actuators incorporate slip clutches that allows the pilot to overcome an internal jam in the actuator. The force required to break through the clutches are 80 lbs. maximum in yaw and 13 lbs. maximum in roll.

The SAS 2 FPS computer constantly monitors the trim actuators when FPS is turned on. If a failure is detected Master Warning Lights, the FPS and the Trim caution lights will illuminate. One or more of the failure advisory capsule lights on the AFCS panel will also illuminate. If these lights illuminate, the pilot should note the failure advisory indications. If the malfunction is of an intermittent nature, a power on reset should clear the malfunction. If the malfunction is of a continuous nature, the affected axis can be controlled manually with no FPS or force gradient in that axis.

STABILATOR

The stabilator is a variable angle of incidence airfoil that enhances the handling qualities and longitudinal control of the aircraft. The automatic mode of operation positions the stabilator to the best angle of attack for existing flight conditions. The stabilator amplifiers receive inputs from various sensors on the aircraft to perform its' functions.

These sensors are:

- Airspeed/Air data Transducers – Located by the pilot's and co-pilot's pedals, these sensors provide an electronic airspeed signal.

- Collective Position Sensor – Located on the MMU, they detect pilot collective displacement.

- Lateral Accelerometers – Located in the cabin bulkhead over the crew chief's stations, they sense an out of trim condition.

- Pitch Rate Gyros – Located in the stabilator amplifiers, they sense pitch attitude changes.

The stabilator is programmed to its optimum angle to provide the following functions:

S 1. **S**treamline with main rotor downwash at low airspeed (30 KIAS and below) to minimize nose-up attitudes resulting from the main rotor downwash on the stabilator. (Airspeed/air data transducer)

C 2. Provide **C**ollective coupling to minimize pitch attitude excursions due to collective inputs. An increase in the collective would result in the helicopter pitching up. The stabilator will program trailing edge down to prevent the nose from pitching up. As collective is reduced the stabilator will program trailing edge up to prevent the nose from pitching down. (Collective position transducer)

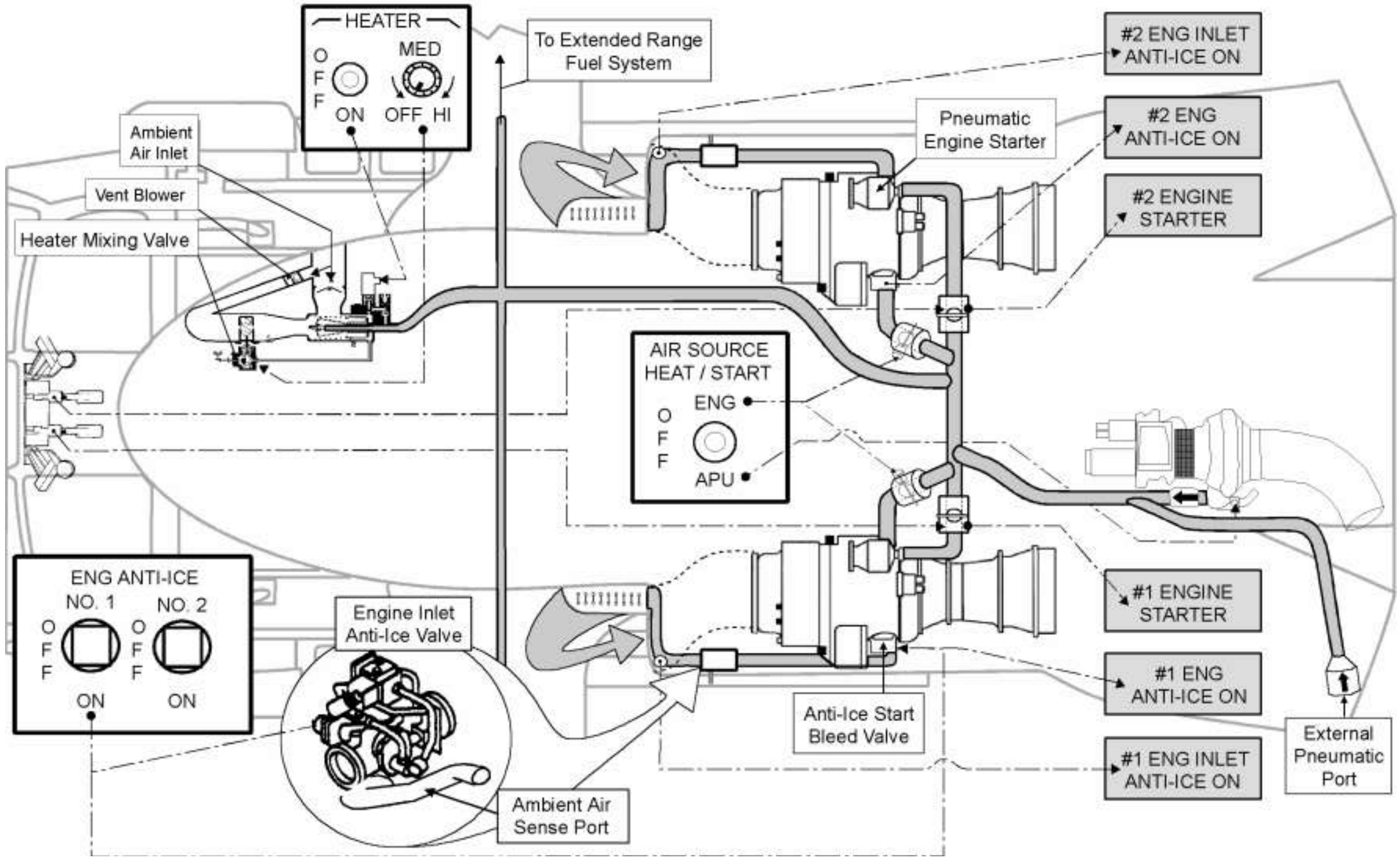
A 3. Decrease **A**ngle of incidence (program up) as airspeed increases to enhance static and longitudinal stability. (Airspeed/air data transducer)

L 4. **L**ateral Sideslip to Pitch Coupling to reduce susceptibility to gusts. Also compensates for downwash on the stabilator and tail rotor efficiency. In forward flight, the downwash on the retreating side is weaker than the downwash on the advancing side. If a right sideslip is entered (left pedal applied), the stabilator encounters increased downwash and the nose tends to pitch up, therefore the stabilator programs down to prevent the nose up attitude. A right sideslip condition also results in increased induced flow through the tail rotor and a corresponding decrease in the amount of lift provided by the tail rotor. In a left sideslip (right pedal applied), the stabilator is positioned in a reduced downwash condition and the nose tends to pitch down. The stabilator programs trailing edge up to prevent the nose from pitching down. A left sideslip condition also results in decreased induced flow through the tail rotor and a corresponding increase in the amount of lift provided by the tail rotor. (Lateral Accelerometers)

