

FINAL REPORT

FOR

ATS F&G

(PHASES B & C)

(17 OCTOBER 1969)

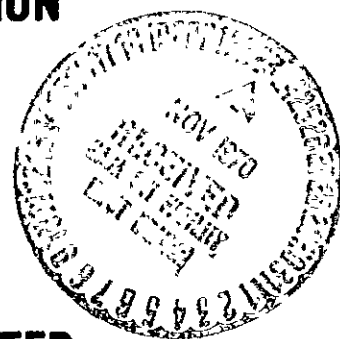
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PREPARED BY

FAIRCHILD HILLER CORPORATION
GERMANTOWN, MARYLAND

FOR

GODDARD SPACE FLIGHT CENTER
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(PHASES B&C)

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Contract Number NAS5-11609

GODDARD SPACE FLIGHT CENTER

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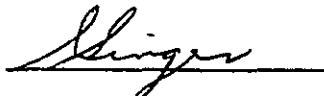
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Greenbelt, Maryland

Approved: 

9.0

ATTITUDE CONTROL SUBSYSTEM

9.1 FUNCTIONAL DESCRIPTION

9.1.1 OBJECTIVES OF THE ATS-F&G MISSION

The attitude control subsystem functional and performance requirements are established by the ATS-F&G mission objectives. These objectives may be briefly stated as follows:

- Demonstrate the feasibility of a deployable, 30-foot parabolic antenna useful to 10 GHz (X-band).
- Demonstrate a spacecraft attitude control capability for directing the antenna beam to any point on the earth's surface, to an accuracy consistent with the smallest antenna beam width and consistent with the angular measurement capabilities of a sensitive radio interferometer.
- Provide a stable oriented platform at synchronous attitude for advanced technology experiments.
- Demonstrate a minimum two-year life capability for a spacecraft of the ATS-F&G size and complexity with an overall objective of maintaining useful performance for five years.
- Demonstrate orbit repositioning and stationkeeping control in synchronous orbit - required for communication and navigation aid purposes.

The requirements imposed on the attitude control subsystem, a description of the functional operation of the subsystem, and the degree to which the subsystem meets the requirements are summarized in the paragraphs below. Details of the technical development and a description of the detailed characteristics of the subsystem components may be found in Volumes I-C, I-D, and I-M of the ATS-F&G proposal.

9.1.2 ATTITUDE CONTROL SUBSYSTEM FUNCTIONAL PERFORMANCE REQUIREMENTS

The 30-foot parabolic reflector antenna to be used on the ATS-F&G satellite is designed to provide a beam width of 0.3 degree or better. To make full use of an antenna of this precision, the spacecraft attitude control

system must be capable of slewing the orbiting spacecraft to direct the antenna to any desired point on the earth and maintain the pointing direction to an accuracy of ± 0.1 degree. The evaluation of antenna degradation with time is accomplished by an ACS-controlled slewing of the spacecraft and antenna in a prescribed pattern with respect to the ground station and plotting the signal strength.

The control system plays an important part in numerous advanced technology operations and experiments. One of these uses a high-resolution camera, which requires a highly stable platform while maintaining a desired pointing direction. The attitude control subsystem design goal for this application is to limit spacecraft angular motion or jitter about the desired pointing direction to angles of ± 0.003 degree. Yet another communication-related experiment involves an ACS-controlled slewing of the spacecraft such that its antenna is tracking a low-altitude orbiting satellite. In other operations and experiments, onboard receivers (interferometer, monopulse, and laser supplied by other contractors) generate attitude error signals with respect to the line-of-sight to a ground transmitting station. This information is then used by the ACS for nulling of the attitude error signal.

In addition to the communications-related experiments, advanced control technology-related experiments are also planned. These will use onboard sensors and torquers, together with ground-based computation, for optimal and self-adaptive control.

An experiment involving the ACS is the commandable gravity gradient subsystem (COGGS). In this experiment, a boom orientation with respect to the spacecraft is commanded so that prescribed torques can be applied to the vehicle.

The combined attitude control/auxiliary propulsion subsystem is required to impart velocity changes to the spacecraft to effect station-keeping and repositioning. These velocity changes will be used to position the spacecraft above South America and Africa during different periods of the mission.

The sensors, controllers, and torquing devices to be used for the ATS-F&G mission must be selected on the basis of demonstrated ability to meet mission requirements.

Satisfying the requirements for a two-year-minimum mission life requires that backup sensors, controllers, and torquers be provided in the attitude control subsystem and that backup modes of operation be available. The backup modes must provide a pointing accuracy of at least ± 0.5 degree in the event of loss of primary control capability.

9.1.2.1 Modes of Operation

The modes of operation which the attitude control subsystem is required to perform may be grouped in two categories, with a number of modes in each:

1) Acquisition modes

Rate damping

- Sun acquisition
 - Earth acquisition
 - Polaris acquisition
- 2) Operational and experimental modes
- Orbit repositioning and stationkeeping
 - Reference orientation (local vertical)
 - Offset pointing (from local vertical)
 - Slew/antenna pattern maneuvers
 - Station pointing (nulling) -- interferometer, mono-pulse, and laser
 - Satellite tracking
 - Low jitter
 - SAPPSAC/SAMOC (self-adaptive precision pointing spacecraft attitude control/spacecraft attitude maneuvering optimal control)
 - Gravity gradient (COGGS)

9.1.2.2

Components Selected for the Attitude Control Subsystem

Design requirements for the ATS-F&G attitude control subsystem have been fulfilled through the use of the following assemblies:

- Two digital operational controllers (DOC) (parallel redundant)
- One analog backup controller (ABC)
- One earth sensor assembly (ESA)
- Four coarse sun sensor (CSS) assemblies
- Two fine sun sensor (FSS) assemblies
- Three sun sensor electronics assemblies (SSEA)
- Two Polaris sensors (PS) (parallel redundant) with external conditioning electronics assemblies
- One inertial reference assembly (IRA)
- One interferometer

- One actuator control electronics (ACE) assembly
- Three inertia wheels
- Auxiliary propulsion subsystem (APS)
- Commandable gravity gradient subsystem (COGGS)

Table 9.1-1 lists the sensors and torquers used, under the control of the digital operational controller, for the several control modes employed.

Table 9.1-1. Summary of ACS Modes

Control Mode	Sensors	Torquers
1) Acquisition/reacquisition Rate nulling Sun acquisition Earth acquisition Polaris acquisition	IRA (derived rate) IRA, CSS, FSS IRA, FSS, ES, or VHF IRA, ES, PS	A/C jets
2) Reference orientation	ES, PS Interferometer, PS	Wheels - A/C jets or jets alone
3) Station point (nulling)	 Interferometer, PS Monopulse (X, S or VHF), PS Laser, PS	Wheels - A/C jets or jets alone
4) Offset point	ES, PS Interferometer, PS	Wheels - A/C jets
5) Low jitter	ES, PS	Wheels
6) Antenna pattern slew maneuvers	ES, PS Interferometer, PS	Wheels - A/C jets or jets alone
7) Satellite tracking	ES, PS S-band monopulse, PS	Wheels - A/C jets or jets alone
8) Gimbale gravity gradient	Same sensor combina- tions as operating modes 2 and 3	Wheels - boom
9) Orbit control	ES, PS Interferometer, PS	A/C jets alone

Note: IRA = inertial reference assembly; CSS = coarse sun sensor;
FSS = fine sun sensor; ES = earth sensor; PS = Polaris sensor,
A/C jets = attitude control jets.

During Phases B-C, the design effort for the above assemblies followed the general outline of the proposed program work statement, using as a departure point the preferred approach for the attitude control subsystem as described in GSFC document entitled "Preferred Approach for ATS-F&G" dated 20 November 1967, and the subsequent revised preferred

approach as described in Fairchild-Hiller Report No. ATS-990-001 entitled "Assessment of the Preferred Approach" dated 17 December 1968. Several modifications to that approach were developed during the program as a result of analyses made of alternate subsystem configurations involving design trade-offs of reliability, cost, size, weight, and power. Principal changes made include the use of parallel redundant digital operational controllers, parallel redundant star sensors, and the substitution of analog sun sensors for the digital units originally designated.

