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SUBSYSTEM D  
ENGINEERING ANALYSIS REPORT  
FOR  
GEMINI AGENA TARGET VEHICLES

U

Contract AF 04(695)-129

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ENGINEERING ANALYSIS REPORT  
FOR  
GEMINI AGENA TARGET VEHICLES (U)

Contract AF 04(695)-129

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## FOREWORD

This report describes the guidance, control, and flight-programming system of the Gemini Agena Target Vehicle. The report was prepared by Lockheed Missiles and Space Company for the National Aeronautics and Space Administration in compliance with Contract AF 04(695)-129.

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Section 1  
INTRODUCTION

The guidance, control, and flight-programming system (Subsystem D) for the Gemini Agena Target Vehicle is comprised of an inertial reference package (IRP), a velocity meter, horizon sensors, a sequence timer, hydraulic servos, pneumatic thrust valves, and associated electronics. (See Fig. 1-1.)

This equipment performs the following functions:

- a. Orbital boost guidance and stabilization
- b. Attitude reference with respect to the local vertical and orbit plane
- c. Engine thrust initiation and termination
- d. Performance of orbital corrections to attain rendezvous capture volume
- e. Stabilization and control in docked mode with Gemini spacecraft
- f. Postdocked maneuvering.

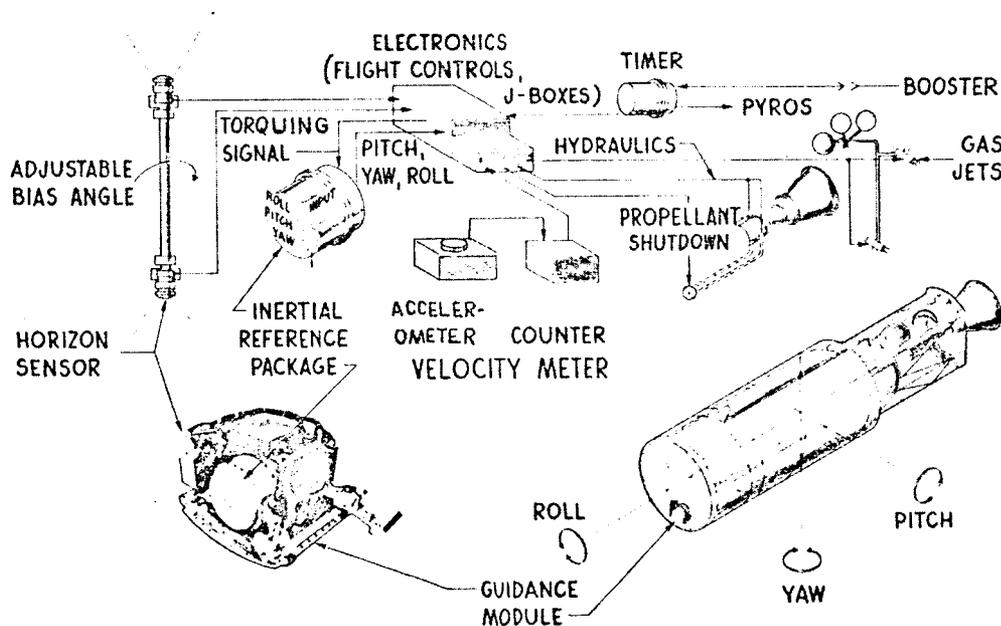


Fig. 1-1 Subsystem D for Gemini Agena Target Vehicle

## Section 2 SYSTEM OPERATION

### 2.1 GENERAL

A functional block diagram of the guidance and control system for the Agena D Gemini missions is shown in Fig. 2-1. Ascent-phase events of the mission are sequenced by a preset timer. After the Agena Target Vehicle is injected into orbit, the sequence of events is a function of the maneuver necessary to realize rendezvous with the Gemini spacecraft. This flexibility is exercised through the Agena command system.

### 2.2 AGENA ASCENT TRAJECTORY

The trajectory of the Agena into orbit can be divided into five discrete phases as follows:

- a. Atlas boost
- b. Ascent coast
- c. Ascent boost (Agena first firing)
- d. Orbital coast
- e. Orbital adjustments (Agena firings).

#### 2.2.1 Atlas Boost

The activation of the sequence timer is the first event of Agena control. (See Appendix A.) The time this function is performed is computed on the ground and initiated through the Atlas radio guidance system. Sequence timer activation is not a preset event, because it is a function of the Atlas performance. After the start of the sequence timer and the cutoff of the

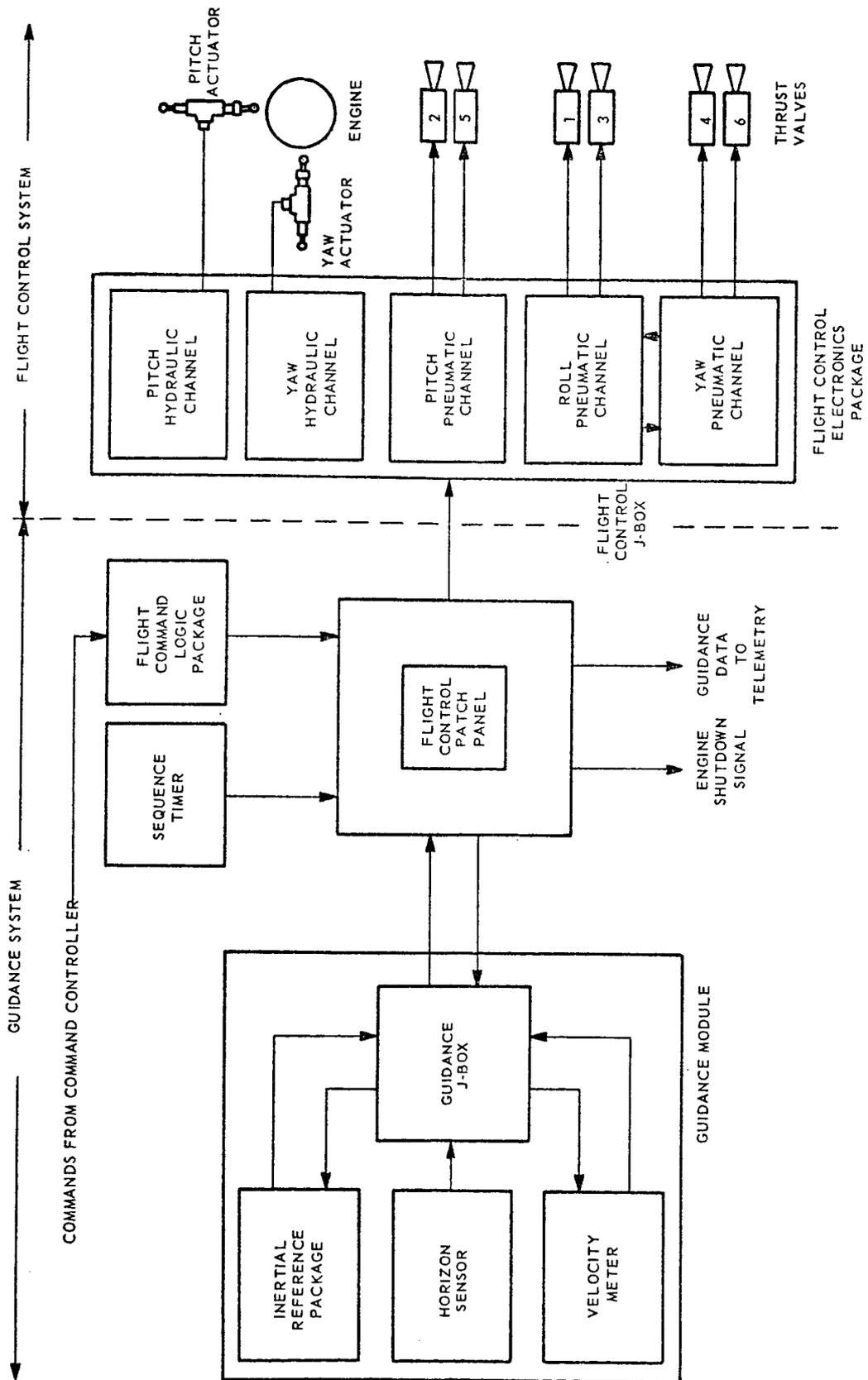


Fig. 2-1 Guidance System Block Diagram

Atlas vernier engine, the gyros are uncaged, the horizon sensor fairings and L-band thermal cover are jettisoned, the separation circuit is armed, and premature separation portion of the flight-termination system is disarmed.

### 2.2.2 Ascent Coast

During the separation maneuver sequence, the Agena pneumatic control system is fully energized, and any residual rates imparted by the Atlas booster are removed. The first function is to connect the roll horizon sensor control. This erects the roll gyro reference in the desired local vertical plane to minimize the azimuth error resulting from a pitch maneuver. The vehicle is pitched down 18 degrees at a programmed rate of -1.5 degrees per second. After completion of the pitchover maneuver to the local vertical, the pitch horizon sensor control is activated along with a nominal geocentric rate.

The Agena vehicle is thus positioned for the ascent boost phase of the trajectory. Prior to the ascent boost, the velocity meter, which is used to measure a predetermined increase in velocity, is enabled.

### 2.2.3 Ascent Boost

The first operation in the ascent boost phase is the firing of the secondary propulsion system (SPS) units to orient propellants for the primary propulsion system (PPS). The pitch and yaw pneumatics are disabled, and the main engine is then ignited and continues to fire until terminated by the velocity meter cut-off signal when the proper velocity has been gained. Throughout ascent boost, both pitch and yaw stability are maintained by controlling the engine thrust vector; roll control is maintained by the pneumatic system.

At termination of engine burn, control is recovered by the pneumatic system, and the Agena is switched to an orbital control mode. This includes activation of the gyrocompass circuit to give complete three-axis referencing. The pneumatic pressure and system gains and deadbands are set to the gas saving mode.

#### 2.2.4 Orbital Coast and Orbit Adjust

This phase of the mission is completely dependent on the necessary maneuvers to effect rendezvous with the Gemini spacecraft. Either plane changes or phase changes can be made by means of the multiple restart capability of the Agena engine. The necessary maneuvers cannot be properly determined until both vehicles are on orbit. However, provisions are made for any anticipated maneuvers.

The following two modes of orbit operation, which are functions of the system deadband and gain commanded from the ground, are possible with the Agena system:

- a. Gas Saving Mode. This mode will be utilized while the Agena is waiting in orbit.
- b. High Accuracy Mode. This mode will be utilized in any maneuver sequence.

### Section 3 ATTITUDE CONTROL ANALYSIS

#### 3.1 GENERAL ANALYSIS

The control parameters for system objectives were established through the use of well defined analytic methods. Digital computer programs were used to obtain the roots of the characteristic equation for the autopilot and air-frame dynamics. These programs also yielded the closed loop transfer function. From this information, the root locus of the control loop was plotted as a function of gain to establish the stability and response.

The analysis has been divided into two basic sections, pneumatic (coast control) and hydraulic (boost control). During any coast phase or SPS operation of this mission, attitude control is accomplished by three-axis pneumatic control. During PPS powered flight, control about the pitch and yaw axes is transferred to the hydraulic system.

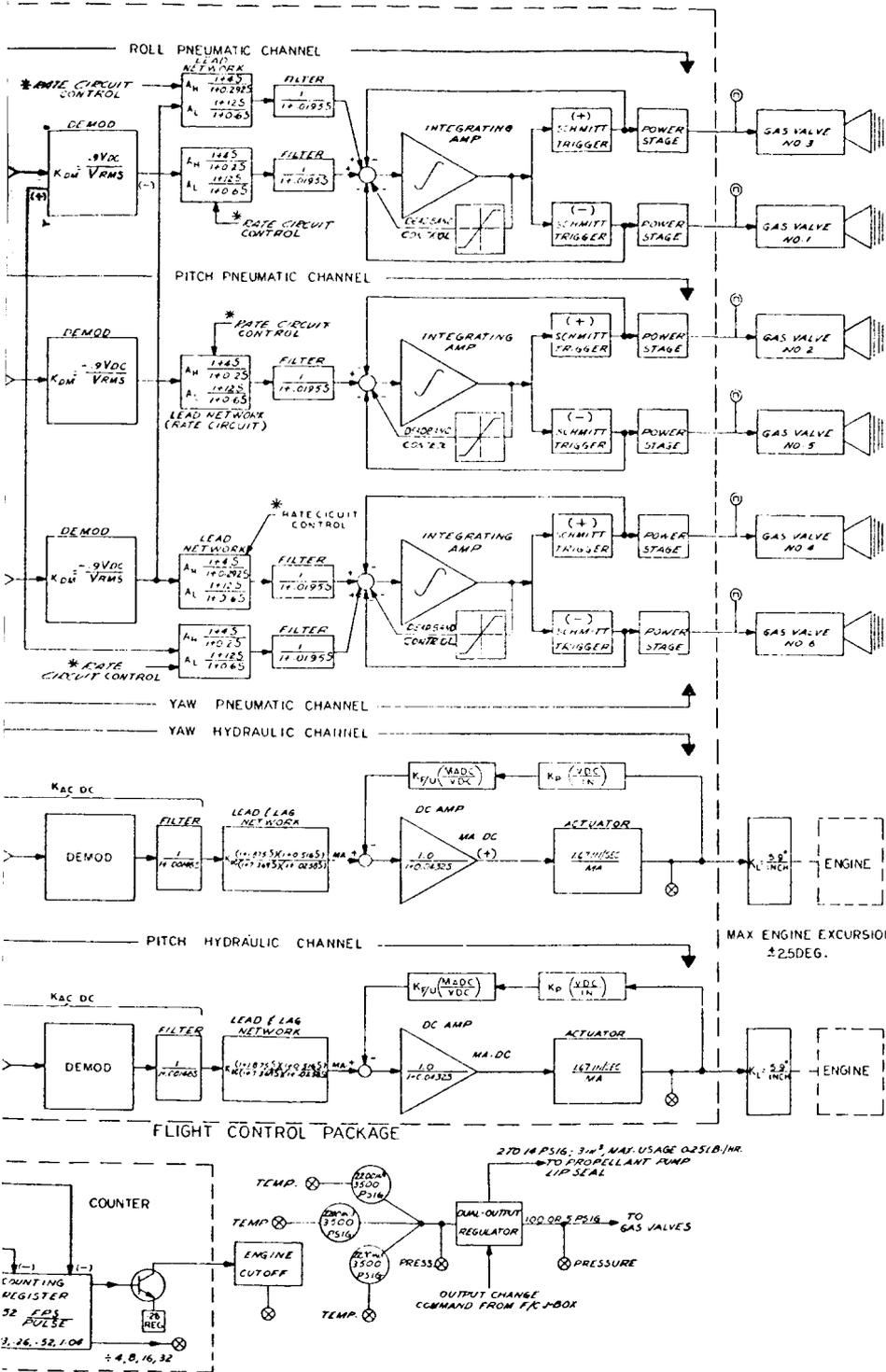
#### 3.2 COAST CONTROL SYSTEM

The coast control system for this vehicle is a fully pneumatic, pulse-width, pulse-frequency modulated system utilizing a rate integrating gyro and lead-lag network in each channel. Figure 3-1 presents the general block diagram for this control. The characteristics of the pulse valve are shown in Fig. 5-39.

##### 3.2.1 Ascent Coast

The three-axis pneumatic system is activated at booster separation. To study the stability and response of this control, the nonlinear pulse valve



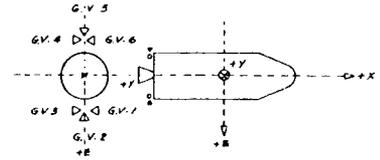


5. SUMMARY OF LOOP CHARACTERISTICS

CHANNEL	TYPE	TIME CONSTANT (SEC)	INTEGRATING	LAG (SEC)	LEAD (SEC)	PHASE (DEG)	MAGNITUDE (DB)
ROLL PNEUMATIC	INTEGRATING	0.025	+	0.0125	0.0125	0	0
PITCH PNEUMATIC	INTEGRATING	0.025	+	0.0125	0.0125	0	0
ROLL HYDRAULIC	INTEGRATING	0.025	+	0.0125	0.0125	0	0
PITCH HYDRAULIC	INTEGRATING	0.025	+	0.0125	0.0125	0	0
YAW PNEUMATIC	INTEGRATING	0.025	+	0.0125	0.0125	0	0
YAW HYDRAULIC	INTEGRATING	0.025	+	0.0125	0.0125	0	0

- MAX GAS VALVE REP. RATE 10RPS.  
MAX PULSE WIDTH 20MS.  
MAX. IMPULSE - 100 PSIG IN - 0.75015 LB.-SEC.  
MAX. IMPULSE - 5 PSIG IN - 0.0325 ± 0.0118 SEC.

3. VEHICLE AXIS AND GAS VALVE (GV) ARE LOCATED AS SHOWN BELOW



2. THE FOLLOWING ARE DEFINITIONS OF SYMBOLS USED ON THE FFD

SYMBOL	DEFINITION
KAC DC	ELECTRONICS ACTUATOR GAIN (2.4 V/PS)
KD	VOLTAGE DIVIDER SCALE FACTOR
K <sub>D</sub>	ACTUATOR FEED BACK SCALE FACTOR (1/3)
K <sub>EN</sub>	SIGN AMP FEEDBACK SCALE FACTOR (1/3)
K <sub>L</sub>	HYD LINKAGE GAIN
K <sub>FC</sub>	HYD CHAN FORMING STATIC GAIN (1/3)
K <sub>IN</sub>	AC AMP INPUT GAIN
S.F.	SCALE FACTOR
S <sub>TG</sub>	TORQUE GEN
S <sub>SG</sub>	SIGNAL GEN
A <sub>N</sub>	MINI SIGNAL NOTE (RESISTOR VALUE)
A <sub>L</sub>	LOW V VALUE
T <sub>C</sub>	IRP CASING LOOP TIME CONSTANT
*	INDICATES ANTENBOARD CONTROL
S <sub>I</sub>	IRP INPUT SENSITIVITY
T.G.	TORQUE GENERATOR
S.G.	SIGNAL GENERATOR
Δ, □	SELECTED FIXED RESISTORS
A <sub>I</sub>	PNEU INTEG AMP GAIN
⊗	TEMP PICK OFF POINT
⊙	GAS VALVE CURRENT TLM, SEE NOTFI.

1. AMA GAS VALVE THRUST WITH REGULATOR IN AMA PRESSURE MODE IS 107.5 LBS., IN LOW AMP MODE IS 15.075 LBS. VALVE CURRENT TLM IS SUPPLIED BY VALVE PAIRS (2, 5, 3, 1, 4, 6) TO PROVIDE THREE CHANNELS.

NOTES:

Fig. 3-1 Subsystem D Block Diagram

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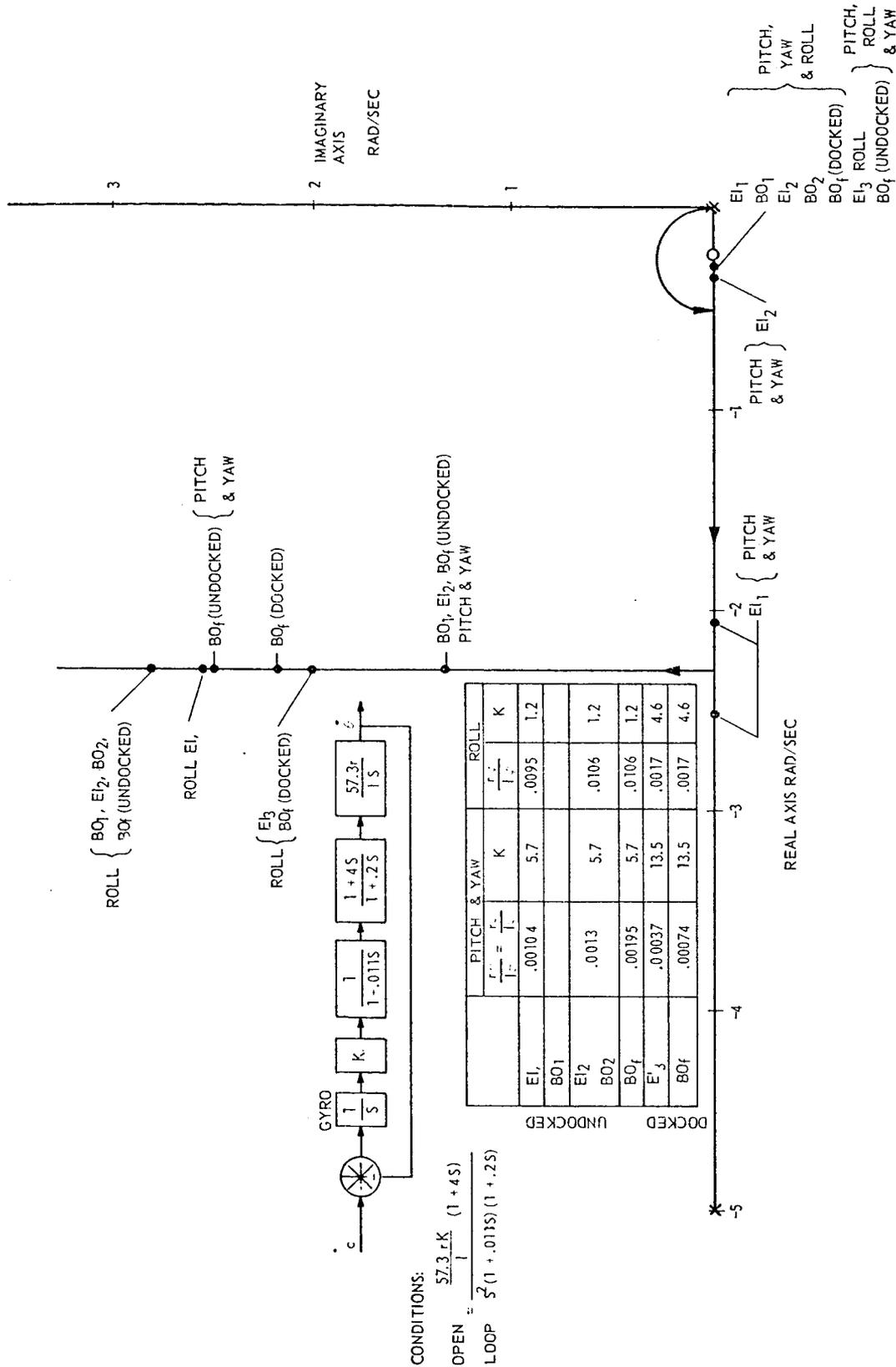


Fig. 3-2 Pneumatic (1 + 4S) Root Locus, Pitch, Yaw, and Roll

system has been reduced to an equivalent linear system. This is valid since the average force level of the valve is proportional to the output of the lead-lag circuit. Study of this control is further facilitated by the fact that each control channel can be represented by the same dynamic terms. One characteristic control channel is studied as a function of dynamic gain, and the results are shown, in root locus form, in Fig. 3-2. The block diagram for this control is shown in Fig. 3-3.

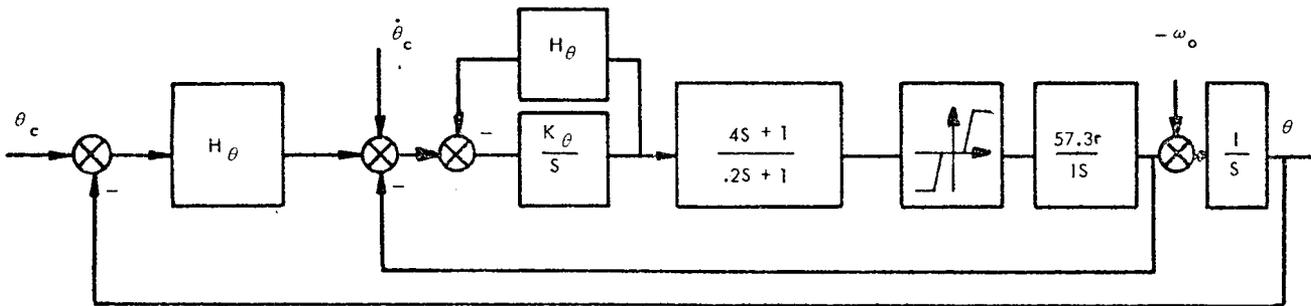


Fig. 3-3 Typical Control Channel, Block Diagram

The dominant nonlinearities associated with this control are deadband and gas valve limiting. The deadband allows this system to operate in a limit cycle resulting in a control gas saving. The root locus operating points are based on the linear region of this gas valve transfer function and will be representative of small signal operation for this system. Operation within the saturation region will result in a reduction of the loop gain. This condition would only result in a different response than that predicted by the root locus, but the response would be absolutely stable.

The root loci for the complete control channel, including horizon sensor, are shown in Figs. 3-4 and 3-5. These represent the roll and pitch channels in the ascent phase of the mission. An external yaw reference is not obtainable with this system throughout the ascent phase of this mission.

### 3.2.2 Orbital Coast System

The orbital coast system utilizes the same basic configuration as that of the ascent coast system. The principle difference in this mode is the increased deadband and high gain rate circuit. Both of these modifications contribute to a reduction of the gas consumption for this control. (See Paragraph 3.2.3.) The root locus and block diagram for this control mode are shown in Fig. 3-6. The similarity to the ascent locus is readily seen, and again absolute system stability is assured. Generation of a yaw reference by means of the gyro-compass technique is also available in the orbital coast system.

### 3.2.3 Control Gas Consumption

The predicted control gas consumption for the Gemini Agena Target Vehicle is as follows:

a. Ascent mode (single burn)	13 lb
b. Orbit mode (undocked)	
(1) Fine control (10 min)	1.8 lb
(2) Coarse control (4 days)	16.1 lb
(3) Orbit adjustments (4)	19.9 lb
c. Docking transients (5)	12.5 lb
d. Orbit mode (docked)	
(1) Coarse control (1 day)	2.7 lb
(2) Maneuver (3 deg/sec)	6.5 lb

## 3.3 BOOST CONTROL SYSTEM

During Agena engine burn, the pitch and yaw autopilot signals command the hydraulic servo actuators, which position the engine thrust vector to generate control moments. Roll control is maintained by the pneumatic system. The operation and dynamics of the thrust vector control are presented in the following subparagraphs.

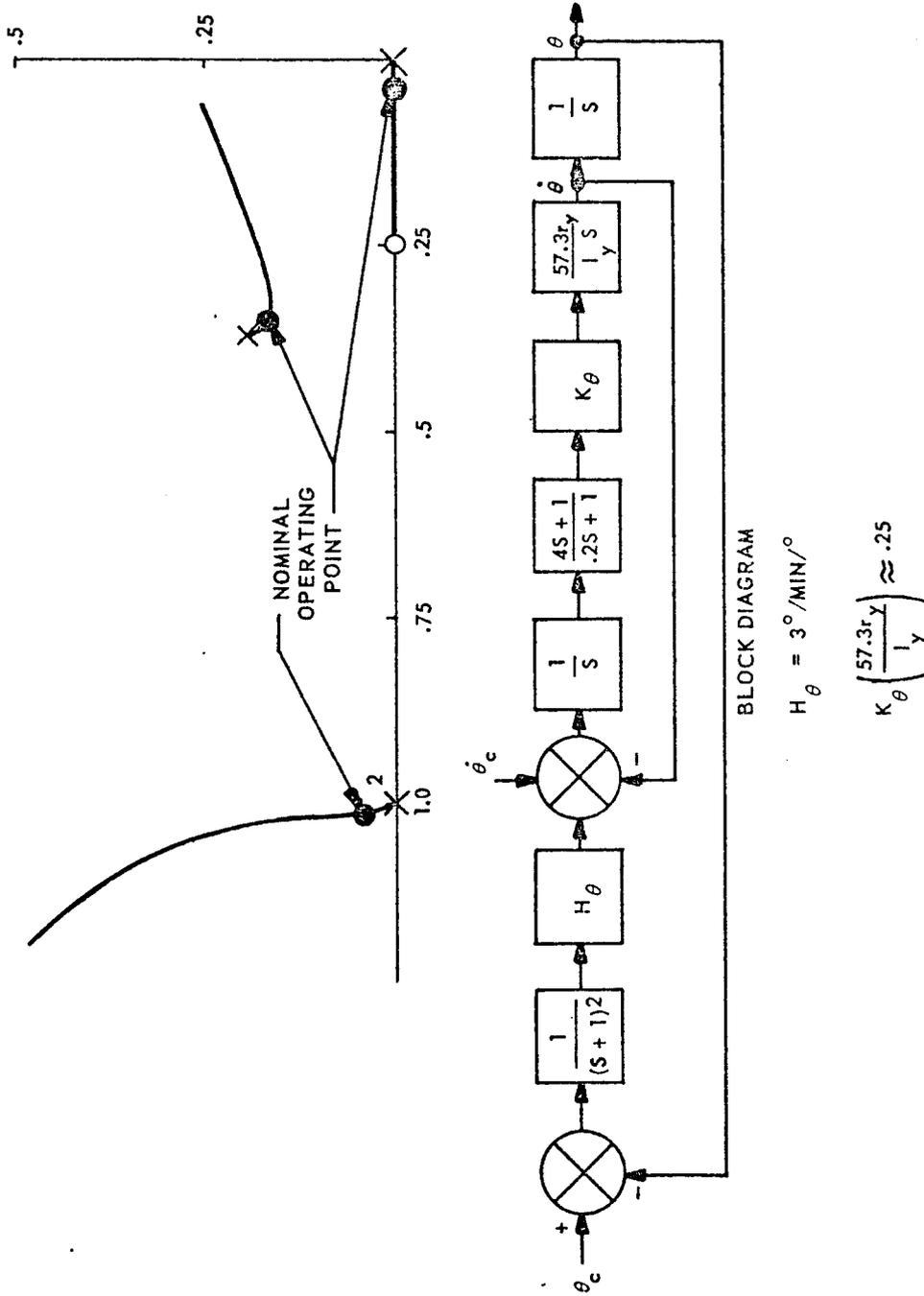


Fig. 3-4 Horizon Sensor Control - Pitch Channel Root Locus

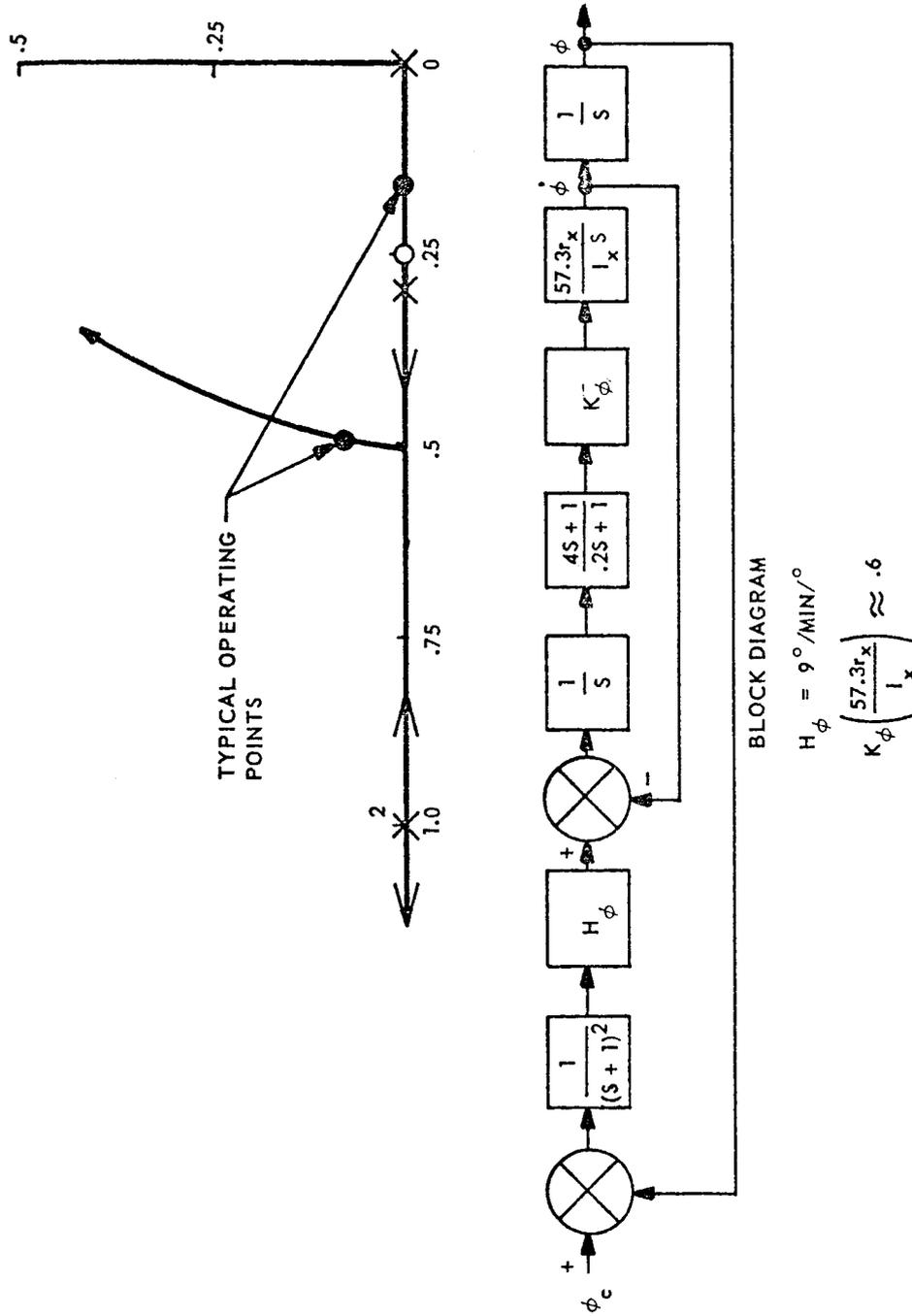


Fig. 3-5 Horizon Sensor Control - Roll Channel Root Locus

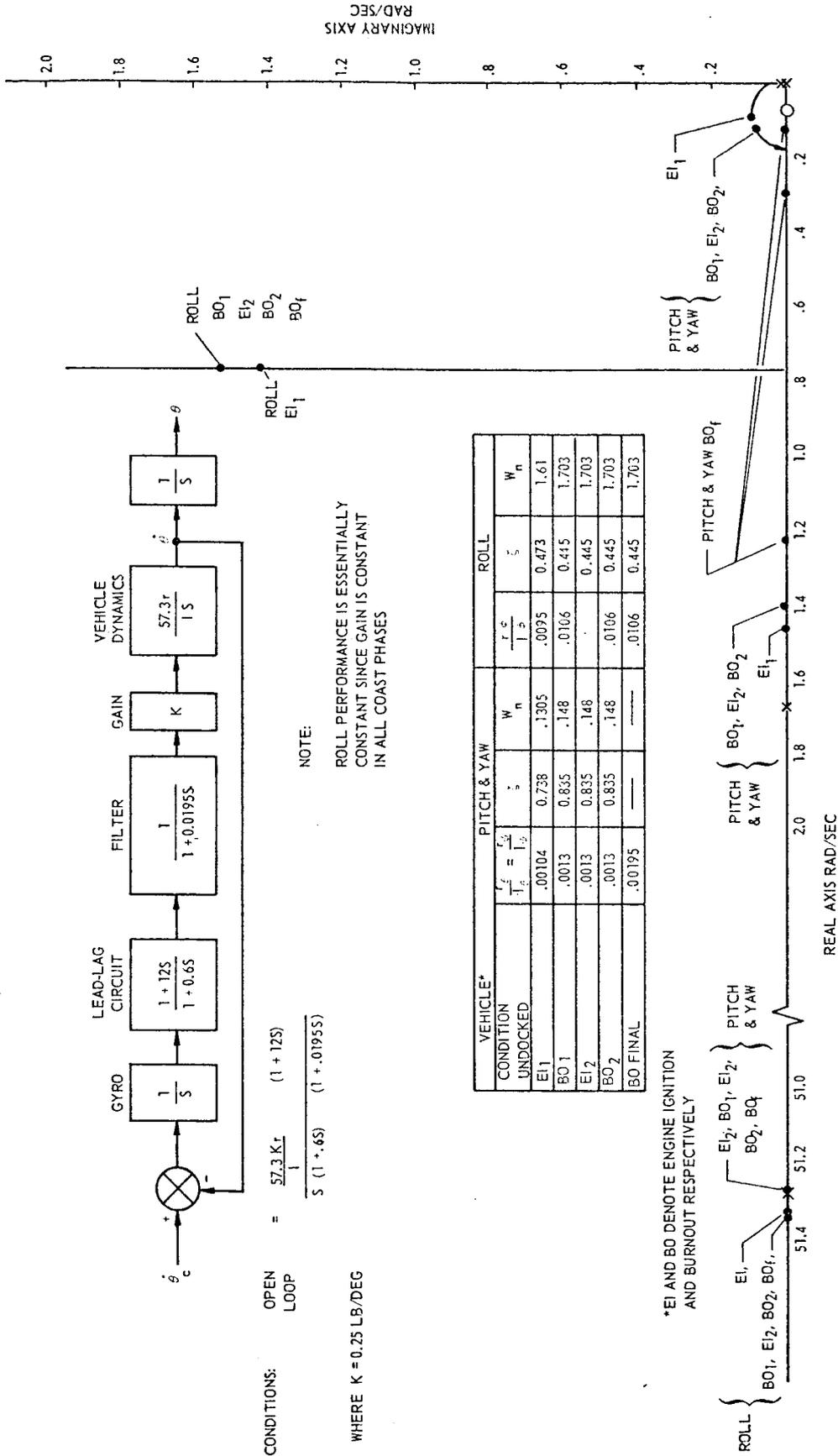


Fig. 3-6 Rate Circuit (1 + 12S) Root Locus - Low Gas Consumption

### 3.3.1 Basic System

The single axis hydraulic control system for the undocked powered flight phase of the Gemini Agena mission is shown in Fig. 3-7. The root locus for this control is given in Fig. 3-8. This is representative of both the pitch and yaw channels. The dynamic effects of the engine backup structure, swiveling engine, and fuel slosh are neglected in this phase of the mission since they introduce negligible effect on the system's dominant mode. The only dynamic effects considered are those introduced by the gyro, rate-integral network, servo actuator, and airframe treated as a pure inertia system. Because of a shift in center of gravity and moment of inertia during an Agena burn, the operating characteristics of this system vary. A satisfactory condition is maintained with one value of autopilot gain throughout the powered phase of this flight.