P 5. Provide **P**itch rate feedback to improve longitudinal stability and to reduce susceptibility to wind gusts. Pitch rate is sensed by pitch rate gyros in each stabilator amplifier and corrections are made to help maintain level pitch attitudes during turbulent conditions. The stabilator also programs as “G” loading increases in turns. “G” forces acting upon the fuselage tend to pull the nose down in a turn, and the stabilator programs trailing up to prevent the nose from dropping. (Pitch rate gyros) The stabilator system incorporates a test circuit to verify proper operation of the fault monitoring of the stabilator system. When the stabilator test button, located on the AFCS panel is depressed, the #1 stabilator actuator begins to move. Once the allowable miss-compare signal is reached the automatic mode of operation is deactivated. (The test circuit is disabled above 60 KIAS) Maximum miss-compare range is 10º at airspeeds up to 30 KIAS and tapers to 4º at 150 KIAS. Each stabilator amplifier processes its own airspeed, collective position, lateral acceleration, and pitch rate information. At airspeeds below 80 KIAS, the larger of the two airspeed signals are used. At airspeeds above 80 KIAS, each stabilator amplifier uses its own airspeed signal.

COMPONENT	FUNCTION	BRAINS	MUSCLE	CONTROL AUTHORITY	AXIS	REMARKS
Stabilator	Helps maintain level flight attitudes. Enhances handling qualities. Improves C.G.	2 Stabilator Amplifiers	2 Stabilator Actuators	N/A	N/A	Programmed for 5 basic functions
SAS 1	Provides short term P/R/Y correction and dampening. (Dynamic stability)	SAS Amplifier	P/R/ Y SAS Actuators	5% → 10%	Pitch Roll Yaw	SAS feedback is eliminated by the pitch boost servo. The Pitch Boost is on when either SAS is on. SAS will not move the flight controls.
SAS 2	Provides short term P/R/ Y correction and dampening. (Dynamic stability)	SAS 2/ FPS computer (digital)	P/R/ Y SAS Actuators	5%		
Trim	Force gradient	Monitored by the SAS / FPS Computer	3 Trim Actuators	100% with FPS	Pitch Roll Yaw	Pitch Trim is Hydro-Electromechanical. Roll / Yaw servos are Electromechanical.
FPS	Provides long term P/R/ Y correction and dampening. (Static stability)	SAS / FPS Computer	3 Trim Actuators	100% with Trim	Pitch Roll Yaw	For FPS to work you must have SAS 1 or SAS 2, Trim, and Boost on.
	BELOW 60 KIAS> ABOVE					
	ATT HOLD - P - ATT & A/S HOLD ATT HOLD - R - ATT HOLD HEAD HOLD - Y - HEADING HOLD OR TURN COORD					

UH-60 PNEUMATIC SYSTEM



PNEUMATIC SYSTEM

The pneumatic system of the UH-60 operates from bleed air furnished by the main engines, APU, or an external air source. The pneumatic system consists of the engine start system, the anti-ice system, the heating system, and the extended range fuel system.

Bleed air supplied by a main engine, the APU, or an external air source can be used for starting purposes. The normal means of starting is through use of the APU as an air source. The APU can provide sufficient air pressure and volume to accomplish single engine starts throughout a wide range of ambient conditions and to accomplish a dual engine start when conditions permit. Chapter 5 of the operator's manual lists single and dual engine start envelopes for given ambient conditions.

Engine starts may also be accomplished by using an operating engine as an air source, and are known as cross bleed starts. Cross bleed starts can be accomplished providing the Anti-ice light on the source engine is off, NG for the source engine is above 90%, and RPMR is 100%. If an engine is to be used as an air source, the Air Source Heat Start switch must be placed in the engine position.

An external inlet is provided to allow an external air source to pressurize the bleed air manifold. If a JASU (jet aircraft starting unit) is used, it must meet the requirements listed in Chapter 5 of the operator's manual. The external pneumatic inlet contains a check valve to prevent air leakage from the manifold when an external source is not connected.

The heater system is pressurized by bleed air from any of the sources listed above. The heater system mixes ambient air with heated bleed air to provide warm air to the crew stations and to help prevent the overhead windows, and gunner's windows from fogging. The heater will automatically disengage when the starter is engaged. The external range fuel system utilizes bleed air from the sources listed above to pressurize the external tanks for fuel transfer.

The engine anti-ice systems consist of the engine anti-ice and engine inlet anti-ice systems. Both anti-ice systems utilize bleed air from their respective engine only. The APU and external sources are not used for the engine anti-ice systems. The engine anti-ice system utilizes bleed air extracted from the compressor of the engine. This bleed air prevents ice formation on the vanes in the inlet. Anti-ice functions are controlled by the pilot through a switch labeled **#1 Eng Anti-ice** or **#2 Eng Anti-ice**. Placing the switch "on" results in the anti-ice start bleed valve remaining open to direct heated air to inlet vanes. An advisory light labeled **#1 Eng Anti-ice On** or **#2 Eng Anti-ice On** will illuminate when the anti-ice start bleed valve opens. These lights will also illuminate during low power conditions (88-92% NG), because the anti-ice start bleed valves open to dump excess compressor discharge pressure preventing the possibility of an engine flameout.

The engine inlet anti-ice is controlled by the same switch listed above, but utilizes air from a separate bleed air tap. Bleed air is routed directly to the engine inlet anti-ice modulating valve. When the engine anti-ice switch is placed on, the anti-ice modulating valve samples ambient temperature through the insulated ambient sense port. If the ambient temperature is above 13º C (Celsius), the modulating valves may not open. If the ambient temperature is between 4º and 13º C, the modulating valves may open. If the ambient temperature is below 4º C, the modulating valves must open. When the modulating valve opens, heated bleed air is forced through the airframe engine inlet and out the slits on the inlet. When the inlet temperature reaches 93º C, an advisory light labeled **#1 Eng Inlet Anti-ice On** or **#2 Eng Inlet Anti-ice On** will illuminate.

FIRE DETECTION AND EXTINGUISHING SYSTEM

The fire detection and extinguishing system of the UH-60 is comprised of IR (Infra-Red radiation) sensors, fire bottles, a control module, and fire warning arming levers (T-handles). There are five IR sensing detectors installed on the UH-60. Each engine compartment has one firewall-mounted sensor and one deck mounted sensor, the APU compartment has one firewall-mounted sensor. When IR radiation is sensed, a signal is sent from the fire extinguisher logic module to illuminate the FIRE light on the master warning panel, and light the appropriate T-handle. The warning light and the T-handle will go off when the IR source is no longer present.