One of the most important tasks in designing the ATS-F&G attitude control subsystem is that of selecting an earth horizon sensor which will make possible the achievement of an overall spacecraft pointing accuracy of 0.1 degree. Honeywell has conducted extensive horizon definition studies in connection with the design of control systems for orbital and suborbital horizon scanning satellites. These studies indicated that a sensor designed to detect radiation in the CO₂ band (14- to 16-micron range) provided the best overall performance at low (approximately 150 nm) satellite altitudes. The earth's radiance profile at synchronous altitudes, however, is sufficiently different from that seen at low altitudes to require further earth sensor development for the ATS-F&G mission. This was accomplished during Phases B-C and resulted in the modification of a commercially available and space-application-proven sensor to detect radiation in the 13- to 25-micron range which will assure the achievement of the required spacecraft pointing accuracy by the specified attitude control subsystem.

Design fabrication of the other major attitude control subsystem component assemblies falls into three categories:

- 1) The favored star sensor, sun sensors, and inertia wheels are all modifications of commercially available units with proven performance in aerospace applications.

- 2) The digital operational controller, analog backup controller, and actuator control electronics are all designed and built by Honeywell. Circuit and packaging designs are based on those used in Honeywell-built devices proven in many aerospace applications.

- 3) The inertial reference assembly is a standard Honeywell built device built for the P95 program. Significant modification to this device will not be necessary for the ATS-F&G applications.

A simulation of the attitude control subsystem using sensors, digital operational controller, and torquers similar to those FHC/HI recommends for the ATS-F&G was conducted during Phases B-C. A demonstration of this simulation with the components mounted on the Honeywell six-foot-diameter air bearing table was given to representatives of Goddard Space Flight Center, NASA-Headquarters, and other agencies.

9.1.3 REVIEW OF ATTITUDE CONTROL SUBSYSTEM FUNCTIONAL OPERATION

The following paragraphs summarize the operation of the overall attitude control subsystem and its constituent subsystem components. Control action is performed with respect to the ATS reference coordinate

system as illustrated in Figure 9.1-1. This reference coordinate system is defined as follows:

- 1) X-axis positive in the direction of flight and in the orbit plane (roll)
- 2) Z-axis positive toward the earth and coincident with local vertical (yaw)
- 3) Y-axis perpendicular to the X- and Z-axes (pitch)

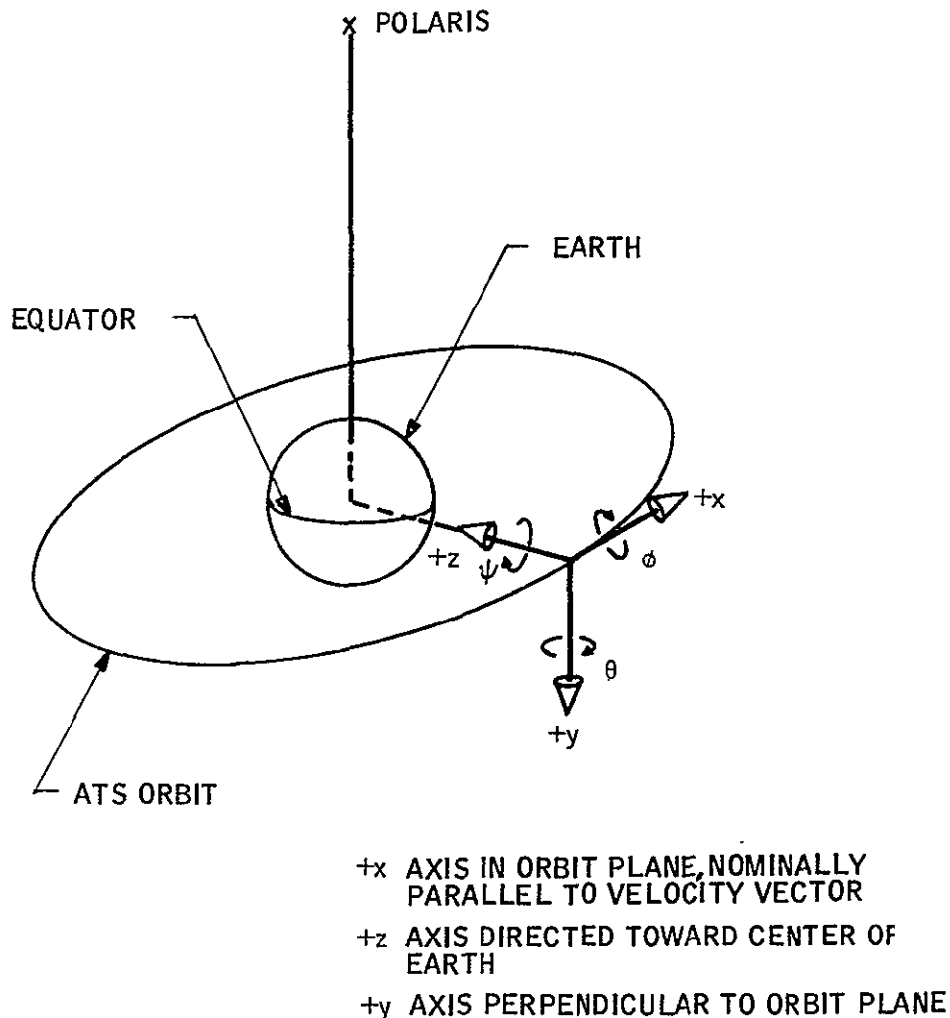


Figure 9.1-1. ATS Reference Coordinate System

9.1.3.1 Overall Subsystem

A block diagram of the acquisition control modes is shown in Figure 9.1-2. The sensors used are shown in their order of use. The IRA provides rate damping signals. The coarse and fine sun sensor signals are used to acquire the sun. The earth sensor is used for earth acquisition and provides the desired local vertical reference (roll/pitch) for subsequent

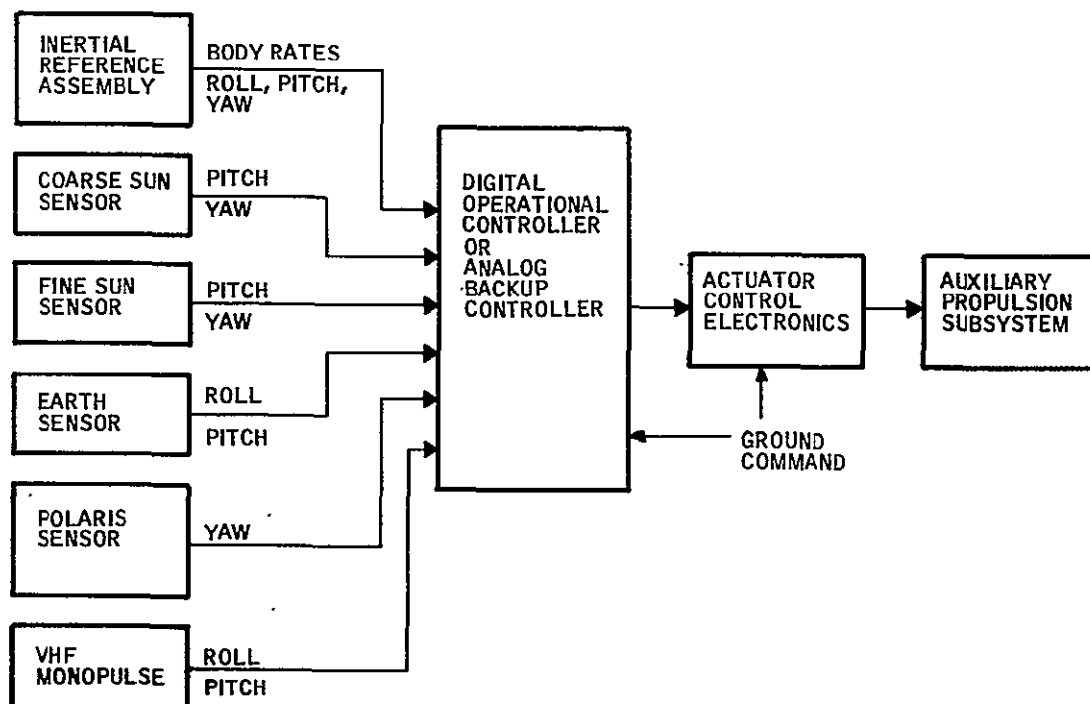


Figure 9.1-2. Acquisition Control Modes Block Diagram

operational control modes. The Polaris sensor is used for Polaris acquisition and subsequently provides a reference for the control of the yaw axis. The VHF monopulse provides a backup capability for roll and pitch acquisition of the earth. A standby redundant Polaris sensor provides for a yaw backup. The acquisition modes use the auxiliary propulsion subsystem for torquing the spacecraft. Backup control modes at reduced power levels include an analog backup controller and ground command of the wheels/jets based on telemetered sensor data.