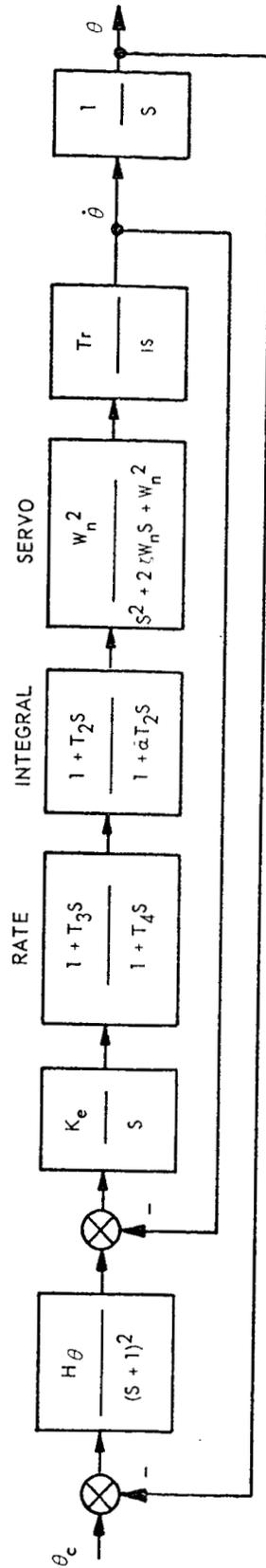
The rigid body mode shows a frequency of less than 0.45 cps while the higher frequency hydraulic servo mode has a frequency of 2.18 cps.

Tests conducted with flight equipment have shown that the Model 8247 rocket engine is sufficiently nulled before the thrust level is of any magnitude that would cause a vehicle instability.

The attitude errors resulting from any misalignments in this system are maintained at a tolerable level with the static gain ( $K_e$ ) of the system.

### 3.3.2 Engine Servo

The servo actuator block in Fig. 3-9 is itself a feedback control system that can be represented by a second order lag. An additional second order lag is introduced because of the flexibility of the engine backup structure. This mode is neglected in analyzing the Agena ascent phase since it is nearly two decades in frequency away from the dominant rigid body mode.



- $K_e = 2.1 \text{ DEG ENG / DEG VEH}$
- $T_3 = 0.516 \text{ SEC}$
- $T_4 = .0258 \text{ SEC}$
- $T_2 = 1.875 \text{ SEC}$
- $\alpha T_2 = 7.37 \text{ SEC}$
- $W_n^2 = 428 \text{ RAD}^2/\text{SEC}^2$
- $\zeta = 0.56$

- $\frac{Tr}{I} = 10.68 \text{ RAD/SEC}^2 - \text{ENGINE IGNITION}$
- $\frac{Tr}{I} = 14.5 \text{ RAD/SEC}^2 - \text{BURNOUT}$

Fig. 3-7 Agena Undocked Boost Control System

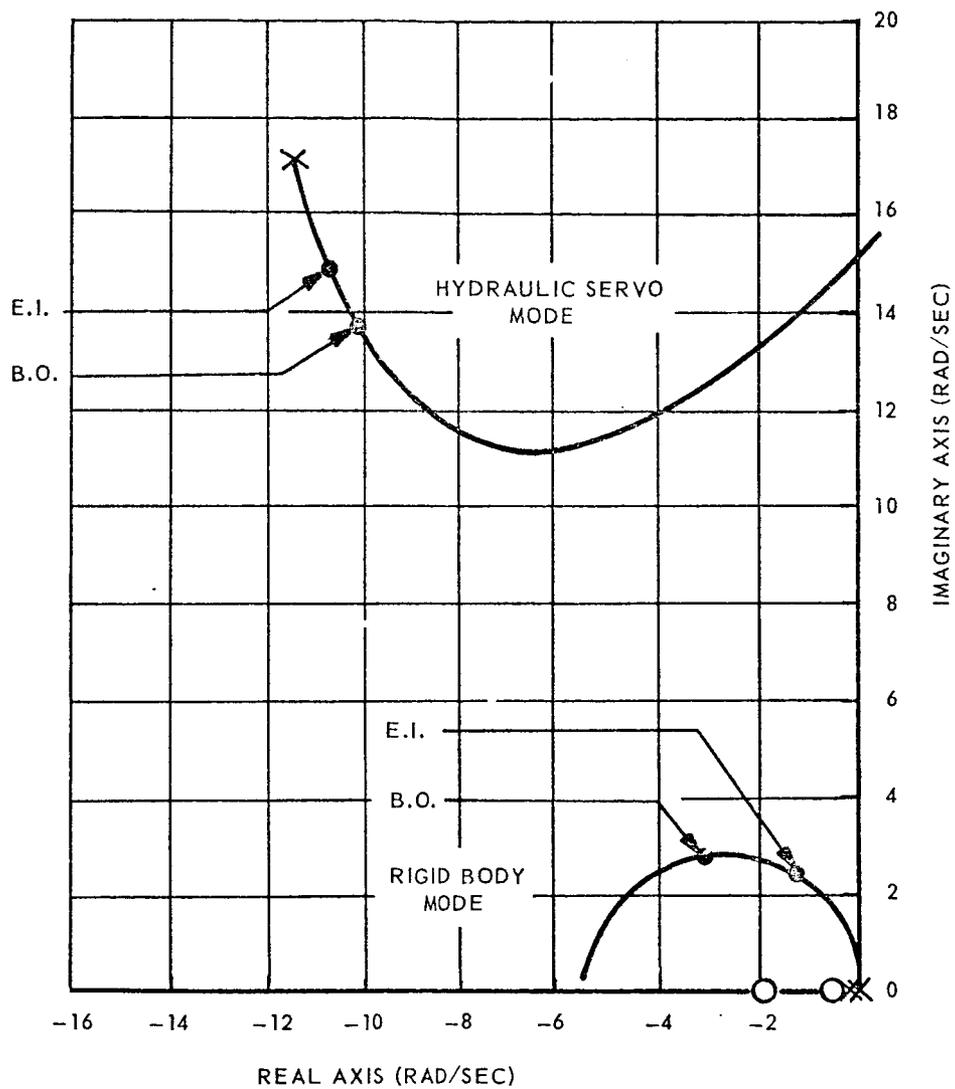
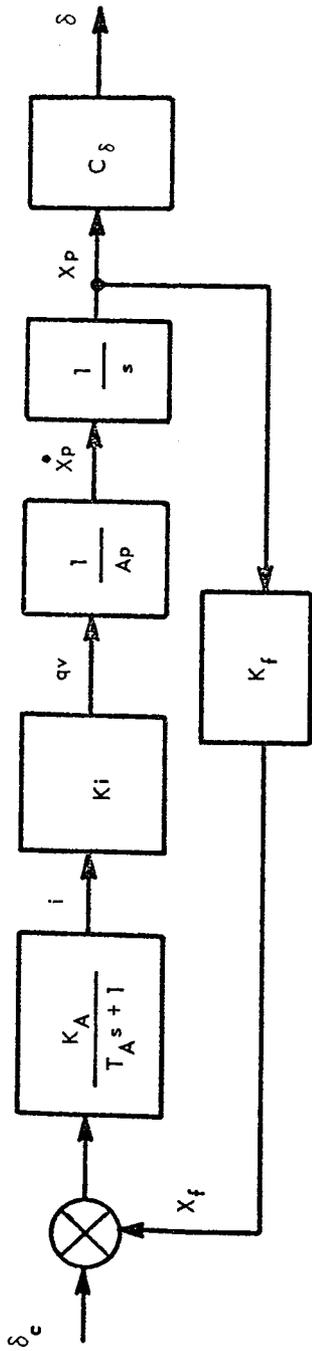


Fig. 3-8 Hydraulic Channel Root Locus (Without Horizon Sensor)



$K_A = 0.66 \text{ MA/VOLT}$   
 $T_A = 0.43 \text{ SEC}$   
 $K_i = 0.64 \text{ IN}^3/\text{SEC/MA}$   
 $A_p = 0.456 \text{ IN}^2$   
 $K_f = 19.8 \text{ V/IN}$   
 $C_\delta = 5.88 \text{ DEG/IN}$

3-14

$$\text{TRANSFER FUNCTION} = \frac{\delta}{\delta_c} = \frac{C_\delta K_A K_i K_f}{K_f T_A A_p} \frac{1}{s^2 + \frac{1}{T_A} s + \frac{K_A K_i K_f}{T_A A_p}} = K_H \frac{W_n^2}{s^2 + 2\delta W_n s + W_n^2}$$

$$W_n = \sqrt{\frac{K_A K_i K_f}{T_A A_p}} \text{ RAD/SEC}$$

$$\delta = \frac{1}{2W_n T_A} \frac{C_\delta}{K_f}$$

$$K_H = \frac{C_\delta}{K_f} \text{ DEG/VOLT}$$

Fig. 3-9 Servo Actuator Block Diagram

Significant parameters associated with the engine servo system are as follows:

Natural frequency	20.7 rad/sec
Damping ratio	0.56
Engine gimbal freedom	$\pm 2.5$ degrees
Maximum gimbal rate	20 deg/sec

A root locus for the engine servo is shown in Fig. 3-10.

### 3.3.3 Effect of Fuel Slosh on Agena (Undocked) Hydraulic Autopilot

During the Agena undocked powered flight phase of the ATV mission, the propellant slosh mass is very slightly coupled to the rigid body mode. Qualitatively, this effect can be seen by the frequency spread between the pole and zero associated with the slosh mode. The root locus for the hydraulic control (undocked) is shown in Fig. 3-11.

The dominant rigid body mode is not influenced by propellant dynamics at engine ignition but is affected near engine burnout. However, satisfactory response and stability margins are still maintained at these stages during the powered phase of flight.

### 3.3.4 Effects of Bending Dynamics and Swiveling Engine

The effects of the bending dynamics in the undocked mode do not present a problem since the resonant frequency is in excess of 30 cps. Even though the structural damping is as low as 1 percent, the attenuation at these frequencies is sufficient to insure system stability.

The swiveling engine represents nearly 1.5 percent of the total system mass. Its dynamic effect is to introduce a pair of zeros on the imaginary axis at approximately 5 cps. This can be seen on the root locus plot, Fig. 3-11. Although this effect contributes to the dynamics of the dominant mode, it is still possible to maintain the desired response and stability.

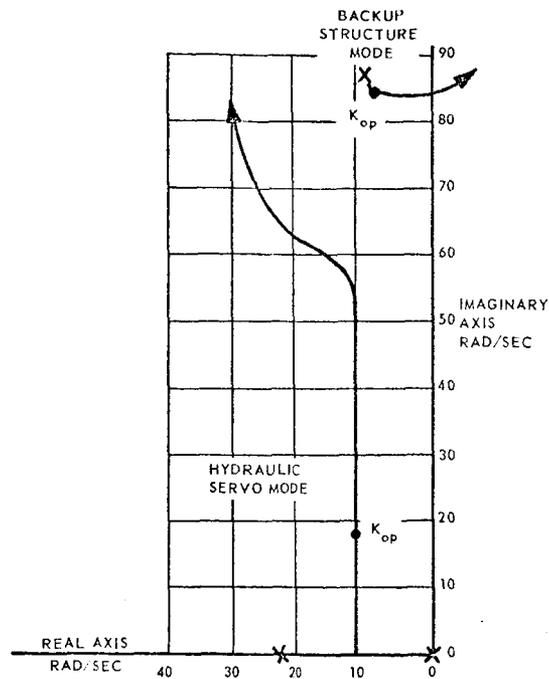


Fig. 3-10 Engine Servo Root Locus

### 3.3.5 Hydraulic Control System (Docked)

During a powered phase of the docked mode, stabilizing the hydraulic control system while maintaining the response becomes a formidable task. The first bending natural frequency has been decreased to approximately 2.7 cps. This is a result of the relatively low effective spring rate between the Agena and spacecraft. To give adequate stability and response for the docked mode, it has been proposed that a notch filter be introduced in the auto-pilot compensation network. This will give the required attenuation of the lightly damped first bending mode without a resulting degradation of the dominant rigid body response.

At present it appears that even with this notch filter, the slosh mode may produce an unstable operating region. It is anticipated that this region will be sufficiently short in duration to present no absolute instability. Since slosh nonlinearities are also present, it is safe to presume low criteria on stability margins without encountering sustained oscillations of the system.

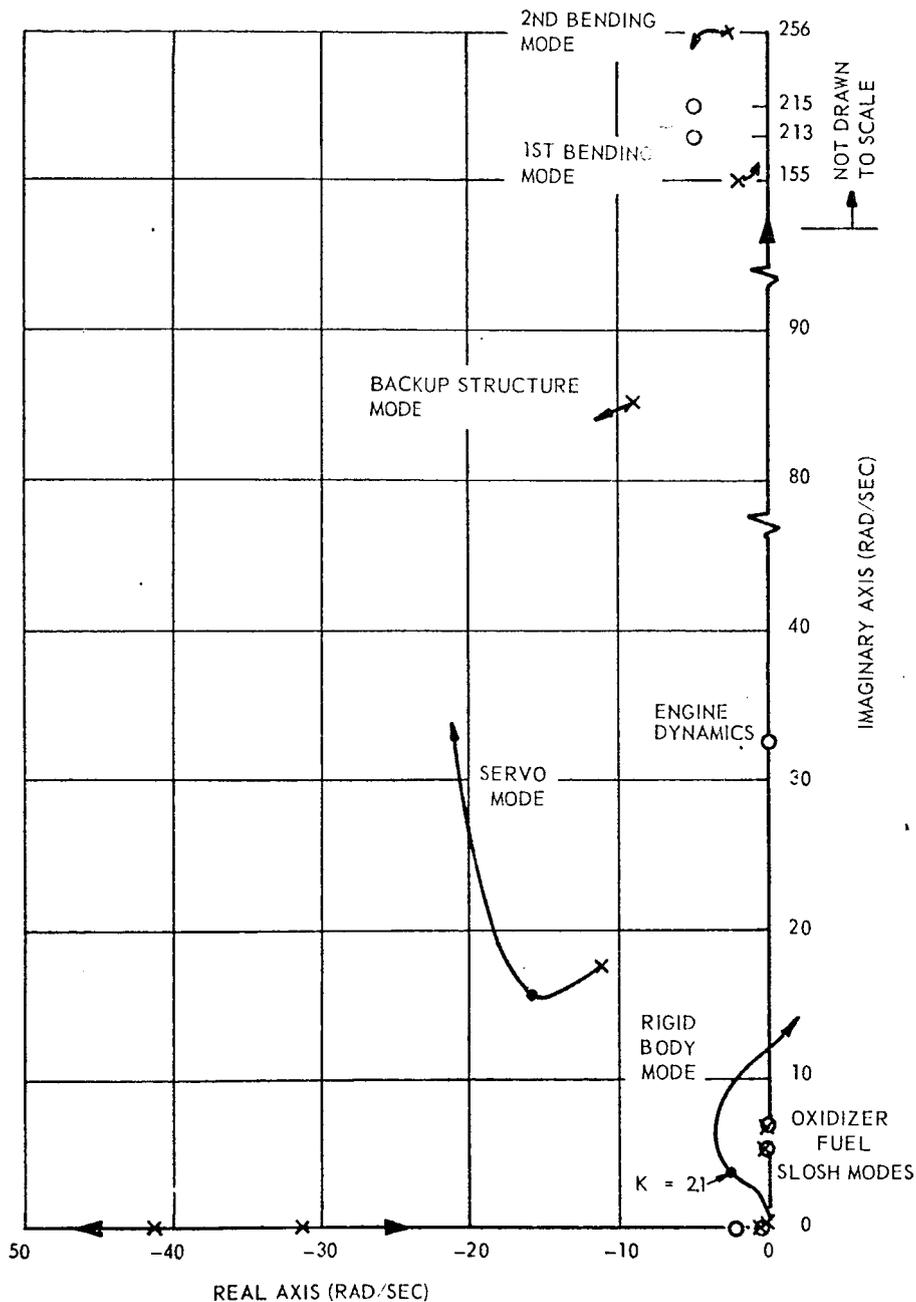


Fig. 3-11 Hydraulic Channel Root Locus (Undocked at Engine Ignition)

## Section 4 GUIDANCE ANALYSIS

The primary objective of this section is to outline the Atlas-Agena guidance system and establish the error sources associated with the Agena system. The deviations in injection and orbit parameters that result from these guidance errors have not been presented here since this type of information is beyond the scope of this analysis.

At the time of writing of this document there is no explicit specification of the accuracy required for the Atlas-Agena ascent system other than that of orbit injection with a maximum of velocity change capability for the rendezvous maneuver.

### 4.1 GUIDANCE SYSTEM DESCRIPTION

#### 4.1.1 Booster Guidance

The Atlas boost phase of the ATV mission utilizes a GE/Burroughs radio command guidance system. The ground based equipment consists of a track radar, doppler radar, and digital computer while the airborne equipment consists of a doppler beacon transponder, pulse beacon transponder, and decoder with associated antenna and electronics.

The ground based digital computer samples the radar data and performs the necessary calculations to generate steering commands in pitch and yaw, discrete functions, and engine cutoff commands. These commands are transmitted via the track radar in a suitable pulse code. The Atlas command receiver transmits these signals to the decoder, which processes them, generating the appropriate autopilot command and/or discrete function.

The technical requirements for this system are established by the desired coast ellipse orbit defined by the major and minor axes and inclination angle. The Atlas guidance system must also provide the correct initial reference angles for the Agena gyros when they are uncaged at vernier engine cutoff.

#### 4.1.2 Agena Guidance System

The Agena guidance system incorporates the velocity-meter, inertial reference package, horizon sensor, ascent timer, flight control electronics, pneumatic system, and hydraulic servo actuators for thrust vector control. During the orbital phase of this mission, it is necessary to include the command and telemetry system to realize the guidance function satisfactorily.

The Agena ascent boost phase of this mission is controlled by a preprogrammed timer. This timer, described in Section 5 of this report, initiates Agena engine burn and other discrete functions necessary for orbit injection. The time of execution of these functions is dependent upon variations in the Atlas boost; therefore the timer is started by ground command through the Atlas guidance system.

The Agena is to be injected into a nominal 161 nm circular orbit with an inclination angle of 28.87 degrees. Subsequent to orbit injection, the guidance function will be dependent on the relative position between the Gemini and Agena vehicles and the maneuver required to effect rendezvous.

#### 4.2 INDEPENDENT ERROR SOURCES - AGENA D

The following is a tabulation of all the error sources associated with the Agena Target Vehicle guidance system. All quoted numbers are assumed to be  $3\sigma$  values. Variations in the guidance component performance is rigidly controlled by specifications that require compliance to standards under a variety of test conditions. It is assumed that each variable has a zero mean.

Parameters that affect the guidance accuracy are summarized as follows:

#### 4.2.1 Guidance Component Errors

##### 4.2.1.1 Horizon Sensor

Alignment	0.1 deg
Instrument uncertainty	0.3 deg
Sensor-to-gyro gain	7 %
Horizon noise	0.1 deg

##### 4.2.1.2 IRP Gyros

	<u>MIG</u> *	<u>HIG</u> **
Drift at null	1 deg/hr	6 deg/hr
Drift off null	1 deg/hr/deg	10 deg/hr/deg
Mass unbalance	10 deg/hr/g	25 deg/hr/g
Alignment of gyros to case	5 min of arc	5 min of arc
IRP alignment to module	0.1 deg	0.1 deg
Module alignment to vehicle	0.1 deg	0.1 deg
Elastic restraint	1 deg/hr/deg	10 deg/hr/deg
Anisoelasticity	4.4 deg/hr/g <sup>2</sup>	1.6 deg/hr/g <sup>2</sup>

##### 4.2.1.3 Velocity Meter

Bias	$2 \times 10^{-4} g$
Bias under vibration (g's)	$10^{-4} g$
Scale-factor linearity	0.015%
Resolution ( ÷ 4 mode)	0.52 ft/sec
Null stability	$1.5 \times 10^{-4} g$
Scale-factor stability	0.02%/month

\*Miniature integrating gyro

\*\*Hermetic integrating gyro

4.2.1.4 Sequence Timer

Switching time	0.1 sec
Repeatability	0.2 sec
Elapsed time error	0.01%
Resolution	1.0 sec

4.2.1.5 Programmer Accuracy

Elapsed time accuracy	0.006 %
Switching time	$2 \times 10^{-6}$ sec
Repeatability	$2 \times 10^{-3}$ sec
Resolution	$8 \times 10^{-3}$ sec

## 4.2.2 Alignment Errors

The required alignment tolerances are taken as the uncertainty values and summarized as follows:

Horizon sensor to module	5 minutes of arc
Velocity meter to module	6 minutes of arc
IRP to module	6 minutes of arc
Module to vehicle centerline	6 minutes of arc
Horizon sensor bias angle	8 minutes of arc
Engine-actuator null position (including thrust chamber anomalies)	12 minutes of arc
Engine-gimbal point displacement from principal longitudinal axis	0.062 in.
Engine-turbine exhaust (producing roll moment)	15 minutes of arc

The relationship of the Agena D center of gravity from the longitudinal axis is also of significant importance to the guidance accuracy. The location of the center of gravity function of fuel depletion is a well established parameter and is periodically updated to include any weight modifications.

Section 5  
DESCRIPTION OF COMPONENTS

5.1 INERTIAL REFERENCE PACKAGE

The inertial reference package (IRP) is a three-axis stabilization and control unit. It is a "strapped down" or nongimbaled inertial system used to provide an attitude reference about the major vehicle axes.

The IRP uses hermetic integrating gyros in the pitch and yaw axes and a wide-angle miniature integrating gyro in the roll axis to supply rate signals or displacement signals as required. (See Fig. 5-1.) When the gyros are operated in a caged or closed loop mode, rate signals are supplied; when they are operated in an open-loop mode, displacement signals are supplied. In addition, the gyros contain torque generators that can be operated to cause displacement of the gimbal.

Rates sensed by the gyros appear as precessional torques on the gyro gimbal. These torques are summed with torques developed in the torque generator by programmed or horizon sensor inputs. A signal proportional to gimbal displacement, which is developed by the signal generator, is amplified and used as an input to the flight control electronics.

The dimensions and weight of the IRP (Fig. 5-2) are as follows:

- a. Weight. 34 pounds (maximum)
- b. Height. 12.5 inches (maximum)
- c. Diameter. 12.03 inches (maximum)

A schematic diagram of the inertial reference package is included in Appendix B (Fig. B-1).

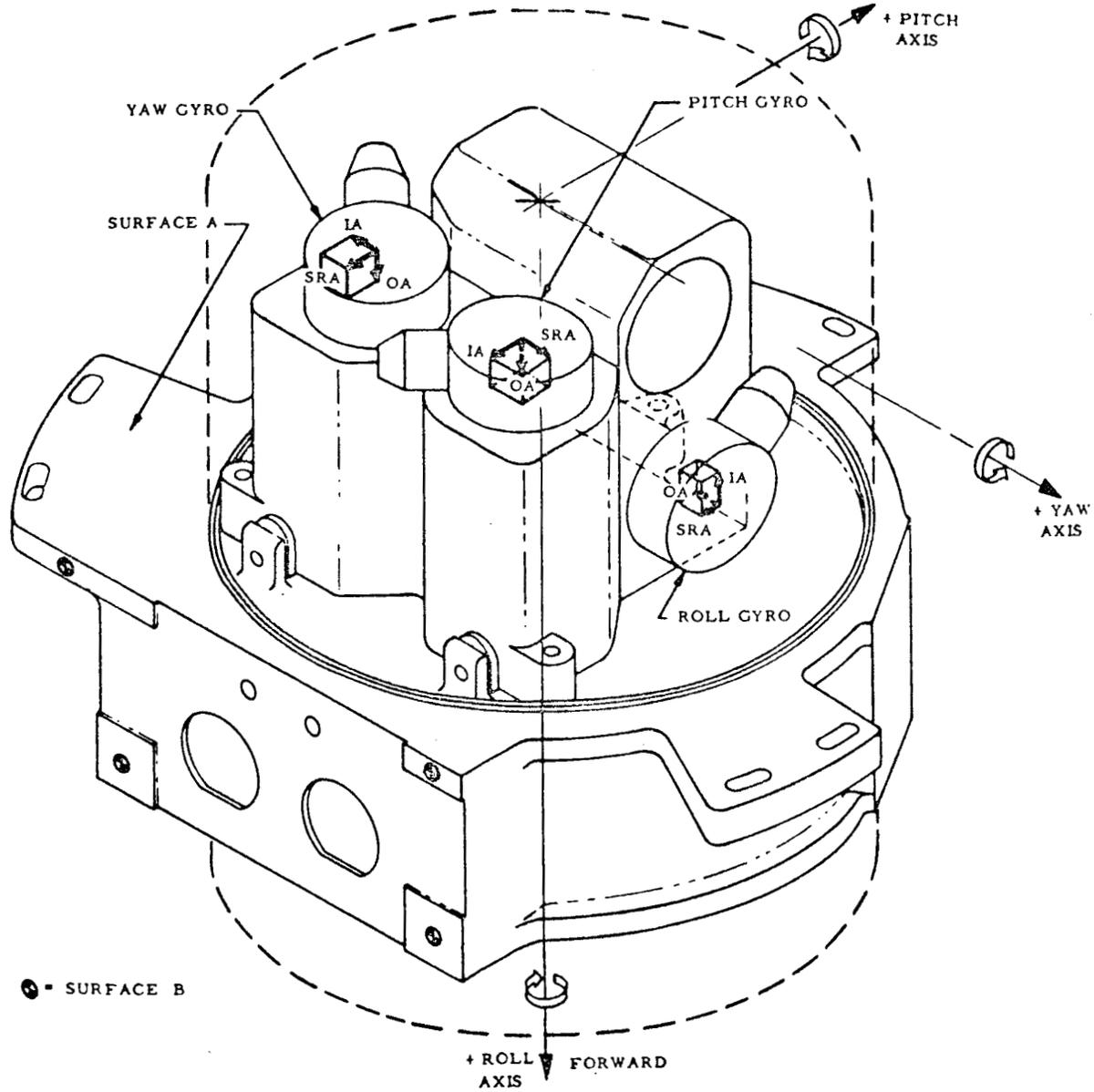


Fig. 5-1 IRP Axes and Phase Relationships

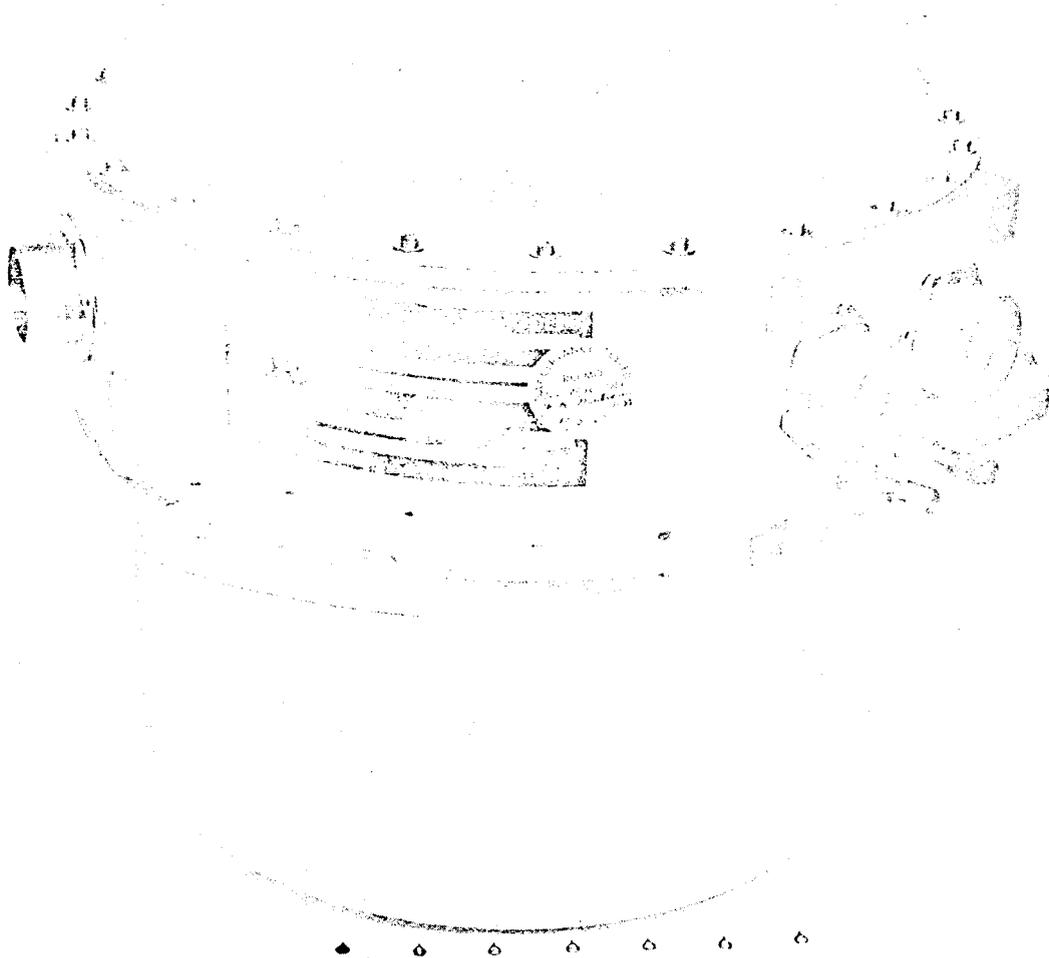


Fig. 5-2 Inertial Reference Package (IRP)

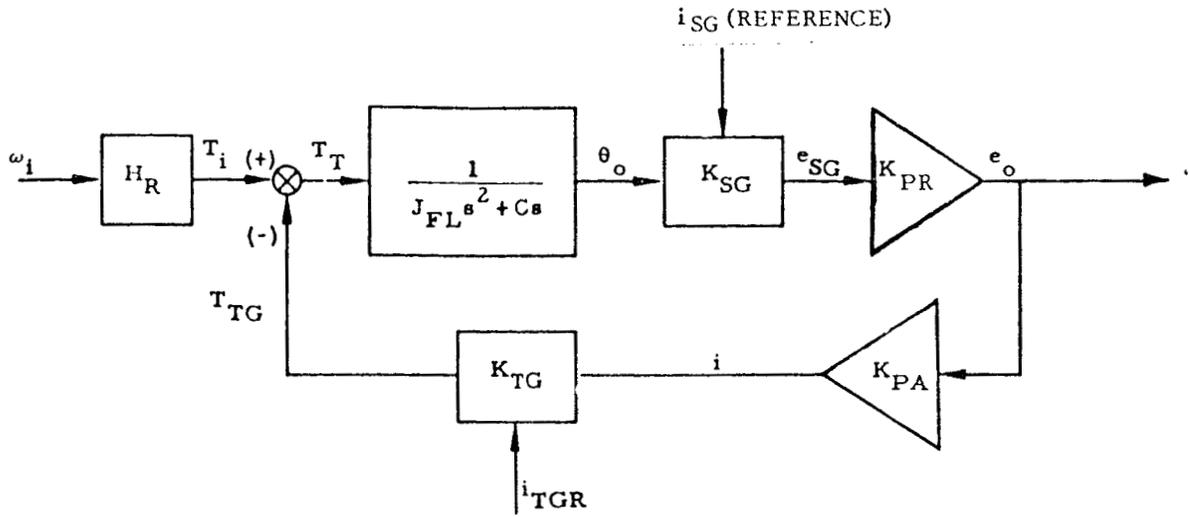
### 5.1.1 System Operation

The IRP contains yaw, roll, and pitch channels. Each channel uses a gyroscope operating at a controlled temperature and associated electronic equipment to provide control signals. The IRP is operated in one of two modes with relays accomplishing the mode switching in response to external signals. In Mode I, the gyros are caged to the vehicle reference axes. In Mode II, the gyros are operated open loop. (See Fig. 5-3.)

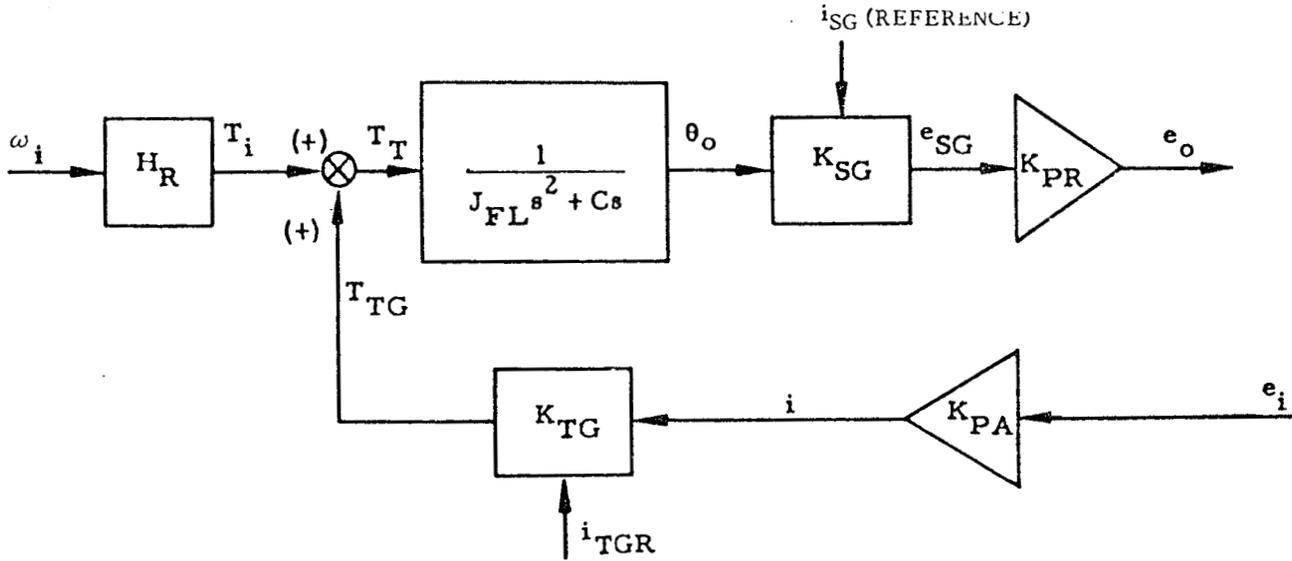
In Mode I operation (A, Fig. 5-3), the cage relay feeds a signal from the gyro signal generator back to the gyro torque generator through the IRP amplifiers. This provides the caged or closed loop operation. Although the output signal is not used in Mode I, it is proportional to the rate of rotation about the input axes. The time constant of each gyro channel in Mode I is determined by the ratio  $C/K$ , where  $C$  equals gyro viscous damping and  $K$  represents the loop gain. The time constant is specified to be 3 seconds.

In Mode II, (B, Fig. 5-3), the gyros are uncaged by opening the caging feedback circuit. This changes the gyro operation to that of a rate integrating (displacement) gyro. In this mode of operation, a signal is supplied when a deviation of the vehicle occurs. Rebalance can only be made by changes in the vehicle attitude, which is accomplished by applying the preamplifier output signal to the flight control electronics.

In Mode II operation, the orbital reference is maintained for the pitch and roll gyros by applying the pitch and roll horizon sensor signals to the respective torque generators. The yaw reference is maintained by gyrocompassing. This use of horizon sensor signals and gyrocompassing nullifies gyro drift.



A. Caged Loop Mode



B. Open Loop Mode

- |   |  |
|---|--|
| C = DAMPING COEFFICIENT OF GYRO FLUID   | J <sub>FL</sub> = GYRO GIMBAL MOMENT OF INERTIA        |
| e <sub>i</sub> = INPUT VOLTAGE (e.g. FROM HORIZON SENSORS OR PROGRAM RATE VOLTAGES) | K <sub>PA</sub> = POWER-AMPLIFIER GAIN                 |
| e <sub>o</sub> = OUTPUT VOLTAGE FROM GYRO PREAMPLIFIER                              | K <sub>PR</sub> = PREAMPLIFIER GAIN                    |
| H <sub>R</sub> = ANGULAR MOMENTUM OF GYRO ROTOR                                     | K <sub>SG</sub> = SENSITIVITY OF GYRO SIGNAL GENERATOR |
| i <sub>TGR</sub> = REFERENCE CURRENT OF GYRO TORQUE GENERATOR (FOR HIG GYROS ONLY)  | K <sub>TG</sub> = SENSITIVITY OF GYRO TORQUE GENERATOR |
|   | s = LAPLACE TRANSFORM VARIABLE                         |
|   | ω <sub>i</sub> = ANGULAR RATE INPUT TO GYRO            |

Fig. 5-3 Gyro Loop Block Diagram

In Mode II, programmed signals can also be applied to the torque generators that will cause predetermined rates to be applied to the vehicle.