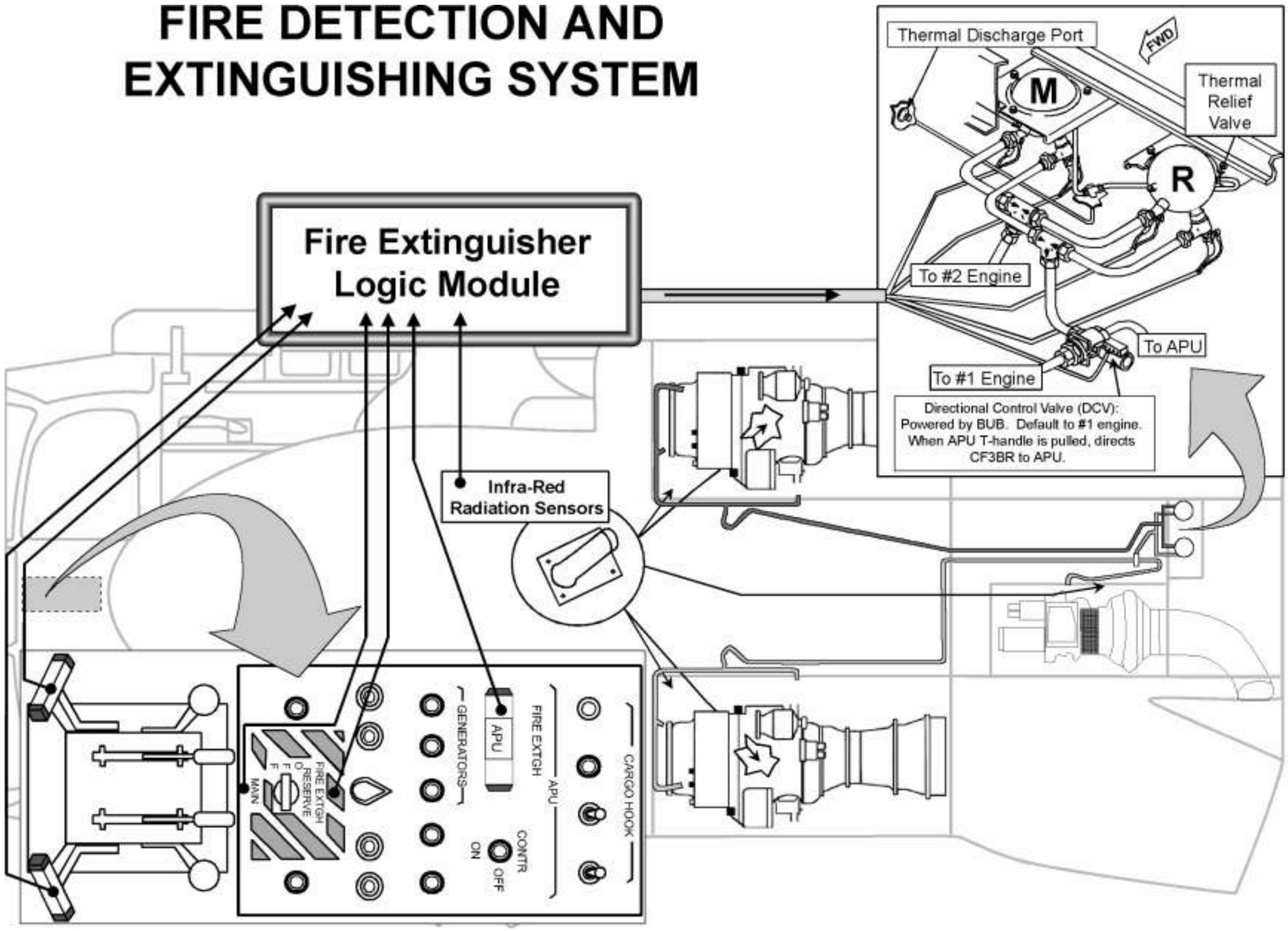
Each engine compartment has an associated emergency off T-handle, which arms the logic module. When either the #1 or #2 Emergency Off T-handles are pulled, the fuel selector valve is moved to the off position by that handle. When the APU emergency off T-handle is pulled, fuel is shutoff electrically. When the **Fire Extinguisher Switch** is placed to **Main** or **Reserve** and a **T-handle** has been pulled**,** fire-extinguishing agent is discharged into the compartment selected by the T-handle. In the event of two T-handles being pulled, extinguishing agent will be discharged into the compartment which of the last lever pulled.

Without AC (alternating current) available, fire-fighting capability for the #2 engine compartment does not exist. AC power converted to DC (direct current) must be available to arm the logic module with the #2 emergency off T-handle. The #2 emergency off T-handle uses DC primary power from the #2 DC primary bus.

Provisions for fire extinguishing during crash sequence are provided via a 10 ”g” impact switch mounted in the left-hand relay panel. Activation of this omni-directional impact switch results in fire extinguishing agent being discharged into both the #1 and #2 engine compartments. The Battery Utility Bus provides power for this impact switch.

Each fire bottle container is filled with liquid and charged with gaseous nitrogen. The containers are mounted above the upper deck, behind the right engine compartment. Each container has a pressure gage, easily viewable for preflight inspection. The system also has a thermal discharge safety port that will cause a visual indicator to rupture, indicating one or both containers are empty. Discharging the extinguishing agent from the cockpit has no effect on this visual indicator.

FIRE DETECTION AND EXTINGUISHING SYSTEM



COMMAND INSTRUMENT SYSTEM (CIS)

The CIS is part of the Electronic Navigation Instrument Display System. The instrument display system provides displays for navigation and command signals on a vertical situation indicator (VSI) and a horizontal situation indicator (HSI) for pilot visual reference. The system has a command instrument processor (CISP), two HSI/VSI mode select panels, and one CIS mode select panel. The CIS *processes* navigational data, aircraft attitude, and airspeed information appropriate for mode selected to provide visual command indications for roll, pitch, and/or collective control, as needed, to maintain heading, altitude, airspeed and to navigate during en route and approach operations. The CISP utilizes three bars on the VSI to display *processed* information; roll command bar, pitch command bar, and the collective position indicator.

The CIS mode selector allows the pilot to select one of three modes of operation to direct navigational signals to the CISP for command signal display.