The block diagram of Figure 9.1-3 shows the primary operational mode control modes and the modes in which each of the sensors may be used. Only one digital operational controller is shown; however, a duplicate set of interfaces is provided so that either controller may perform the functions. The inertia wheels are the prime means for torquing the spacecraft, with the auxiliary propulsion subsystem providing torques for unloading the wheels. The APS is also capable of performing all modes except low jitter without the wheels. For orbit control, the APS is the prime means of producing torque. Backup control modes at reduced power levels include an on-board analog backup controller and ground command of the wheels/jets based on telemetered sensor data.

A block diagram for the experiment modes is shown in Figure 9.1-4. These are the experiments which have a direct interface with the ACS. The gravity gradient experiment will use the DOC to compute the control laws required for unloading the wheel or torquing the vehicle. The SAMOC/SAPPSAC experiment bypasses the DOC. The attitude information from the earth sensor and Polaris sensor is telemetered to the ground for this mode. Computations are made in a ground computer and commands are sent directly

to the actuator control electronics which interface with the APS to provide spacecraft control torques. The multistation interferometer also makes use of a ground loop to provide attitude error commands.

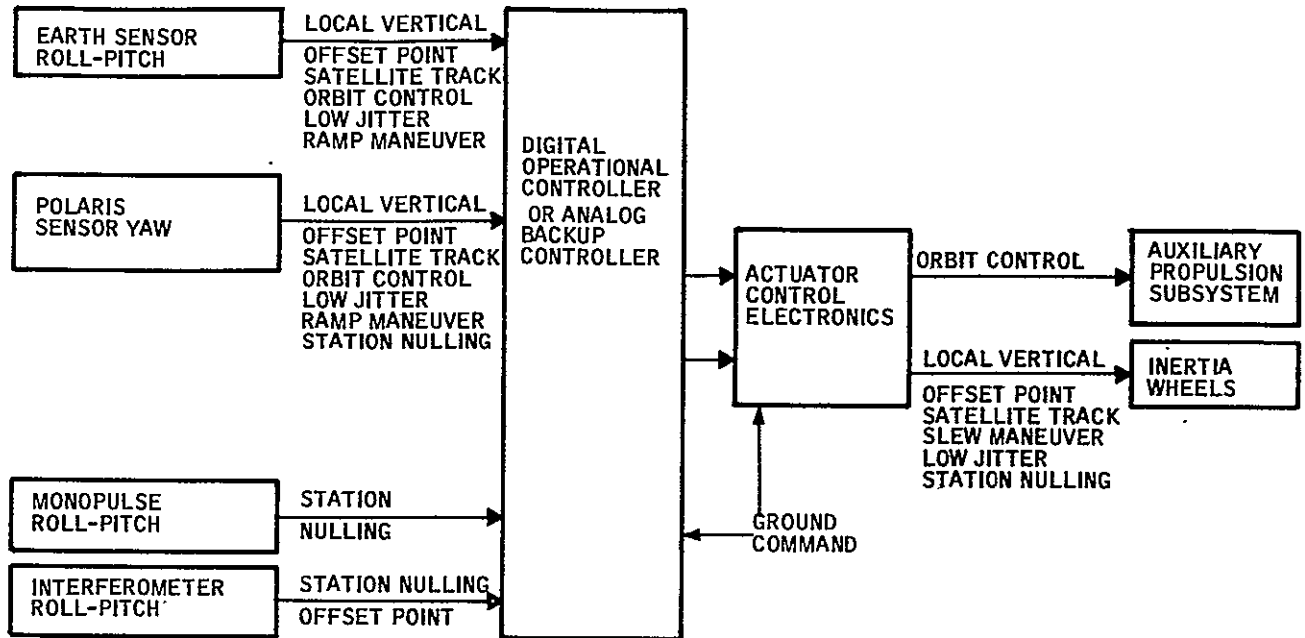


Figure 9.1-3. Operational Control Modes Block Diagram

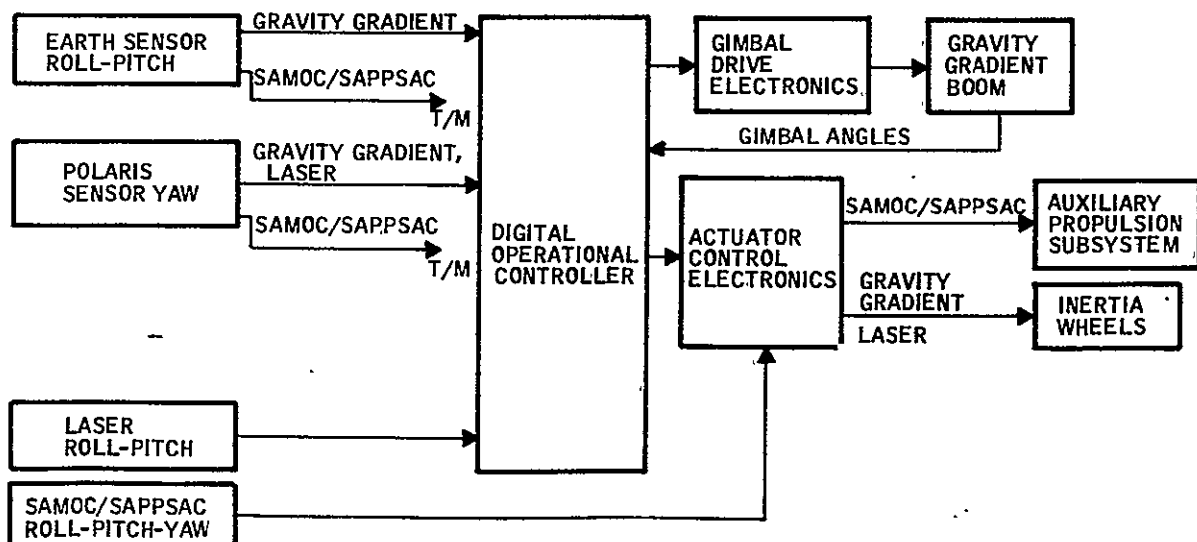


Figure 9.1-4. Experiment Modes Block Diagram

9.1.3.2 Controllers

9.1.3.2.1 Digital Operational Controller

Two digital operational controllers are provided in the attitude control subsystem. The ground will choose the controller to be used. The digital operational controller serves as the prime controller for all modes of operation. In addition to performing control loop computations for stabilizing the spacecraft, the DOC provides the following functions:

- 1) Executes acquisition logic to acquire the sun, earth, and Polaris
- 2) Accepts commands for offset pointing (angle and ground coordinate commands)
- 3) Computes compensations required for pointing commands to account for orbit eccentricity, orbit inclination, and Polaris diurnal motion
- 4) Computes commands for performing satellite tracking
- 5) Computes commands for performing slew maneuvers for antenna pattern measurements
- 6) Provides momentum wheel unload signals
- 7) Provides necessary A/D and D/A conversion
- 8) Provides a self-test routine within the DOC

The basic DOC consists of a power supply, central processor, a memory, and an input/output section. The DOC provides for block transfer data to the telemetry data acquisition and control unit (DACU) or complete data transfer from memory on command from the ground.

The heart of the DOC is the central processor which performs programs stored within the memory. The input/output (I/O) electronics accepts both analog and digital signals and can supply monitoring signals to telemetry, and control signals to sensors and to the ACE for controlling the wheels and jets.

The favored DOC is a suitably packaged version of the Honeywell SDC III computer which was designed for space systems applications. It represents experience Honeywell has gained in developing control systems for Mercury, Gemini, Apollo, and several classified programs. The SDC III is a compact, low-power, high-performance, 16-bit binary processor. It has a memory capacity expandable up to 16K words. The memory is the plated-wire-type which Honeywell has been developing and refining since 1963.