5.1.2 Description of Components

5.1.2.1 Heater Block and Gyro Assembly. The heater block contains the three gyros, heating elements, and temperature monitoring devices. (See Fig. 5-4.) It has a large thermal inertia that insures temperature stability. The heater block is used for the following purposes:

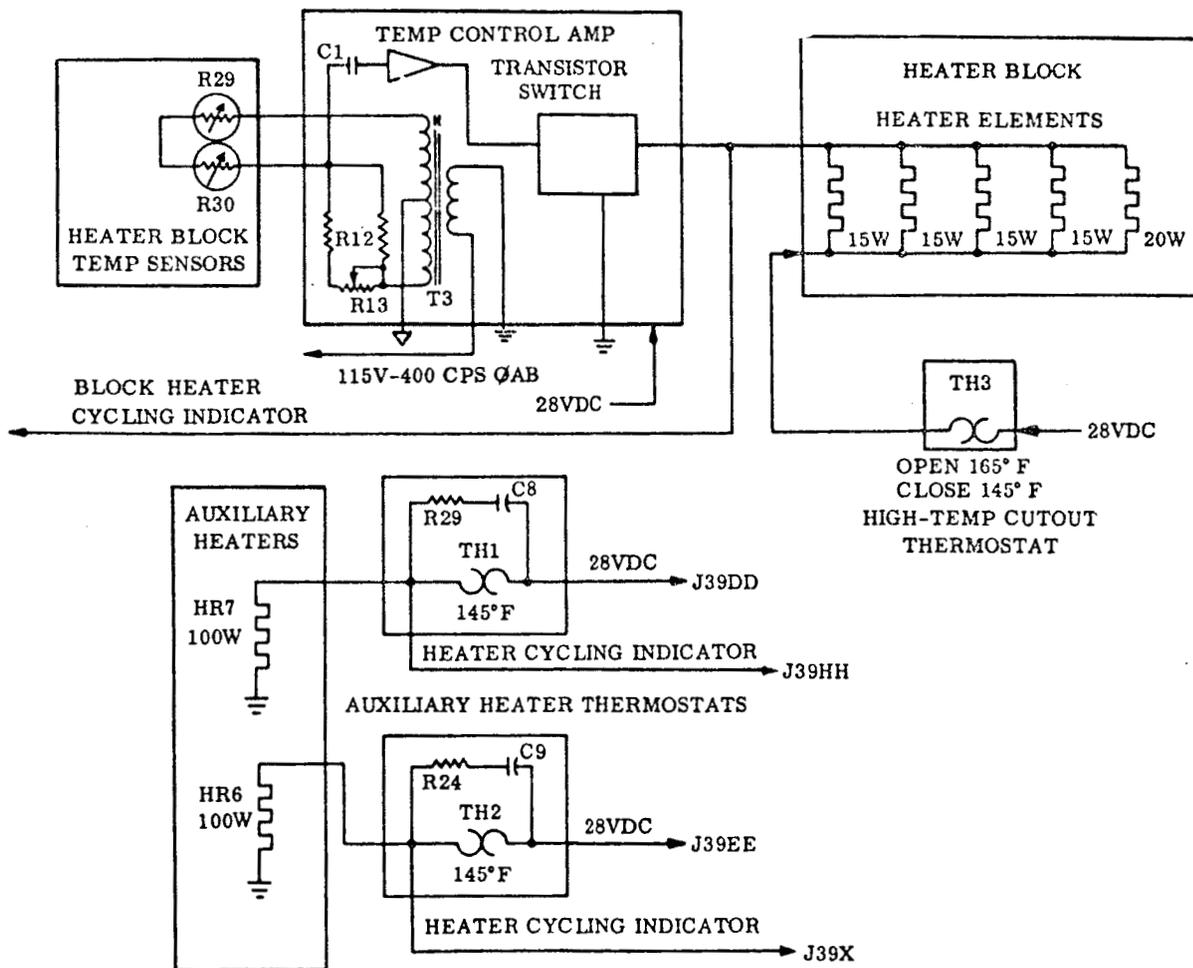


Fig. 5-4 Block Temperature Control Circuits

- a. To mount the gyros mechanically, maintaining alignment with respect to machined reference surfaces A and B (Fig. 5-1)
- b. To maintain the pitch and yaw gyros at a constant temperature of 145°F
- c. To provide a 145°F ambient temperature for the roll gyro.

There are two sets of heating elements in the heater block; one set consists of two elements, and the other of five elements.

The elements in the two-element heating set are used as auxiliary heaters. Each element is controlled by a thermostat that opens when the heater block reaches 145°F and closes when the temperature drops to 130°F.

The elements of the five-element heating set are used as the main heaters. These elements are controlled by an electronic circuit incorporating two sensing elements and a control amplifier. A high temperature cutout thermostat set to open at 165°F prevents overheating in the event of a malfunction in the electronic control system.

The main heaters provide all of the heat required to maintain the heater block at 145°F. The auxiliary heaters are used to decrease the warmup period or to keep the gyros warm during ground or orbital storage.

The roll gyro is thermally isolated from the heater block and is maintained at 180°F by a separate control system. (See Fig. 5-5.)

In addition to the temperature control systems, a thermistor is used to provide heater block temperature information for telemetry.

5.1.2.2 Gyros. The IRP uses hermetic integrating gyros (HIG) in the pitch and yaw axes and a miniature integrating gyro (MIG) in the roll axis. A cut-away view of the HIG gyro is presented in Fig. 5-6.

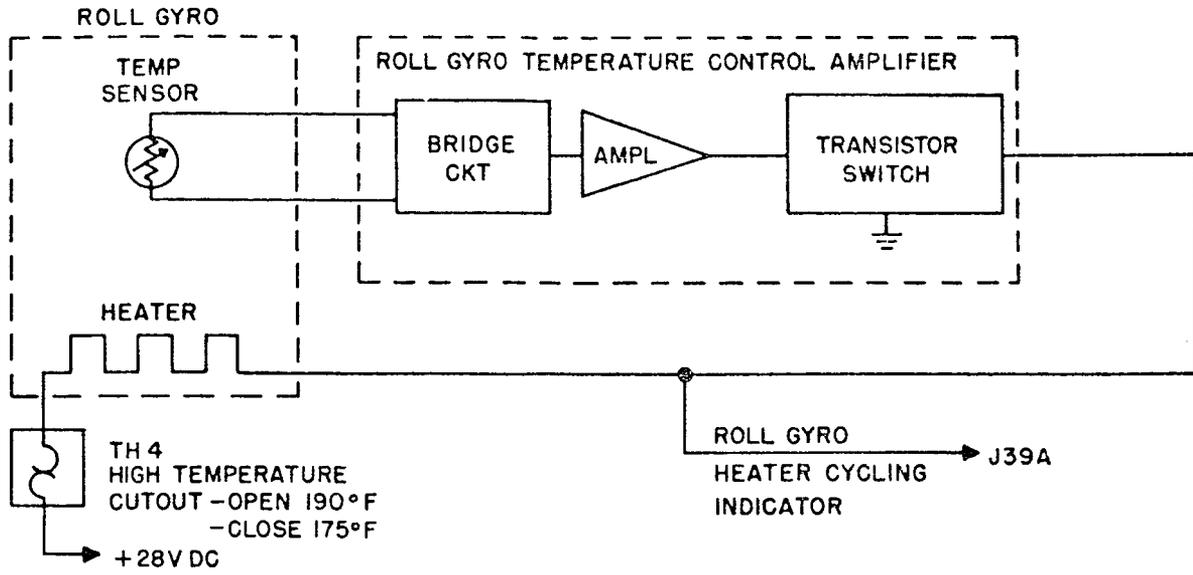


Fig. 5-5 Roll Temperature Control Circuits

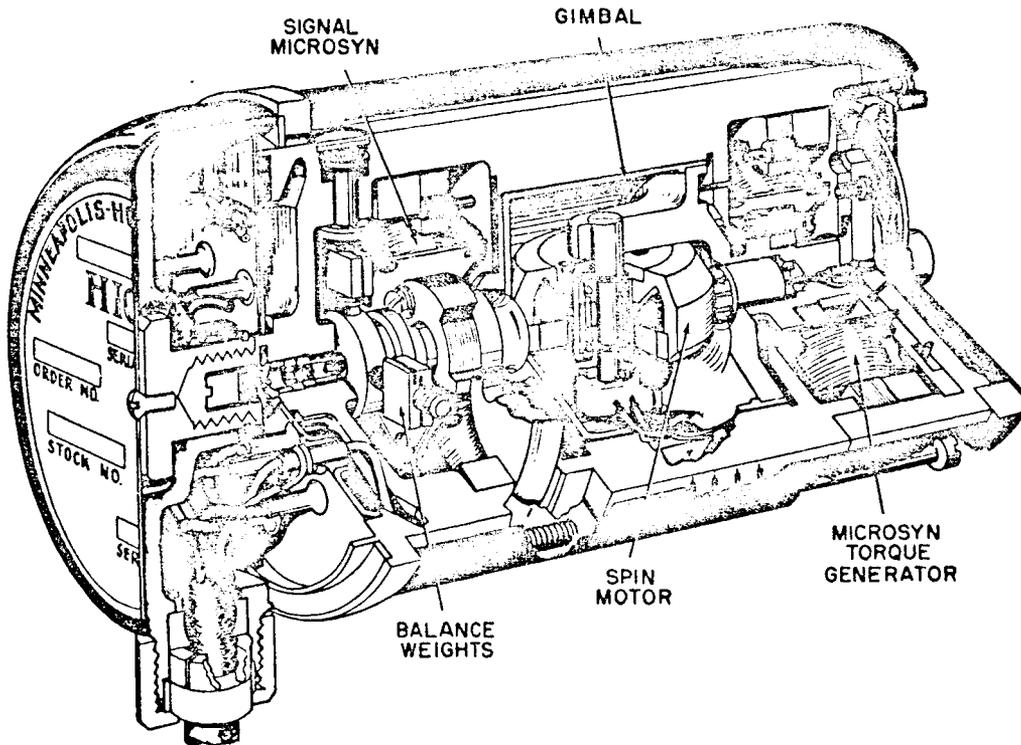


Fig. 5-6 Cutaway View of HIG-4 Gyro

Performance characteristics of HIG and MIG gyros are as follows:

	<u>HIG</u>	<u>MIG</u>
Input axis freedom	±10 deg	±10 deg
Angular momentum	$1.5 \times 10^4$ gm-cm <sup>2</sup> /sec	$10^5$ gm-cm <sup>2</sup> /sec
Maximum commanded rate	4.0 deg/sec	1.67 deg/sec
Drift uncertainty	3 deg/hr	1 deg/hr
Total mass unbalance drift	25 deg/hr/g	10 deg/hr/g
Signal generator sensitivity	27 volts/radian	12.5 volts/radian
Torque generator sensitivity	7.28 dyne-cm/ (ma) <sup>2</sup>	825 deg/hr/ma

The gyros are rigidly mounted in the IRP, which is rigidly mounted in the vehicle. If the vehicle rotates about any of its axes, the corresponding gyro also turns about its input axis. This causes a precessional torque to be developed about the gimbal output axis equal to the product of the gyro wheel angular momentum and the input rate. The torque is algebraically summed with torque produced by the torque generator.

Since gimbal inertia and friction are small, the only restraint to gimbal motion is that provided by the action of the damping fluid if no excitation is supplied to the torque generator. As a result, gimbal displacement is proportional to the time integral of the input turning rate.

The rotor of the signal generator microsyn (Fig. 5-7) rotates with the gimbal. The output is a voltage proportional to the product of gimbal displacement and the frequency and magnitude of the current used to excite the microsyn. The voltage per degree of gimbal angle is known as the signal generator transfer function.

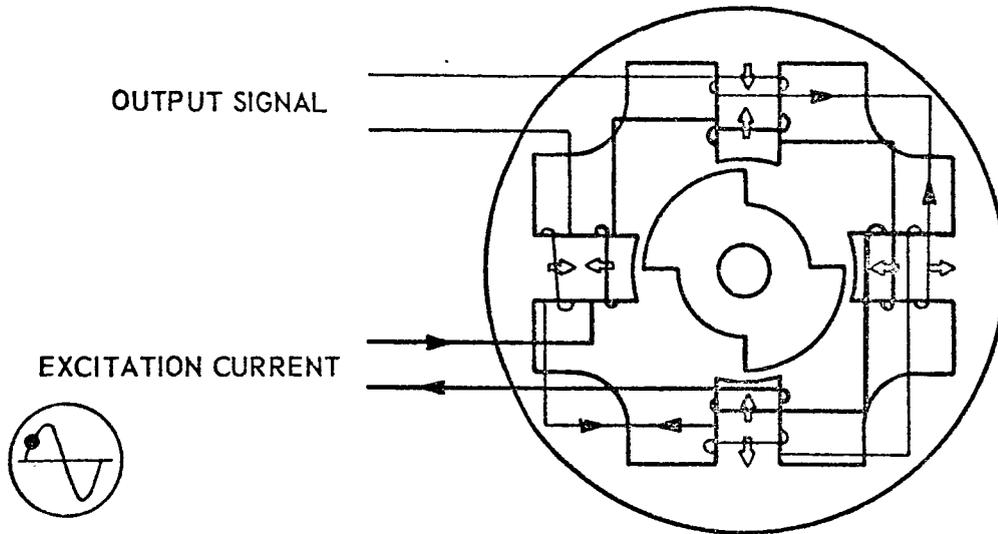


Fig. 5-7 Functional Diagram of Signal Generator Microsyn

The rotor of the torque generator microsyn (Fig. 5-8) is also fixed to the gimbal to which it applies precessional torques. The torque generator is physically similar to the signal generator, and the torque it develops is proportional to the product of the currents applied to its two sets of windings.

The torque generator and signal generator are at opposite ends of the gyro gimbal in the HIG. In the MIG, the two functions are performed by a dualsyn at one end of the gimbal. The torque generating portion of this dualsyn uses a permanent magnet in place of the conventional electromagnet so that developed torque is proportional to the d-c input.

The gyro is filled with a viscous damping fluid that also floats the gimbal, provides shock protection, and reduces gimbal bearing friction. Damping is directly proportional to viscosity, which is directly proportional to temperature. This is why accurate temperature control is required.

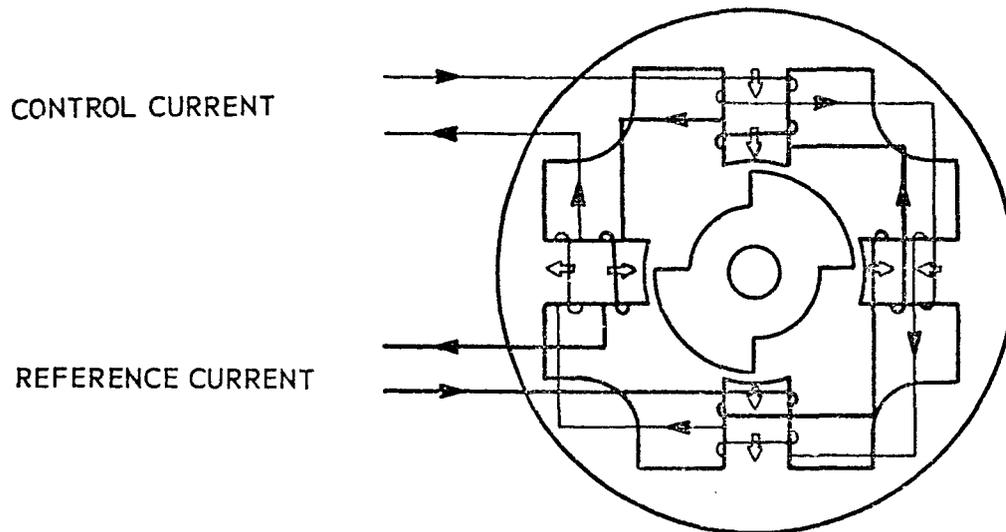


Fig. 5-8 Functional Diagram of Torque Generator

5.1.2.3 Amplifiers. Both temperature control amplifiers are basically the same. The temperature sensing elements form one leg of an a-c bridge. The bridge output is fed into a two-stage, direct-coupled amplifier that drives an amplifier and half-wave demodulator section. The amplified demodulator output is applied to a trigger circuit, which is essentially a d-c amplifier with positive feedback, to obtain bistable operation. The trigger switches the current into the heater elements by switching the power transistor on or off.

The preamplifier serves as a gain stage in each gyro loop and consists of three direct-coupled silicon-junction transistors. Each of the three transistors utilizes a common emitter configuration designed for Class A operation.

The output transformer has a split output winding that is isolated from ground. The output scale factor is 1.67 volts per degree for pitch and yaw and 1.0 volts per degree for roll.

The gyro power amplifier is used as a power gain stage in the yaw and pitch channels. The voltage gain stages of the amplifier consist of two silicon-junction transistors, the output of which feeds a transformer input to a push-pull stage. The output transformer converts from push-pull to single ended operation. The amplifier utilizes fixed current feedback to control the current through the gyro torque generator in a fixed ratio to the input voltage. The amplifier is designed to accept seven separate inputs that are summed through their respective input resistors. Some of the input scale factors are 100 mv/deg/min, and the remainder are 150 mv/deg/min.

The gyro torquing amplifier, used with the MIG in the roll channel, performs a function similar to the gyro power amplifier in providing a signal to drive the gyro torque generator. It consists of a seven input summing amplifier plus a phase-sensitive demodulator. The gain of the three-stage direct-coupled summing amplifier is stabilized by a-c voltage feedback from the output transformer secondary to the input summing point. This amplifier feeds a full-wave demodulator that uses two pairs of matched transistors in a configuration that minimizes output drift.

5.1.2.4 Spin Motor Rotation and Direction Detection (SMRDD). Spin motor rotation and direction is monitored by a special circuit. This consists of a pickup coil on the gyro case and magnets placed in the gyro rotor. Voltages induced in the coils are applied to the SMRDD amplifier.

The SMRDD amplifier requires the presence of four d-c signals on its "AND" gate to light a monitor lamp external to the IRP. The lighted lamp indicates that the three gyro spin motors are running forward. One signal comes from

the three-phase motor supply voltage. The presence of this signal indicates a proper phase sequence. Each of the remaining three signals is supplied from the pickup coils on the gyros. Each of the four signals is filtered, amplified and rectified, then fed into the four "AND" gate inputs. Loss of any of the four signals will cause the lamp to extinguish.

## 5.2 HORIZON SENSOR

### 5.2.1 System Operation

The Model IIC horizon sensor system (Fig. 5-9) is a device manufactured by the Barnes Engineering Company to determine vehicle attitude by sensing the infrared radiation emitted by the earth. The system comprises two sensor heads and a mixer-computer. The heads contain the optical system that receives the radiation, the detector that converts the radiation to an electronic signal, and a preamplifier. The signals from the two heads are processed in the mixer-computer to yield pitch and roll attitude signals.

The system produces pitch and roll outputs that are proportional to the vehicle deviation from the local vertical. The pitch and roll outputs are applied to the inertial reference package to reorient the vehicle. The system also provides telemetry outputs.

The two sensor heads look out and down from the sides of the Agena. Each head sweeps a 75-degree field of view through a conical scan that intersects the earth's thermal horizon. In this way, the sensor head receives the infrared radiation from the earth during part of the scan cycle and radiation from space during the remainder of the cycle. This produces a sharp discontinuity in the intensity of the radiation received. (See Fig. 5-10.)

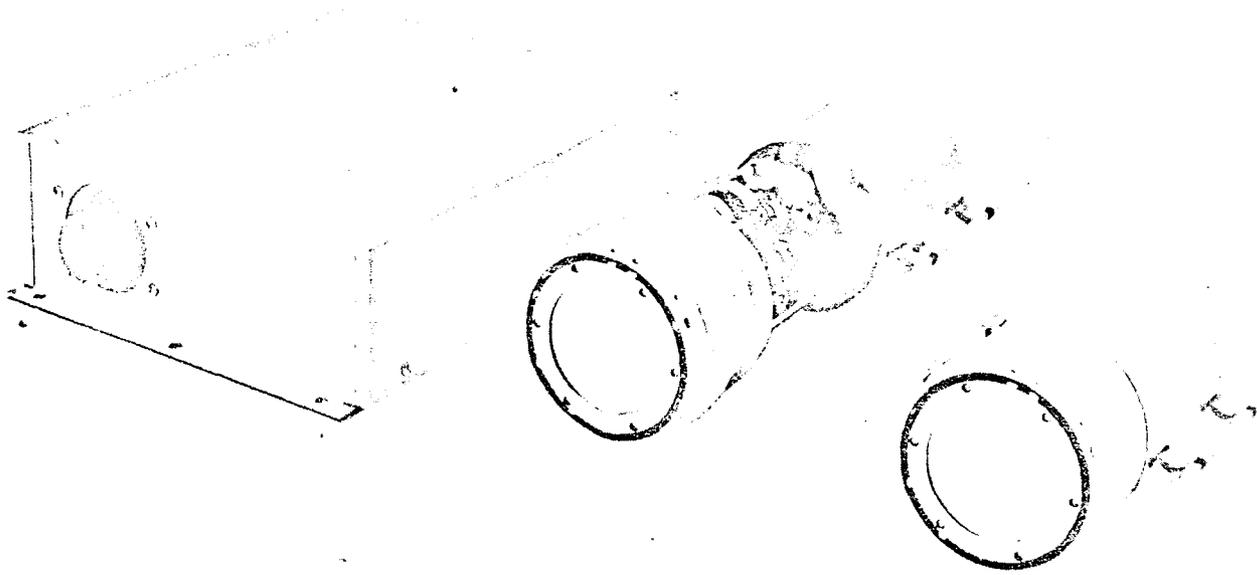


Fig. 5-9 Horizon Sensor System

When no attitude error exists, the two sensor heads see equal portions of the earth, and each earth pulse is centered around a reference built into the system (A, Fig. 5-10). When a roll error occurs, the heads see unequal portions of the earth, but the earth pulses are centered around the reference pulse (B, Fig. 5-10). When a pitch error occurs, the earth pulses are not centered around the reference pulse, but the heads see equal portions of the earth (C, Fig. 5-10).

The electronic circuitry in the mixer-computer determines roll error by comparing earth pulse width from the two heads; it determines pitch error by comparing the center of each pulse with the position of the reference pulse.

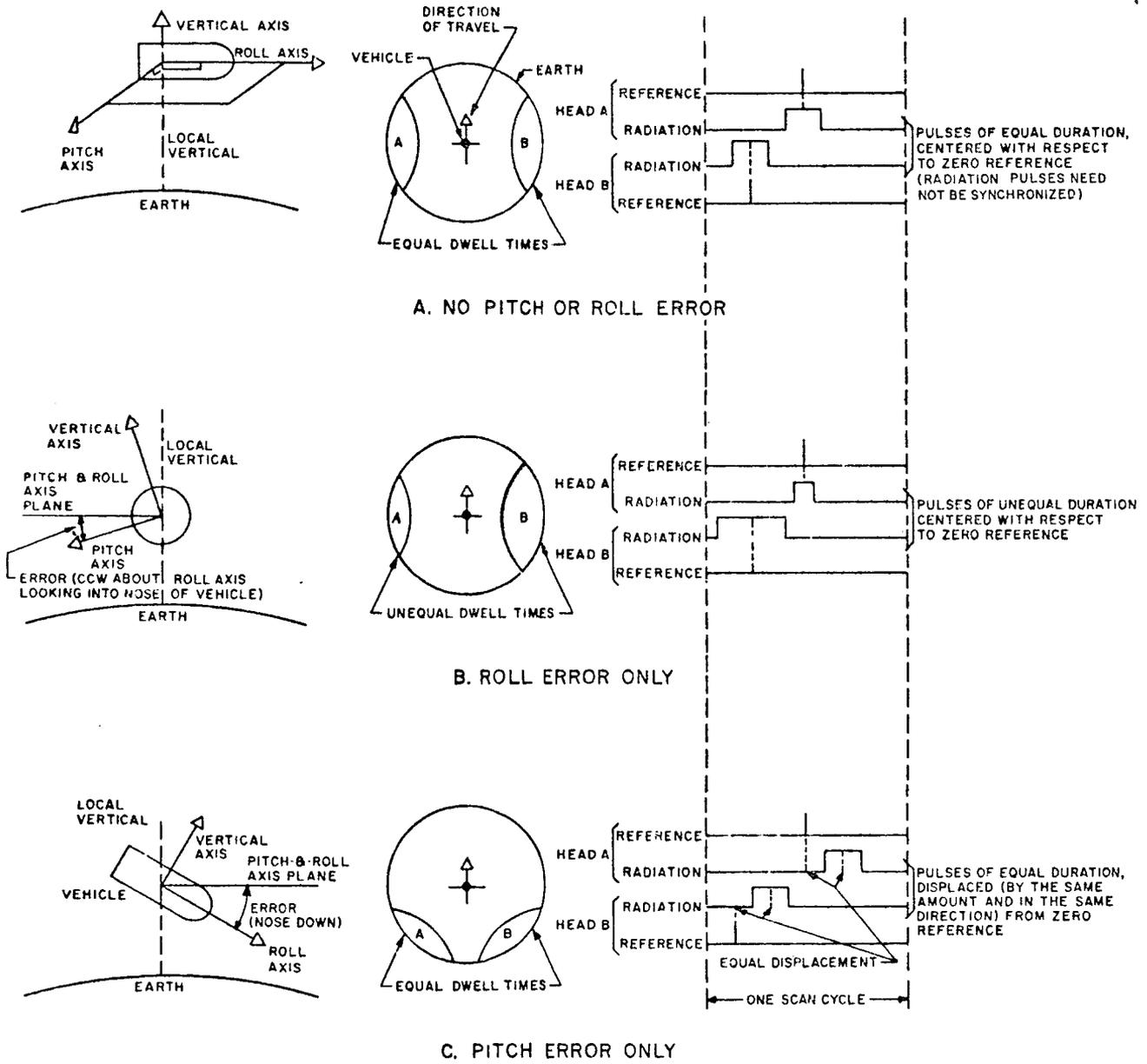


Fig. 5-10 Horizon Sensor Scan Pattern

### 5.2.2 Operational Environment

The optical system of the Model IIC sensor is sensitive to the bandpass in the 12 to 20 micron region. Bandpass frequencies are lower than that of predecessor systems in order to reduce the interfering signal from the sun by a factor of 4 and permit sensor operation in a spectral band where the range of maximum to minimum earth-space height is much less. Since the minimum earth-space pulse level is caused by cold clouds, the shift in spectral band gives better protection from interference by cold clouds. Contrast is reduced since the blackbody curves differ less at the longer wave lengths. Further reduction would be expected because a 12 to 20 micron system sees only upper parts of the atmosphere where temperatures are less variable.

### 5.2.3 System Optics

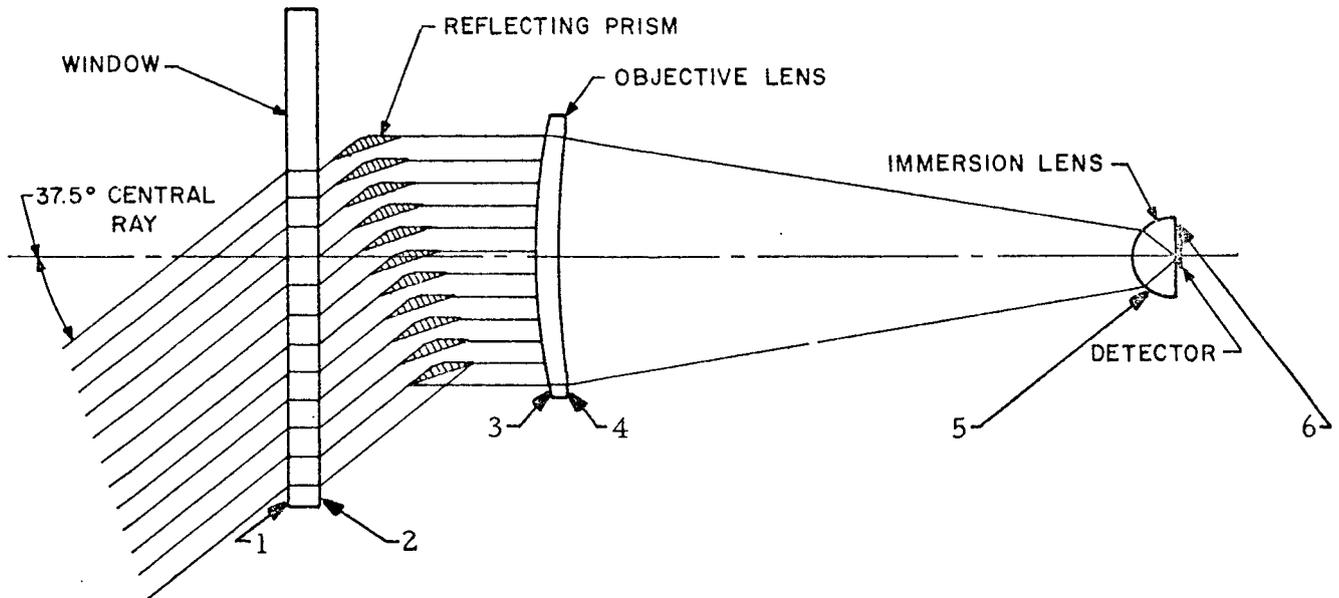
The optics portion of the system (Fig. 5-11) includes the window-scanning prism, the objective lens, and the immersion lens - all made of germanium. The window is coated with an interference type sun filter. The 12 micron cut-on point is determined by the filter, which gives four times the attenuation to the radiation of the sun in comparison with the Model IIA sensor sun filter (8 micron cut-on used on previous Agena vehicles).

The surfaces of the objective lens are antireflection coated to one-fourth wave-length thickness at 15 microns with zinc sulfide. In addition, the immersion lens to which the bolometer flake is cemented has the front surface antireflection coated.

Figure 5-12 shows the resulting spectral transmission curve for the IIC horizon sensor compared to the IIA. It is readily seen from these curves why the IIC has an advantage in cold cloud rejection over the IIA, since the shift is toward the longer wave lengths in the IIC.

OPTICAL PARAMETERS

FIELD OF VIEW ..... 1.8 x 3.6 DEGREES  
 DETECTOR SIZE ..... 0.2 x 0.4 mm  
 FOCAL RATIO ..... /0.2  
 CLEAR APERTURE ..... 1.25 INCHES



Surface	Radius	Clear Aperture	Thickness	Material	Element
1	$\infty$	2.20	0.180	Germanium	Window
2	$\infty$	2.20		Air	
3	4.174	1.25	0.125	Germanium	Objective lens
4	6.490	1.23	3.012	Air	
5	0.140	0.20	0.174	Germanium	Immersion lens
6	$\infty$	0.20			

Fig. 5-11 Optical System for Model IIC Horizon Sensor

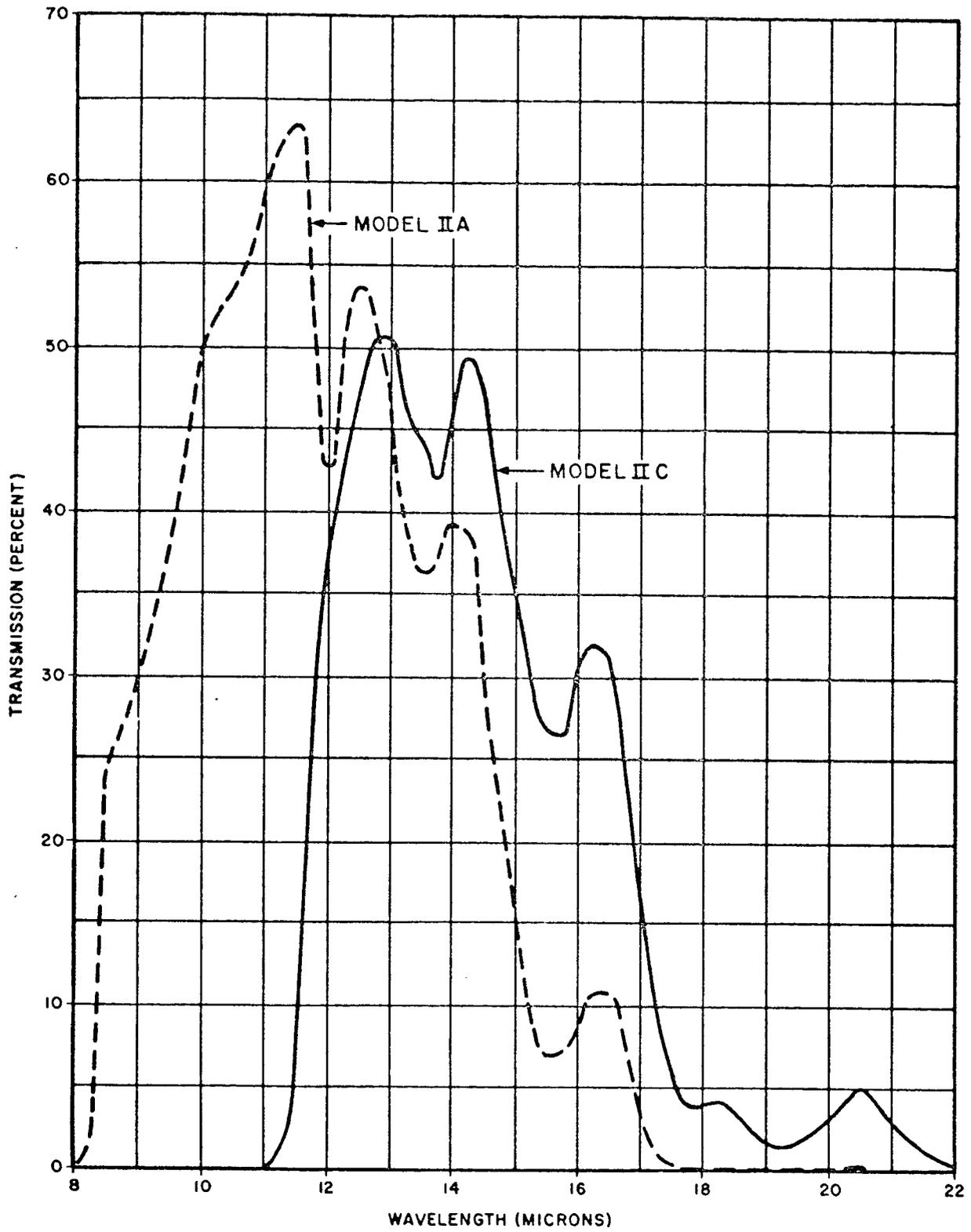


Fig. 5-12 Spectral Transmission of Models IIA and IIC Horizon Sensor

The motor-driven scanning prism deflects the field of view 37.5 degrees to generate the 75 degree conical scan previously described. The radiation exposed to the field of view is focused on the bolometer via the reflecting prism, the objective lens, and the immersion lens.

The prism consists of a rotatable assembly of seven reflecting surfaces situated to achieve no obscuration for incoming radiation 37.5 degrees off axis. This design tends to reduce the effect of sun interference.

#### 5.2.4 Detector

The detector is a thermistor flake cemented to the back side of the germanium immersion lens. It is mounted so that its field of view is aligned with the optical axis of the sensor head. This unit is also referred to as the bolometer.

Selection of the immersion lens thickness depends on two effects that work against one another. Decrease of thickness brings increased power density on the flake, because there is less absorption and because optical gain increases. However, decrease of thickness results in increased vignetting that reduces total power falling on the flake. (Vignetting means that points off axis on the flake do not see the entire entrance aperture and therefore receive a smaller total radiation from the outside.) Figure 5-13 shows how power density on the flake is affected by vignetting and lens thickness. The product of power density and vignetting factor has a peak at the optimum thickness (0.250 inch).

A thickness of 0.174 inch (with  $R = 0.140$  inch) was selected because it represents the closest standard bolometer radius. Operation at 0.174 inch instead of 0.250 inch results in a loss of 5 percent in signal amplitude, which did not prove to be significant in the qualification test system.