1. *HDG ON*-the roll command bar, when followed, causes the helicopter to acquire and track the heading manually selected on either pilot's HSI, this will provide 1 degree of roll command for each degree of heading error up to a roll command limit of 20 degrees. Switches off automatically within 10 degrees of selected course (VOR) and 2.5 degrees (localizer).

2. *ALT ON*-the collective position indicator, when followed, causes the helicopter to maintain altitude to within +/- 50 feet. Three parameters must be met before selecting ALT ON:

A. Vertical rates no greater than +/- 200 feet per minute.

B. Altitude between –1,000 and +10,000.

C. Airspeed between 70 to 150 KIAS.

3. *NAV ON*-causes the CISP to enter the VOR NAV, ILS NAV, DPLR NAV, or FM HOME mode as selected on the pilot's VSI/HSI mode selector. The CIS will then provide steering commands based upon the course selected on either the pilot or copilot's HSI, dependent on the mode select CRS HDG selection of PLT or CPLT. NOTE: “Command” of course and heading information can only be TAKEN, it cannot be given.

A. VOR NAV Mode: engaged by selecting the VOR/ILS switch on VSI/HSI mode selector and pressing NAV switch on CIS mode selector. The roll command bar, if followed, will cause helicopter to acquire and track course manually selected on the HSI. If the helicopter is in excess of 10 to 20 degrees (outside the capture zone) from selected radial this will cause the initial course intersection to be made in the heading mode. Once the helicopter is within 10 to 20 degrees (within the capture zone) of the selected course HDG ON will drop off and the roll command bar will provide a 45-degree intercept for the final course. When passing over the VOR, the CISP reverts to a *station passage sub-mode* for approximately 30 seconds. Cyclic roll commands during this time will be obtained from the HSI course datum signal.

B. *ILS NAV Mode*: engaged by selecting the VOR/ILS switch on VSI/HSI mode selector, tuning a localizer frequency and selecting the NAV switch on the CIS mode selector. NAV ON, ALT ON, HDG ON lights will illuminate. If the helicopter is in excess of 2.5 to 5 degrees from localizer course this will cause the initial course intersection to be made in the heading mode. Once the helicopter is within 2.5 to 5 degrees of the localizer course the HDG ON will drop off and the roll command bar will provide a 45-degree intercept for the final course. Selecting the inbound course on the HSI will give pictorial correct representation of approach. Pitch command bar, if followed, will result in maintaining an airspeed that should not deviate more than +/- 5 knots from the existing airspeed when ILS NAV mode engaged. Collective position indicator, if followed, will work as described in altitude hold mode. When the glide slope is intercepted ALT ON will drop off and the collective position indicator, if followed, will cause the helicopter to acquire and track the glide slope. This is the *APPROACH Mode*, a sub-mode of the ILS NAV Mode. *BACK COURSE Mode* is also a sub-mode of the ILS NAV Mode. It is engaged by selecting BACK COURSE ON on the VSI/HSI mode selector. The roll command bar, if followed, will allow the pilots to fly the localizer back course in the same manner as the front course ILS.

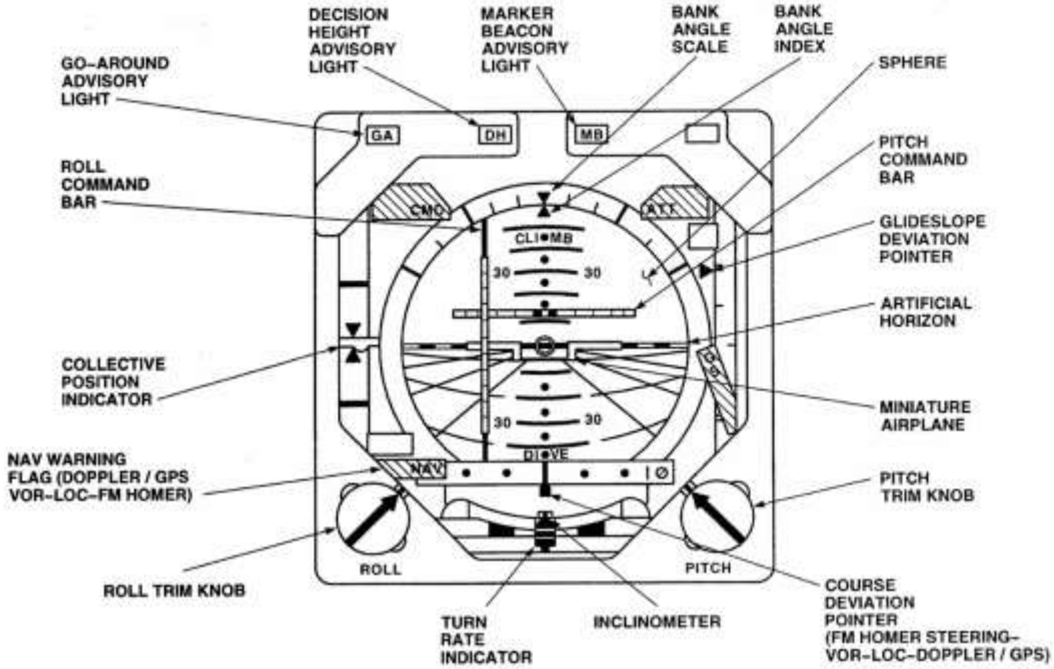
C. *DOPPLER/GPS NAV Mode*: engaged by selecting DPLR/GPS switch on the VSI/HSI mode selector and the NAV switch on the CIS mode selector. The roll command bar, if followed, will provide a straight line, wind corrected course to the selected waypoint. The course deviation bar and course deviation pointer provide a visual display of where the initial course lies in relationship to the helicopter's position. The initial course is the course the Doppler/GPS computes from the helicopter's position to the destination waypoint at the time the fly to destination thumbwheel is rotated (or entered from keyboard). To achieve pictorially correct view of the course, rotate the course knob to the head of the No. 1 needle.

D. *FM HOME Mode*: engaged by selecting the FM HOME switch on the pilot's VSI/HSI mode selector and the NAV switch on the CIS mode selector. FM homing signals are only sent to the VSI, other NAV modes will be retained on HSI if previously selected. The CISP provides cyclic roll commands to home in on a radio station selected on the No. 1 VHF-FM radio. When followed, the roll commands result in not more than two overshoot heading changes before maintaining a tracking error not to go over 3 degrees.