The present design incorporates many parts that have been qualified or are presently being qualified for programs requiring high-reliability components including the plated-wire memory. This development has been supported by several government agencies.

9.1.3.2.2 Analog Backup Controller

The analog backup controller serves as a backup controller to the DOC in certain modes at reduced power levels. The ABC provides the following functions:

- 1) Stabilizes the spacecraft
- 2) Acquires the sun, earth, and Polaris with aid from the ground
- 3) Maintains local vertical hold mode
- 4) Maintains a stationpointing mode

The ABC has no digital interfaces. The basic control loop will consist of switching amplifiers with lag feedback compensation to obtain rate information (when not using gyros) for stabilizing the loop.

The ABC is a simple analog control unit. It consists of a power supply, mode logic, and switching amplifiers. The packaging is designed for the ATS-F&G. However, the planned circuitry uses techniques which have been proven in Mercury, Gemini, Apollo, and many other space applications. Use of high-reliability qualified parts will provide a highly reliable unit.

9.1.3.3 Sensors

9.1.3.3.1 Coarse-Null Sun Sensors

Four coarse-null sensors are provided, offering two-axis sensitivity. Each sensor consists of silicon photovoltaic diode eyes. Each eye provides an analog output signal related to the angle between the sun line and a vector normally incident on the sensor. When used in conjunction with the electronics, the output from two differentially connected eyes will provide a null when the sun is at equal angles relative to each of them.

The functions of the coarse-null sun sensors are to enable the spacecraft $\pm X$ -axis to be brought to within ± 8 degrees of the sun line, where the fine sun sensor can take over control of the spacecraft.

The coarse sun sensors are located on the hub and the earth viewing module (EVM), providing a 4π steradian field-of-view (Figure 9.1-5). The yaw coarse sun sensor field-of-view experiences some blockage by the solar panels, requiring two additional coarse eyes on the hub to provide for sun line location capability throughout the yaw plane.

The coarse sun sensors, through their electronics, must interface with the DOC and ABC for onboard use and with the DACU for telemetry to the ground.

The photovoltaic eyes considered for this application are the C-105 coarse solar sensors manufactured by Ball Brothers Research Corporation and used on the Orbiting Solar Observatories for the past seven years.

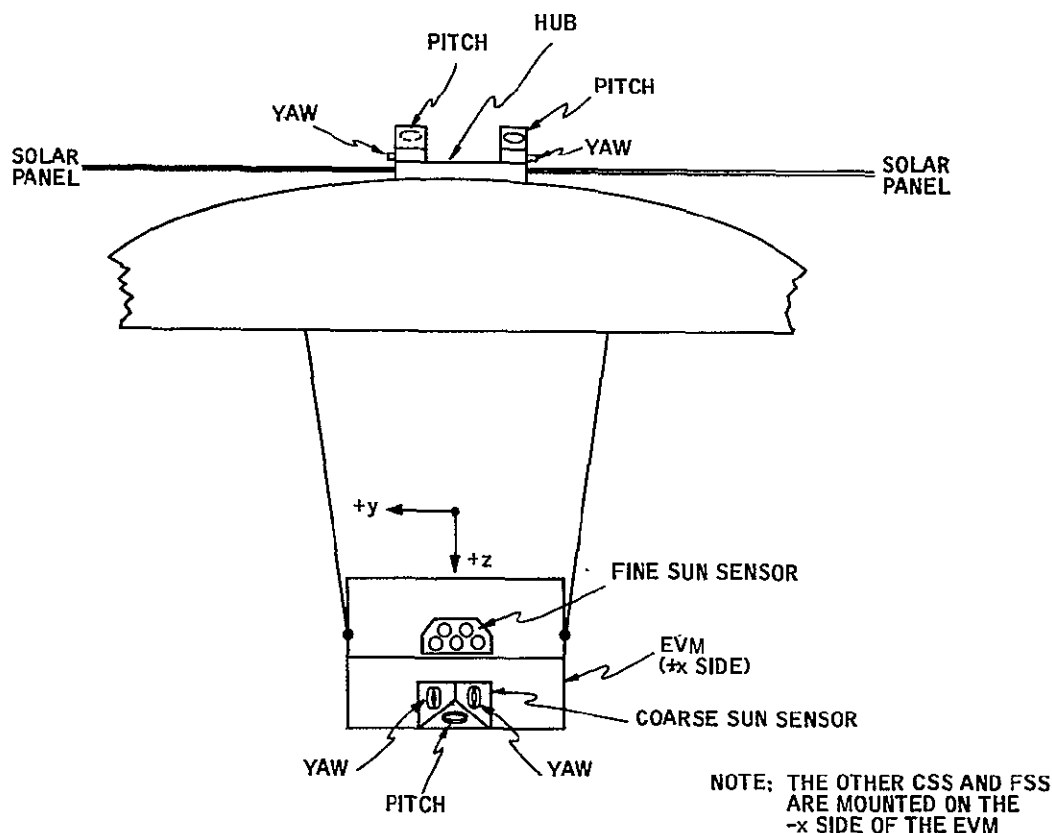


Figure 9.1-5. Sun Sensor Configuration

They have also been used on Aerobee rockets. They are simple, inexpensive sensors which have been well proven in space applications.

9.1.3.3.2 Fine-Null Sun Sensor

Two fine-null sun sensors are provided, each with two-axis sensitivity. Each sensor consists of four silicon photovoltaic diode eyes which are interconnected to obtain a null output when the sensors are pointing to the sun. In addition, a target eye is included to provide an indication to the controllers for switching control from the coarse sensors to the fine sensors.

The sensors are mounted on the EVM, with one sensor's null axis coinciding with the +X-axis and the other with the -X-axis (Figure 9.1-5). The fine sun sensors, through their electronics, must interface with the two DOCs and the ABC for onboard use and with the two DACUs for telemetry to the ground. The fine sun sensor considered for this application consists of the F-125 fine solar sensor eyes manufactured by Ball Brothers and used in their SS-100 series. An available (catalog) unit shows a nominal linear range of 5 degrees and saturated level to 10 degrees, with the output dropping to zero beyond 15 degrees. (Recent data from Ball Brothers on an SS-109-type sensor shows a linear range of ± 12 degrees to as much as 10 degrees off-axis.) A target-type (T-120) is used in conjunction with the fine eyes to provide an indication when control should be switched to the fine eyes. The target eye has a field-of-view of approximately ± 8 degrees. The fine sun sensor can provide an accuracy of ± 2 arc minutes. Since this accuracy is not

essential to ATS performance, it may be possible to extend the field-of-view with some reduction in accuracy.

9.1.3.3.3 Earth Sensor Assembly

The earth sensor is used for earth acquisition during the acquisition modes and serves as the primary roll/pitch sensor for all operational modes. The earth sensor assembly consists of two sensing heads (one roll, one pitch) plus an electronics unit. Each head contains a sensing element, electronics, scan mirror, and an offset mirror. The scan mirror is driven back and forth to provide a scan amplitude of ± 13 degrees. An encoder is mounted on the mirror to indicate center crossing and provide pulses to a counter. Earth radiance signals gate the pulses to the counter. The scan mirror is biased by the attitude angle signals to keep the scan centered on the earth. The roll offset mirror is biased by the pitch attitude angle signals to keep the roll scan near the full diameter of the earth. Likewise, the pitch offset mirror is biased by the roll attitude angle signal to keep the pitch scan near the full diameter.

The earth sensor attitude angle output is formed by scanning the instantaneous field-of-view of the bolometer telescope assembly across the earth's disk. During the scan, encoder pulses representing incremental angle changes are gated into an up/down counter in the earth sensor electronics based on the following events:

- Space-to-earth crossing
- Center crossing on the encoder (Z-axis null plane)
- Earth-to-space crossing

The attitude output is determined after the scan has completed a full cycle. Thus, with a scan frequency of 5 Hz, new outputs are available every 0.2 second. This output is available as a digital output. An analog signal is also available as a result of the signals generated to bias the scan and offset mirrors.

The attitude output of the earth sensor is supplied to the two DOCs (digital) and ABC (analog), as well as the two DACUs (digital) for telemetering to the ground.