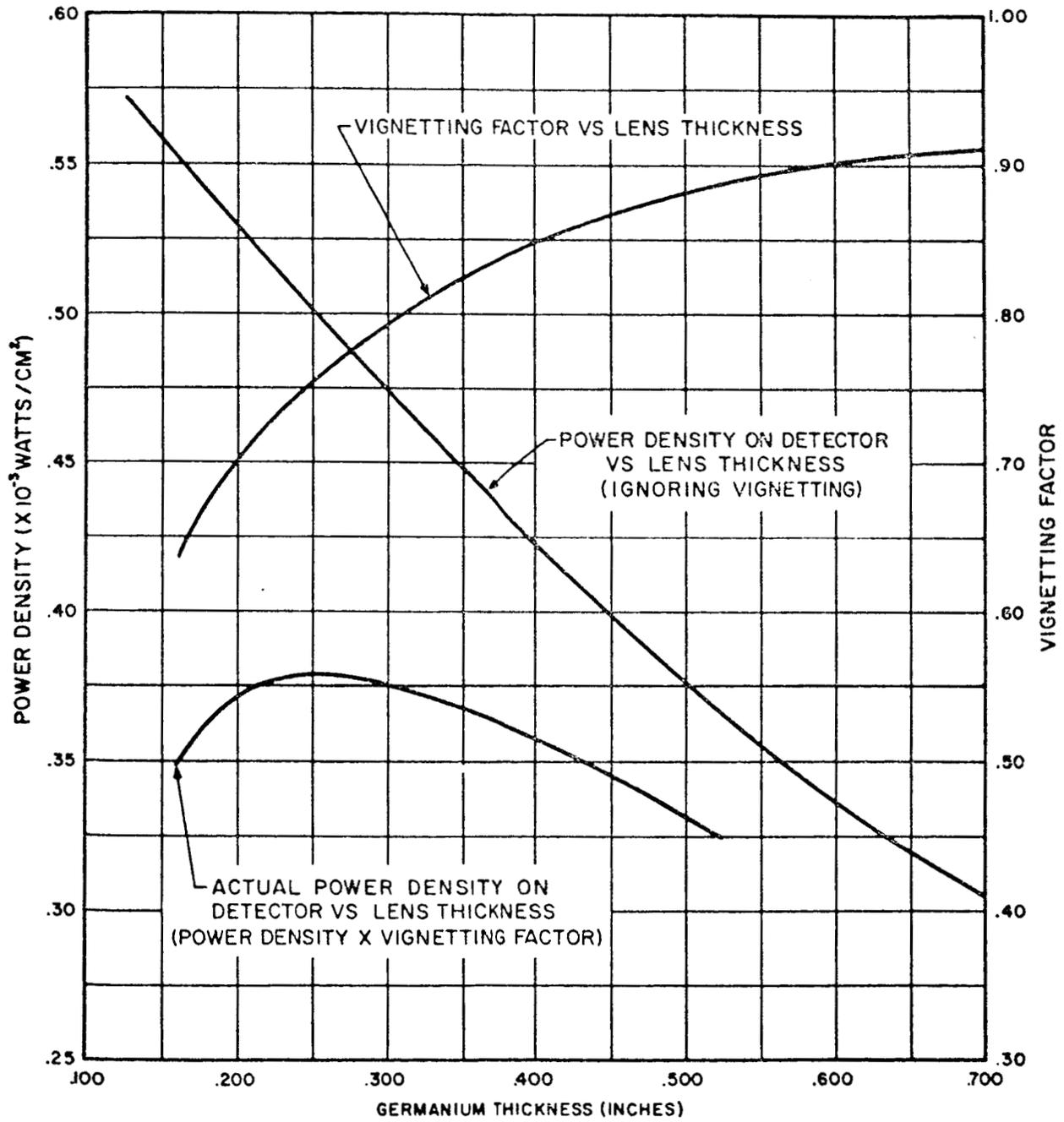


Fig. 5-13 Power Losses Due to Immersion Lens Absorption and Vignetting

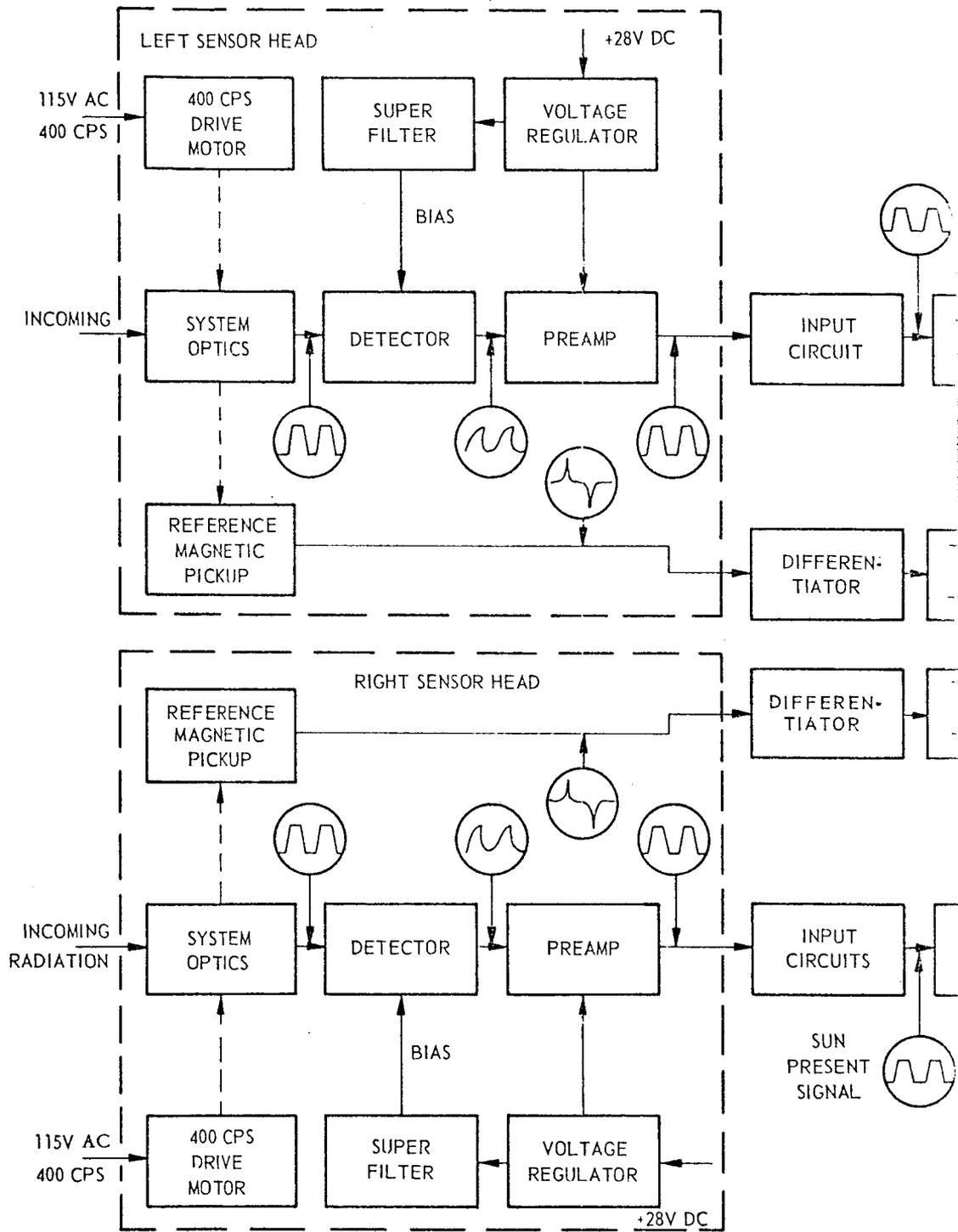
### 5.2.5 Electronic Design

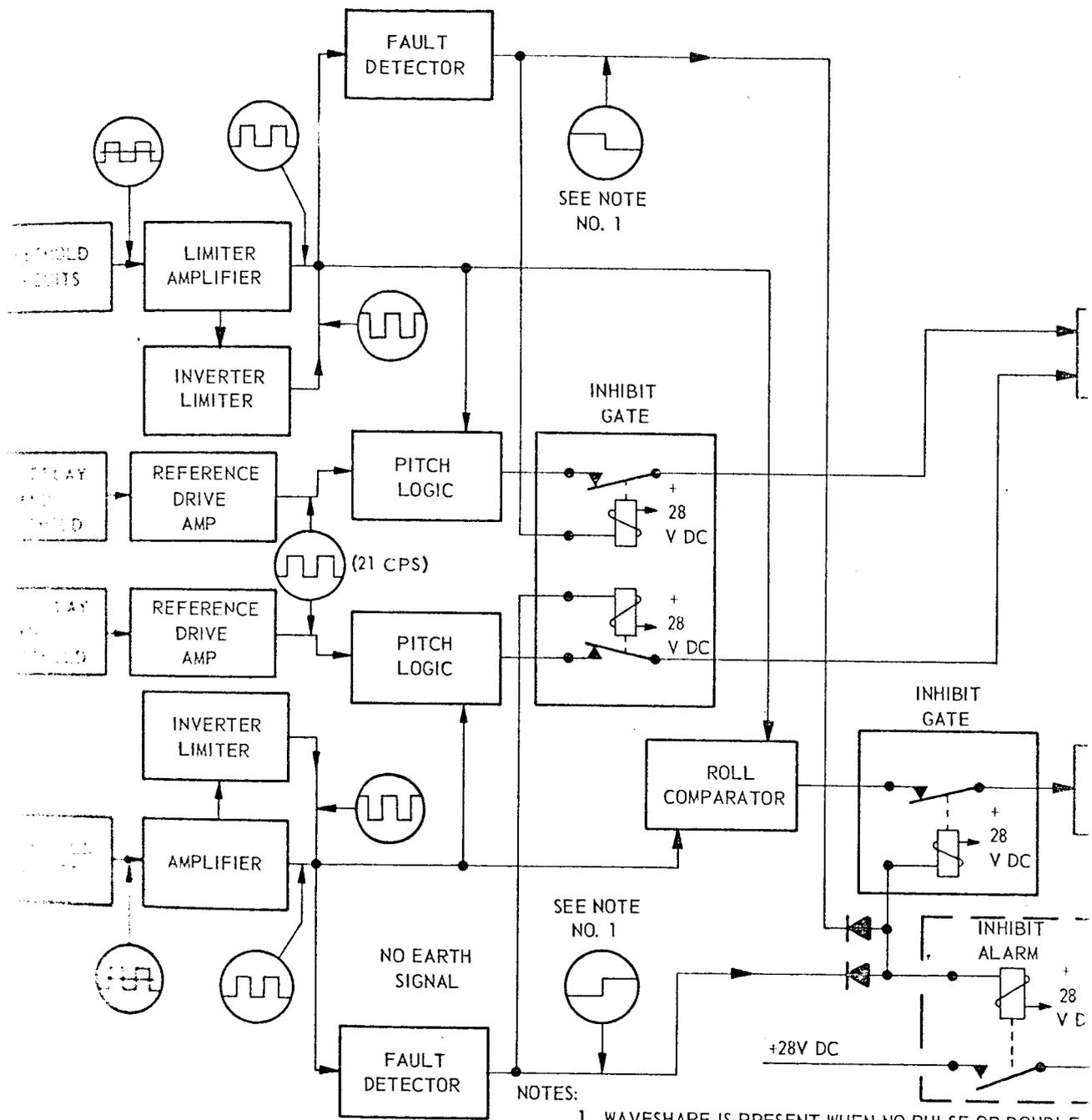
Figure 5-14 presents the block diagram of the Model IIC horizon sensor electronic system. The electronic schematic diagram of the system is included in Appendix B (Fig. B-2).

**5.2.5.1 Preamplifier.** The preamplifier, which is located in the sensor heads, is an operational amplifier. It has independent a-c and d-c feedback paths to provide sufficient high frequency boost to restore the radiation waveform at the output. The earth space contrast is more faithfully reproduced, since the preamplifier is used in the Model IIC with a 21 cps scan rate instead of the 30 cps rate of the Model IIA. Low frequency noise has been further reduced by utilizing a new type of PNP transistor in the bias supply. The preamplifier also contains three stages of amplification, with an amplification factor of 100 per stage, and an emitter follower. An internal a-c feedback network has been incorporated between the stages to prevent any oscillation. In addition, there is an overall external a-c loop.

The overall preamplifier characteristics compensate for detector frequency response nonlinearities, thus recreating the square wave radiation input. The overall a-c feedback reduces the preamplifier gain by 40, and the resulting level is approximately 2 volts.

**5.2.5.2. Input Circuits.** These circuits (Figs. 5-15, 5-16, 5-17, and 5-18) limit the input signal to prevent saturation of subsequent stages; they also contribute an important part in the elimination of the effects of cold clouds and sun. The effect of clouds is reduced by limiting the earth portion of the signal to that nominally produced by a 200°K blackbody and clamping the space portion of the signal to zero volts. This space clamp allows the use of a ground clamp to eliminate sun pulses and to establish a fixed voltage signal threshold. Thus, the circuit assures that in the processing circuitry the contrast between cold clouds (170°K) and the earth can never exceed the contrast between a 170°K and a 200°K blackbody.





NOTES:

1. WAVESHAPES ARE PRESENT WHEN NO PULSE OR DOUBLE PULSE IS PRESENT AT LIMITING AMPLIFIER OUTPUT.
2. WAVESHAPES ARE IDEALIZED. OPERATING CONDITIONS SHOW NO PITCH AND ROLL ERROR.

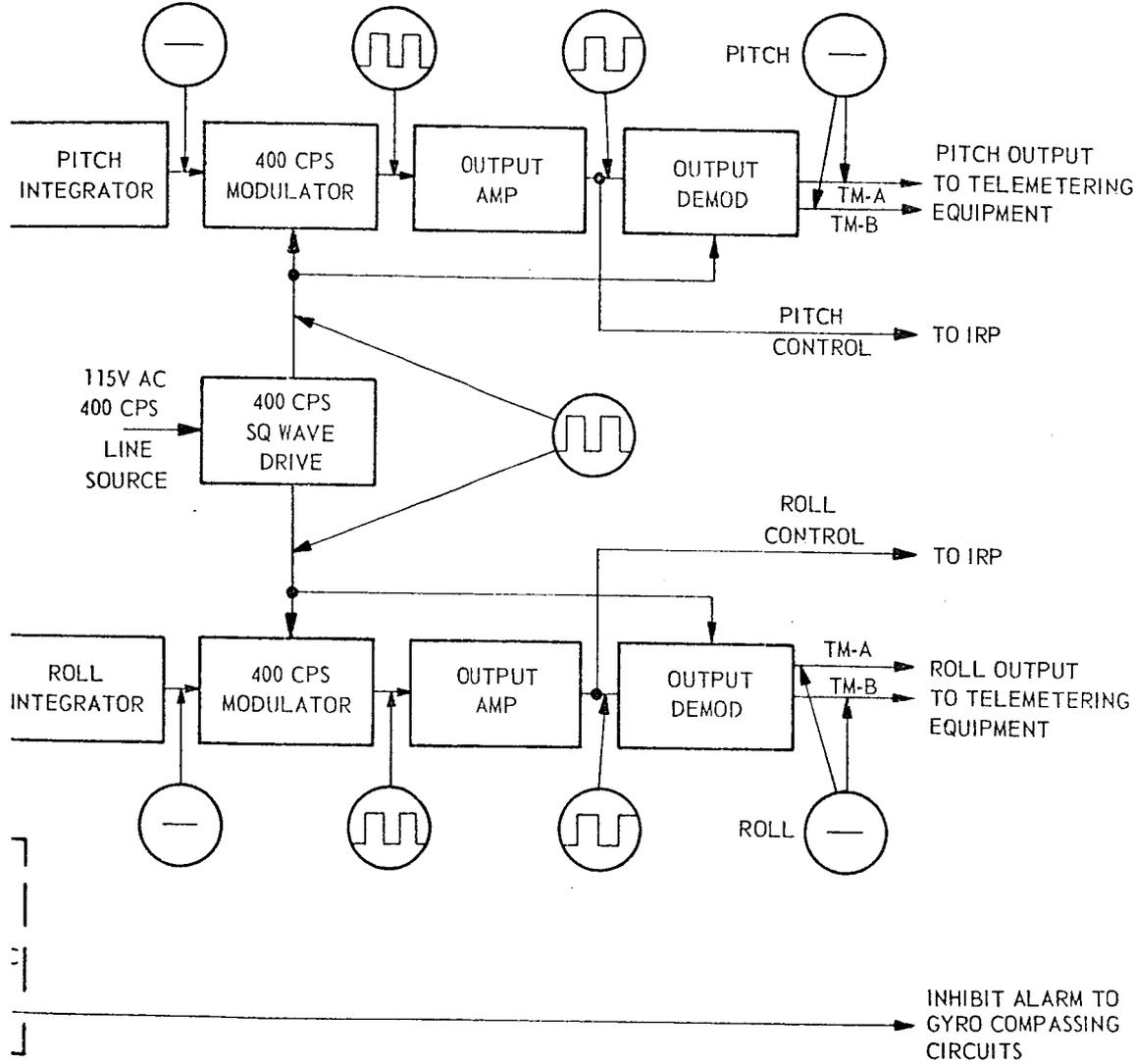


Fig. 5-14 Horizon Sensor System Block Diagram

The sun gate circuit effectively eliminates a sun pulse in the space portion of a scan cycle. The sun gate depends on the sun pulse exceeding the nominal level of a 320°K blackbody. When this occurs, the signal is clamped to the space level until the radiation input drops below the level required to operate the circuit. This reduces the sun pulse to small pulses at the leading and trailing edges. These pulses are easily eliminated in later processing.

The input signal from the preamplifier is isolated by transistor Q402 in the input limiter circuit (Fig. 5-16). The emitter output is clipped at 1.2 volts by a following diode and divider network and represents the earth-space contrast signal with the peak-to-peak amplitude limited to a fixed level corresponding to a 200°K earth.

The output of the input limiter is shorted to ground by sun gate transistor Q403 (Fig. 5-17) whenever a sun signal is present, thus removing the sun signal. The amplitude of the residual pulses on either side of the removed sun pulse are attenuated by the pi filter.

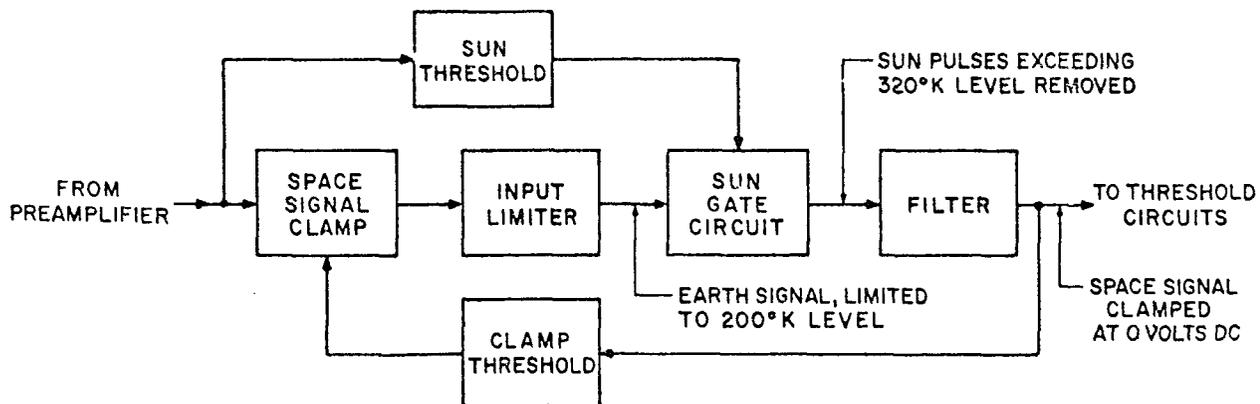


Fig. 5-15 Input Circuits - Block Diagram

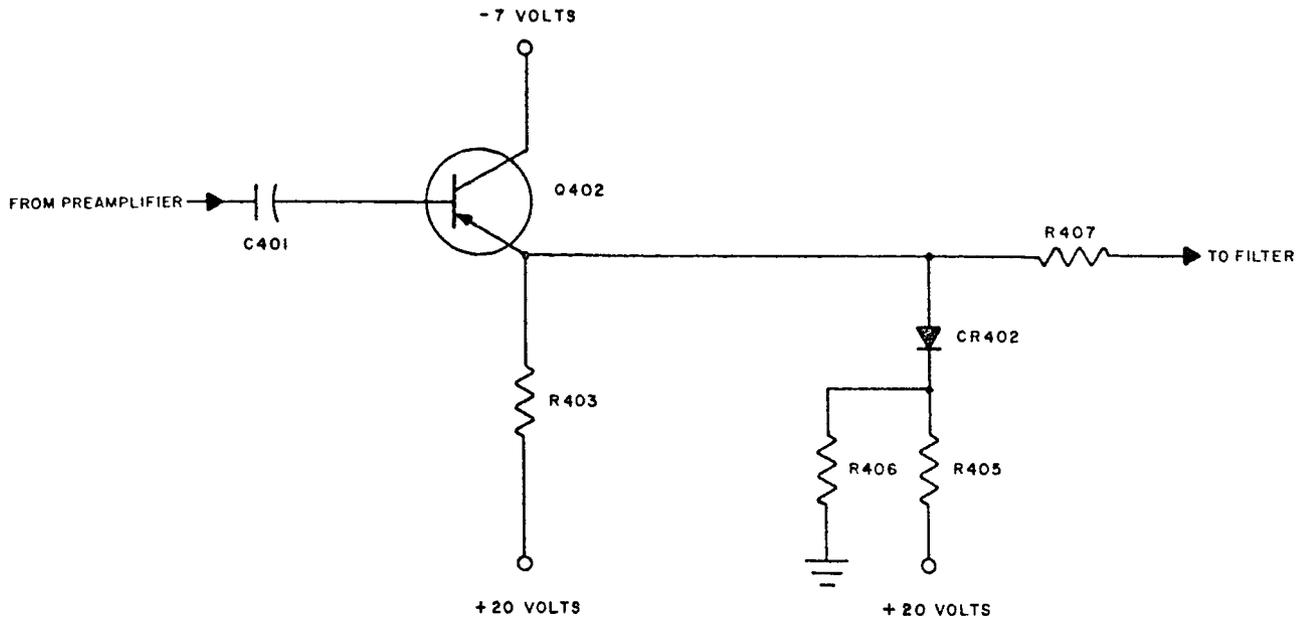


Fig. 5-16 Input Limiter - Simplified Diagram

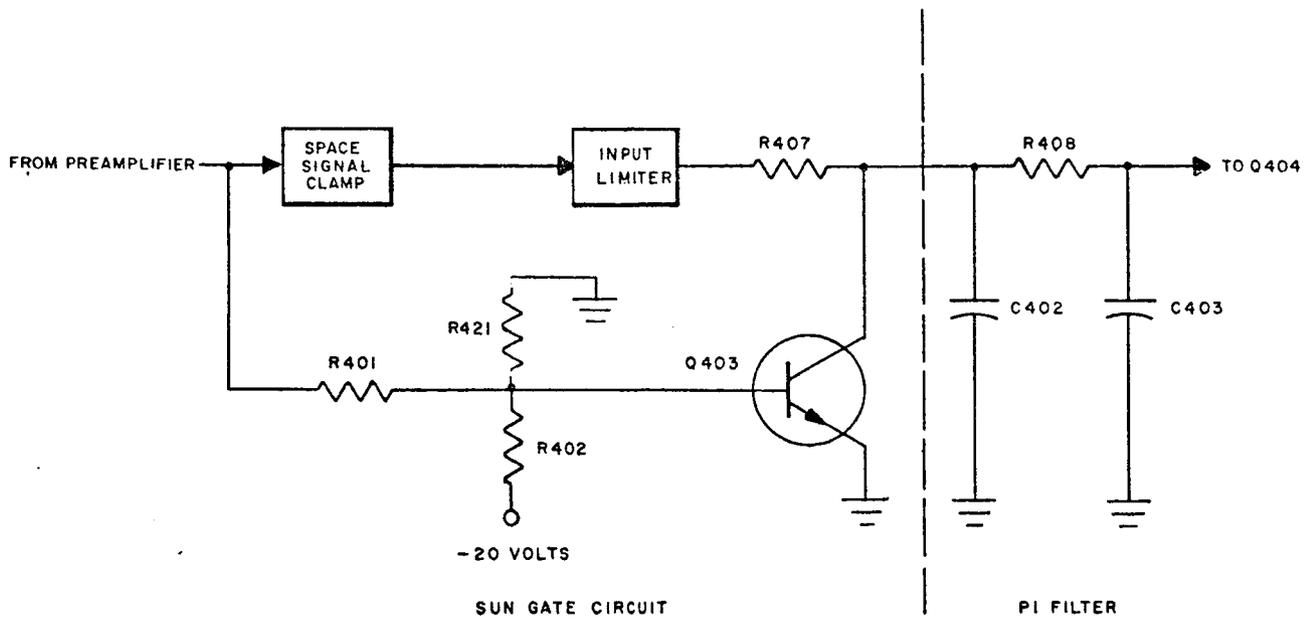


Fig. 5-17 Sun Gate Circuit and Pi Filter - Simplified Diagram

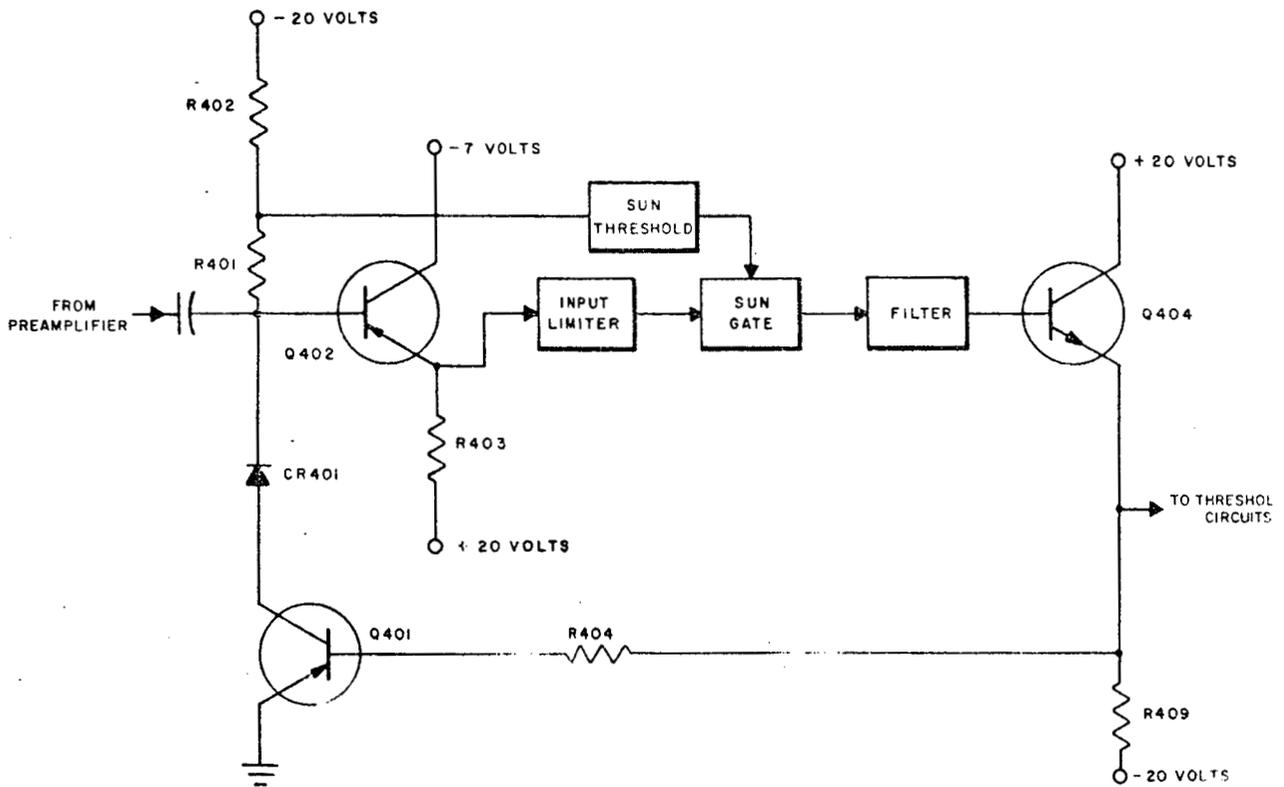


Fig. 5-18 Space Signal Ground Clamp – Simplified Diagram

The space signal ground clamp circuit (Fig. 5-18) provides a zero volt reference level for the space portion of the input signal. It employs a differential sensing feedback circuit from output to input (Q401). It is essential to maintain this zero volt space reference for proper operation of the sun gate circuit.

5.2.5.3. Threshold Circuits. The threshold circuits (Fig. 5-19) detect a fixed level (slice level) above the bottom of the ground clamped space signal. Since noise and residual sun pulses may exceed this level, a time discriminator removes short pulses and allows only the earth signal to reach the final limiter. The booster amplifier at the input provides sufficient signal for the remainder of the circuits to operate.

The output signal (earth-space contrast) from the input circuit is applied to the booster amplifier (Fig. 5-20) and amplified by a factor of 4. This signal is then converted into a sharp square wave by turning transistor Q503 on and off whenever the booster amplifier output crosses a 2-volt level. The 2 volts (when divided by the booster amplifier gain) corresponds to a 0.5 volt (170°K) slice level at the preamplifier output.

Figure 5-21 presents the noise and sun pulse rejection circuit, which utilizes a capacitor's charging time to eliminate sharp spikes. The earth signal leading edge becomes delayed 1.4 milliseconds by this capacitor, and the reference signal must also be delayed to retain the proper reference relationship for pitch signals.

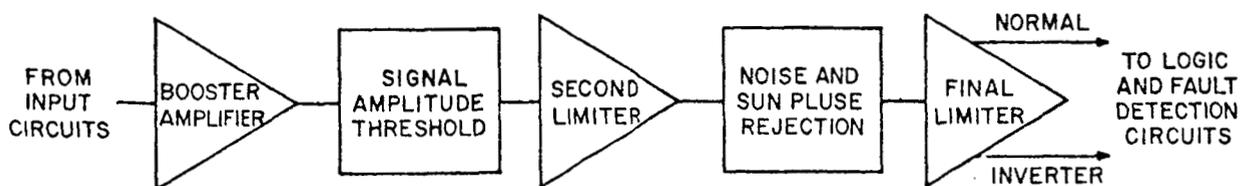


Fig. 5-19 Threshold Circuits - Block Diagram

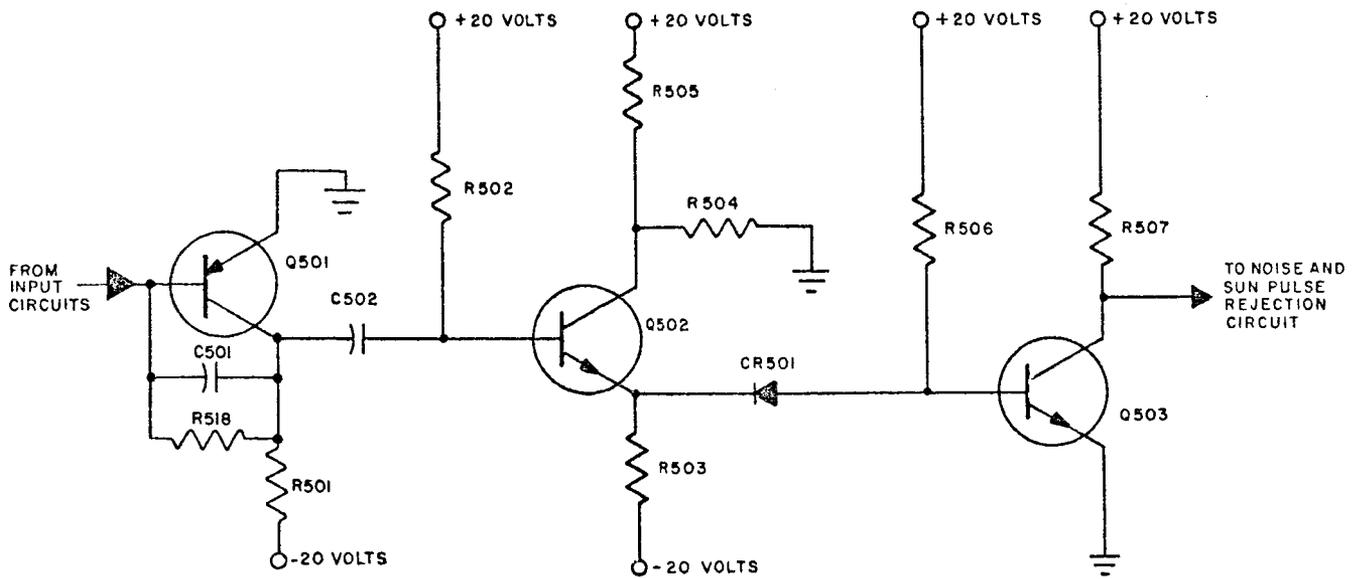


Fig. 5-20 Booster Amplifier and Signal Amplitude Threshold Circuits - Simplified Diagram

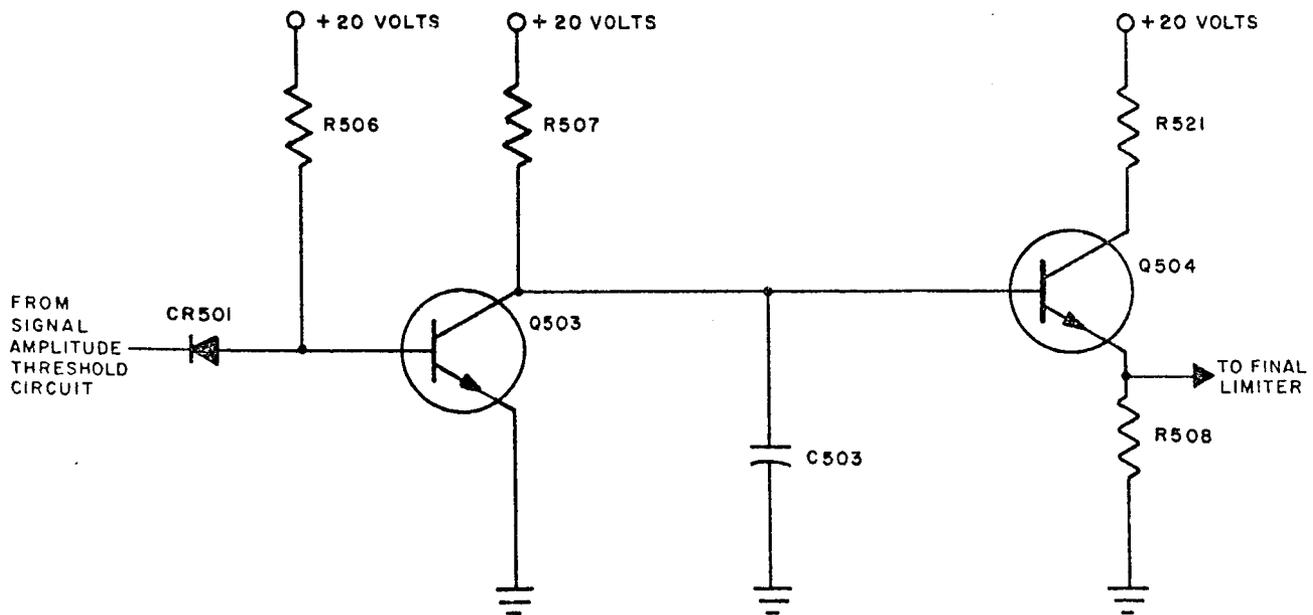


Fig. 5-21 Noise and Sun Pulse Rejection Circuit - Simplified Diagram

5.2.5.4 Reference Signal Time-Delay Circuit. The reference signal time-delay circuit was introduced as a result of the change in motor speed to 21 cps for the Model IIC and increased signal time delay.

To maintain a precise time reference, the reference pickup pulses are differentiated. (See Fig. 5-22.) The pulses come from a coil that is energized by a magnet rotating on the induction motor armature; pulse spacing dependent on motor speed variation results. The pulses drive a flip-flop circuit that triggers on the peak of each pulse and produces a sharp square-wave output. The square wave is delayed and then applied to a threshold circuit. Since the delayed square wave has an exponential rise characteristic, the desired delay time is selected by setting the threshold level. The threshold circuit controls a driven amplifier that supplies the necessary power to drive the switches in the logic circuits.

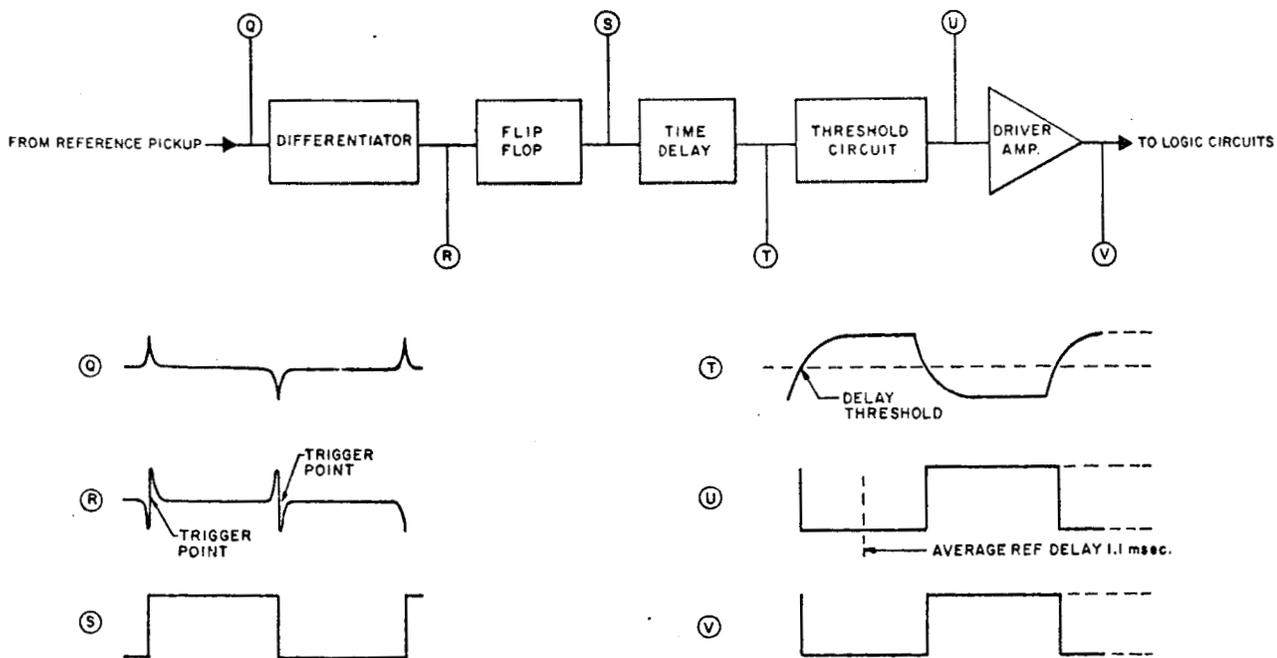


Fig. 5-22 Reference Signal Processing - Block Diagram

5.2.5.5 Spurious Signal Detector (Figs. 5-23 and 5-24). The Model IIC horizon sensor system uses an inhibit circuit that is sensitive to abnormal scan signals. The circuit detects limiter amplifier outputs, since all sun and cloud pulses have presumably been eliminated before this point, and uses a pulse-count-per-cycle concept to detect signal deficiencies. As long as one earth pulse per cycle is detected, the signal is acceptable. When zero, two, or more pulses per cycle are detected, an inhibit signal is produced, and the input to the integrator is grounded. (See Fig. 5-14.) An inhibit signal is also produced when the scan rate exceeds its normal range of from 12 to 28 cps.

As soon as the input fault disappears, the integrators are again connected normally. This circuit produces rapid reaction to the faulty signal, acting within three scan cycles on both the removal and restoration of the signals. Extremely narrow pulses that will not materially affect the horizon sensor output will not trigger this fault detection circuit.

5.2.5.6 Switching and Mixing Units. The normal and mirror image outputs of the final limiter circuit are applied to the switching unit. The phase difference denoting pitch error is detected by feeding the signals through syncroverters (a type of relay) and summing the result in a filter network. Syncroverters used in the pitch logic alternately sample normal and inverted signals. The syncroverters are driven by the reference signals that key the scan cycle to vehicle orientation. If there is no pitch error, the net output of the integrator capacitors is zero. Any pitch error results in a voltage difference in the output levels of the capacitors and, hence, an error signal.