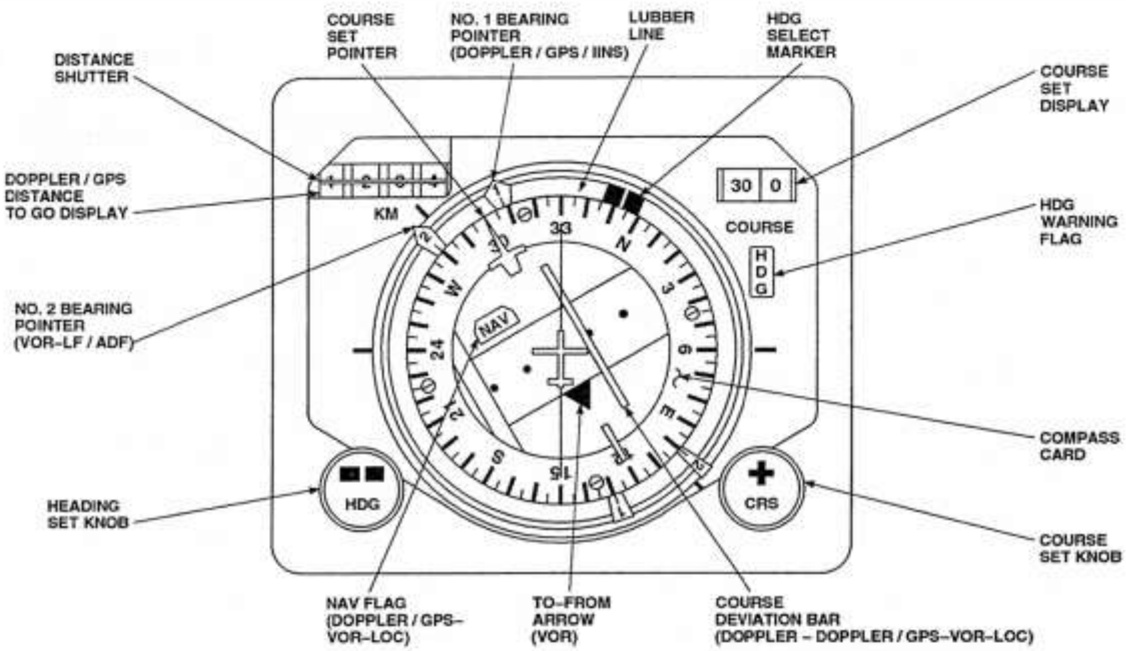
E. *GO AROUND Mode*: engaged when either pilot presses the GA (go around) button on his cyclic control grip. When engaged, the CISP immediately provides a collective position indication, which, when followed, will result in a 500 +/- 50 fpm rate of climb at zero bank angle (roll command bar). Five seconds after the GA button is pressed, the CISP will provide cyclic pitch commands, which, when followed, will result in 80 KIAS for the climb out. The GO AROUND Mode is disengaged by changing to any other mode on the CIS mode selector.

Operational "RULE OF 5":

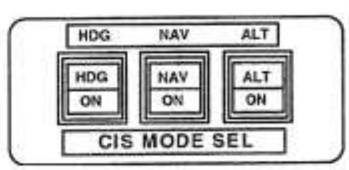
1. No. 1 Bearing Pointer. (Doppler/GPS). 2. No. 2 Bearing Pointer. (A) ADF. (B) VOR 3. Course Deviation Bar. (HSI) (A) Doppler/GPS. (B) VOR. (C) Localizer	4. Course Deviation Pointer (VSI). (A) Doppler/GPS. (B) VOR. (C) Localizer. (D) FM home. 5. Roll Command Bar (VSI). (A) Doppler/GPS. (B) VOR. (C) Localizer. (D) FM home. (E) Heading.



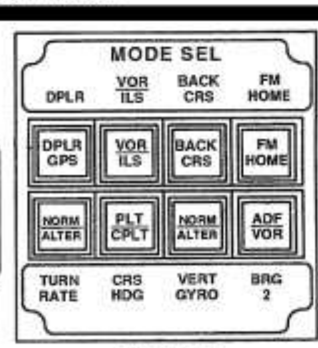
VERTICAL SITUATION INDICATOR



Horizantal Situation Indicator



CIS MODE SELECTOR



HSI / VSI MODE SELECTOR

WEIGHT & BALANCE CLEARANCE FORM F

DD Form 365-4
(TM 55-1500-342-23)
17 September 1996
Current to Change 8

1. The information presented here is an extract from TM 55-1500-342-23 dated 29 August 1986. It is provided as a reference to the aviator since the DD Form 365-4 is not presented in the Operators Manual.

a. This form, referred to as the Form F, is used to derive the gross weight and C. G. of an aircraft. The Form F furnishes a record of the aircraft weight and balance status at each step of the loading process. It serves as a worksheet on which to record weight and balance calculations and any corrections that must be made to ensure that the aircraft will be within weight and C.G. limits. Sufficient completed FORMS F must be onboard the aircraft to verify that the weight and C.G. will remain within allowable limits for the entire flight. Sufficient forms can be one (for that specific flight) or it can be several. Several FORMS F for various loading of crew, passengers, stores, cargo, fuel, external loads, etc., which will result in extreme forward and extreme aft C.G. locations and variations in gross weight, but will remain within allowable limits, may be used to verify that a particular loading which is clearly between these extremes would remain within limits.

b. The basic weight and moment obtained from the CHART C serve as the basis for the calculations on the FORM F. AR 95-1 provides for some minor exceptions to this rule. Small changes in Basic Weight and Moment due to removal or installation of aircraft equipment or other actions may be allowed to accumulate on the CHART C without changing the FORM F for 90 days or less. Also, the FORM F can be utilized to record certain items of aircraft equipment, which is part of Aircraft Basic Weight when it is temporarily added to, removed from, or relocated within the aircraft because of maintenance or specific mission requirements, etc. Procedures for this situation are described in the CHART C discussion.

c. There are two versions of this form, transport and tactical. Instructions for completing the TRANSPORT SIDE of the form are as follows:

(1) Insert necessary identifying information at the top of the form.

(2) REFERENCE 1. Enter aircraft basic weight and moment/constant (or index). Obtain this information from the last entry on Chart C (Form 365-3).

NOTE

If a load adjuster is used in loading the aircraft enter opposite Reference 1 the index figure Obtained from Chart C and use index figures throughout the form. Enter plate number of load adjuster (located on the left end of base) on the Form F. If Chart E or -10 operators manual are used instead of a load adjusted, enter moment/constant values throughout the form.

(3) REFERENCE 2. Leave blank (oil is included in the basic weight).

(4) REFERENCE 3. Enter number, weight and moment of flight crew (pilot, co-pilot, and observer).

(5) REFERENCE 4. Enter weight and moment of crew's baggage.

(6) REFERENCE 5. Enter weight and moment of steward's equipment, if applicable.