The ATS earth sensor assembly provides a high-accuracy means of determining pitch and roll attitude with respect to the local vertical vector. The ATS sensor represents a significant improvement in the state of the art in scanning earth sensors. The fundamental design of the favored TRW sensor has been flight-proven on the Vela program at an accuracy of ± 0.1 degree, for a mission duration equivalent to ATS. This second-generation sensor, capable of ± 0.05 -degree accuracy over the mission life, has been made possible through improved radiance gain electronics, product improvements in detectors, and a simple, rugged, scan mirror assembly. The design and performance analysis has been guided by worst-case assumptions relative to the earth model which are significantly more severe than the expected nominal earth radiance levels.

9.1.3.3.4 Polaris Sensor

Two Polaris sensors and their respective conditioning electronics are provided. The ground will choose which sensor is to be used. The Polaris sensor is used during the Polaris acquisition mode and serves as the primary yaw sensor for all operational and most experimental modes. The Polaris sensor responds to irradiation from Polaris and provides an analog output voltage proportional to the angle between the line-of-sight to Polaris and the optical null plane of the tracker. The sensor has a linear range of ± 3.5 degrees in the sensitive axis. The field-of-view in the nonsensitive direction is greater than ± 14.0 degrees. To make use of the full tracking field, the sensor has the capability, on command, of electronic gimbaling of the instantaneous field-of-view to retain the star in the field-of-view during roll maneuvers.

A sun shutter is incorporated as a protective device which inhibits the input radiant energy to the sensor when an intense light source is in the field-of-view. A solar detector is incorporated to automatically actuate the shutter. The baffle provides for attenuation of stray light due to reflections and tracking near the sun.

The Polaris sensor interfaces with both DOCs, the ABC, and the two DACUs.

The Polaris sensor baseline favored for this application is the JPL Mariner Mars '69 Canopus tracker. Laboratory tests have shown that the tracker currently has the ability to track a star of lesser brilliance than Polaris. To provide further design margin, some minor design modifications are proposed. They include:

- 1) Increasing the gain of the preamplifier
- 2) Lowering dynode voltage to decrease noise
- 3) Increasing light transmission of the optics

9.1.3.3.5 Inertial Reference Assembly

The three-axis inertial reference assembly is used during the acquisition mode and acquisition of Polaris. It can also provide some temporary attitude hold functions. The IRA consists of three gas bearing spinmotor gyros, with associated electronics, which provide output pulses corresponding to incremental body angle changes about each of the three axes. By counting the pulses, the total attitude can be calculated. The DOC also uses these pulses in its computations to provide body rate signals for rate damping and acquisition modes. The IRA also provides an analog output signal for ease of interface with the ABC. Filtering of this signal is accomplished in the ABC.

The electronics for the IRA include:

- A power supply which converts input power to supply voltages required for proper operation.

- Four temperature control amplifiers, one for the oven and three for the gyros. Although it is intended to operate the gyros without application of heater power for the nominal temperature ranges encountered during orbit, they are in the present design. There is also a possibility that there may be conditions during the mission where a fast warmup capability is desired.
- Three pulse-rebalance loops. These electronics provide the incremental pulse output. The IRA provides an analog rate output to the ABC and DACU. The attitude outputs will be transmitted to ground as a part of the data block transfer from the DOC.

The three-axis IRA proposed for use in the ATS-F&G attitude control subsystem is an adaptation of a unit presently in production. A production prototype was subjected to acceptance testing and was delivered to the customer on 1 December 1968. A design-proof test program has been completed. The first qualification unit is now in acceptance testing.

9.1.3.3.6 Interferometer

When illuminated by RF energy from a ground transmitter, the interferometer will measure the direction cosines between the spacecraft roll and pitch axes and the line-of-sight vector from the spacecraft to the transmitter. Digital outputs representing these two direction cosines are provided to the DOC and to telemetry, covering the ± 17.5 -degree roll/pitch angle range of the interferometer. When two or more ground transmitters are used, all three spacecraft attitude angles can be computed. The latter mode is implemented via telemetry and ground computation only.

A detailed description of the interferometer subsystem is provided in the Interferometer Subsystem Report, Volume I-K.

9.1.3.4 Torquers

9.1.3.4.1 Actuator Control Electronics

The actuator control electronics will be energized at all times for all modes of operation. They serve as the power amplifier between the controllers and the inertia wheels or the APS. The electronics for the variable-frequency a-c wheels consist of an integrator, voltage-to-frequency converter, and power amplifier. The output to the wheels is provided at a frequency which changes at a rate determined by the magnitude of the input signal. The voltage level of the a-c output is varied to maintain a constant volts-per-cycle ratio except near the low-speed range where the ratio is increased. Outputs are supplied to both phases of the motor.

The electronics for the APS consist of the logic to accept the jet firing commands from the two DOCs, the ABC, the two command decode and distribution (CDD) units, and the power drives to energize the appropriate

solenoids in the APS for attitude control. The electronics also provide the power required for switching the latching valves in the APS. Additional electronics are required to supply heater and solenoid power for the orbit control thrusters in the APS.

The ACE provides outputs to 12 attitude control thrusters and 4 orbit control thrusters. The wheel drive section has outputs to three inertia wheels. The ACE also processes the wheel tachometer signals for telemetry to ground. Although the ACE is a new package, the circuitry and packaging techniques used are the same as used on previous space programs such as Gemini, Apollo, and Mariner Mars.

9.1.3.4.2 Inertia Wheels

The inertia wheels serve as the prime torquers for all modes except acquisition, orbit control, and SAMOC/SAPPSAC. Three wheels form a part of the ACS (one roll, one pitch, and one yaw). A pulse-type tachometer is provided for use by the DOC to obtain spacecraft rate information for the control loop. The tachometer output is also used to indicate the need for inertia wheel unloading.

The inertia wheels have an a-c, low-resistance cage motor which is driven by a variable frequency and voltage source. The voltage is reduced as the frequency is decreased. The drive electronics provide an output whose frequency changes at a constant rate which the wheel will follow. This provides a constant acceleration, resulting in a torque which reacts against the spacecraft to rotate it.

Each wheel is designed to provide 0.03 ft-lb or torque at the following momentum storage levels:

- Roll: 3.7 ft-lb-sec
- Pitch: 4.7 ft-lb-sec
- Yaw: 3.7 ft-lb-sec

The roll wheel requirement was determined to be 3.2 ft-lb-sec. However, it was increased to 3.7 ft-lb-sec to be the same design as the yaw wheel.

The wheels are packaged in a hermetically sealed container.

The basic mechanical structure and motor considered for the ATS is that which has been designed for the OGO roll and pitch wheel. Some winding modifications would be made to eliminate the need for transformers. The copper alloy bars in the rotor would be changed to pure copper to achieve the desired motor characteristics.

9.1.3.4.3 Auxiliary Propulsion Subsystem

The auxiliary propulsion subsystem provides thrust of suitable magnitude and direction to torque the spacecraft about the three axes for attitude control and thrust to impart a change in orbital velocity of the spacecraft for orbital position control in four directions.

The APS is an ammonia system which operates on ammonia gas at 40 to 100°F for attitude control and high-temperature resistojets for orbit control. It consists of an ammonia propulsion assembly containing two (standby redundant) regulated pressure feed assemblies, 12 unheated attitude control thrusters, and two resistively heated resistojet orbit control thruster modules, all controlled by electronics in the actuator control electronics.

The regulated pressure feed assembly contains tankage for propellant storage, an evaporator or preplenum for conversion of propellant from liquid to gas, a plenum chamber for gas storage which is approximately 20 times the volume of the preplenum chamber, a regulating solenoid valve, a latching on-off solenoid selector valve, a temperature sensor transducer, a pressure sensor transducer, an electrical control with feedback to maintain the regulated set feed pressure, and distribution tubing with latching valves for controlling propellant supply to several groups of thrusters.

The 12 unheated attitude control thrusters are mounted on the perimeter of the EVM, with the solenoid valves located on the feed assembly. The two heated resistojet orbit control thruster assemblies each contain two jet nozzles with solenoid valves located in the feed assembly and electrical heaters immediately upstream of each jet nozzle.

A more complete description of the APS is contained in the Auxiliary Propulsion Subsystem, Section 12.0.

9.1.3.4.4 Commandable Gravity Gradient Subsystem

As a part of the ACS, the commandable gravity gradient subsystem is included as a technology experiment that will evaluate, in orbit, the use of a two-axis gimbale gravity gradient boom to supply torques for attitude control of the spacecraft.