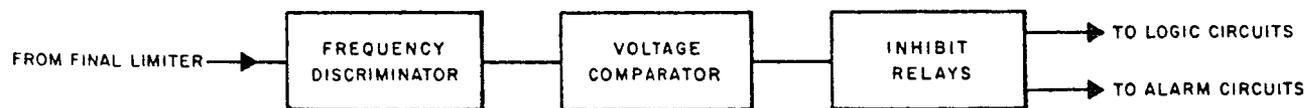


Fig. 5-23 Fault Detection-Block Diagram

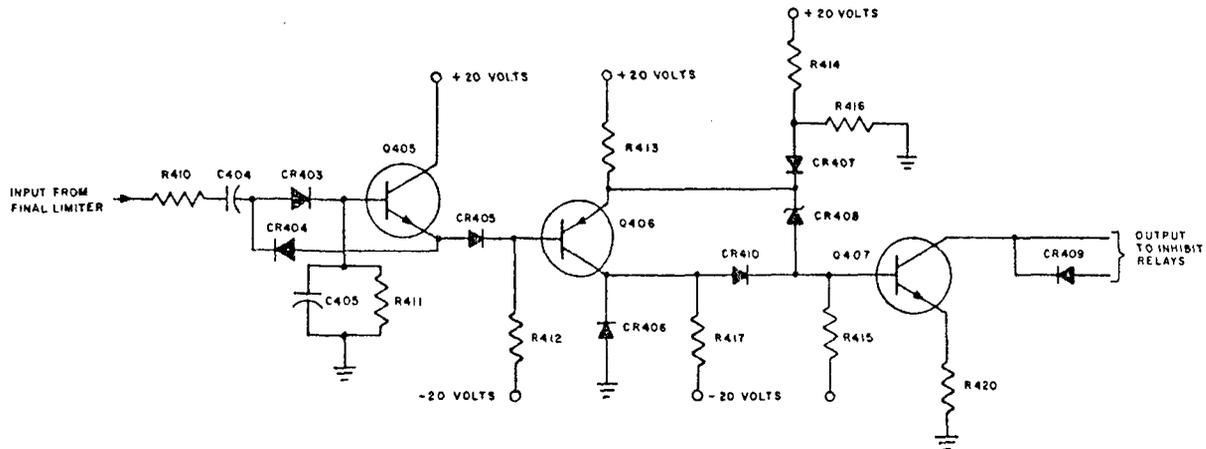


Fig. 5-24 Spurious Signal Detector – Simplified Diagram

When the vehicle experiences a change in pitch, the points at which reference signals are generated change with it; that is, with a nose-up (positive) attitude, reference signals would still be generated at the top and bottom of the scan, referenced to the vehicle's vertical axis. Since the reference signals are shifted to agree with the nose-up alignment of the vehicle, their relationship to the incoming signals is also changed.

To the pitch comparator, pulsed by the reference signals, it appears as though the sensors are seeing earth longer in the backward scan and less on the forward scan. The relationship between horizon points and vehicle orientation is changed, even though the actual scan pattern remains the same. This shift or phase difference results in differing voltage sums on the integrator capacitors; an error signal, proportional to pitch error, results.

In the roll comparator, the integrator capacitors and their associated resistors make up the filter network. In this circuit, the difference in pulse widths determines the charge on the capacitors. With any roll error, one sensor head has a longer dwell time on earth while the other decreases. Clockwise roll produces a positive difference voltage, and counterclockwise roll, a negative difference voltage.

5.2.5.7 Output Circuits. The d-c outputs of the pitch and roll comparators are fed into their respective output stages. The pitch and roll output stages are identical.

The output of the integrator network is applied to a 400-cycle full transistor chopper that sees symmetrical transistor pairs. This balances input impedance and leakage currents. The modulated 400-cycle output is filtered and applied to a three-stage amplifier with a push-pull output driver stage. The output is transformer-coupled to the vehicle control system, and the transformer has a second output that is used to convert the a-c signal back into d-c information for transmittal to the necessary telemetering equipment.

### 5.3 VELOCITY METER

The velocity meter (Fig. 5-25) is an instrument furnished by Bell Aerosystems Company to measure the velocity gain of the Agena during primary propulsion system (PPS) or secondary propulsion system (SPS) burn in order to give an engine cutoff signal.

The major parts of the velocity meter are as follows: accelerometer, electronics, and counter (Fig. 5-26). The accelerometer senses acceleration along its sensitive axis and produces an output proportional to the acceleration. (See Fig. 5-27.) The electronics convert this output to pulses at a rate proportional to acceleration. The time integral of this acceleration is the desired velocity and is equal to the total number of pulses. These pulses are divided by 4 upon entering the counter, before application to the counting register.

The counter is a subtract-one type of digital unit that energizes the cutoff relay two pulses after the number loaded into the counting register has been counted down to zero. The number for loading the register is obtained by



Fig. 5-25 Velocity Meter – Accelerometer and Accelerometer Electronics

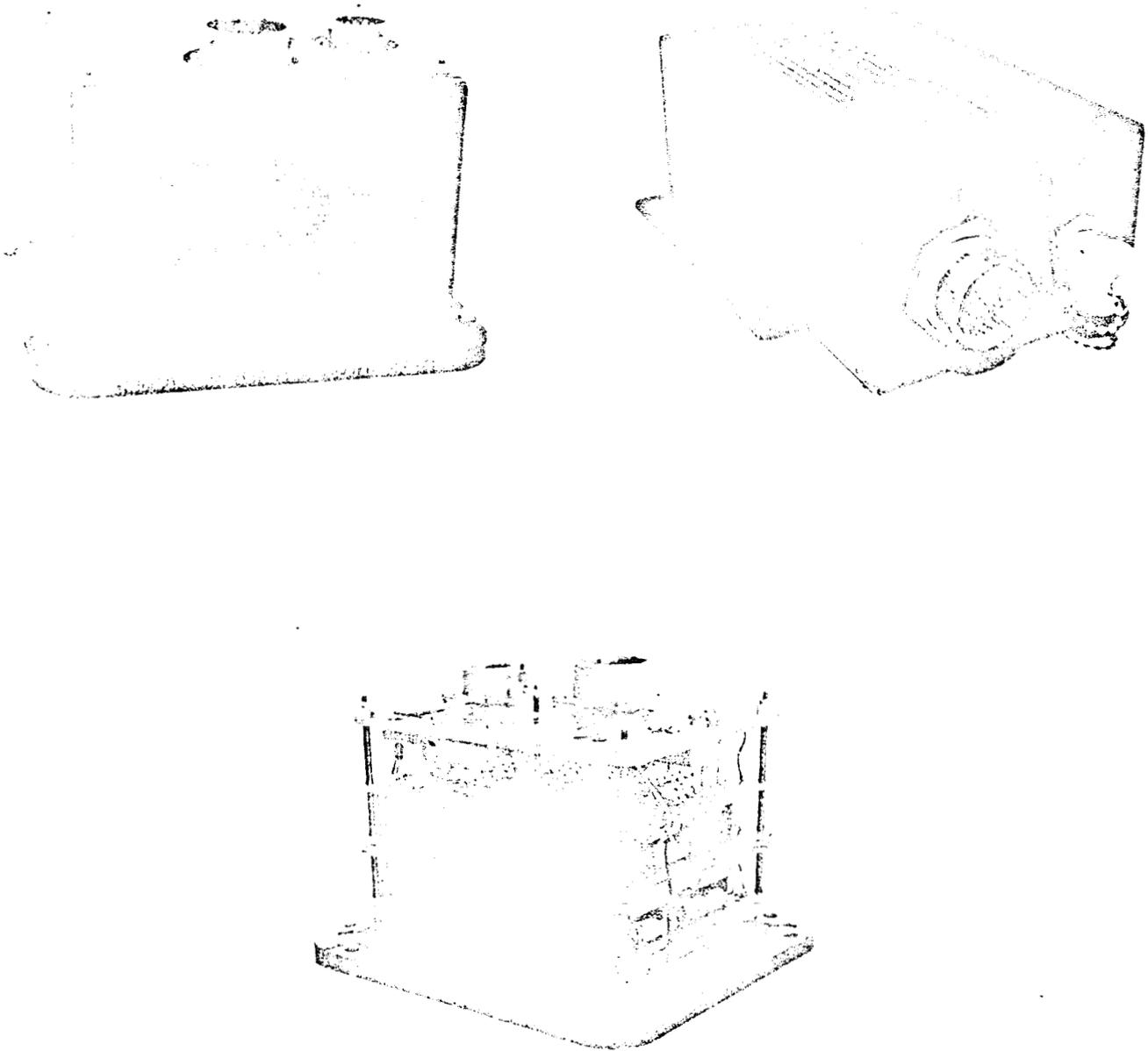


Fig. 5-26 Velocity Meter Counter

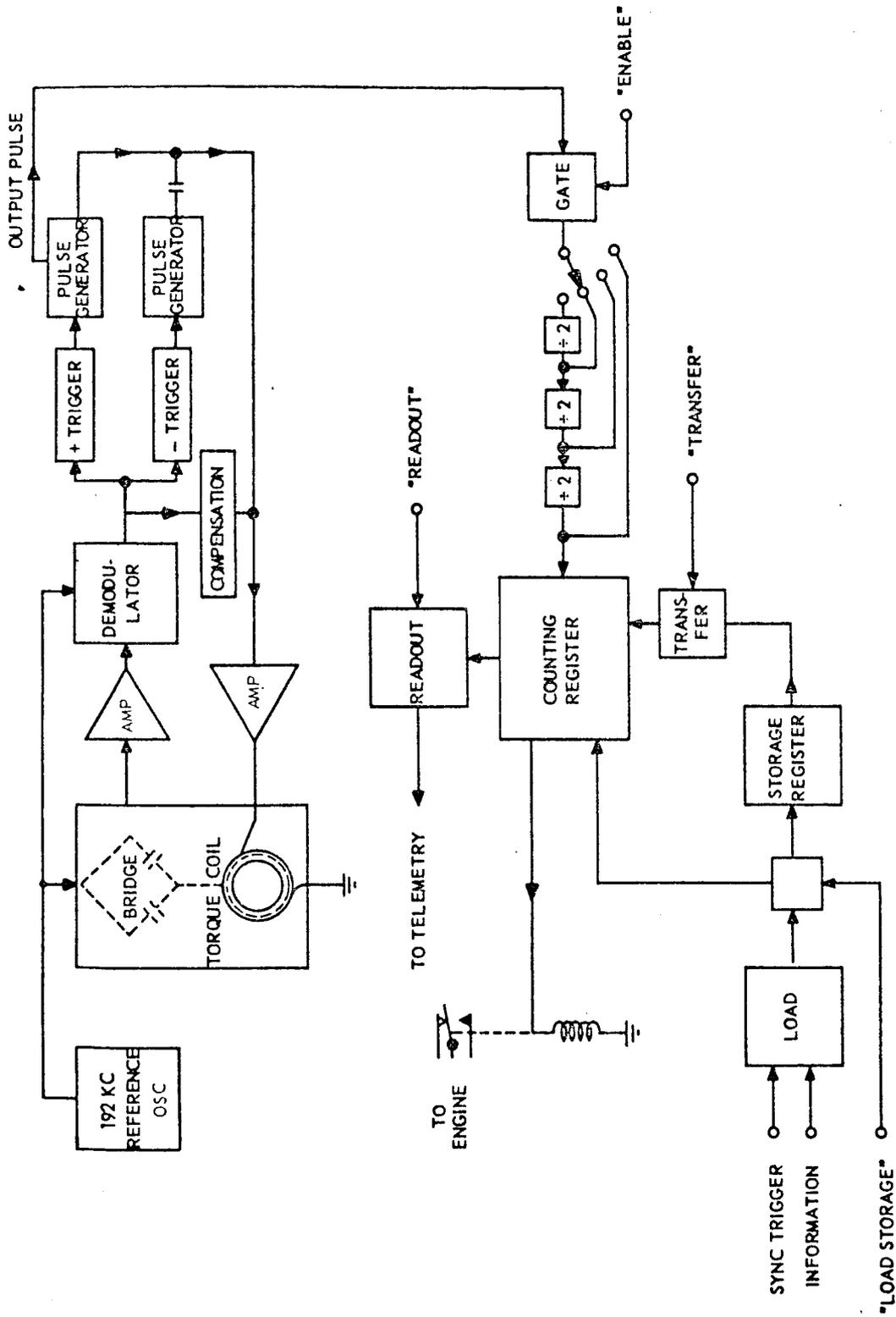


Fig. 5-27 Velocity Meter Block Diagram

converting the velocity-to-be-gained (VTBG) in feet per second into a 15-bit binary number as follows: Multiply the accelerometer scale factor of the individual velocity meter (approximately 0.13 feet per second per pulse) by the VTBG times 4 and then subtract 2. The resulting number is then converted directly to binary form. The number is loaded least significant bit first after putting in an index "one" bit. The number is also read out least significant bit first, but the index or "carry" bit is read out last.

The accelerometer utilizes a force-balance pendulum that is returned to null by pulses generated in a circuit that senses the displacement of the pendulum. The pulses are also transmitted to the counter.

The electronics unit provides signal processing and temperature control circuits for thermally stabilized elements.

The accelerometer is of the pendulous type that employs the force balance principle. It consists of a pendulum with two legs, supported by springs, and located in a field provided by two permanent magnets. The coil form, representing the main mass of the pendulum, is the center plate of a capacitive pickoff. This pickoff is used in a bridge network that produces a signal proportional to pendulum deflection. This signal is amplified, demodulated, and fed into a trigger circuit that develops a pulse rate proportional to its input signal. This signal is fed back through an amplifier to the torquer coil located in the magnetic field to balance the acceleration force action on the pendulum. These pulses to the accelerometer coil produce a d-c component inversely proportional to their spacing in time, which results in a situation such that the pulse frequency is proportional to the d-c current constraining the accelerometer. As a result, the pulse rate is proportional to acceleration, and the time integral of the pulse rate is proportional to velocity. Simply, the time integral of the pulse rate is equal to the total number of pulses.

To maintain accuracy, a constant temperature is a requirement. Two ovens controlled by proportional temperature controls are employed. One controls the accelerometer temperature, and the other controls the temperature of critical electronic components. The ovens control to  $\pm 1^\circ\text{C}$ .

There is a third temperature control used to maintain the temperature of the magnetic core used in the pulse generator at  $\pm 0.1^\circ\text{C}$ . This core enables the pulse generator to maintain a constant pulse amplitude that is the prime requirement for accuracy.

The capacitive pickoff, which consists of three plates, is excited by a 192 kc carrier. The nominal value of the capacitors in the arms of the bridge is 15 picofarads. Under zero acceleration, the bridge is balanced, and no output signal is present. Under acceleration, the pendulum is deflected, the bridge becomes unbalanced, and a 192 kc signal results. The pendulum constraint is chosen so that the maximum deflection is about 0.1 milliradian, which changes the capacitance in the order of 0.25 picofarads.

The oscillator is a modified Colpitts oscillator. Crystal control is employed by placing the crystal in the base lead of a transistor; thus the feedback to the emitter is only effective at the crystal frequency.

The reference amplifier uses overall feedback to produce a stable gain of 33. The push-pull output stage provides 100 volts peak-to-peak excitation to the demodulator reference and up to 5 volts peak-to-peak accelerometer excitation.

The preamplifier is a low-level shielded amplifier having a stable gain of 52, located adjacent to the accelerometer. The amplifier-demodulator consists of an amplifier similar to the reference amplifier and a double-ring demodulator, which is capable of supplying a filtered output of greater than  $\pm 14$  volts. The a-c rms to d-c gain is 9.1. The demodulator supplies the trigger circuits, rate generator, and negative spring input to the d-c amplifier.

The pulse generator is a trigger circuit that generates a signal when the demodulated input signal reaches 7 volts. The input signal drives a Schmitt trigger through an emitter follower. When the Schmitt trigger is switched on, a gated multivibrator produces a pulse which flips a flip-flop. The output transistors, in turn, change state, and an accurate pulse is generated and rectified.

The rate generator produces a current in the torquing coil that is proportional to the derivative of the pendulum motion. The negative spring produces a current in the torquing coil to offset the effect of the springs used to restrain the pendulum.

The d-c amplifier employs chopper stabilization to minimize d-c offset and drift. This amplifier sums the various signals fed into it.

#### 5.4 SEQUENCE TIMER

The sequence timer (Fig. 5-28) originates switching signals for guidance and control functions during ascent. The ascent sequence of events shows which functions these are.

The complete timer cycle is 10,000 seconds; however, events can occur only during the first 6000 seconds, after which the switches are reset in the remaining 4000 seconds. Normally, the timer is reset backwards to zero rather than run through the complete 10,000 seconds, since the Gemini sequence lasts only to the point at which the timer removes its own motor power (approximately 900 seconds).

The timer operates 72 switches in 24 groups of 2 to 4, at 24 possible event times that are ordinarily adjustable from 1 to 6000 seconds. Certain exceptional events that are common to all Agena vehicles are set to fractional values of 1 second, because of system requirements. The switch groups are

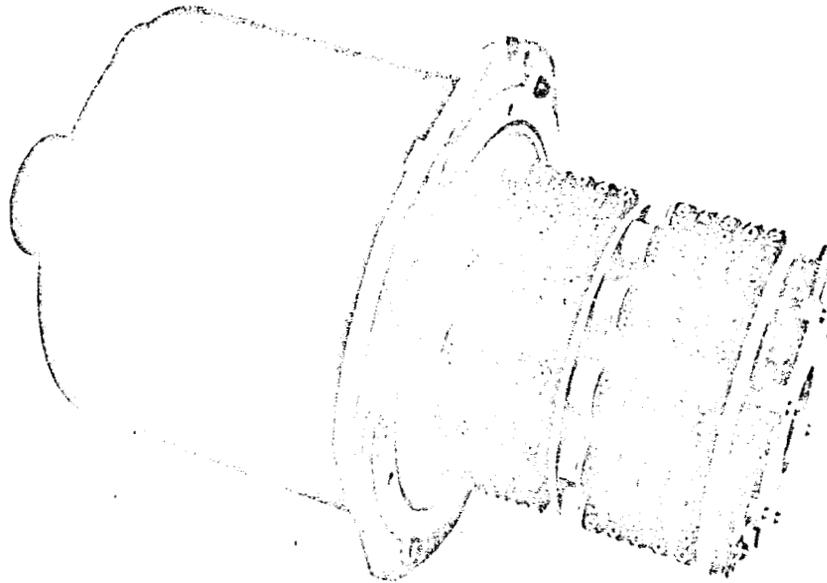


Fig. 5-28 Sequence Timer

actuated by 24, five-figure star-wheel revolution counters, each adjustable to 1 second within the range of event times. The counters are gang-driven at 1 revolution per second by a synchronous motor having an integral gear reduction. Also integral with the motor, is a d-c clutch/brake device that declutches the motor and brakes the load when it is energized.

#### 5.4.1 Motor

The motor accelerates its output shaft to full speed within 0.1 second, while driving a 25 ounce-inch torque load and maintaining full speed at no more than 0.1-percent slip. It is connected to a 115-volt, 400-cps, 3-phase delta power supply. Twenty-eight volt d-c power is supplied to the

clutch/brake, which is capable of bringing the load up to full speed with running motor, within 0.1 second after being de-energized. The clutch-brake is also capable of stopping a running load within 0.1 second after being energized.

#### 5.4.2 Counters

The 24 counters are driven in pairs by 12 shafts that are geared to the motor to run at 1 revolution per second. Unit wheels of all counters are pinned to the 12 drive shafts so that resetting can only be accomplished by whole revolutions of the unit wheels (the equivalent of 1-second increments). The wheels are fitted with cams containing roller followers. The followers rotate stub shafts containing pins, which operate groups of one to five switches. Desired event times are manually set into each of the 24 counters, while their unit wheels are unpinned from their drive shafts.

#### 5.4.3 Switches

Switches are of the two circuit type, in which one circuit is normally open and one is normally closed. During manufacture, the switch assembly is closely controlled to prevent cross-connection of circuits. When tested with a 24-ampere, 28-volt d-c load for 5 seconds, the switches demonstrate no failures below 1200 operating cycles; current capacity rating with an inductive load is 5 amperes at 115 volts 400 cycles, or 3 amperes at 28 volts dc. Contact resistance for each circuit is less than 0.05 ohm. Breakdown voltage under a pressure of 0.5 atmosphere exceeds 250 volts ac rms.

### 5.5 FLIGHT COMMAND LOGIC PACKAGE

The flight command logic package (Fig. 5-29) performs logic matrix functions on command from the Type IV controller on all guidance and control equipment, on the PPS and SPS, and on the target docking adapter equipment. Primarily, the logic matrices are formed by electromechanical (relay) action, but some electronics are used for timing and decision circuitry.

The logic circuitry is contained in seven modules (Fig. 5-30) mounted to the plate of the logic package. An RFI shielding gasket forms part of the assembly. The module locations are keyed to preclude mounting the wrong module in a given location on the plate.

As presently designed, the logic package accepts 55 commands, and from these commands forms over 200 separate events. A typical example of the logic involved in the guidance and control circuitry is illustrated in Fig. 5-31. It will be noted that eight commands are involved, but over 40 separate events occur.

A typical example of the propulsion system control circuitry is illustrated by Fig. 5-32. Here, six commands are involved, but over 30 events occur.

Another typical example is shown in Fig. 5-33. Although only three commands are involved, over 20 events are initiated to control various circuits in the Agena status display panel on the target docking adapter.

The present design of the flight command logic package will accommodate a command capability expansion of approximately 10 percent.

## 5.6 GUIDANCE JUNCTION BOX

The guidance junction box (Fig. 5-34) performs the switching functions that control the IRP torquing rate scale factors from the horizon sensors and the gyro decoupling networks. It also controls power to the IRP, horizon sensor, and velocity meter systems. In addition, the guidance junction box routes other signals such as interrogate, transfer storage register, and so forth to the velocity meter system. All switching is accomplished by relay action.

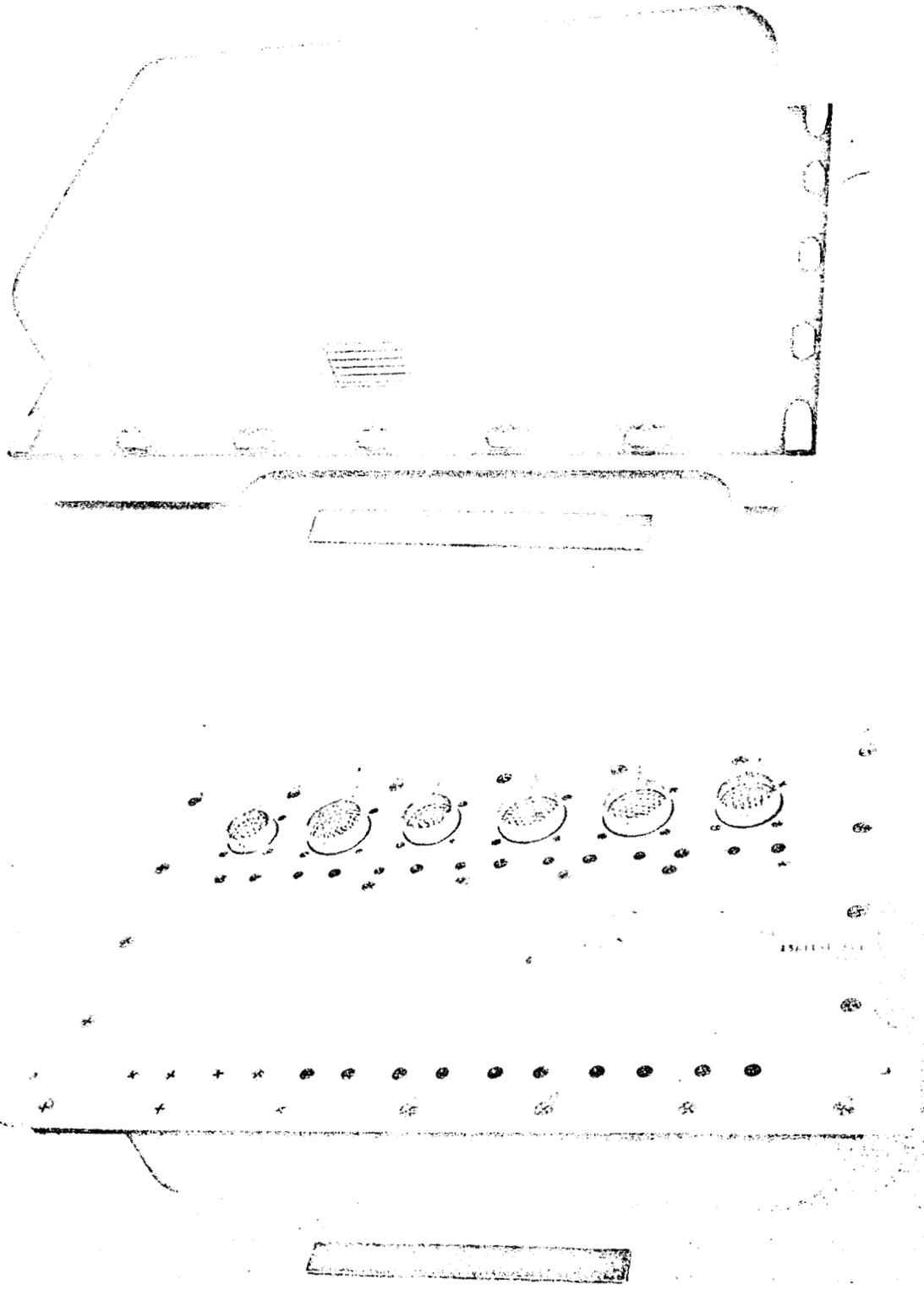


Fig. 5-29 Flight Command Logic Package

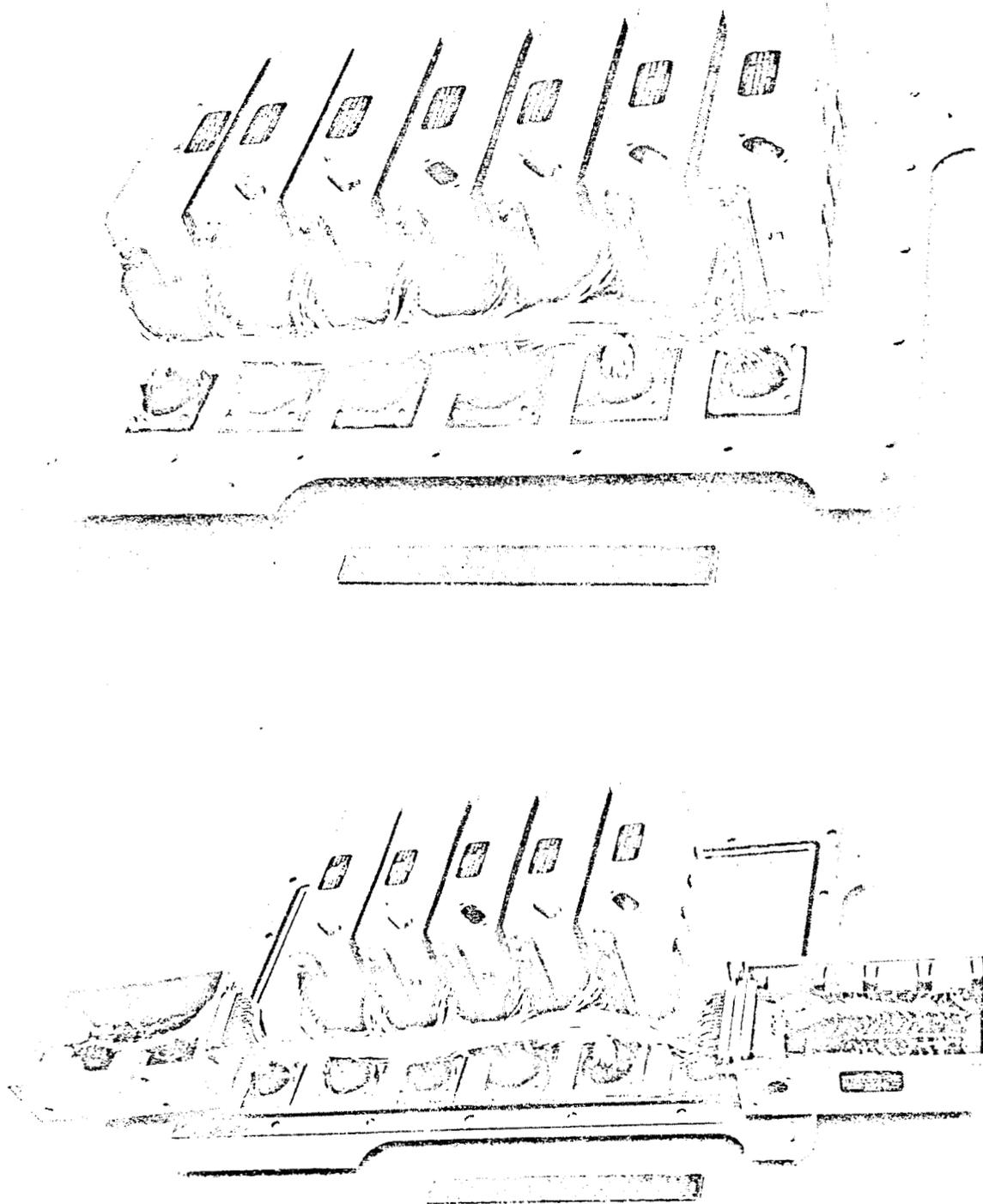


Fig. 5-30 Internal Construction, Flight Command Logic Package

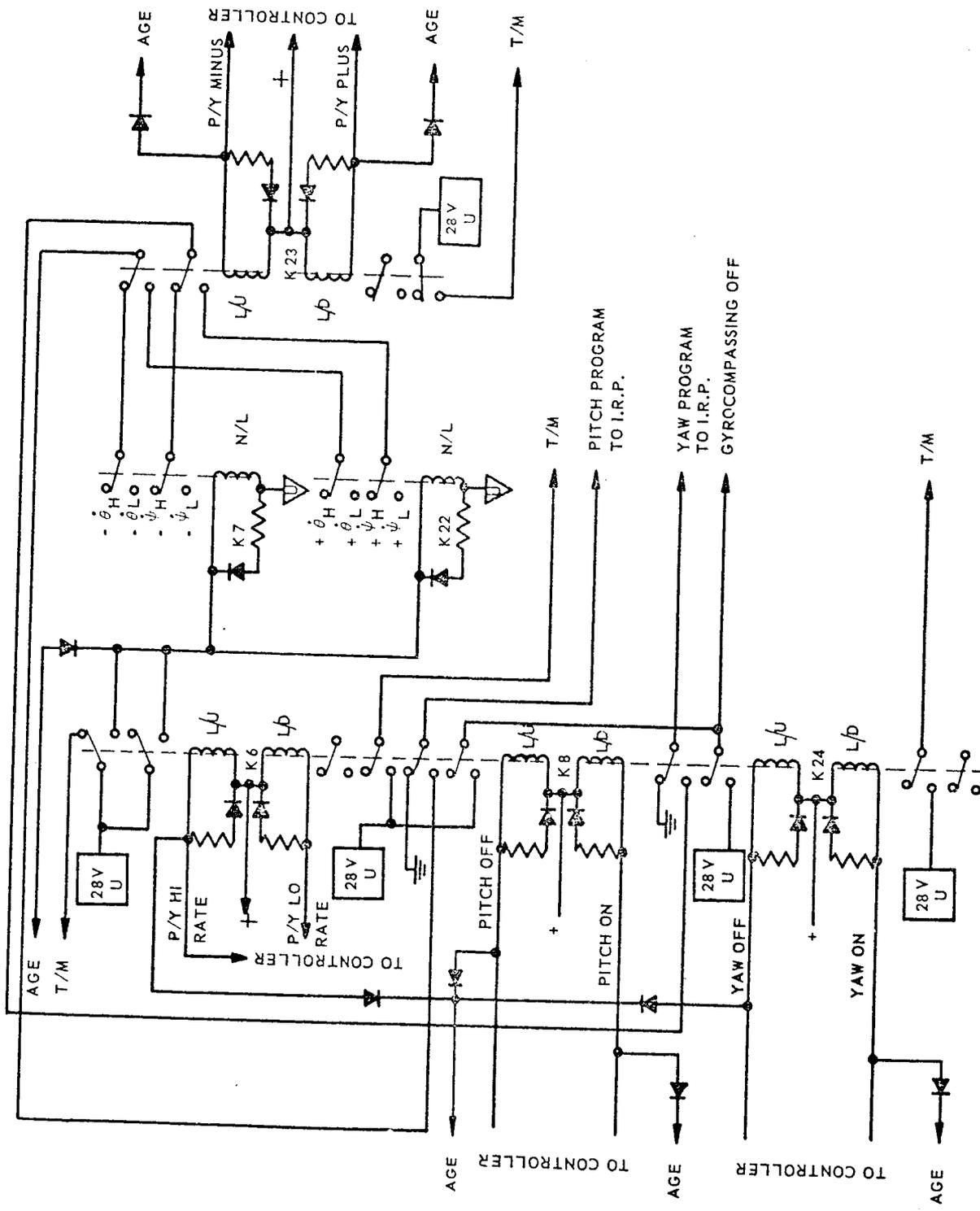


Fig. 5-31 Typical Logic Circuit, Flight Command Logic Package

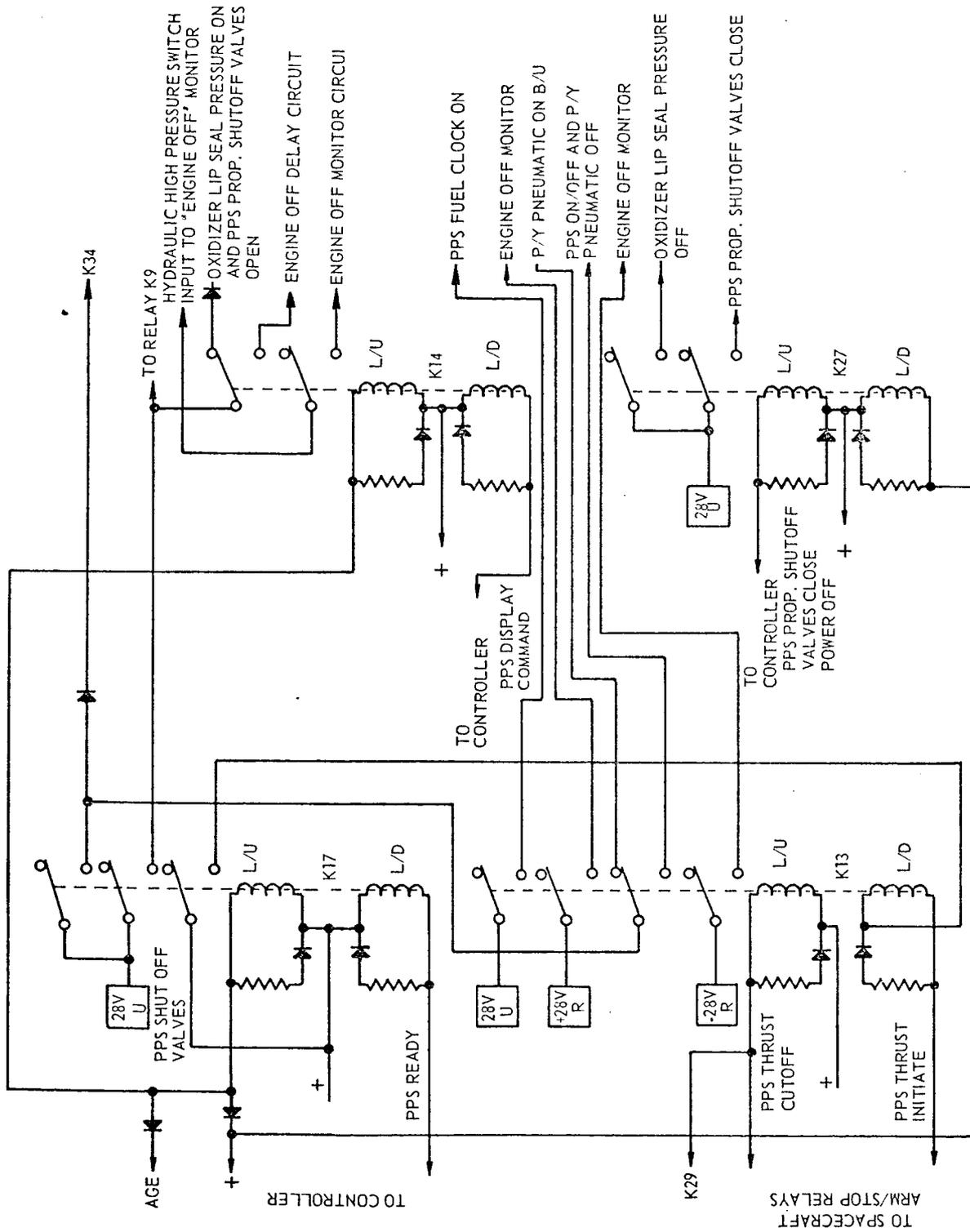


Fig. 5-32 Primary Propulsion System Control

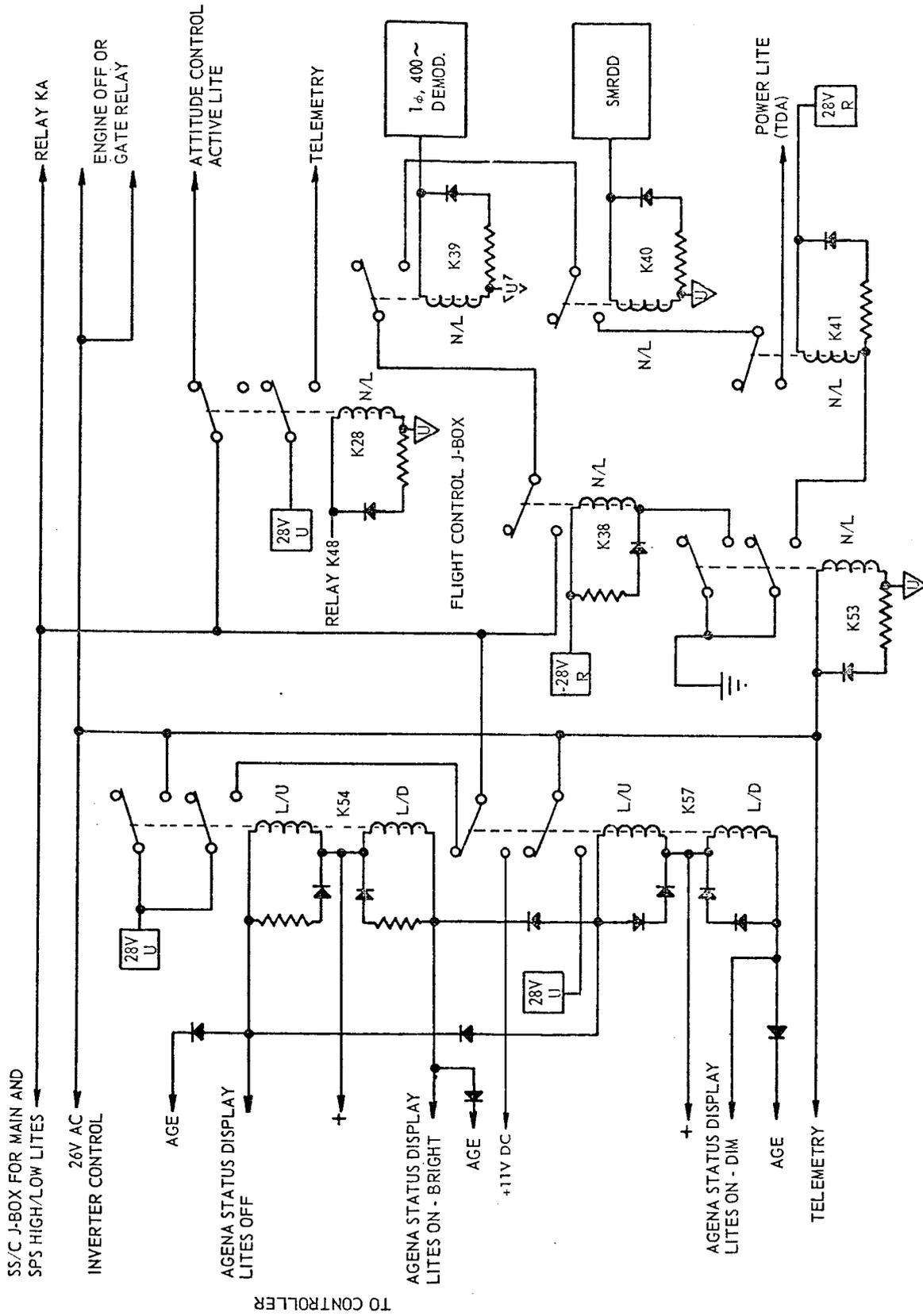


Fig. 5-33 Agena Status Display Lights and Relay Bus Control

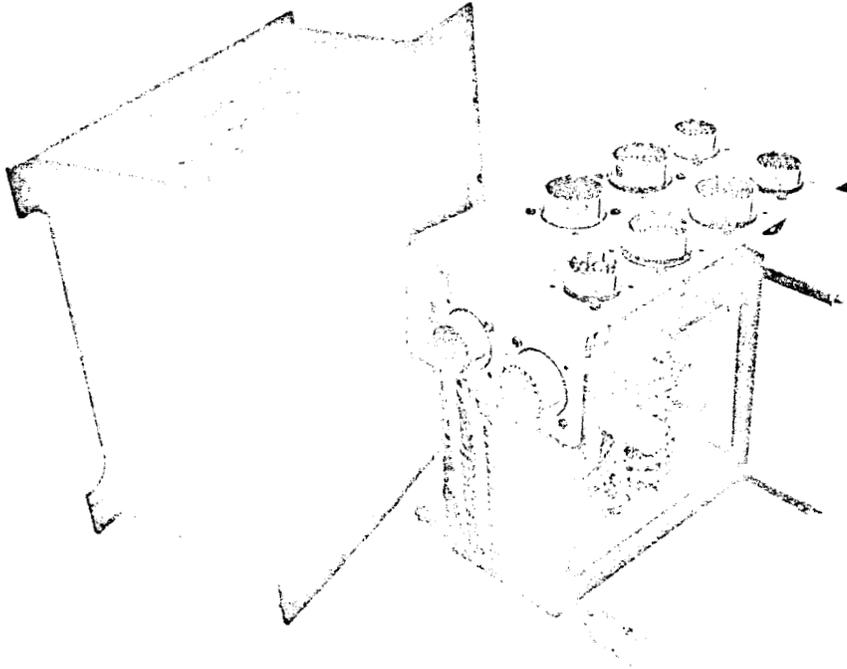


Fig. 5-34 Guidance Junction Box

In the junction box, gyrocompassing is provided for nose aft, forward, left, and right configuration. The regulated d-c voltage monitors and the line drivers for reading out the velocity meter counter counting register to the PCM telemeter registers are accommodated in the junction box by a new printed circuit board. Conversion of 400 cps output signals from the IRP to d-c voltage to the telemeter is accomplished by a signal conditioner assembly.