(7) REFERENCE 6. Enter weight and moment of emergency equipment not included in the basic weight.

(8) REFERENCE 7 and 8. Enter weight and moment of any extra equipment included in the basic weight.

(9) REFERENCE 9. Enter the sum weights and moments for REFERENCE 1 through REFERENCE 8 inclusive, to obtain OPERATING WEIGHT.

(10) REFERENCE 10. Enter the number of gallons, weight and moment of the fuel on board at takeoff List under REMARKS the fuel tanks involved and the amount of fuel in each tank (as required).

(11) REFERENCE 11 Enter the number of gallons, weight and moment of water injection fluid, if applicable.

(12) REFERENCE 12. Enter Sum of weights and moments for REFERENCE 9 through REFERENCE 11, inclusive, to obtain TOTAL AIRCRAFT WEIGHT.

(13) LIMITATIONS (Lower Left Corner) The maximum ALLOWABLE LOAD based on takeoff landing and limiting fuel restrictions determined by the -10 operators manual of Chart E (aircraft diagram) loading data. (In the case of most helicopters, the takeoff and landing gross weight limitations are the same, and there are no "ZERO FUEL" restrictions.) These values are computed in the LIMITATIONS table on the lower left hand corner of the Form F as follows:

(a) Enter the ALLOWABLE GROSS WEIGHT for TAKEOFF and LANDING. If the aircraft can have a gross weight restriction above which all weight must be fuel in the wings (ZERO WING FUEL GROSS WEIGHT), enter the ALLOWABLE GROSS WEIGHT for LIMITING WING FUEL in the last column of the LIMITATIONS table.

(b) If the aircraft's ALLOWABLE GROSS WEIGHT can be limited by a taxing and/or ground handling gross weight, use the REMARKS section for subtracting the warm up and/or taxi fuel from the maximum permissible ground handling gross weight. The resulting value will be entered in the ALLOWABLE GROSS WEIGHT FOR TAKEOFF block if the LIMITATIONS table and a statement similar to the following will be noted in the REMARKS section: ALLOWABLE GROSS WEIGHT FOR TAKEOFF LIMITED BY MAXIMUM TAXI GROSS WEIGHT.

(c) Determine the ALLOWABLE LOAD for TAKEOFF by subtracting the TOTAL AIRCRAFT WEIGHT (REFERENCE 12) from the TAKEOFF ALLOWABLE GROSS WEIGHT. (For most helicopters, this is the only ALLOWABLE LOAD calculation required.) Determine the ALLOWABLE LOAD for LANDING by subtracting the OPERATING

WEIGHT (REFERENCE 9) PLUS ESTIMATED LANDING FUEL WEIGHT (REFERENCE 23) from the LANDING ALLOWABLE GROSS WEIGHT. Determine the LIMITING WING FUEL ALLOWABLE LOAD by subtracting the OPERATING WEIGHT (REFERENCE 9) from the LIMITING WING FUEL ALLOWABLE GROSS WEIGHT.

(14) REFERENCE 13. Using same compartment letter designation as shown on CHART E (Aircraft diagram) or on load adjuster, enter the number, weight, compartment, and total weight and total moment of passengers. Then enter weight, compartment, total weight, and total moment of cargo.

(15) REFERENCE 14 and 15. Not applicable unless specifically required by command policy.

(16) The area to the right of REFERENCE 13 is provided for aircraft requiring Zero Fuel Weight, Zero Fuel Moment, and Zero Fuel C.G. computations. For most helicopters, these blocks are not used. The required values are determined as follows:

(a) Add the weights and moments of OPERATING WEIGHT, (REFERENCE 9) and DISTRIBUTION OF ALLOWABLE LOAD (PAYLOAD), (REFERENCE 13). Enter the calculated total weight in the ZERO FUEL WEIGHT block. Enter the corresponding moment in the ZERO FUEL WEIGHT MOMENT block.

(b) Compute Zero Fuel C.G. for that weight and enter in the ZERO FUEL % MAC block (Cross out % MAC and enter value in inches (IN).

(c) Enter on the LIMITATIONS table in the ALLOWABLE GROSS WEIGHT (FUEL) block any Zero Fuel or Limiting Wing Fuel limitations set forth in the -10 operators manual or Chart E loading data. This figure must be compared with the calculated value in the ZERO FUEL WEIGHT block. If the calculated weight exceeds the limits, adjust the load accordingly.

(d) The Zero Fuel C.G. cannot exceed the forward and aft C.G. limits at the Zero Fuel Weight. These may be found in the -10 operators manual or Chart E loading data. If it is within limits, enter the PERMISSIBLE C.G. ZERO FUEL WEIGHT forward and aft limits at the Zero Fuel Weight in the LIMITATIONS table. If it is not, adjust the load accordingly, and repeat the process.

(e) Enter the Zero Fuel weight and moment in REFERENCE 21.

(17) REFERENCE 16. Enter sum of REFERENCE 12 and the compartment totals under REFERENCE 13 opposite TAKEOFF CONDITION (Uncorrected).

(18) REFERENCE 17. Enter the TAKEOFF C.G. (Uncorrected) as determined from weight and moment values of REFERENCE 16.

(19) The weight value from REFERENCE 16 must be compared with the allowable GROSS WEIGHT TAKEOFF as shown in the LIMITATIONS table to ensure it is within limits. Use the REFERENCE 16 TAKEOFF CONDITION (Uncorrected) gross weight to determine the PERMISSIBLE C.G. TAKEOFF forward and aft C.G. limits from the -10 operators manual or Chart E loading data. If the takeoff C.G. of REFERENCE 17 is within these PERMISSIBLE C.G. TAKEOFF limits, and no other corrections are necessary, (i.e.

temporary equipment changes), enter the permissible limits in the uncorrected weight and C.G. values from REFERENCE 16 and REFERENCE 17 into the blocks at REFERENCE 19 and REFERENCE 20 respectively.

NOTE

The C.G. charts and tables in the Chart E and the -10 operator's manual are not accurate enough to use near the forward and aft C.G. limits. In those instances when the actual C.G. is very close to the aircraft limits, the C.G. MUST be arithmetically calculated to ensure the necessary accuracy.

(20) REFERENCE 18. When the takeoff weight of REFERENCE 16 and/or the takeoff C.G. of REFERENCE 17 are not within permissible takeoff weight and/or C.G. limits, changes in the amount or distribution of load (REFERENCE 13) are required. The necessary load adjustments must be noted in the CORRECTIONS columns on the left hand portion of FORM F. Enter a brief description of the necessary load adjustment in the left hand column with the weight and moment listed in the columns provided. Sum all the weight and moment increases and/or decreases to obtain the net change (+ or -) in the amount or distribution of the load. Transfer the total weight and moment adjustment to the spaces provided for CORRECTIONS (if required) at REFERENCE 18.