The COGGS consists of a mechanical assembly (boom, gimbal, and tip-mass deployer) and an electronics package to provide the necessary interface with the ACS. The mechanical assembly is mounted on top of the reflector hub. The tip mass is deployed away from the earth when this experiment mode is initiated. The boom is gimbale in roll and pitch and driven by roll/pitch torquers. Attitude control is achieved by torquing against the inertia of the boom, based on gimbal command signals provided by the DOC. Attitude control in yaw is accomplished by the DOC using inertia wheel control. The roll and pitch wheels are used only to provide loop damping torques.

The COGGS will interface with the DOC in a mode that uses the earth sensor and Polaris sensor as a source of attitude information. Off-set and maneuver commands will be computed in the normal method. The control loop will be modified to provide torquer commands to the gimbals and damping commands to the roll and pitch inertia wheels. Optical encoders will provide gimbal angle information to the DOC.

9.1.3.5 Backup Control Operation

As previously stated, the attitude control subsystem is equipped to provide backup modes of control in the event of the malfunction of

primary component assemblies. Table 9.1-2 summarizes the options available.

Table 9.1-2. Backup Capability -- Operational

Sensors		Controllers	Torquers
Roll/Pitch	Yaw		
Earth sensor	Polaris sensor 1	Digital operational controller 1	Wheels - APS 1
Interferometer	Polaris sensor 2	Digital operational controller 2	Wheels - APS 2
Monopulse (X, S, VHF)			APS 1 APS 2
Earth sensor	Polaris sensor 1	Analog backup controller	APS 1
Monopulse (X, S, VHF)	Polaris sensor 2		APS 2
Earth sensor	Polaris sensor 1	Ground computation and command	Wheels - APS 1
Interferometer	Polaris sensor 2		Wheels - APS 2
Monopulse (X, S, VHF)			APS 1 APS 2

9.1.4 PERFORMANCE VERSUS REQUIREMENTS

9.1.4.1 Overall Attitude Control Subsystem

The performance requirements imposed on the attitude control subsystem and the performance capabilities of the subsystem design for the various modes of operation are summarized in Table 9.1-3. In addition to the quantitative items listed in this table, the subsystem design incorporates the following features:

- Provides the mechanism for a gravity gradient experiment
- Provides the capability of six months operation on jet only backup control
- Provides operational capability for a minimum two-year mission with a five-year design goal

The orbit control characteristic velocity and impulse requirements are shown in Table 9.1-4.

An additional 2600 lb-sec are required to meet attitude control requirements as established by analysis and simulations of the various control modes for a two-year mission for the design weight spacecraft, and 2900 lb-sec for the maximum weight spacecraft for which the APS propellant tanks are designed.

Table 9.1-3. ACS Requirements/Performance Summary

Mode of Operation	Requirement	System Performance
Damp initial body rates of 1 deg/sec	± 0.05 deg/sec 10 minutes	± 0.05 deg/sec 3.5 minutes
Acquire sun	± 1 deg of sun line 20 minutes to 30 degrees of sun line 30 minutes to 1 degree of sun line	1 deg 11.0 minutes 15.0 minutes
Acquire earth	0.2 deg/sec search rate ± 0.3 deg of local vertical 80 minutes to local vertical	0.25 deg/sec 0.3 deg 27.7 minutes
Acquire star	± 0.5 10 minutes to 0.5 degree of star	0.5 deg 5.8 minutes for solstice
Static pointing	± 0.1 degree in roll and pitch ± 0.1 degree in yaw Offset angle of ± 10 degrees roll and pitch	0.08 deg 0.08 deg* ± 10 deg
Satellite tracking	Peak value ± 11 degrees Frequency 0.07 rad/min 0.5-deg peak error	± 11 deg 0.07 rad/min 0.3 deg
Slew maneuver	Amplitude - Accuracy (deg) (deg) ± 5.0 0.9 ± 15.0 1.55 ± 3.0 0.45 ± 2.7 0.42 ± 2.2 0.33 ± 4.0 0.2 ± 3.0 0.2	 0.2 0.2
Low jitter	0.003 deg (design goal) 0.0003 deg/sec (design goal)	0.003 deg 0.0003 deg/sec
Orbit control	0.5 deg	0.5 deg

*Not for end-of-life.

Table 9.1-4. APS Orbit Control Requirements

Item	Characteristic Velocity (ft-sec)	Impulse (lb-sec)	
		Design S/C (2000 lb)	Max Wt. S/C (2450 lb)
Initial orbit adjust	30	1875	2285
East-west stationkeeping (2 years)	14	870	1070
North-south stationkeeping (1 month)	14	870	1070
East-west station repositioning	60	3750	4570
Additional orbit maneuvers	10	635	805
Total	128	8000	9800

9.1.4.2 Spacecraft Characteristics

The spacecraft inertias and jet lever arm characteristics used in the ACS stability and performance analyses and simulations reported herein are:

<u>Item</u>	<u>Stowed</u>	<u>Deployed</u>
Spacecraft weight (lb)	2000	2000
Moments of inertia (slug-ft ²)		
Roll	2860	6690
Pitch	2830	3650
Yaw	580	3560
Jet lever arm (ft)		
Roll	5.75	7.0
Pitch	5.75	7.0
Yaw	3.75	3.75

Recently updated inertias and jet lever arms were used for determining APS and wheel sizing. The new values for the deployed condition are:

	<u>Inertia (slug-ft²)</u>	<u>Moment arm (ft)</u>
Roll	7842	5.61
Pitch	4207	5.61
Yaw	4224	2.00

9.1.4.3 Disturbances

The dominant disturbances sources for the ACS are summarized in Table 9.1-5.

Table 9.1-5 ACS Dominant Disturbances

<u>Item</u>	<u>Roll</u>	<u>Pitch</u>	<u>Yaw</u>
Initial tumbling rates (deg/sec)	1.0	1.0	1.0
Maximum solar pressure torque (10 ⁻⁴ ft-lb)			
Equinox	±0.38	±2.6	-0.32 to +0.18
Solstice	-0.29 to -0.92	±2.2	-0.11 to -0.26
Cyclic momentum-solar torque (ft-lb-sec)	3.2	4.7	3.7
Gravity gradient torque (10 ⁻⁶ ft-lb)	1.3	4.5	---
(-6 deg offset pitch and roll)			
Radiometer mirror maximum torque (10 ⁻³ ft-lb)	2.23	---	---
(scan frequency 0.5 Hz)			

9.1.4.4 Components

The requirements imposed on each of the attitude control subsystem components and the degree to which these requirements are fulfilled by the components selected are summarized in Tables 9.1-6, 9.1-7, and 9.1-8 for controllers, sensors, and torquers, respectively.

9.1.5 INTERFACES BETWEEN ACS AND OTHER ATS SUBSYSTEMS

For the ACS to perform its many control modes and functions, numerous interfaces with other spacecraft equipments are required. Such equipments include:

- Power subsystem
- Command subsystem
- Telemetry subsystem
- Gravity gradient experiment

The power subsystem supplies the necessary voltage and power to operate the ACS. Power interfaces are shown in Figure 9.1-6.

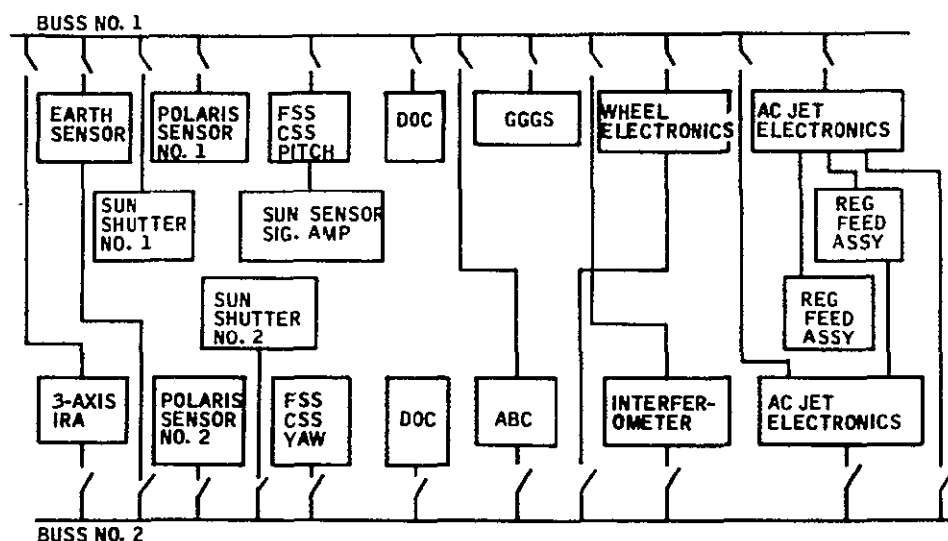


Figure 9.1-6. Power Distribution for ACS

To maintain control of operating modes and functions, the ground must have a command capability. This capability is provided through the command decode and distribution unit. Commands are either discrete or nondiscrete, depending on whether the command is to switch modes or to supply maneuver data for execution in the DOC. The data input to the DOC will be in data block form, each block containing a number of data words.