#### 5.7 FLIGHT CONTROLS JUNCTION BOX WITH ASSOCIATED GUIDANCE PATCH PANEL

The flight controls junction box (Fig. 5-35) contains switching functions for the sequence timer, pneumatic system gains and deadband controls, hydraulic system gain controls, and various power controls for the flight control

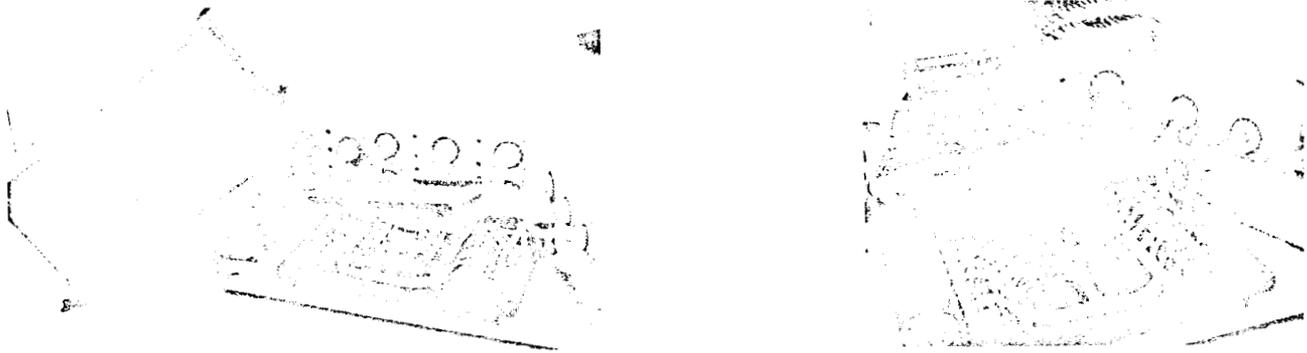


Fig. 5-35 Flight Controls Junction Box

electronics and sequence timers. The junction box also contains two transformers for torquing rate programs, switching networks, and integrating capacitors for program convenience. The sockets for the patch panel plugs are mounted within the flight controls junction box.

The guidance patch panel is mounted in the flight controls junction box. It provides interconnection, switching, isolation, and voltage diversion from point to point within the flight control junction box, thereby providing the listed functions for other components in the flight controls system. The patch panel contains the resistor voltage divider networks used to develop the program torquing rate signals applied to the IRP. These voltage dividers are powered by transformers in the flight controls junction box.

## 5.8 FLIGHT CONTROL SYSTEM

The flight control system consists of three basic subassemblies:

- a. Flight control electronics
- b. Hydraulic system including the hydraulic power package and two hydraulic actuators (pitch and yaw)
- c. Pneumatic system including the pneumatic regulator and two thrust valve clusters.

The flight control package processes and converts the output signals from the inertial reference package into quantities compatible with the requirements of the pneumatic and hydraulic control elements.

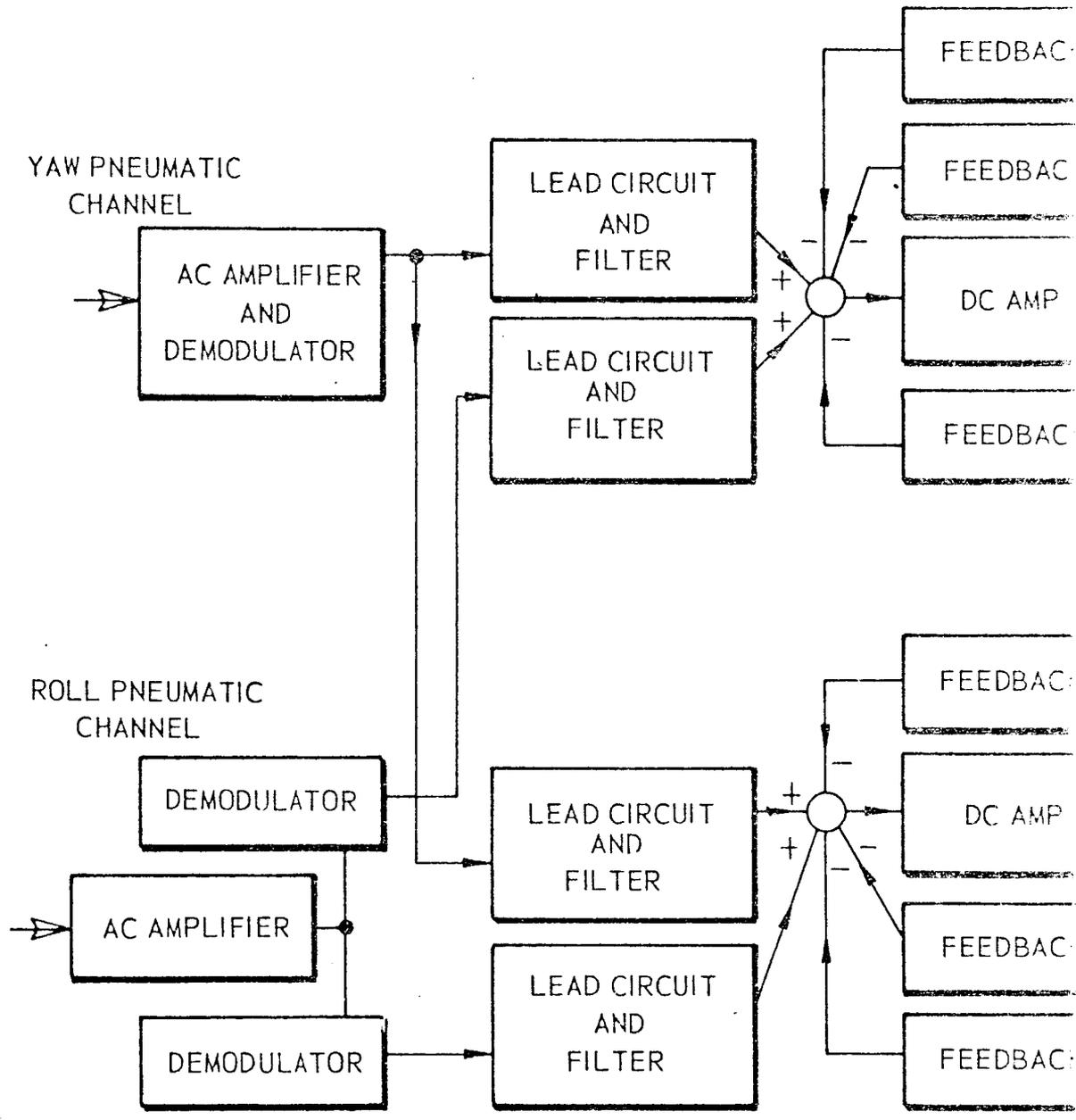
Figure 5-36 is a functional block diagram illustrating the mechanization that provides signal flow through the control channels. Attitude-input error signals, which consist of yaw, pitch, and roll commands in the form of 400-cps suppressed-carrier-signal voltages, are received by these channels.

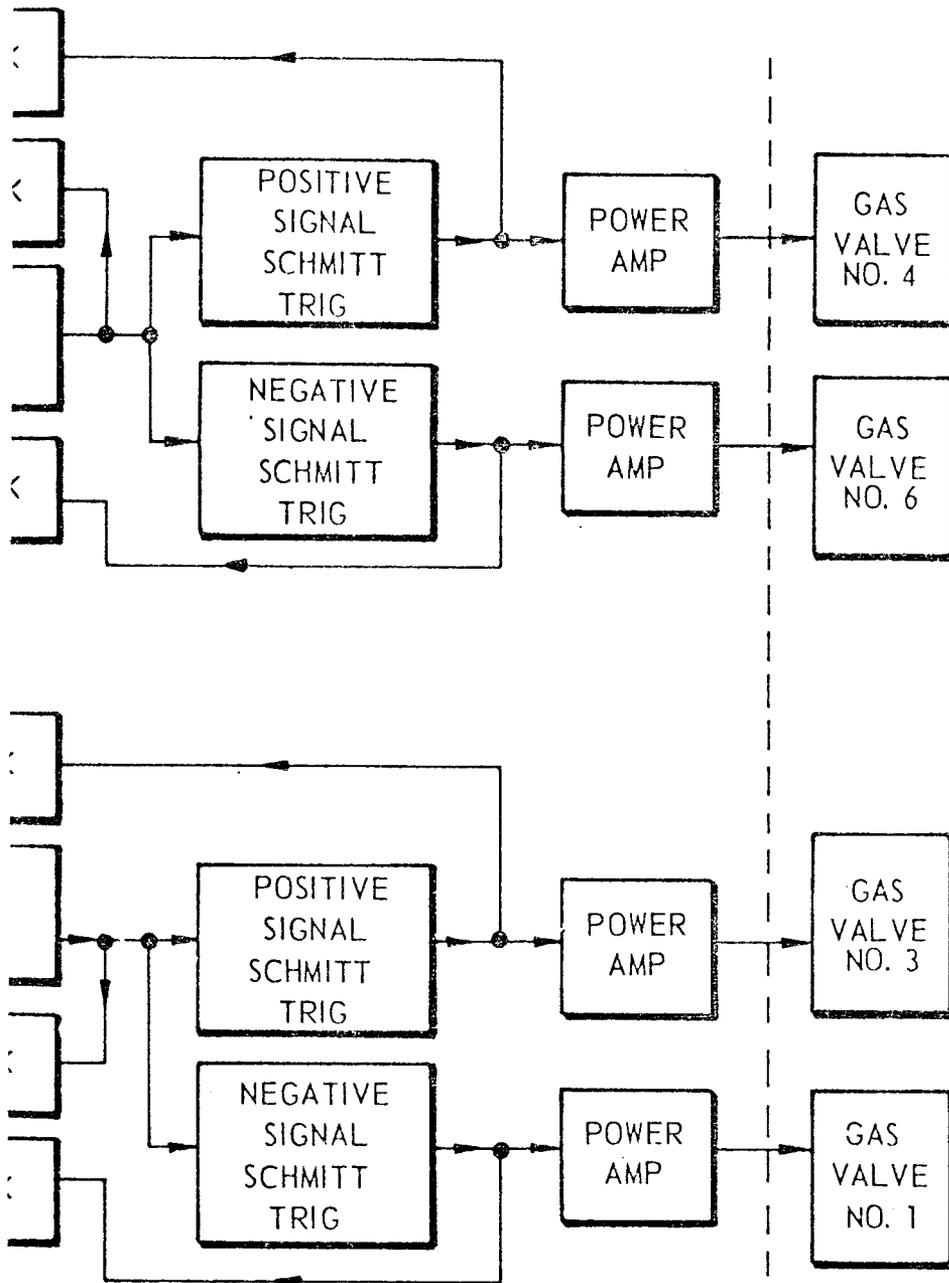
### 5.8.1 Flight Control Electronics

The five electronic control channels consist of two hydraulic engine and three pneumatic control channels. (See Figs. 5-37 and 5-38.)

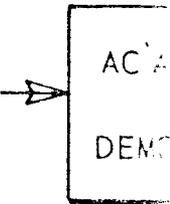
Pneumatic attitude control is first activated during the ascent phase at Atlas-Agena separation; separation switches on the Agena aft rack remove grounds from points in the pitch, roll, and yaw pneumatic channels to allow signal flow to the attitude control valves. During primary propulsion system (PPS) operation, pitch and yaw control are transferred to the hydraulic channels. Roll control always remains in the roll pneumatic channel.

The green ATT light in the status display panel indicates that the attitude control system is operating.





PITCH PNEU  
CHANNEL



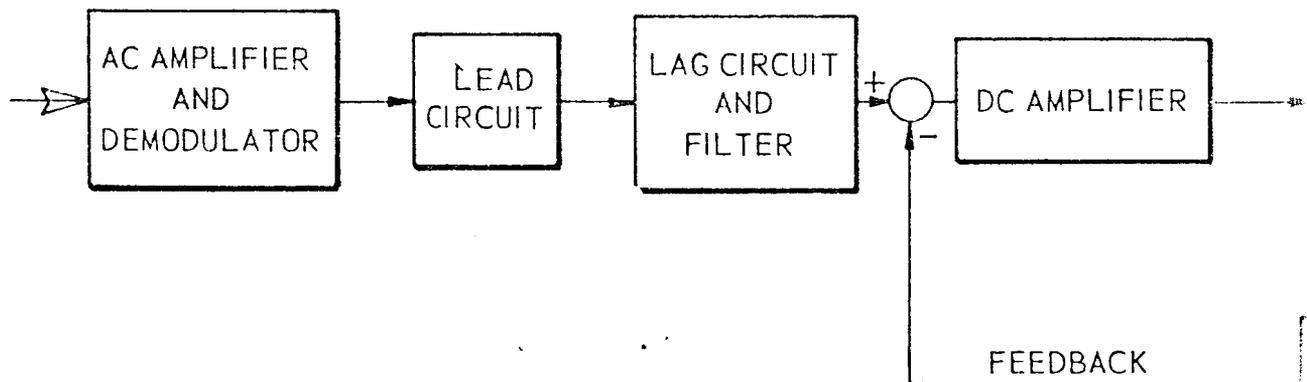
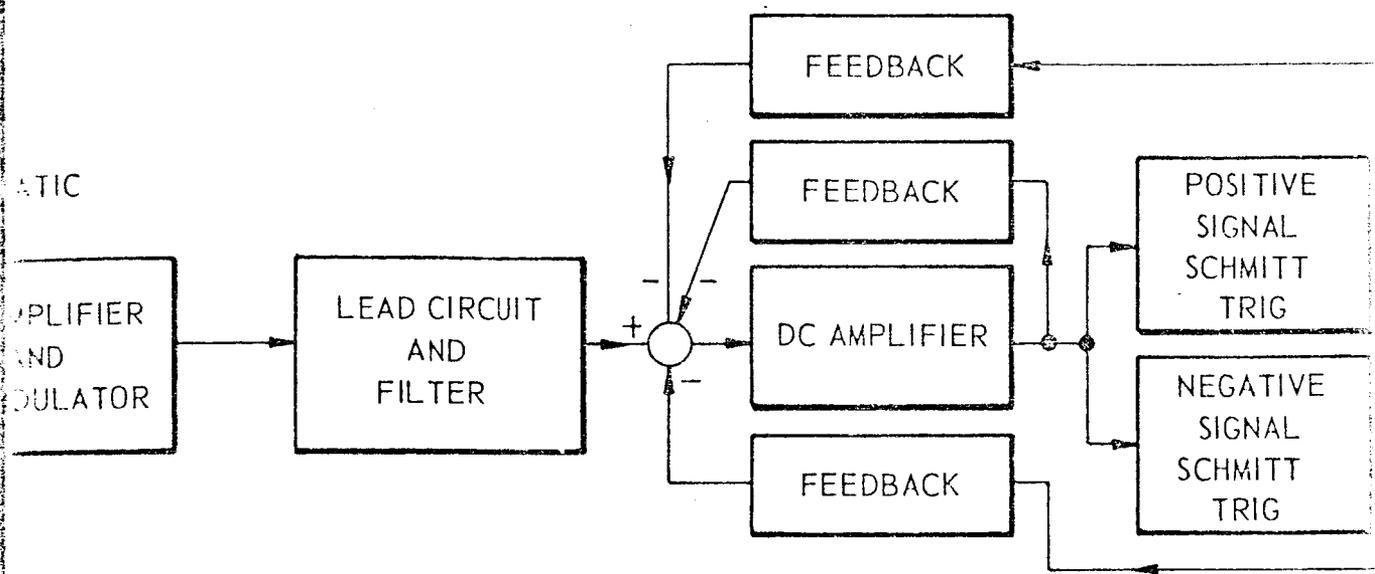
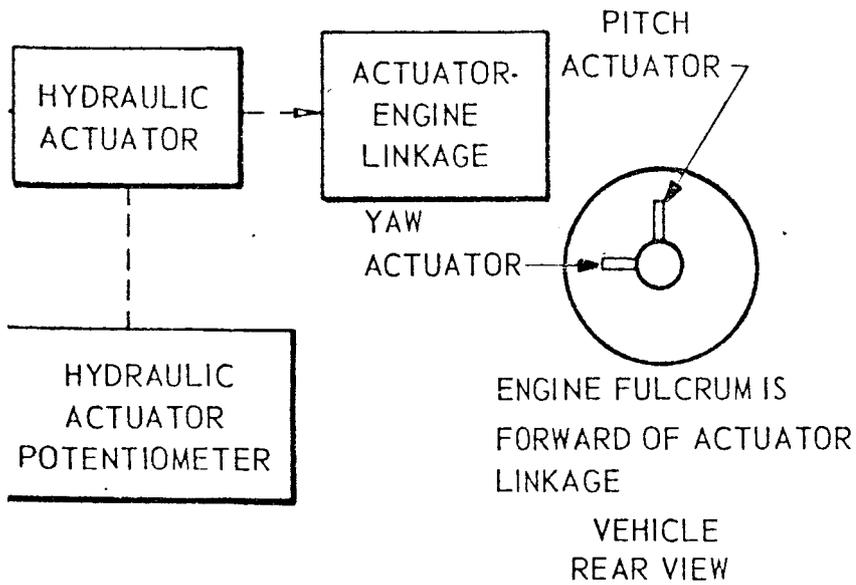
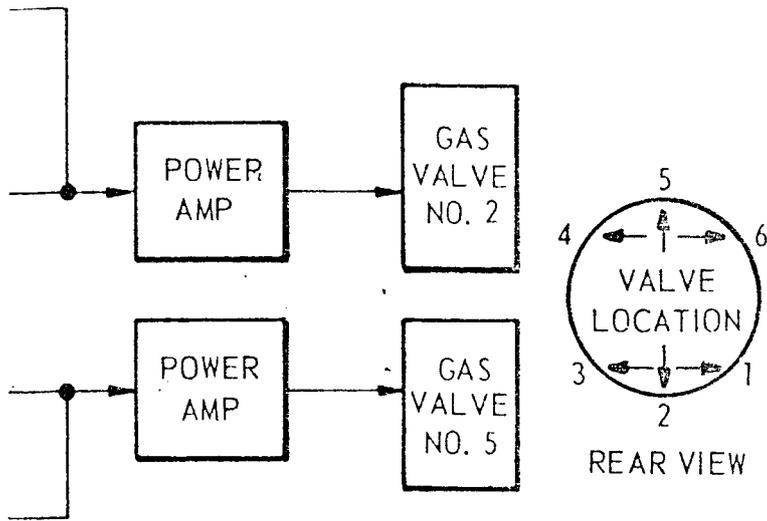


Fig. 5-36



Functional Block Diagram of Flight Control Package

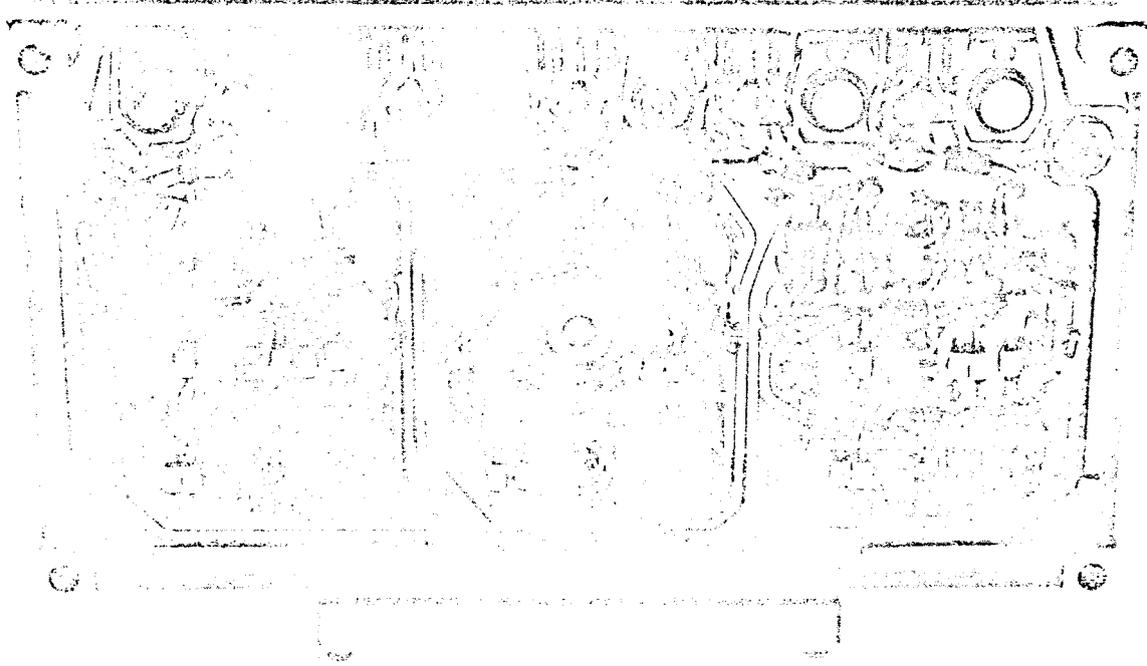


Fig. 5-37 Circuit Board, Flight Control Electronics

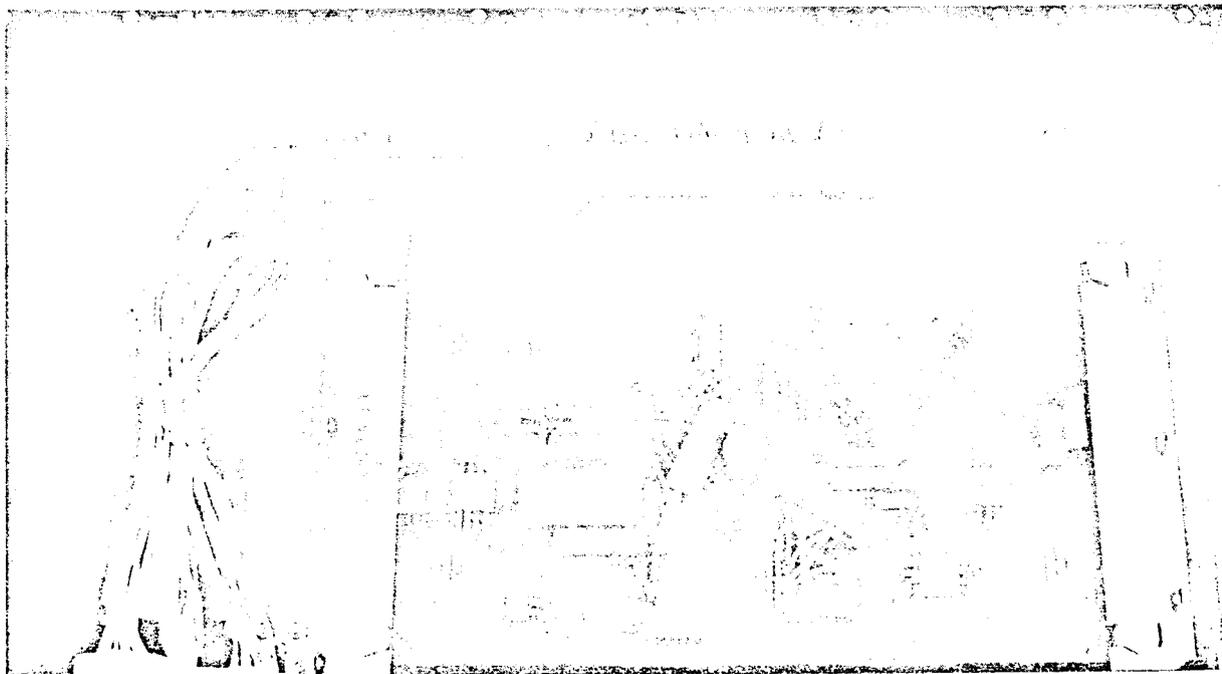


Fig. 5-38 Flight Control Electronics Package

5.8.1.1 Hydraulic Control Channels. Each of the two hydraulic channels contain an a-c amplifier, dual bridge, keyed diode, phase-sensitive demodulator, passive RC lead-lag circuit, and a chopper-stabilized d-c amplifier as shown in Appendix B (Fig. B-3).

The input to the hydraulic channel is a 400-cps amplitude-modulated suppressed-carrier error signal that is converted to d-c error currents that drive the hydraulic actuator assemblies. The hydraulic-channel-input error signals are amplified and demodulated. They are then differentiated in the lead circuit to yield rate information that is summed with the attitude signal at the d-c amplifier. Prior to summing, the attitude signal is shaped in the lag circuit to meet the stability criteria. The signal is inverted in the d-c amplifier to drive the hydraulic valve assembly. Feedback is employed from the hydraulic position potentiometer to the d-c amplifier, closing the hydraulic servo loop.

5.8.1.2 Pneumatic Control Channels. The three pneumatic control channels are used to control the gain and deadband operations of six control valves. These channels are electronically identical except that the a-c gain differs and there is cross-coupling between the roll and yaw channels as shown in Appendix B (Fig. B-4). The attitude errors are processed in a manner similar to the processing of errors in the hydraulic channels. The lead time constant required for the ascent mode of operation is made to differ from that of the orbit mode by shunting the rate capacitors to ground through a resistor.

Rate and attitude signals are summed at the input to the operational amplifier; the resulting signal is inverted by the amplifier while it is functioning as an integrator. Feedback loops around the operational amplifier, consisting of a diode bridge in series with the feedback resistor, are employed to produce the required orbit deadbands. The output of the operational amplifier is applied to a pair of polarity-sensing Schmitt triggers that drive the power amplifiers of the attitude control valves.

The attitude control valves provide thrust in response to width and frequency modulated electronic pulses. (See Fig. 5-39.)

5.8.1.3 A-C Amplifier and Demodulator. The a-c amplifier, which is used in the hydraulic and pneumatic channels, is a direct-coupled, three-stage amplifier. Bias stability is provided by d-c feedback from the emitter of the last stage into the base of the first stage. A-C feedback is achieved from the winding of the transformer to improve the gain and phase characteristics of the amplifier.

The output from the a-c amplifier is applied to a dual-bridge, keyed-diode, phase-sensitive demodulator such that a signal in phase with the reference voltage yields a negative output and a signal out of phase with the reference yields a positive output.

5.8.1.4 Lead-Lag Circuit. The output of the demodulator is differentiated in the lead-lag circuit to yield rate information that, with the attitude signal, is shaped in the lag circuit to meet the stability criteria. The effect of the lead-lag network on the attitude signal provides sufficient phase lead to achieve the necessary damping characteristics for the system.

5.8.1.5 Operational Amplifier. The basic design of the hydraulic channel operational amplifier is similar to that of the pneumatic channel operational amplifier. Operation of each amplifier is as follows:

- a. Pneumatic Channel Operation. Feedback from the output of the Schmitt trigger is applied to the input of the amplifier. The polarity of this feedback current is opposite that of the input signal. When the trigger is turned on, the step current is integrated and converted to a ramp voltage for the ascent mode of operation. The slopes of this ramp voltage are opposite to that produced by the error signal current; the result is the resetting of the Schmitt trigger. The change in ramp voltage from the on to off condition of the trigger is approximately

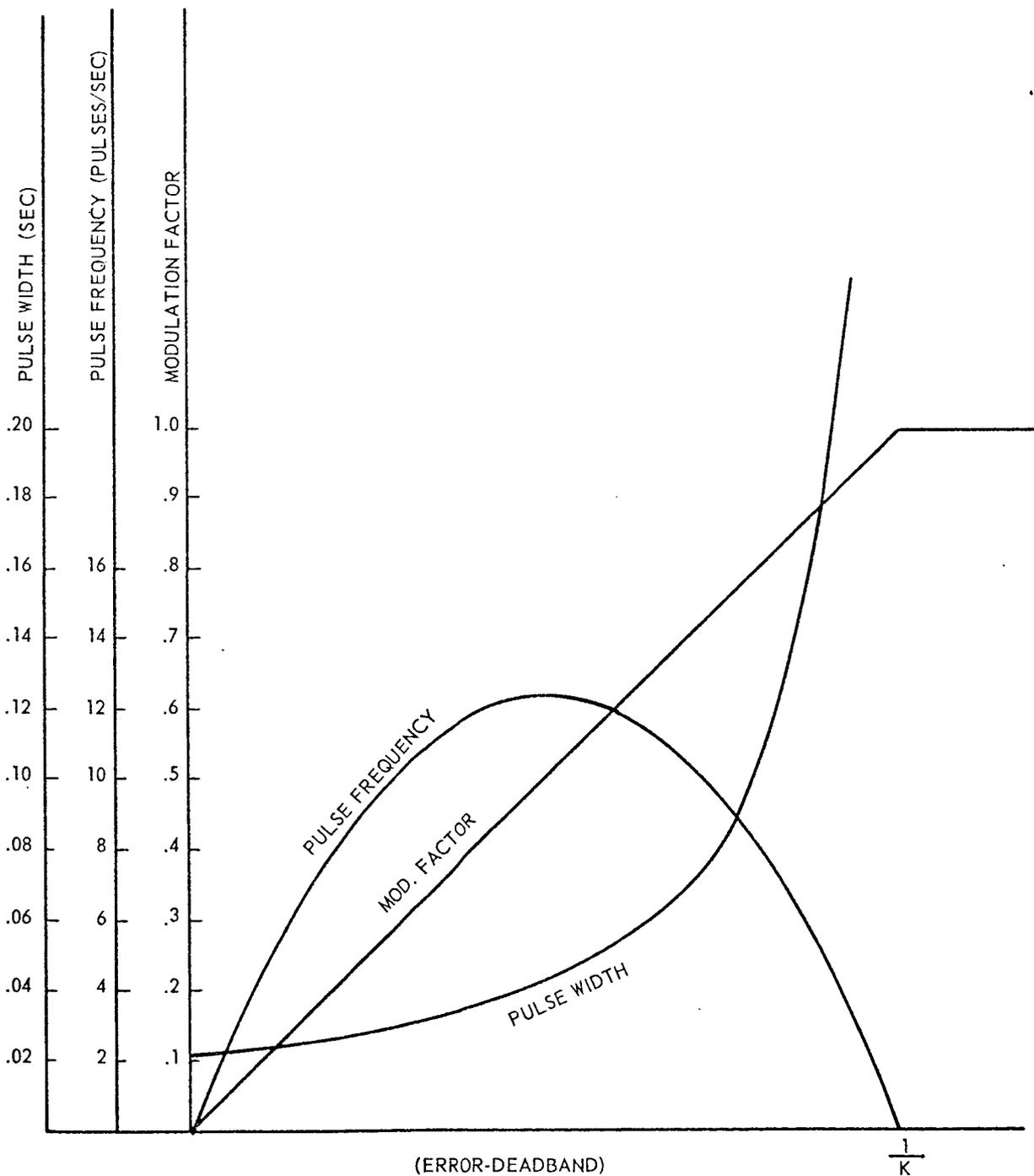


Fig. 5-39 Modulation Factor Computation

3.4 volts, which represents the hysteresis of the Schmitt trigger.

- b. Hydraulic Channel Operation. The operational amplifier employed in the hydraulic channel is used as a proportional d-c servo amplifier with the servo loop feedback current provided from a potentiometer that is physically connected to the hydraulic actuator. The error current from the amplifier is applied directly to the driving coil of the actuator.

### 5.8.2 Hydraulic System

The hydraulic system consists of a hydraulic power package and two servo actuators that provide control of the vehicle during PPS firings. The engine and gimbal mount configuration are illustrated in Figs. 5-40 and 5-42.

5.8.2.1 Hydraulic Power Package. The Mark III hydraulic power package that supplies hydraulic pressure is located on the Bell rocket engine gimbal ring and thrust chamber. Fuel pressure from the Bell rocket engine is transferred to the hydraulic power package through a fixed orifice to control the flow and pressure to the unsymmetrical dimethylhydrazine (UDMH) motor.