NOTE

If there are any temporary equipment changes listed on DA Form 2408-13-1/2408-13-1-E or DA Form 2408-14-1, they should be considered changes in aircraft loading. These changes should be entered with the notation "EQUIPMENT CHANGES" near the top of the CORRECTIONS table. A brief description, weights and moments should be entered in the columns below this notation. These entries should be treated as a variation in loading and applied to the total entered in REFERENCE 18.

(21) REFERENCE 19. In the space provided for TAKEOFF CONDITION (Corrected), enter the sum of REFERENCE 16 and REFERENCE 18. (Add If REFERENCE 18 is positive. If it is negative, subtract REFERENCE 18 from REFERENCE 16.)

(22) REFERENCE 20. Enter the TAKEOFF C.G. (Corrected), as determined from the weight and moment values of REFERENCE 19.

(23) The weight value from REFERENCE 19 must again be compared with the allowable GROSS WEIGHT TAKEOFF as shown in the LIMITATIONS table to ensure compatibility. At the REFERENCE 19 TAKEOFF CONDITION (Corrected) gross weight, again determine the PERMISSIBLE C.G. TAKEOFF forward and aft C.G. limits from the -10 operators manual or Chart E loading data. Recheck the takeoff C.G. REFERENCE 20 to ensure it is within the PERMISSIBLE C.G. TAKEOFF limits. Enter these limits in the space provided in the LIMITATIONS table.

(24) REFERENCE 21. Enter Zero Fuel Weight and moment. This is normally calculated by subtracting TAKEOFF FUEL (REFERENCE 1)) from corrected TAKEOFF

FUEL (REFERENCE 19). If “Zero Fuel” weight limitations apply, this figure will match the values appearing to the right of REFERENCE 13.

(25) REFERENCE 22. Enter weight and moment of any aerial supply load(s) to be dropped before landing.

NOTE

If the aircraft has no Zero Fuel Weight limitation, but it appears that the C.G. at the Zero Fuel weight may exceed the aircraft’s forward or aft C.G. limits, a further check must be made. The procedures are described in paragraph 16 above. This procedure must be applied to any analogous situation not already taken into consideration. Examples might include the unanticipated jettisoning of external stores, relocation of passengers, etc. Enter the results of the Zero Fuel (or similar) C. G. calculations in the REMARKS section. It should include a notation such as “Center-of-Gravity at the Zero Fuel Weight (or with the auxiliary fuel tanks released or whatever) has to be checked and the C. G. is (is not) within limits.” Amplify the remarks if the C. G. is not within limits.

(26) REFERENCE 23. Determine the ESTIMATED LANDING FUEL weight and moment and enter it in the space provided.

(27) REFERENCE 24. Determine the ESTIMATED LANDING FUEL CONDITION by subtracting the weights and moments of REFERENCE 22 from REFERENCE 21 and adding REFERENCE 23.

(28) REFERENCE 25. Enter the ESTIMATED LANDING C.G. as determined from the weight and simplified moment values of REFERENCE 24.

(29) The weight value from REFERENCE 24 must be compared with the allowable GROSS WEIGHT LANDING as shown in the LIMITATION table to ensure compatibility. Use the REFERENCE 24 ESTIMATED LANDING CONDITION gross weight to determine the PERMISSIBLE C.G. LANDING forward and aft C.G. from the -10 operators manual or Chart E loading data. If the ESTIMATED LANDING C.G. of REFERENCE 25 is within these PERMISSIBLE C.G. LANDING limits, enter them in the spaces provided in the LIMITATIONS table.

(30) When the ESTIMATED LANDING CONDITION of REFERENCE 24 and/or the ESTIMATED LANDING C.G. of REFERENCE 25 are not within permissible landing weight and/or C.G limits, changes in the amount or distribution of load and/or fuel are required. A new Form F will be completed.

(31) Enter signature of person computing the form in COMPUTED BY SIGNATURE.

NOTE

If local requirements exist for the use of the WEIGHT AND BALANCE AUTHORITY SIGNATURE block, the commander will establish policies and procedures.

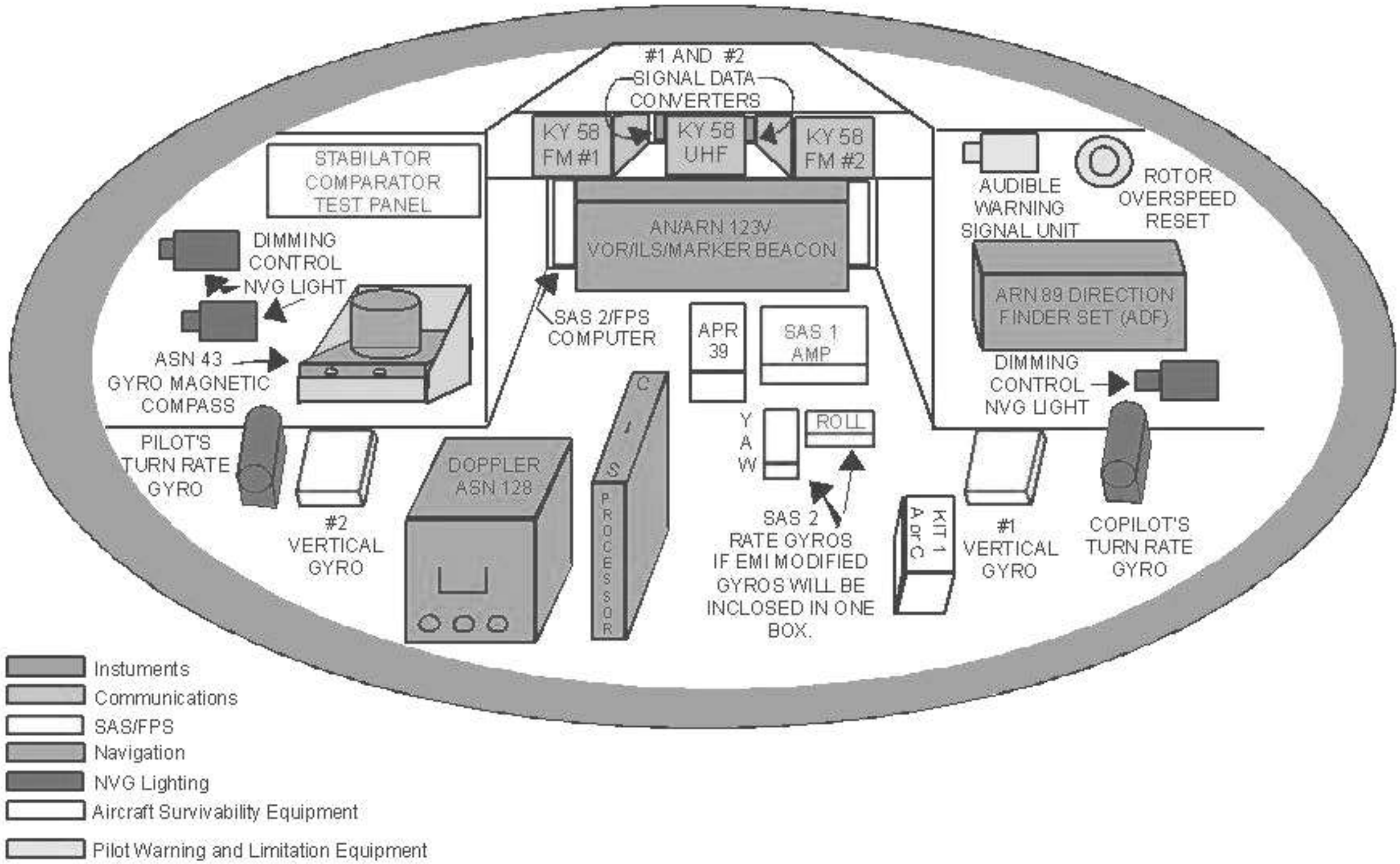
Responsibilities of the PC regarding weight and balance: Comply with the ATM task and AR 95-1.