To provide the ground with information relative to system performance requires that the ACS interface with the data acquisition and

Table 9.1-6. Controller Requirements/Performance Summary

Controller/Parameter	Requirement	Performance
1) Digital operational controller a) Function	Serve as prime controllers for all modes of operation.	Provided
b) Orbital defect compensation	Compensate for pointing errors due to orbital inclination, eccentricity, and star diurnal motion. Uncompensated errors can be as large as: Pitch: 0.7 deg Roll: 0.8 deg Yaw: 2.8 deg	Pointing errors are reduced to less than 0.01 deg in all three axes.
c) Computational errors	Errors due to quantization and computation: • For static modes: <0.01 deg • For maneuvers: <0.03 deg	Word length: • Telemetry and commands: 9 bits • Input/output: 13 bits • Computations: 16 bits (Double precision arithmetic is available.)
d) Computation rate	Rate sufficient to maintain errors less than noted above.	Normal: 100 msec Low jitter: 60 msec
e) Time reference	Pointing errors due to timekeeping shall be less than 0.005 deg.	Internal real-time clock updated daily (for worst-case orbit) by ground command.
f) Memory storage	Estimated program storage requirement is 3000 words.	Up to 8192 words can be provided within DOC package design.
g) Telemetry data	Provide data at a rate consistent with telemetry capability.	Rate: • Normal, 1 word every 3 sec • Dwell, 22.5 msec per word Format: Parallel digital, 9 bits per word. Transmission identification words are followed by data.
h) Command data	Receive data consistent with command capability.	Rate: Up to 2 words/sec. Format: Serial digital, 9 bits of data followed by 6-bit identifier.
i) Function generation	Special functions shall be generated to: • Track substellites with altitude to 600 nm • Slew vehicle in maneuver to enable plotting of antenna characteristics • Orbital defect compensation.	Computed functions include: • Track Ephemeris of satellite or point related to two satellites must be known • Slew: 6 maneuver cases including different excursion limits about antenna boresight, attitude control accuracy, and time period • Point: Vehicle pointing is compensated for orbit defects and Polaris apparent motion.
j) Acquisition	Provide reference coordinate orientation via sun, earth, Polaris acquisition sequence.	Acquire: Fully automatic and manual step-by-step capability is provided.
k) Operational modes	Provide three-axis vehicle static pointing, dynamic pointing maneuvers (slew and track) orbit control, and stable platform (low jitter) using various sensor/actuator combinations.	Low jitter mode requires 13-bit sensor resolution and 60 msec computation rate.
2) Digital operational controller actuator drive a) Reaction jets • Attitude deadbands • Rate stabilization • Minimum on-time • Drive	---	Selectable, +0.05 or +0.5 deg. Pseudo-rate feedback. 100 msec. Logic level control pulses.
b) Inertia wheels • Scale factor • Rate stabilization • Momentum unload • Drive	---	Full-scale output drive for attitude error of ± 0.05 deg. Derived from wheel tachometers. Jets "on" for wheel speeds of $\geq 80\%$ of maximum value (0.5-sec jet on-time, 10-sec jet off-time). Analog proportional.
3) Analog backup controller a) Function	Provide independent path between sensors and jets for backup three-axis S/C control if DOC fails.	Provides for: • Stabilization of spacecraft • Acquisition of sun, earth, and Polaris • Maintain local vertical hold mode • Maintain station pointing mode
b) Controller design	Simple analog mechanization based on flight-proven designs.	All interface signals are analog or discrete. Circuitry consists of power supply, mode logic, and switching amplifiers with lag feedback compensation to obtain rate information.
c) Attitude deadband	± 0.1 deg (X-band monopulse) ± 0.5 deg (VHF, S-band) Use IRA analog rate output.	± 0.1 deg ± 0.5 deg Stabilized to $< \pm 0.05$ deg/sec
d) Rate damping		
e) Command subsystem interface	Discretes only	Provided

Table 9.1-7. Sensor Requirements/Performance Summary

Sensor/Parameter	Requirement	Performance
1) Earth sensor		
a) Attitude measurement	Two-axis (pitch, roll)	Provided
b) Measurement accuracy	- Static-on-earth disc (max offset 11.25 deg) - Dynamic	± 0.05 deg (1-sec averaging) $\leq \pm 0.1$ deg for ABC ± 0.1 deg
c) Linear range	± 11.25 deg	± 11.25 deg
d) Acquisition range (for full-diameter earth)	± 24 deg	± 24 deg/axis
e) Performance with sun in field-of-view	Detection and automatic avoidance	Provided, for on-earth disc pointing
f) Performance with orbital defects	Nonsensitive to orbit altitude changes	Scanning-type sensor; performance met with an earth whose subtended angle is ≤ 22 deg.
g) Ease of test	Ability to test at assembly facility and following spacecraft integration	Bolometer telescope does not require vacuum test facility.
h) Environmental/life	In-spec performance with aging/temperature effects at 100 percent operating duty cycle for two years	Meets requirement for detector responsivity changes of 30 percent. Error computation relatively insensitive to component degradation.
i) Sensor design	Autonomous self-contained computation of pitch/roll angles	Provided
2) Polaris sensor		
a) Attitude measurement	Single axis, yaw	Provided
b) Star magnitude range	Polaris magnitude +2, 1	Track stars as dim as 0.4X Polaris
c) Field-of-view	- Sensitive direction (yaw) - Nonsensitive direction (roll)	± 3.5 deg linear range ± 4.5 deg saturated range ± 14.9 deg
d) Accuracy (2 years)	± 0.078 deg (yaw) over entire field-of-view	± 0.075 deg
e) Bright particle rejection	Minimize reacquisition time after loss of track due to bright particle	Provides rapid, automatic reacquisition of target star by flyback and a single sweep
f) Acquisition range	3-deg x 3-deg field-of-view	9-deg x 11-deg field-of-view
g) Sun protection	Required	Proven sun shutter design
h) Sensor design	Prefer a flight-proven sensor	Mariner Mars '69 Canopus tracker
i) Sensor time constant	Less than 0.5 sec	0.33 sec
3) Sun sensor		
a) Function	Provide: - Sun acquisition - Location of S/C roll axis along sun-line - Control of S/C in pitch and yaw - Definition of center of sun	Provided
b) Acquire sunline	Locate sun from any random orientation in orbit.	Sensor configuration provides 4 π steradian coverage.
c) Accuracy	Locate center of sun within ± 0.5 deg	Coarse sun sensors locate sun within ± 8 deg; fine sun sensors provide required measurement accuracy of $\leq \pm 0.5$ deg.