The hydraulic package consists of the following components (see Fig. 5-41):

- a. Gear Motor. The motor is a constant-displacement, high-speed, gear type. Gears are mounted on each end of an integral shaft by means of roller bearings. Gear side clearances are maintained mechanically for startup and hydraulically through rated operation. This provides limited end plate motion to reduce the possibility of gear seizure due to fuel contaminants.
- b. Flow Control Valve. The valve consists of a spring-positioned bypass flow control spool. The spool ends are ported to sense pressure drop across a flow orifice and subsequently monitor actual motor flow. The bypass flow is directed to the spool's

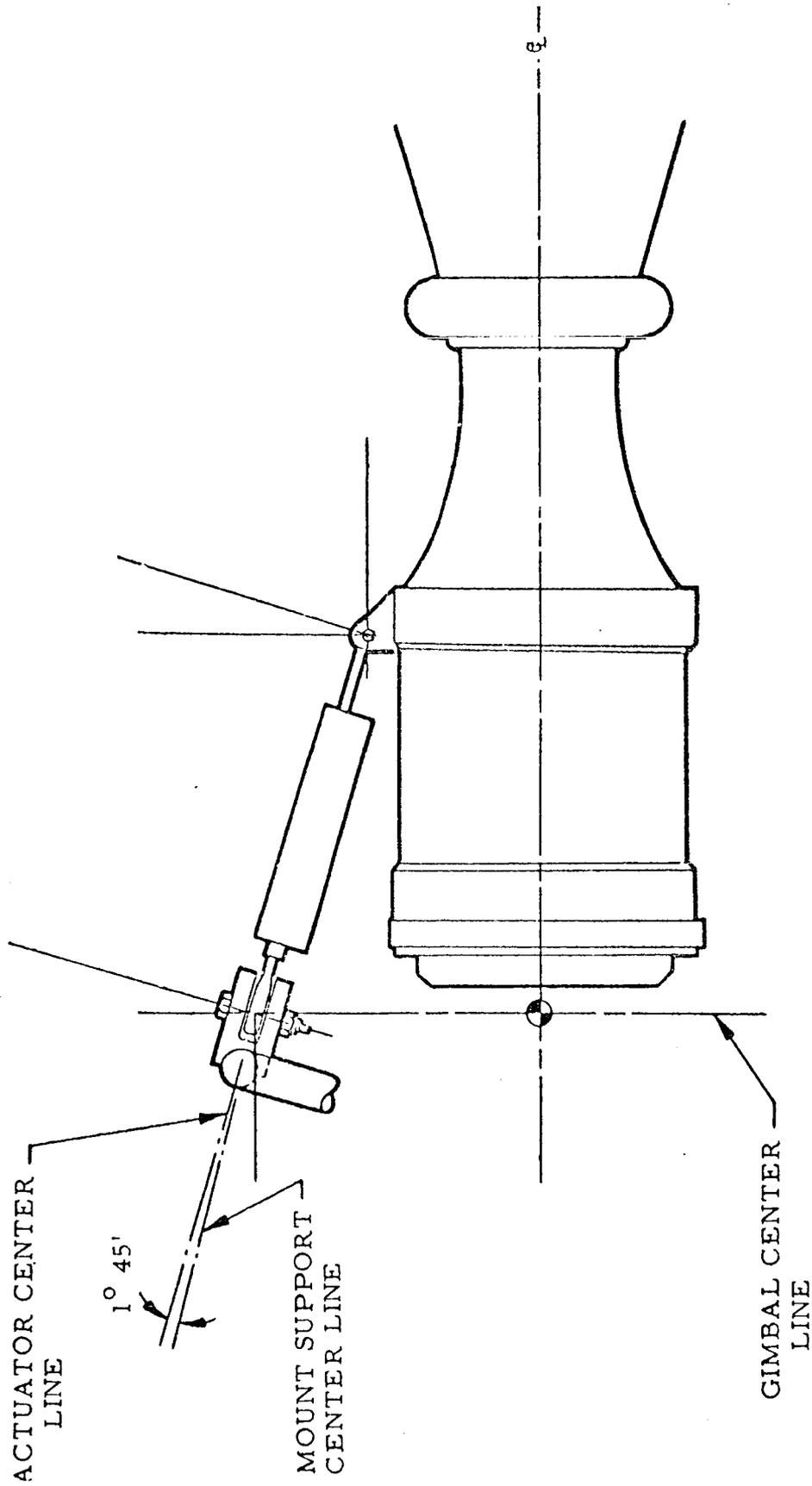


Fig. 5-40 Hydraulic Actuator Installation

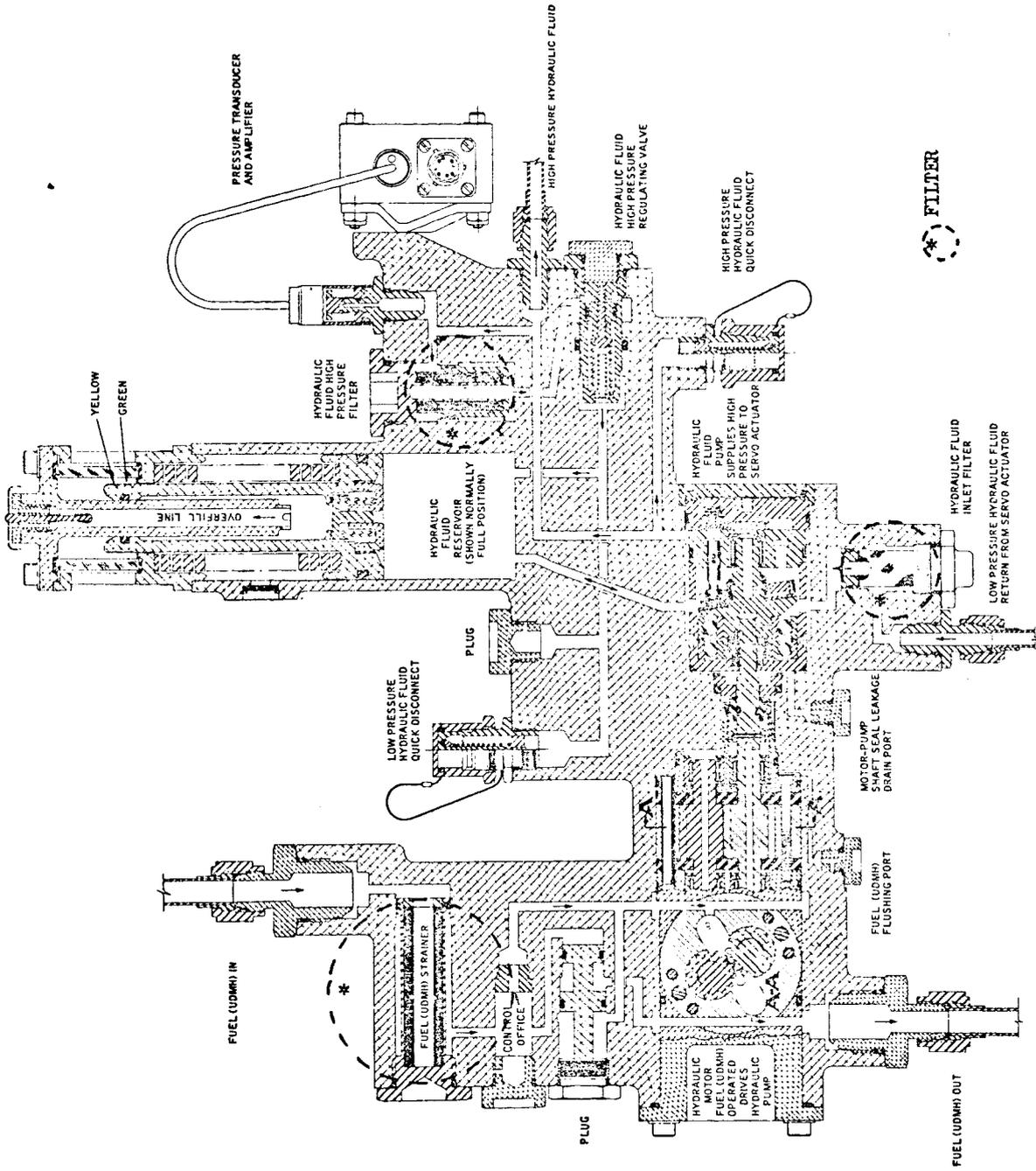


Fig. 5-41 Hydraulic Power Package

hollow center and then radially through small orifice ports to the discharge ports of the motor. The unit is mounted in parallel to the supply motor flow porting. Flow and subsequently unit speed are regulated to within 1 percent of the rated requirement.

- c. Fuel Filter. The filter is a stainless-steel, sintered, wire-mesh cartridge. The unit has an absolute rating of 125 microns. Higher filtration levels could not be reasonably attained due to the plugging nature of the fuel.
- d. Piston Pump. The pump is of an axial piston design with a rotating wobbler plate and stationary cylinder block. Seven pistons of low inertia are used. Piston return is accomplished by a return plate attached to the wobbler plate. The unit represents a design of high speed capability through the use of low inertia components with minimum load unbalances. Spring-loaded check valves are an integral part of the block assembly.
- e. Reservoir. The reservoir is an integral part of the manifold and contains a spring-loaded piston. The useful volume is approximately 7 cubic inches. The reservoir has a mechanical full indicator, which provides information during prelaunch package charging.
- f. Relief Valves. Two relief valves are used in the system as follows:
  - (1) High Pressure Relief Valve. This valve is used to provide constant pump load when the servo actuators are in the null position. It is a spring-loaded poppet type with metal to metal seats.
  - (2) Low Pressure Relief Valve. This valve is placed between the pump inlet and the reservoir. It relieves pressure surges due to thermal expansion and also assures pressure relief during ground checkout and testing. It is a spring-loaded poppet type with metal to rubber seats.

- g. Quick Disconnect Coupling. Conventional self-sealing quick disconnects are used for ground servicing such as bleeding, filtering, and charging the system. They are placed in the high and low pressure sides of the system.
- h. Test Valve. A test valve is located upstream of the high pressure relief valve. It is a spring-loaded poppet type that is used to block out the high pressure relief valve during system ground testing.

5.8.2.2 Design Considerations. This method of engine position control involved special design considerations. The hydraulic package had to present a relatively constant load to the engine fuel pump regardless of actuator gimbaling requirements. In addition to this, hydraulic system performance capabilities had to be stable during changing fuel pump power output levels that occur during rocket engine start transients and also through nominal steady state conditions

Because of performance variables in gas generator, turbine fuel pump, and thrust chamber for any given engine system, the control orifices were sized on an individual basis. The hydraulic package performance level was controlled within very narrow tolerances.

5.8.2.3 System Operation. Thrust chamber ignition is initiated shortly after separation of the Agena from the booster. During this start transient, which is of 1 or 2 seconds, the hydraulic motor pump must accelerate to rated conditions, the thrust chamber must be positioned, and a 90 percent thrust level must be attained.

A constant load is applied to the fuel pump through the use of a control orifice at the pump discharge port. This port is sized such that the pressure drop across it and the hydraulic package is equal to the pressure rise of the fuel pump.

The low breakout torque of the motor pump enables engine positioning prior to rated performance levels. The system reaches stabilized operating levels after gas generator bootstrap has been effected. The fuel-driven gear motor provides the necessary torque required to drive the hydraulic pump, which in turn provides the necessary flow and pressure to the actuators. During the nongimbaling period, pump flow is relieved through the high pressure relief valve. (The valve does not function as a relief valve, as such, but serves to maintain system pressure.)

The hydraulic rating is as follows:

- a. Motor Capacity. 6.75 gpm, 1040 rated psid
- b. Pump Capacity. 0.7 gpm, 2750 rated psid

Figure 5-42 outlines the complete hydraulic system in schematic form.

5.8.2.4 Electro-Hydraulic Servo Actuator. The actuator being utilized for gimbal control is of the balanced area design and is made of three major components — piston, cylinder, and inboard bearing. (See Fig. 5-43.) The stainless steel body contains the hydraulic cylinder, fluid ports, servo valve, position transducer, and filters.

The piston and rod are of integral design and are fitted with O-rings and teflon backing rings.

The servo valve has mechanical feedback from the second stage valve spool to the first stage flapper assembly. It consists of a polarized electrical force motor and two stages of hydraulic power amplification. The polarizing magnetic flux circuit is formed by upper and lower pole plates supported by two Alnico magnets. The motor armature extends between the air gaps of the magnetic flux circuit and is supported in this position by a flexure tube member. The flexure tube acts also as a seal between the electromagnetic and hydraulic sections of the valve. Two torque motor coils are located about the armature, one on either side of the flexure tube.

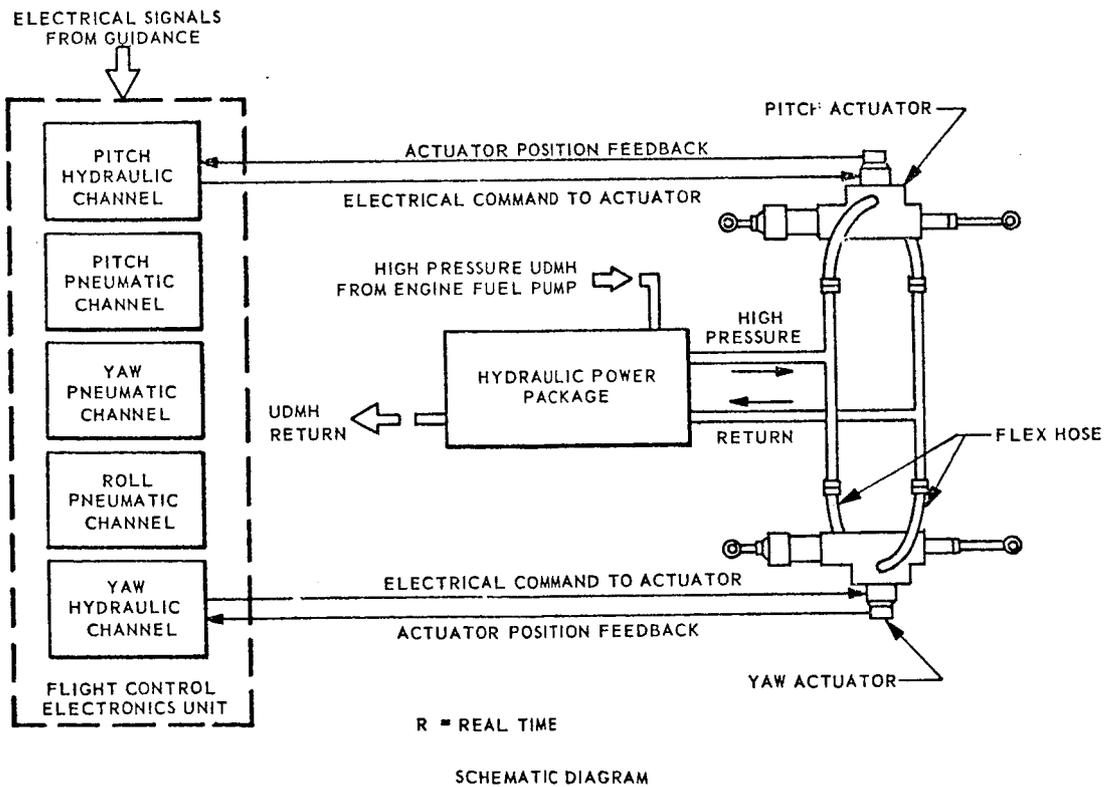
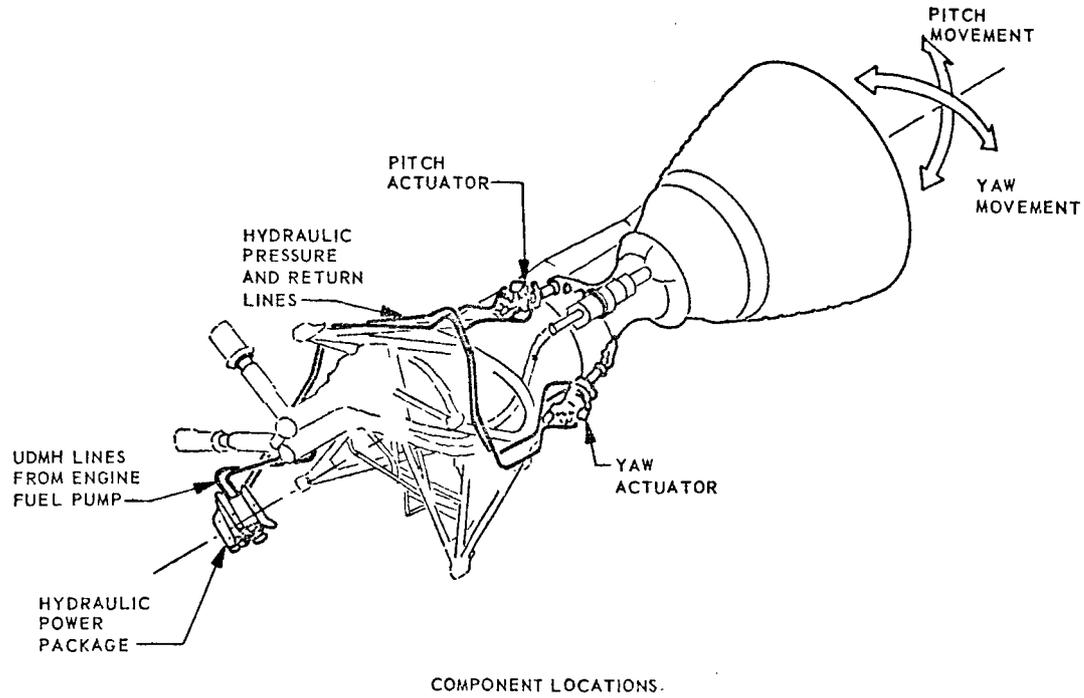


Fig. 5-42 Hydraulic Flight Control System

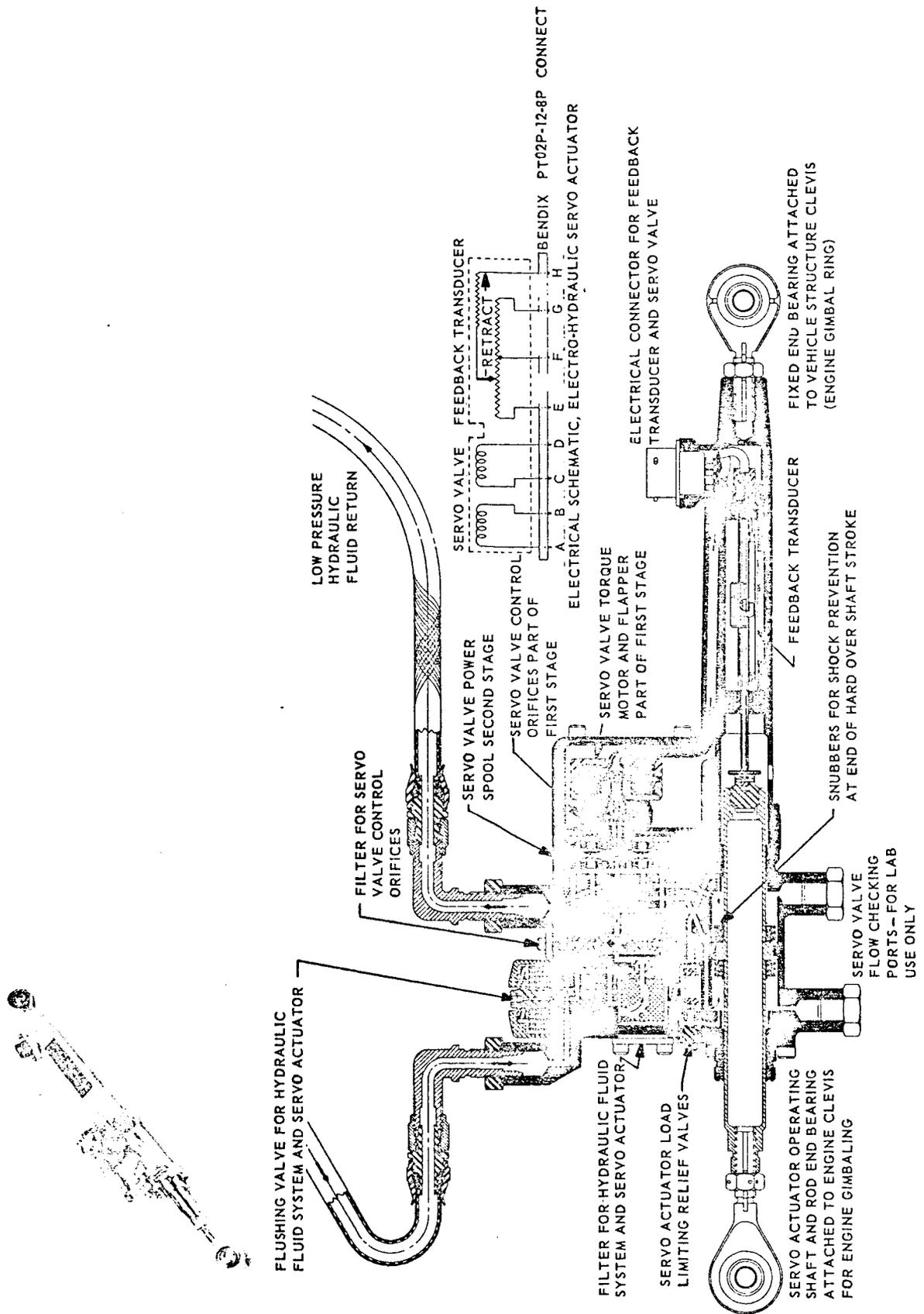


Fig. 5-43 Electro-Hydraulic Servo Actuator (Cutaway View)

Rigidly attached to the armature at the flexure tube support point is the flapper of a hydraulic amplifier. The flapper extends through the tubular flexure member and passes between two nozzles, creating two variable orifices between the nozzle tips and the flapper. Pressure oil is supplied to these orifices through two fixed upstream orifices. The intermediate pressures developed are applied to either end of the output stage sliding spool. The spool is a conventional four-way design so output flow from the valve, at a fixed valve pressure drop, is proportional to spool displacement. A cantilever-spring feedback element relates spool displacement to torque at the motor armature. This feedback spring is fixed to the armature-flapper assembly at the flexure tube support point. The free end of the spring extends through the flapper to engage a slot at the center of the spool.

The load-limiting relief valves are positioned between the cylinder ends and the supply pressure. A compact back-to-back design employs a single spring for two balls.

Piston deceleration is provided by a series of plates perpendicular to the piston centerline that throttles the fluid flow during the last portion of the stroke. When the snubber plate contacts the end wall, a capillary passage is formed that throttles the discharge flow and snubs the piston to a smooth stop.

Two 25-micron (absolute rating) filters are housed in the actuator. Both units are of stainless steel, wire-mesh design. The larger of the two filters total valve flow. The second and smaller filter protects the pressure controlling orifices to the second stage spool.

The flushing bypass valve is an integral part of the actuator body. A cylindrical lapped plug is manually rotated to align bypass flow ports during flushing operations. System oil is thus diverted from passing through the servo valve proper and subsequently contaminating it. The valve returns to a nonbypass position through spring load when released.

The position feedback transducer employs a stationary wire-wound resistive element coil with a movable wiper carriage. The potentiometer shaft is coupled to the piston rod and is housed within the actuator assembly. The designed mechanical stroke of the transducer is 2.00 inches maximum and 1.75 inches minimum. The resistance of the transducer is 10,000 ohm, centertapped to ground at 5,000 ohms  $\pm 2$  percent. Scale factor of the transducer when excited with plus and minus 28 volts is 19.8 volts per inch.

A low pressure switch, which is electrically connected to the status display panel, is located in the return side of the actuator to detect leaks within the hydraulic system. The switch is set for actuation if the low pressure decreases below 35 psig.

An additional switch, located in the hydraulic high pressure line, gives a positive indication on the status display panel when the high pressure reaches 1500 psig.

### 5.8.3 Pneumatic System

The pneumatic system (Fig. 5-44) consists of the following components:

- a. Pneumatic thrust controllers (6)
- b. Pneumatic pressure regulator
- c. Nitrogen spheres (3).

Control of the vehicle is maintained during the coast phase with the pneumatic control system. Two pressures may be selected depending upon the degree of control required. The 100 psig regulated pressure from the regulator is defined as 10-pound thrust from the gas valve; the 5 psig regulated pressure is equivalent to 0.5-pound thrust at the gas valve nozzle.

5.8.3.1 Pneumatic Thrust Controllers. Gas valves 2 and 5 provide corrective pitch torques with negligible vehicle coupling with either roll or yaw. (See Fig. 5-45.) The remaining four valves are shared for roll and

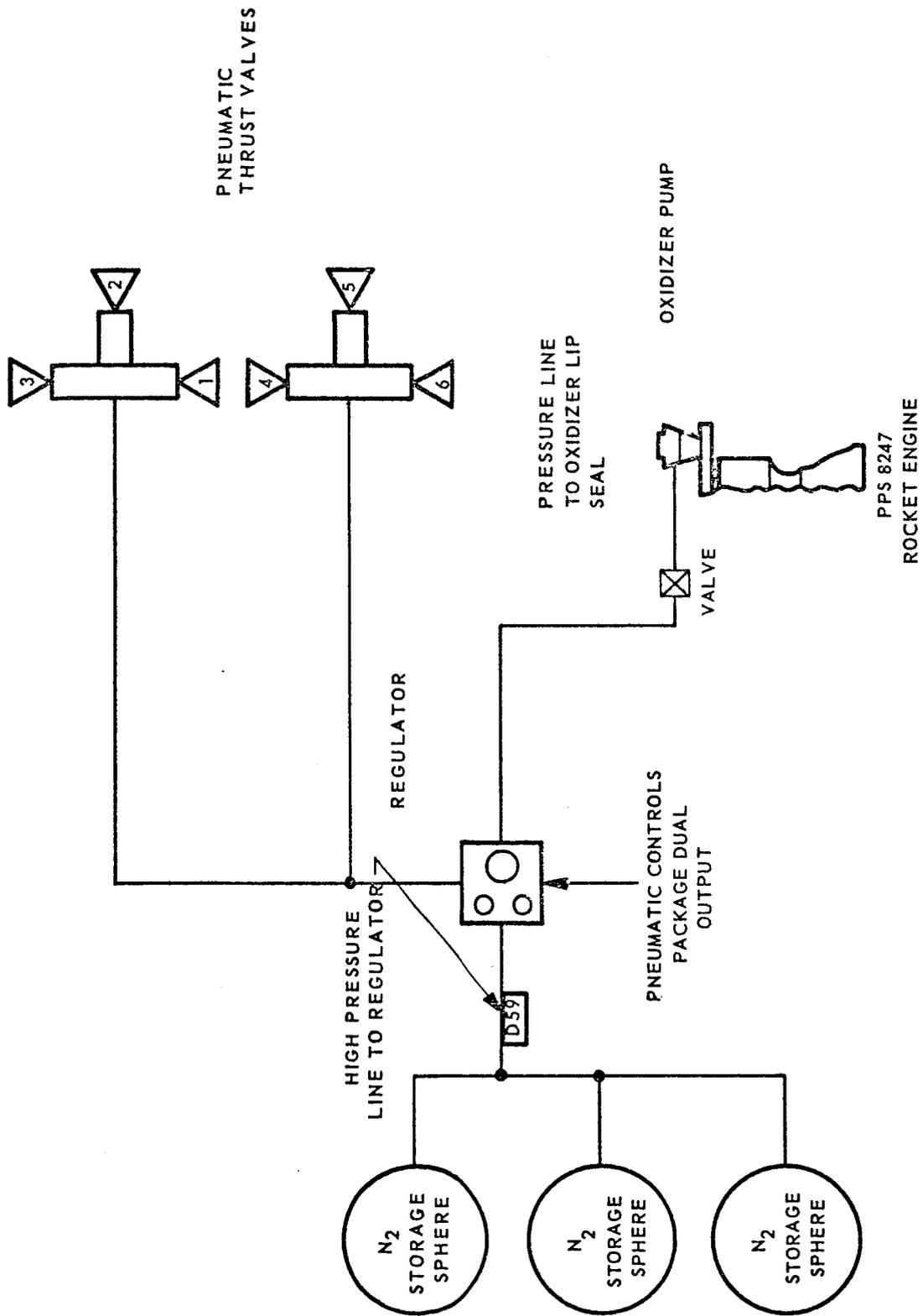


Fig. 5-44 Pneumatic Flight Control System

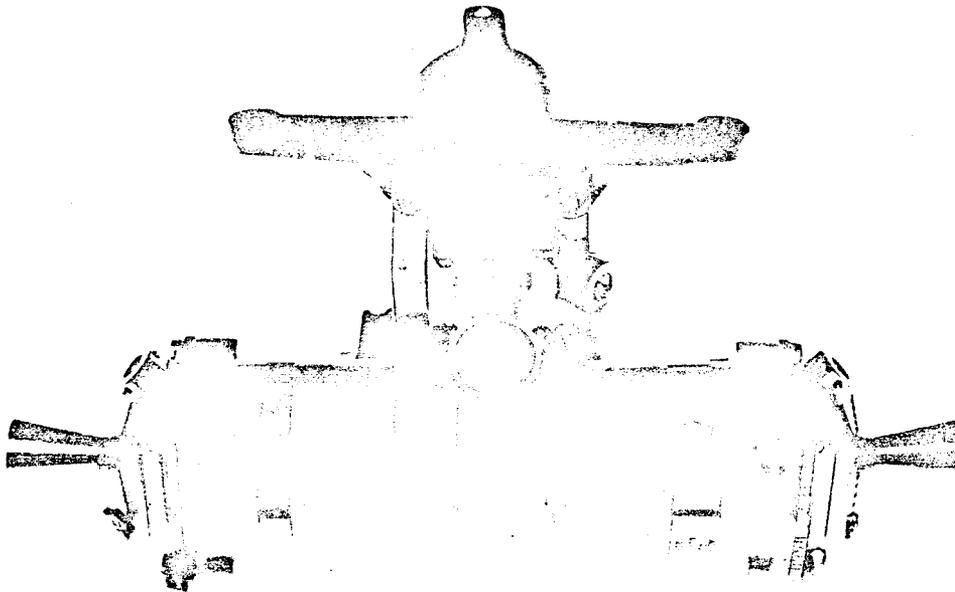


Fig. 5-45 Thrust Valve Cluster

yaw control to conserve gas. For example, a plus yaw error alone (nose right) would fire gas valves 3 and 4, while a negative yaw error would fire gas valves 1 and 6. A simple positive roll error (clockwise, looking forward), would fire valves 3 and 6, while a negative roll error would fire valves 1 and 4. If a positive yaw error (valves 3 and 4) and a negative roll error (valves 1 and 4) occurred simultaneously, valve 4 and either valve 1 or 3, or neither, would fire, depending on the relative sizes of the roll and yaw errors. This feature is the result of one amplifier driving the two "back-to-back" valves, with the error polarity determining which is to fire. The firing of valves 1, 3, 4, and 6 will produce both roll and yaw torques upon the vehicle. The coupling of roll and yaw is appropriate since the channels are coupled dynamically in the vehicle through geocentric rate.

5.8.3.2 Pneumatic Pressure Regulator. The pneumatic pressure regulator (Fig. 5-46) performs three regulating functions: (1) It reduces the nominal

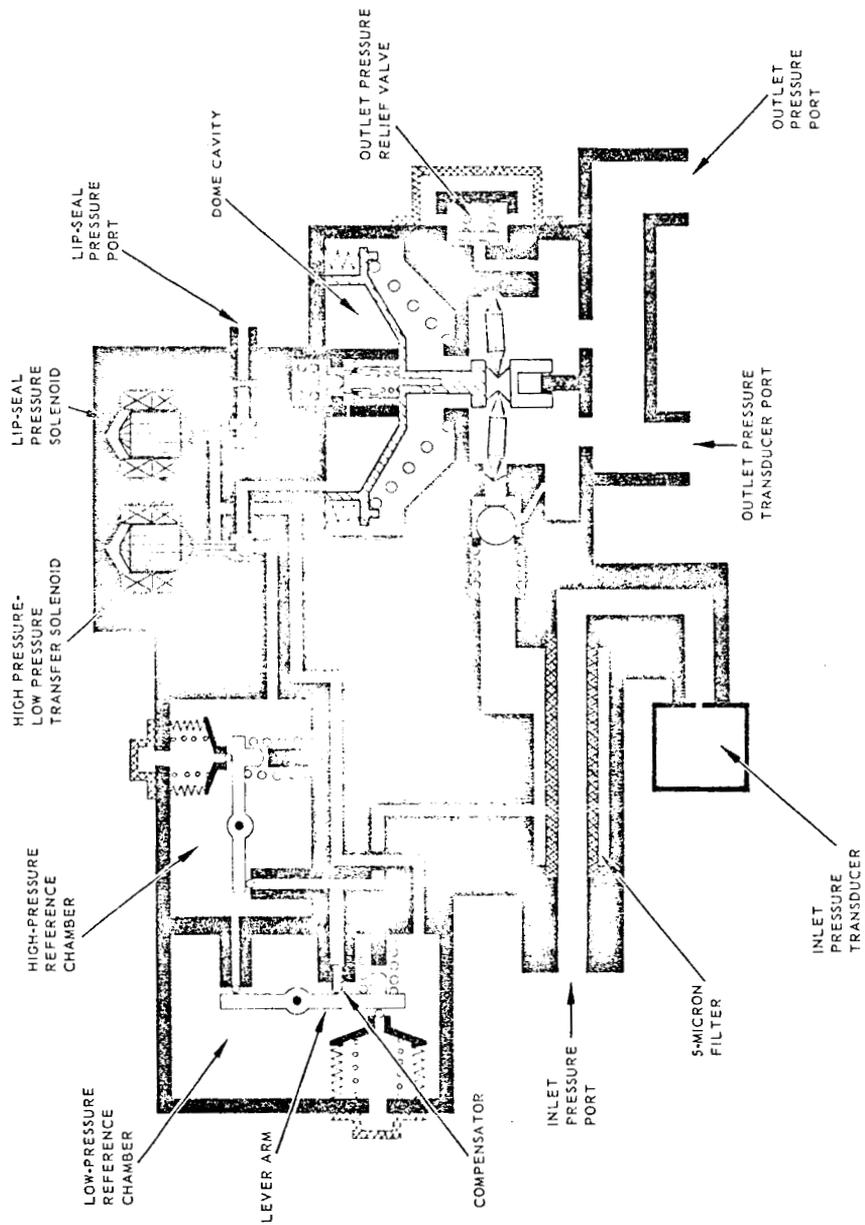


Fig. 5-46 Pneumatic Pressure Regulator

3600 psia from the spheres to a nominal 100 psia (high mode operation) or to a nominal 5 psia (low mode operation) for use by the gas valves. (2) It maintains a constant regulation of the selected pressure while the gas valves are pulsing. (3) It supplies low pressure to the PPS for pressurizing the lip seal in the oxidizer pump.

5.8.3.3 Nitrogen Spheres. The control gas for the system is provided by three 2200 cubic-inch supply spheres containing a total of approximately 140 pounds of a Freon (80 percent) and nitrogen (20 percent) mixture. The pressure in the three spheres is monitored on the status display panel by a meter.

## Section 6 TEST PROGRAM

The test program specified for the guidance and controls subsystem includes acceptance testing from individual components to the integrated system tests. The purpose of such a program is to assure compliance with performance requirements at the component and system level.

### 6.1 ACCEPTANCE TESTS OF COMPONENTS

An acceptance test specification is written for each component or equipment that has been modified from the basic Agena D configuration, based on design parameters and system requirements for the components. This specification details the tests that will determine the accuracy and characteristics of the individual component.

### 6.2 QUALIFICATION TESTS OF COMPONENTS

Environmental requirements for the qualification tests of components or equipment that has been modified from the basic Agena D configuration incorporate the criteria and test methods specified in the applicable portions of LMSC-6117-B, General Environment Specification. A qualification test specification is written for each new major component or a major redesign of any Agena D component in the subsystem.

### 6.3 FLIGHT CONTROLS TESTS

Compatibility tests are performed on all guidance and controls components after modification and installation in the Gemini Agena Target Vehicle. These tests consist of applying attitude and rate signals to the electronics

packages and measuring the response characteristics of the control elements. Movement and response is measured by monitoring the position potentiometer located on the hydraulic actuators. Current is measured through the gas valves to establish proper operation and phasing.

During this test, the gains and dynamic characteristics of the pneumatic and hydraulic channels are established.

#### 6.4 FLIGHT COMMAND LOGIC PACKAGE

The flight command logic package is tested by being subjected to simulations of every possible input and command to the system. To assure proper response of the commands, the flight command logic package is monitored at the controlling function.

#### 6.5 GUIDANCE AND CONTROLS TESTS AND TELEMETRY CALIBRATION

The guidance and controls system is checked in the vehicle and in the caged mode of operation to assure proper system compatibility. The tests determine the gains, system balance, electrical pickup, phase shift, gyro drift, and velocity meter parameters.

All measurements that have been modified or signal conditioning equipment that has been changed are calibrated at this time.

#### 6.6 SANTA CRUZ HOT FIRE VEHICLE 5001

The testing at Santa Cruz Test Base will include a comprehensive simulated mission test. The test will simulate a typical launch ascent and orbit operation; it will include multiple firings of the primary and secondary propulsion system.

## 6.7 LAUNCH READINESS TESTS

The J-FACT (joint flight acceptance composite test) is an integrated check of the vehicle system, the launch complex, and all applicable range support station and facilities. It includes a simulated countdown, launch, and flight mission. The test will validate all items of launch complex aerospace ground equipment (AGE). The J-FACT will follow normal countdown procedure, omitting propellant loading and pressurization.

APPENDIX A  
GEMINI ATV ASCENT SEQUENCE OF EVENTS

This appendix is a typical ascent sequence of events for the Gemini ATV. If specific up-to-date information is required, refer to the latest revision of LMSC-1365396, Gemini Sequence of Events Ascent Model 37205 Vehicle 5001.