The PC will ensure the accuracy of computations on the DD Form 365-4 and that the form is completed and aboard the aircraft to verify that the weight and center-of-gravity will remain within allowable limits for the entire flight. Several DD 365-4 forms completed for other loadings also may be used to satisfy this requirement. In this case, the actual loading being verified must clearly be within the extremes of the loading shown on the DD 365-4 used for verification. (AR 95-1, para. 5-2 h.)

The UH-60 is classified as a Class 2 aircraft for weight and balance purposes. These are those aircraft whose weight or center of gravity limits can readily be exceeded by loading arrangements normally used in tactical operations or those aircraft designated primarily for transporting troops and other passengers. Therefore, a high degree of loading control is needed. Also, all aircraft whose weight and balance class is not stated in the operator's manual will be considered Class 2. (AR 95-1 para. 7-3 b.)

UH-60 NOSE COMPARTMENT COMPONENTS

ver 1.0
9/97



UH-60 Aircraft Survivability Equipment (ASE)

This portion of the supplement is to provide a basic overview of the baseline UH-60 ASE. ASE may vary with specific unit missions and vary by H-60 model. Specialized systems will be trained by the units utilizing them.

Knowledge of the operation and the proper employment techniques of ASE will enhance aircrew survival. The UH-60 has both active and passive ASE equipment systems. When aircrews must stay on station despite warnings, active countermeasures help to ensure their survival. Active countermeasures jam or act as decoys to confuse the fire control or guidance systems of threat weapons. Examples of active ASE are the Common Missile Warning System and the APR-39. Examples of passive design characteristics or equipment are the Hover Infrared Suppressive System (HIRSS), whose signature reduction provides a passive countermeasure. Also, flat plate glass and low infrared reflective paint work together to lower the IR signature of the aircraft.

Another ASE related system is the Transponder AN/APX-100(V)1 or AN/APX-118(V). The transponder provides automatic radar identification of the aircraft to all suitably equipped challenging aircraft and surface or ground facilities within the operating range of the system. Mode 4 is the secure mode used for cooperative combat identification for identifying friend or foe (IFF). There is a transponder computer located in the nose of the aircraft which operates in conjunction with Mode 4. Operation of the Cryptographic Computer Kit (KIT-1C) which uses electronic key loading via Secure Key Loader (SKL) or an Electronic Transfer Device.

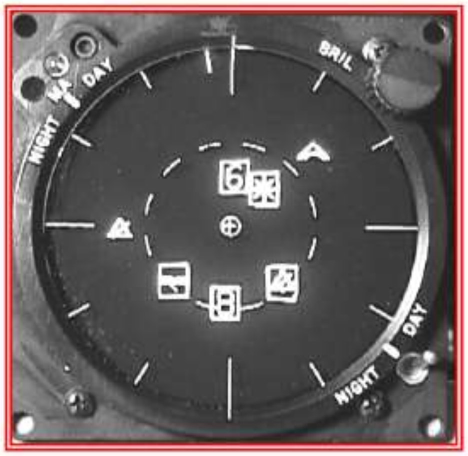
APR-39A(V)1

The purpose of the AN/APR-39A(V)1 is to provide aircrews with a warning of hostile radar-directed weapons which surround the aircraft and advise the crew of the system status. Components of the APR-39A(V)1 are:

- CONTROL UNIT (Located in the cockpit lower console)
- 1 INDICATOR (Located on the instrument panel)
- 1 DIGITAL PROCESSOR (Located in the nose avionics compartment)
- 2 RECEIVERS (One in nose area, the other in the tail)
- 5 ANTENNAS (Located on the nose, tail, and underside)

Digital Processor

CRT display



Control unit

Upgraded version of the APR-39A(V)1 utilizes a digital processor, alphanumeric display and synthetic voice warning for radar directed air defense threat systems. The symbology displays a bearing for each processed emitter signal; it does not provide any range data.

Preflight checks include:

- General condition and security of avionics compartment processor 4 spiral antennas, 1 blade antenna.
- Power up - Software version # Operational Flight Program (OFP upper #) and Emitter.
- Identification Data (EID lower #).

AN/AAR-57 Common Missile Warning System

The AN/AAR-57 CMWS provides missile warning for the UH-60. It is able to detect and declare missile identity before missile burnout. The CMWS consists of an Electronics Control Unit (ECU) developed by Sanders (now BAE Systems, following acquisition in 1999) and up to six Electro-Optical Missile Sensors (EOMS). The ECU processes threat data and provides cueing for automatic launching of countermeasure flares. (Chaff launching is manually performed by the pilot.) The EOMS employs ultra-violet detector technology, to detect radiation from the plume of the attacking missile. Further operational details and capabilities are classified and therefore not covered in this document.

Preflight checks include:

- Check wiring in nose section and verify EOMS are secure and uncovered.
- Ensure CMWS safety pin in cabin is installed and locked.
- Check ECU for condition and security and ensure UDM is installed.
- Check flare and chaff dispensers for condition and security and confirm they contain the appropriate countermeasures.
- Check aft EOMS secure and uncovered.