Event No.	Nominal Time From Liftoff (Seconds)	Event Description	Source of Signal (Other Than Timer)
1		Sequence Timer Reset (Ground Reset Function)	
LO	0	Liftoff	Test Conductor
		Atlas Main Engine Burn	
		Atlas Sustainer Burn	
		Atlas Vernier Burn	
BCO	131.5	Atlas Main Engine Cutoff	Atlas Guidance
SCO	281.0	Atlas Sustainer Cutoff	Atlas Guidance
		Start Atlas Programmer	
		Sustainer Cutoff Subroutine	
		Disarm Self-Destruct	GE Relay K6
SDT	282.0	Start Sequence Timer	GE Relay K1
2	282.0	Timer Reset S1A (Ground Reset Function)	
2	282.0	Timer Safety Input (Ground Function)	
VCO	301.0	Atlas Vernier Engine Cutoff	Atlas Guidance

## APPENDIX A (Continued)

<u>Event No.</u>	<u>Nominal Time From Liftoff (Seconds)</u>	<u>Event Description</u>	<u>Source of Signal (Other Than Timer)</u>
	299.72	Uncage Gyros, Fire Horizon Sensor Door and L-Band Transponder Thermal Cover Squibs, Arm Atlas Agena Separation Circuit, Disarm Self-Destruct (Backup)	GE Relay K5
VCO (BU)	307.0	Atlas Vernier Engine Cutoff (Backup)	Atlas Programmer
SEP	307	Fire Separation Primacord Squibs and Retro Rockets Atlas/Agena Separation	GE Relay K8
PAP	307.5	Start Sequence Timer (Backup)	Pullaway Plug (P700)
SEP (BU)	308.5	Atlas/Agena Separation (Backup)	Atlas Programmer
APAC	308.5	Enable Attitude Control System	Separation Switches (S3 & S4)
	308.5	Roll Horizon Sensor to IRP ON (High Gain) (Alpha Angle 1.57 Degrees)	
3	322.5	Enable Command Separation Backup	
	322.5	Eject L-Band Covers (Backup)	
	322.5	Fire Command Sep Backup Horizon Sensor Door and L-Band Transponder Thermal Cover Squibs & Uncage Gyros (Backup)	
3	322.5	Reset Turbine Overspeed Circuit	
4	323.5	Roll Horizon Sensor to IRP ON (Backup)	
5	325.5	Pitch ON (-1.5 degrees per second)	

## APPENDIX A (Continued)

<u>Event No.</u>	<u>Nominal Time From Liftoff (Seconds)</u>	<u>Event Description</u>	<u>Source of Signal (Other Than Timer)</u>
	325.5	Open SPS Pressurization Start Valve	
	325.5	Remove Horizon Sensor Door Squibs Signal, Remove Uncage Gyro Signal	
	325.5	Remove Eject L-Band Covers Backup Signal	
5	325.5	Remove Reset Turbine Over-speed Signal	
6	337.5	Pitch OFF	
	337.5	Geocentric Rate ON (-2.84 degrees per minute)	
	337.5	Pitch Horizon Sensor to IRF ON (High Gain)	
6	337.5	Enable Velocity Meter (Velocity to be Gained 8150 ft/sec)	
7	347.5	<u>Open SPS 16 lb Bipropellant Valves (SPS 16 lb Thrust Initiate)</u>	
7	347.5	Remove Geocentric Rate ON Signal	
8	365.5	Disable Attitude Control System Pitch and Yaw Pneumatic Channels and Open PPS Gas Generator Valve ( <u>PPS Thrust Initiate</u> )	
9	367	Open Helium Pyro Valve, Fire Fuel, Oxidizer, and Helium Disconnect Isolation Valve Squibs	

## APPENDIX A (Continued)

<u>Event No.</u>	<u>Nominal Time From Liftoff (Seconds)</u>	<u>Event Description</u>	<u>Source of Signal (Other Than Timer)</u>
9	367	Open Helium Pyro Valve; Fire Fuel, Oxidizer, and Helium Disconnect Isolation Valve Squibs	
10	367.5	Close SPS 16 lb Bipropellant Valves ( <u>SPS Thrust Cutoff</u> )	
10	367.5	Close SPS Pressurization Start Valves	
11	371.5	Fire Jettison Nose Shroud Squibs	
12	540.5	Arm PPS Thrust Cutoff	
AECO	549.18	<u>PPS Thrust Cutoff (Enable Pitch and Yaw Pneumatics)</u>	Velocity Meter Signal
13	559.5	Close PPS Gas Generator and Fuel Valves (PPS Thrust Cutoff Backup)	
13	559.5	Enable Pitch and Yaw Pneumatics Backup	
14	564.5	Close PPS Propellant Shutoff Valve and Close Oxidizer Lipseal Pressurization Valve	
14	564.5	Remove Enable Pitch and Yaw Pneumatic Backup Signal	
15	570.5	Power OFF PPS Propellant Shutoff Valves and Oxidizer Lipseal Valve	
15	570.5	Attitude Control System Deadband Wide	
16	605.5	Disable Velocity Meter	
	605.5	Gyro Compassing ON	

## APPENDIX A (Continued)

<u>Event No.</u>	<u>Nominal Time From Liftoff (Seconds)</u>	<u>Event Description</u>	<u>Source of Signal (Other Than Timer)</u>
16	605.5	Pitch, Roll and Yaw Horizon Sensor Gain Low ON	
17	607.5	Antenna Transfer Ascent/Orbit	
17	607.5	Disable Command Destruct Receivers	
18	683.5	Oxidizer Pressurization System Isolation Valve Closed	
19	688.5	Remove Command Destruct Receiver Disable Signal	
	688.5	Attitude Control System Pressure Low	
19	688.5	Attitude Control System Gain Low	
20	693.5	Remove Attitude Control System Pressure Low Signal	
20	693.5	Remove -2.84 deg/min and apply -3.99 deg/min Geocentric Rate Signal	
21	765.5	Shut Down Sequence Timer	
21	765.5	Fire Horizon Sensor 0 degrees Position Squibs	
22	766.0	Remove Shutdown Sequence Timer Signal	

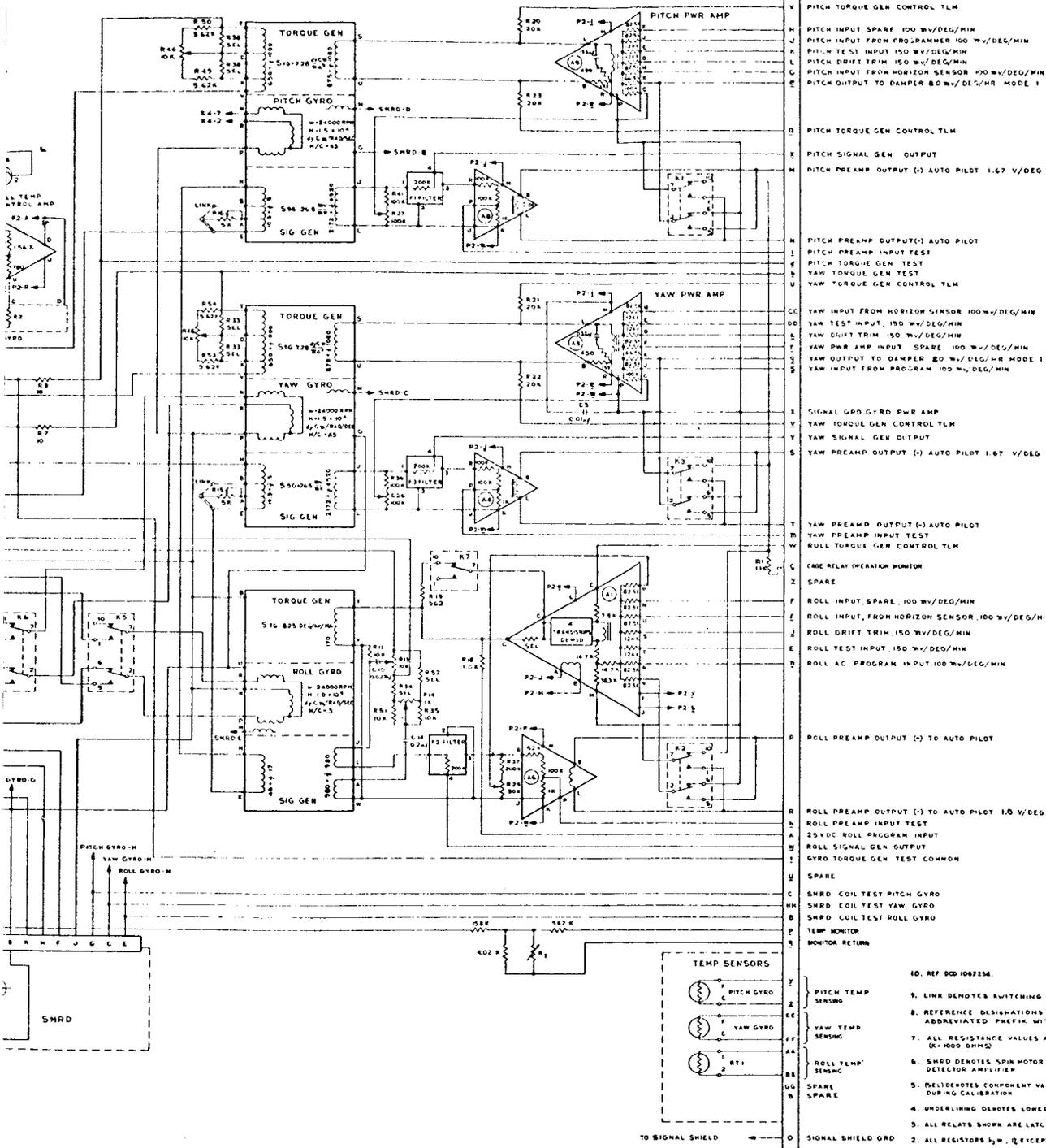
APPENDIX B  
SCHEMATICS

This appendix contains the following schematic diagrams:

- Fig. B-1 Inertial Reference Package Schematic Diagram
- Fig. B-2 Horizon Sensor System Schematic Diagram
- Fig. B-3 Hydraulic Control Channel Schematic Diagram
- Fig. B-4 Pneumatic Control Channel Schematic Diagram



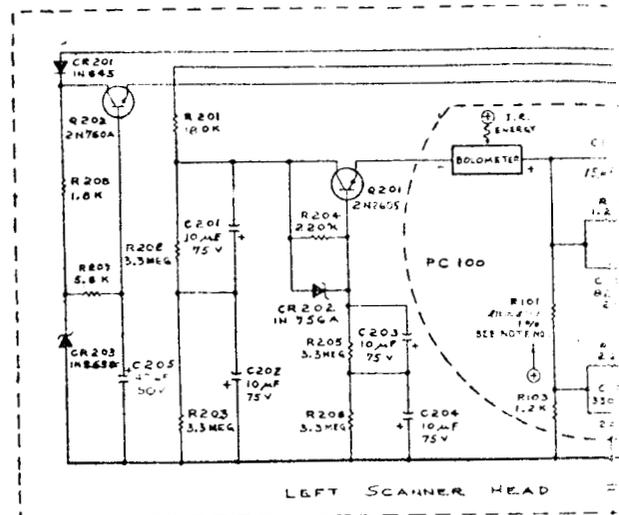
J38



- V PITCH TORQUE GEN CONTROL TLM
- H PITCH INPUT SPARE 100 W/DEG/MIN
- J PITCH INPUT FROM PROGRAMMER 100 W/DEG/MIN
- K PITCH TEST INPUT 150 W/DEG/MIN
- L PITCH DRIFT TRIM 150 W/DEG/MIN
- G PITCH INPUT FROM HORIZON SENSOR 100 W/DEG/MIN
- E PITCH OUTPUT TO DAMPER 80 W/DEG/HR MODE 1
- Q PITCH TORQUE GEN CONTROL TLM
- I PITCH SIGNAL GEN OUTPUT
- M PITCH PREAMP OUTPUT (3) AUTO PILOT 1.67 V/DEG
- N PITCH PREAMP OUTPUT (1) AUTO PILOT
- I PITCH PREAMP INPUT TEST
- F PITCH TORQUE GEN TEST
- U YAW TORQUE GEN TEST
- U YAW TORQUE GEN CONTROL TLM
- CC YAW INPUT FROM HORIZON SENSOR 100 W/DEG/MIN
- DD YAW TEST INPUT 150 W/DEG/MIN
- E YAW DRIFT TRIM 150 W/DEG/MIN
- F YAW PWR AMP INPUT SPARE 100 W/DEG/MIN
- Q YAW OUTPUT TO DAMPER 80 W/DEG/HR MODE 1
- P YAW INPUT FROM PROGRAM 100 W/DEG/MIN
- I SIGNAL GRD GYRO PWR AMP
- Y YAW TORQUE GEN CONTROL TLM
- Y YAW SIGNAL GEN OUTPUT
- S YAW PREAMP OUTPUT (4) AUTO PILOT 1.67 V/DEG
- T YAW PREAMP OUTPUT (1) AUTO PILOT
- B YAW PREAMP INPUT TEST
- W ROLL TORQUE GEN CONTROL TLM
- G CASE RELAY OPERATION MONITOR
- Z SPARE
- F ROLL INPUT, SPARE, 100 W/DEG/MIN
- F ROLL INPUT, FROM HORIZON SENSOR, 100 W/DEG/MIN
- E ROLL DRIFT TRIM, 150 W/DEG/MIN
- D ROLL TEST INPUT, 150 W/DEG/MIN
- D ROLL AC PROGRAM INPUT, 100 W/DEG/MIN
- P ROLL PREAMP OUTPUT (4) TO AUTO PILOT
- R ROLL PREAMP OUTPUT (3) TO AUTO PILOT 1.0 V/DEG
- B ROLL PREAMP INPUT TEST
- A 25VDC ROLL PROGRAM INPUT
- W ROLL SIGNAL GEN OUTPUT
- I GYRO TORQUE GEN TEST COMMON
- U SPARE
- C SHRD COIL TEST PITCH GYRO
- M SHRD COIL TEST YAW GYRO
- B SHRD COIL TEST ROLL GYRO
- P TEMP MONITOR
- Q MONITOR RETURN
- Y PITCH GYRO
- Z PITCH TEMP SENSING
- EE YAW GYRO
- FF YAW TEMP SENSING
- AA ROLL GYRO
- BB ROLL TEMP SENSING
- GG SPARE
- B SPARE
- O TO SIGNAL SHIELD

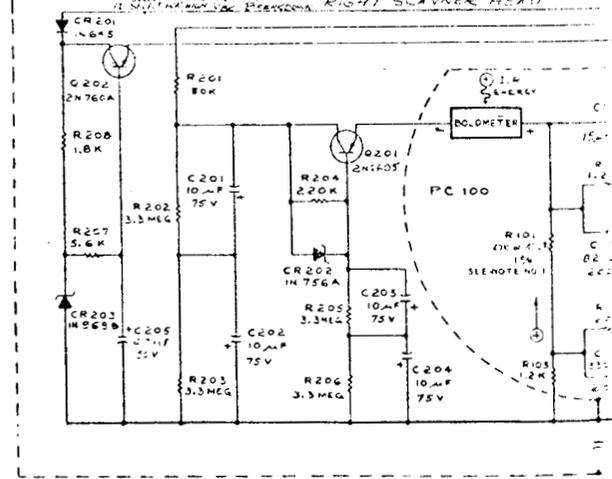
10. REF DOD 108234.
1. LINK DENOTES SWITCHING
  2. REFERENCE DESIGNATIONS ARE ABBREVIATED PREFIX WITH D7
  3. ALL RESISTANCE VALUES ARE IN OHMS (X=1000 OHMS)
  4. SHRD DENOTES SPIN MOTOR ROTATION DIRECTION DETECTOR AMPLIFIER
  5. UNDERLINES COMPONENT VALUES SELECTED DURING CALIBRATION
  6. UNDERLINES DENOTES LOWER CASE LETTERS
  7. ALL RELAYS SHOWN ARE LATCHING
  8. ALL RELAYS 1/2 W. (EXCEPT AS NOTED)
  9. JMW-A BENDIS P107H-22 55P SIGNAL PLUG
  10. J3915-A BENDIS P107H-22 55PW POWER PLUG
  11. SEE DCD DWG NO 1062256 FOR VEHICLE PW CONNECTIONS
- NOTES:

Fig. B-1 Inertial Reference Package Schematic Diagram



- KEY
- ⊕ EARTH PULSE POSITIVE GOING
  - ⊖ EARTH PULSE NEGATIVE GOING
  - ⊕ INCREASING VOLTAGE WITH ↑ PITCH (NOSE UP)
  - ⊖ DECREASING VOLTAGE WITH ↑ PITCH (NOSE UP)
  - ⊕ INCREASING VOLTAGE WITH ↑ ROLL (CLOCKWISE)
  - ⊖ DECREASING VOLTAGE WITH ↑ ROLL (CLOCKWISE)
  - ⊕ SIGNAL GROUND
  - ⊖ CHASSIS GROUND
  - ⊕ INDICATES SELECTED COMPONENTS

- NOTES:
1. (NOMINAL VALUE) ACTUAL VALUE SELECTED AT FINAL ASSEMBLY
  2. (NOMINAL VALUE) ACTUAL VALUE TO BE SELECTED AT FINAL ASSEMBLY
  3. ALL PIN CONNECTIONS ROUTED THRU 500μF FEEDTHRU
  4. ALL RESISTORS 1/2 W 10% UNLESS OTHERWISE SPECIFIED
  5. ALL 1% RESISTORS ARE 1/8 WATT
  6. ALL RELAYS ARE SHOWN IN DE-ENERGIZED POSITION
  7. TRANSFORMER NUMBERING FOLLOWS RESISTOR NUMBERING
  8. TRANSFORMERS ARE ENCASED IN SHIELDING CAN (CAN IS TYPICAL)
  9. STRAY RESISTOR SHALL NOT BE LESS THAN 1 MEGOHM
  10. SELECT AT FINAL ASSEMBLY FOR 100 KΩ SET
  11. SHEET FOR RIGHT SCANNER HEAD
  12. SHEET FOR RIGHT SCANNER HEAD









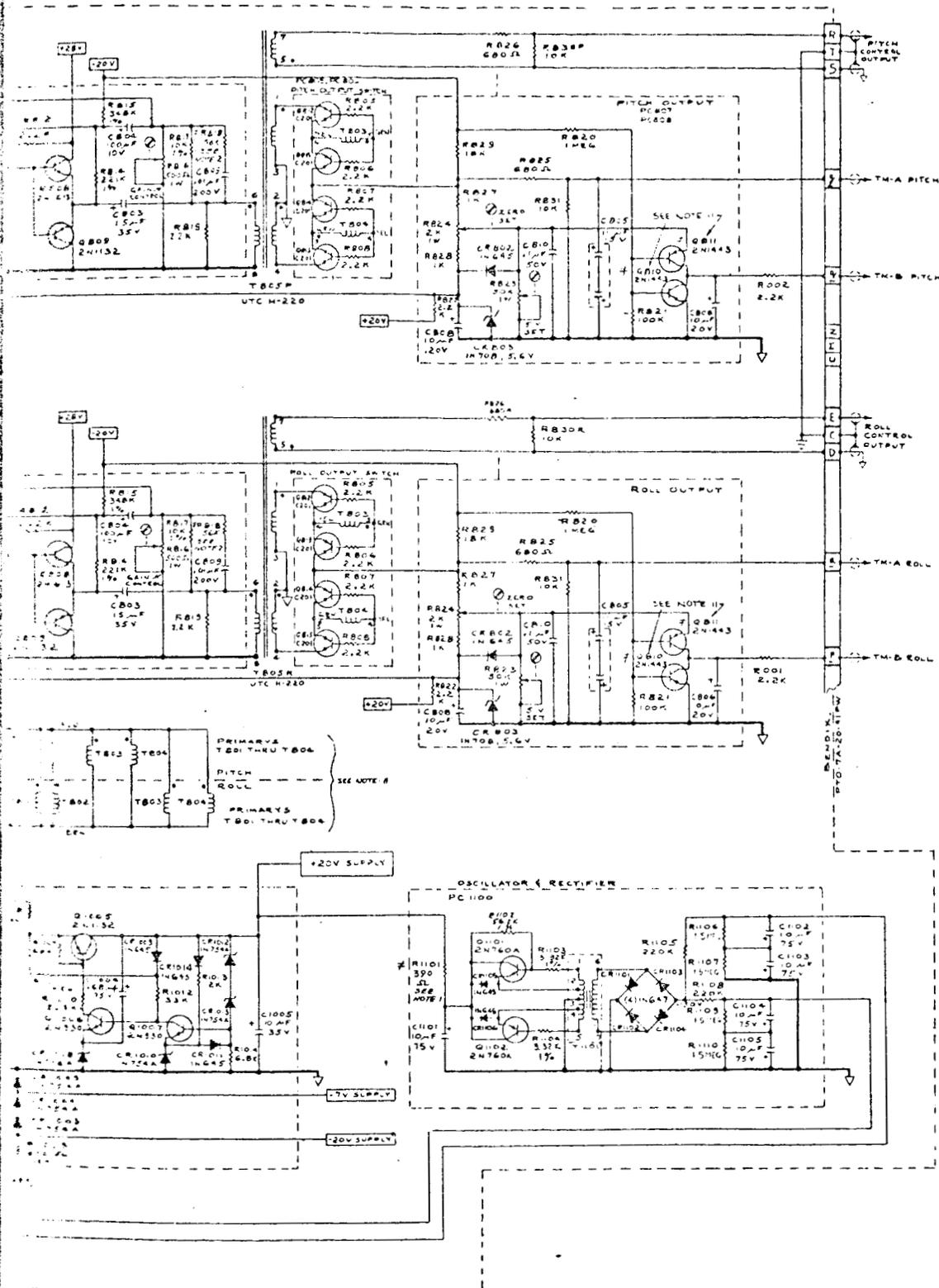


Fig. B-2 Horizon Sensor System Schematic Diagram



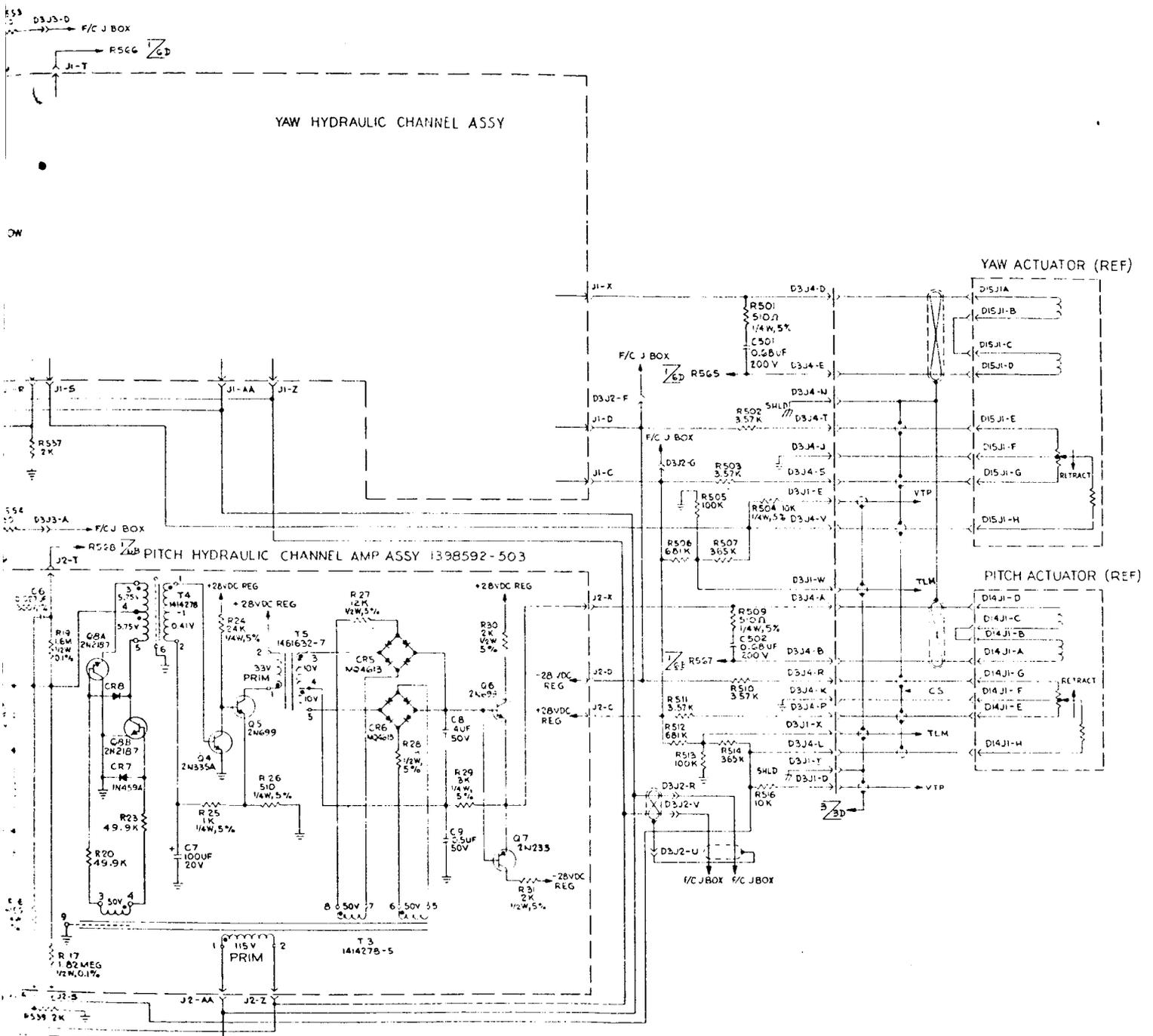
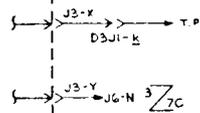
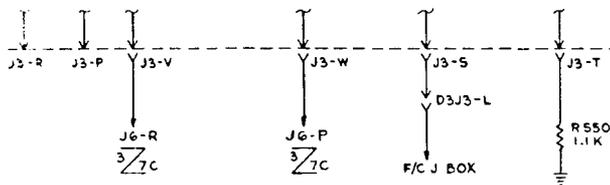
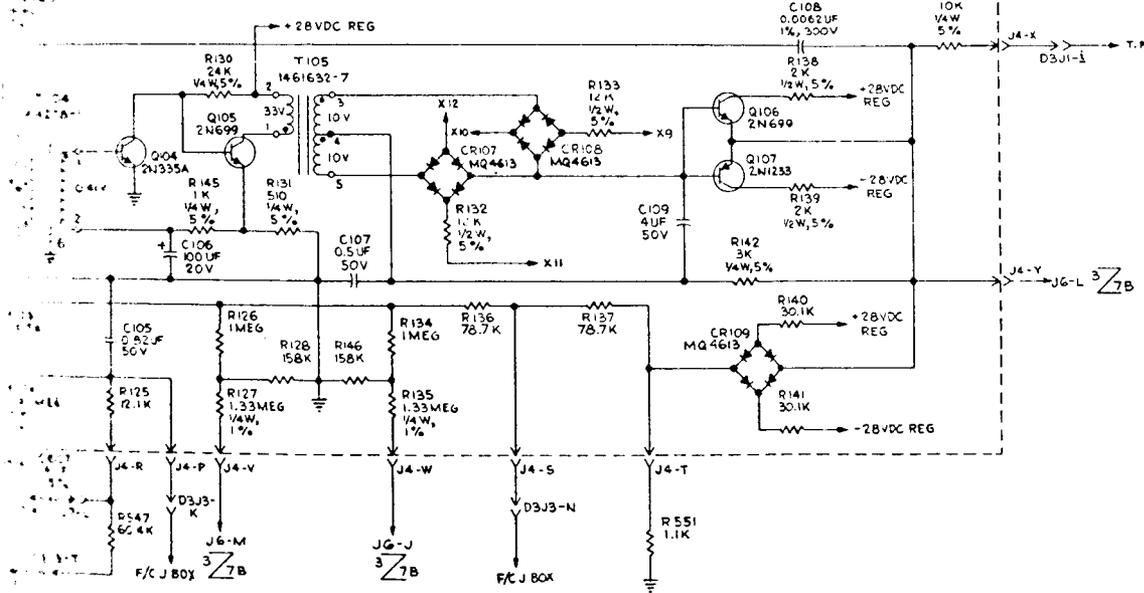


Fig. B-3 Hydraulic Control Channel Schematic Diagram

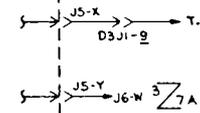
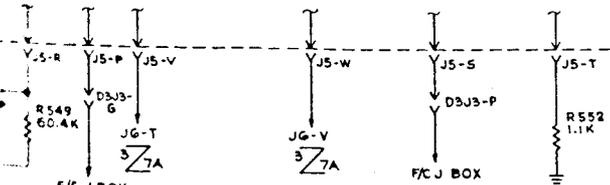
PITCH PNEUMATIC ELECTRONIC AMPLIFIER ASSY  
CIRCUITRY SAME AS BELOW



YAW PNEUMATIC ELECTRONIC AMPLIFIER ASSY



ROLL PNEUMATIC ELECTRONIC AMPLIFIER ASSY  
CIRCUITRY SAME AS ABOVE



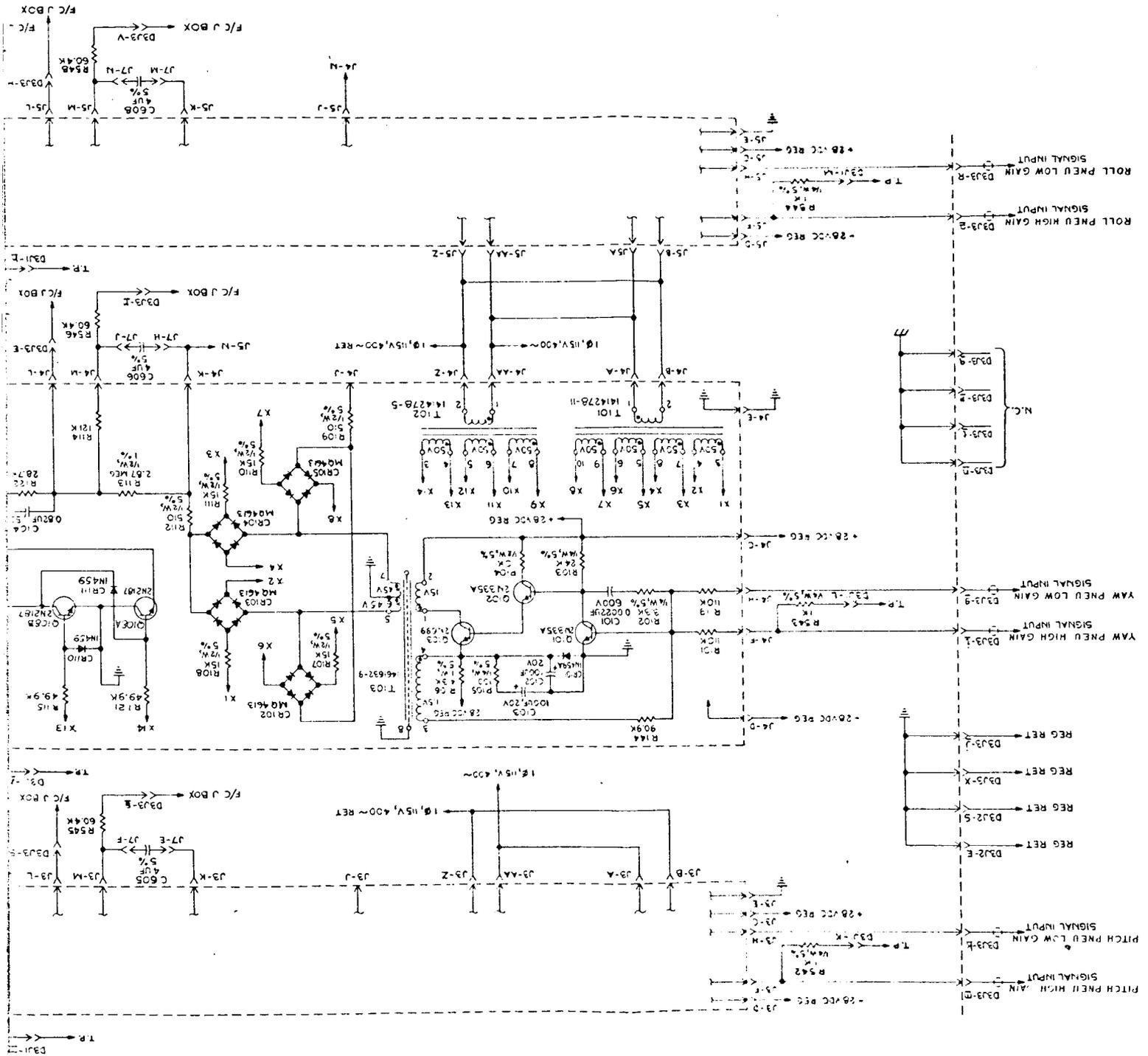
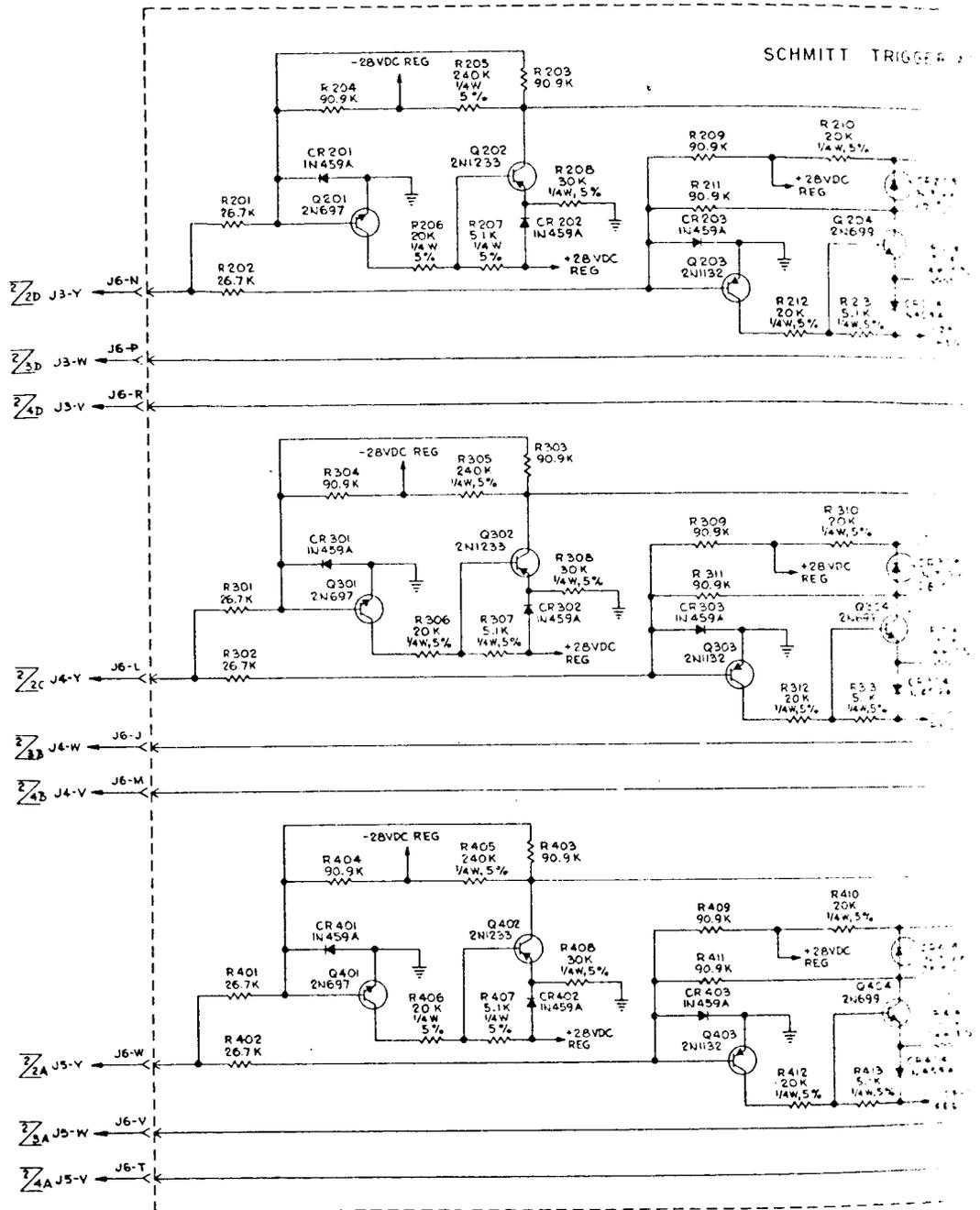


Fig. B-4a Pneumatic Control Channel Schematic Diagram

SCHMITT TRIGGER



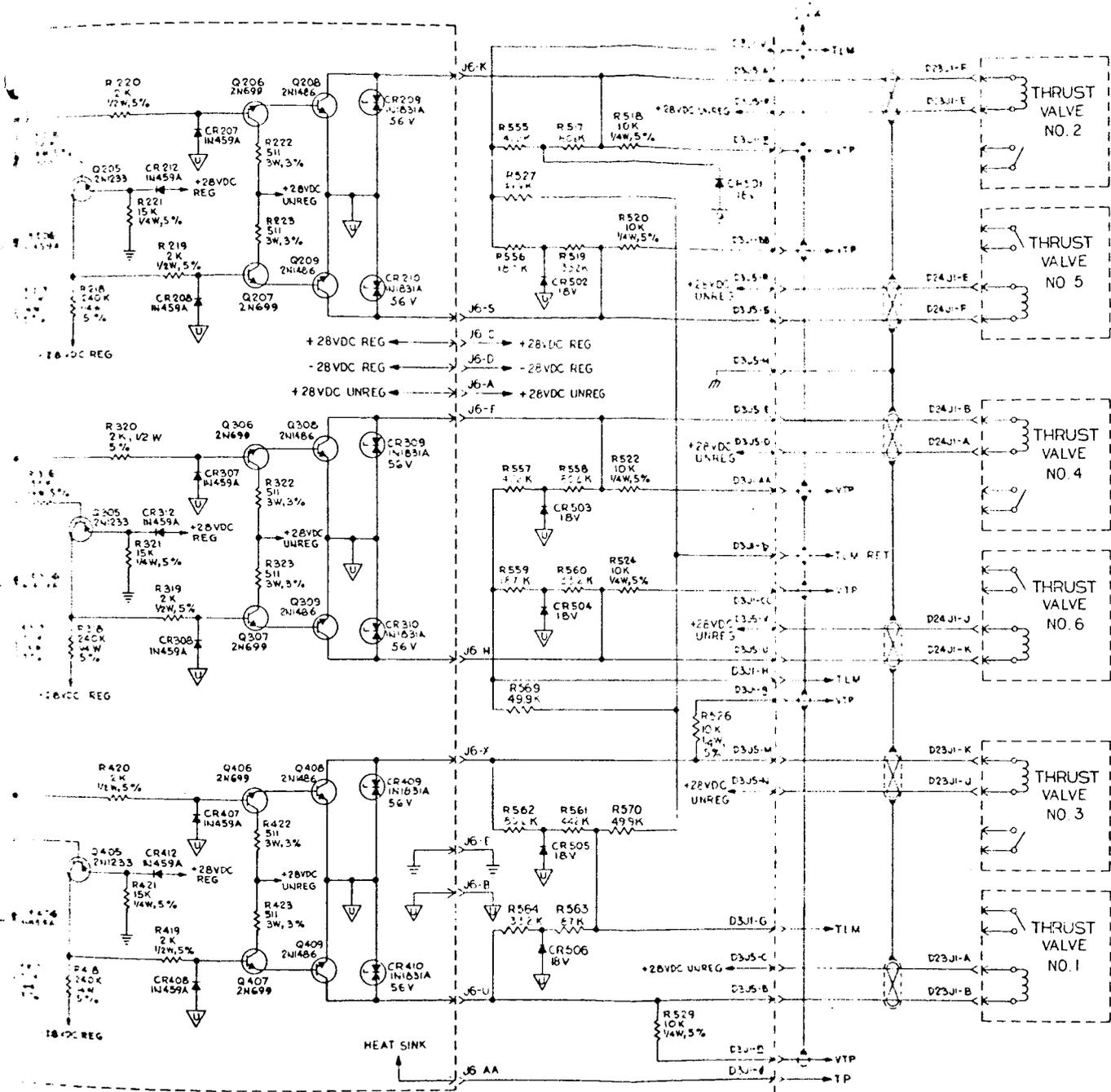


Fig. B-4b Pneumatic Control Channel Schematic Diagram