**Table 9.1-7. Sensor Requirements/Performance Summary
(Concluded)**

Sensor/Parameter	Requirement	Performance
3) Sun sensor (continued)		
d) Spacecraft orientation	Capability of acquiring sun from +X or -X reference	Provided
e) Sensor design	Prefer flight-proven sensors	Sensors have flown on Aerobee rockets and OSO
f) Operating duty cycle	100 percent for two years	Provided
g) Response time	---	Less than 20 μ sec to change output between 10 and 90 percent of full scale.
4) Inertial reference assembly		
a) Rate and attitude sensing	Three-axis rate and attitude sensing during both acquisition and operational control modes	Provided
b) Resolution	0.0856 arc sec/pulse	Provided
c) Maximum rate	± 5 deg/sec	Saturated to 100 deg/sec
d) Linear range	± 1.5 deg/sec	± 1.5 deg/sec
e) Bias drift trim	0.15 deg/hr	Achieved by a precise current into secondary gyro torquers.
f) Gyro heaters	Avoid if possible	Internal electronic compensation eliminates need for heaters.
g) Autonomous operation	Self-contained clock reference, temperature controls and power supply -- only raw power input.	Provided
h) IRA design	Prefer a proven, flight-qualified design	Honeywell P-95 IRA for Lockheed.
5) Interferometer		
a) Attitude measurement	Onboard pitch/roll angle measurement with closed-loop S/C control	Interferometer provides 13-bit coarse/vernier counts representing direction cosines to DOC.
b) Linear range	± 12.5 deg relative to line-of-sight vector	Provided
c) Resolution	0.005 deg	0.0043 deg
d) Measurement accuracy	± 0.03 deg	± 0.03 deg
e) Update rate	160 msec	Updated, averaged counts provided every 160 msec
(For additional requirements/performance data, see Interferometer Subsystem Report)		

3

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control unit of the telemetry subsystem. The DOC provides its data output in block data form. The entire memory content may be transmitted to ground.

The ACS is an integral part of the gravity gradient experiment. This interface and the interfaces described above are treated in detail in the ATS F&G proposal, Volume I-C, Section 1.3.

9.1.6

PHASE D PROGRAM PARTICIPANTS

To allow accomplishment of the ATS Phase D program, Honeywell has assembled a strong group of participants in support of the GSFC-FHC ATS team. Each participant has been selected for the specialized capability and experience which he can apply to this program. Specific participants and their responsibility include:

- TRW -- Earth sensor
- Bendix -- Inertia wheels
- TRW -- Auxiliary propulsion subsystem
- Honeywell Radiation Center -- Polaris sensor
- Honeywell Aerospace Division
 - a. Total ACS/APS system management and integration
 - b. Digital operational controller
 - c. Analog backup controller
 - d. Actuator control electronics
 - e. Inertial reference assembly
 - f. Sun sensors
 - g. System AGE
 - h. Field support

The physical features and characteristics of the components and of the overall ATS attitude control subsystem are described in this section. Weight, volume, and power requirements for the ACS components are listed in Section 9.2.1. Component packaging is described in Section 9.2.2. In addition, details of the operation of the electrical circuits, mechanical devices, and optics used in the subsystem components are presented.

9.2.1 ATTITUDE CONTROL SUBSYSTEM

9.2.1.1 Mass Properties

The mass properties and dimensions of each of the elements in the ACS are provided in Table 9.2-1. The total weight of the ACS is 352.1 pounds.

9.2.1.2 Power

The power consumed by the ACS from the spacecraft 28-volt power supply is presented in Table 9.2.2. The power requirements are listed by the various modes of operation.

9.2.2 COMPONENTS

9.2.2.1 Digital Operational Controller

9.2.2.1.1 Physical Description

The ATS digital operational controller uses packaging techniques which have been used on space programs such as Mercury, Gemini, Apollo, MOL, and Mariner Mars '69. Experience on these programs allows use of the most practical schemes of packaging for mission accomplishments. Delivery schedule risk is minimized by calling on past experience in designing a package for high reliability while maintaining proven packaging schemes.

The packaging approach chosen for the ATS controller uses multilayer printed circuits and reflow soldering of integrated circuit flat packs, complemented by multilayer circuit board mounting of discrete components for maximum density. Solderless wire wrap terminations are used for interconnecting wiring within the device for reliability and light weight. Honeywell has developed and demonstrated capability in all production processes used in the manufacture of these devices. Table 9.2.3 summarizes the digital operational controller physical data and reliability.

Environmental requirements for the ATS mission are met by the structural design, which is derived from other successful Honeywell space flight hardware, using a fabricated electron beam welded housing and cover.

Table 9.2-1. Mass Property Summary

Component	A-Number	Weight (lb)	Overall Dimensions (in.)
Fine sun sensor	A123	0.4	2.8 x 1.7 x 1.9
Fine sun sensor	A124	0.4	2.8 x 1.7 x 1.9
Coarse sun sensor (EVM)	A133	0.6	2.8 x 2.4 x 1.7
Coarse sun sensor (EVM)	A134	0.6	2.8 x 2.4 x 1.7
Coarse sun sensor (hub)	A135	0.5	2.4 x 2.0 x 1.7
Coarse sun sensor (hub)	A136	0.5	2.4 x 2.0 x 1.7
Sun sensor electronics 1	A141	0.8	6.0 x 6.0 x 2.0
Sun sensor electronics 2	A142	0.8	6.0 x 6.0 x 2.0
Sun sensor signal amplifier	A143	0.3	6.0 x 6.0 x 2.0
Polaris sensor 1	A151	8.0	12.0 x 5.6 x 5.2
Polaris sensor baffle 1	---	2.0	12.5 x 7.0 x 5.4
Polaris sensor conditioning electronics 1	A153	3.0	4.8 x 3.0 x 2.0
Polaris sensor 2	A152	8.0	12.0 x 5.6 x 5.2
Polaris sensor baffle 2	---	2.0	12.5 x 7.0 x 5.4
Polaris sensor conditioning electronics 2	A154	3.0	4.8 x 3.0 x 2.0
Inertia wheel - pitch	A161	15.5	12.0 x 12.0 x 4.8
Inertia wheel - yaw	A162	14.5	12.0 x 12.0 x 4.8
Inertia wheel - roll	A163	14.5	12.0 x 12.0 x 4.8
Three-axis inertial reference assembly	A169	18.0	9.25 x 9.0 x 6.5
Earth sensor - pitch head	A176	3.7	5.70 x 6.0 x 4.0
Earth sensor - roll head	A177	3.7	5.70 x 6.0 x 4.0
Earth sensor electronics	A178	6.1	11.0 x 7.0 x 5.0
Digital operational controller 1	A185	17.0	11.5 x 10.5 x 6.5
Digital operational controller 2	A186	17.0	11.5 x 10.5 x 6.5
Analog backup controller	A188	5.4	6.0 x 4.7 x 7.0
Actuator control electronics	A190	10.0	8.0 x 12.0 x 6.5
Auxiliary propulsion subsystem	A101 thru A121	122.2	See APS report
Interferometer subsystem	A272-276	34.6	See Interferometer report
Gravity gradient subsystem	A831	39.0	14 diam. x 20

Table 9.2-2. Attitude Control Subsystem Power Requirements (watts)

Component	Ascent (6.6 hours) IRA & wheels (1 hour)			Acquisition Mode						Operational Mode									Orbit Control (10% duty cycle)			Low Power Backup Mode (5% duty cycle)		
				Rate Nulling(40 min. Coarse Sun Acquisition (20% jet duty cycle)			Fine Sun/Earth/ Polaris (10% jet duty cycle) (90 min.)			Normal Orient (wheels & jets)			Offset Point (wheels & jets)			Normal Orient (jets)(5% duty cycle)								
	Peak	Normal	Low	Peak	Normal	Low	Peak	Normal	Low	Peak	Normal	Low	Peak	Normal	Low	Peak	Normal	Low	Peak	Normal	Low	Peak	Normal	Low
Attitude control jets	0.0	0.0	0.0	8.0	1.6	0.0	8.0	0.8	0.0	8.0	0.0	0.0	8.0	0.0	0.0	8.0	0.4	0.0	8.0	0.8	0.0	8.0	0.4	0.0
Orbit control jets	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	30.0	30.0	30.0	0.0	0.0	0.0
APS feed system	0.0	0.0	0.0	9.0	5.0	5.0	6.0	4.0	4.0	6.0	4.0	4.0	6.0	4.0	4.0	6.0	4.0	4.0	7.0	5.0	5.0	6.0	4.0	4.0
Actuator control electronics	11.0	1.7	0.0	12.7	11.4	11.4	12.7	11.4	11.4	20.3	16.7	16.7	20.3	16.7	16.7	12.0	11.4	11.4	15.5	14.4	14.4	6.0	5.7	5.7
Inertia wheels	16.5	2.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	45.0	16.5	13.5	45.0	16.5	13.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Sun sensor assemblies	0.0	0.0	0.0	3.6	3.6	3.6	3.6	3.6	3.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Polaris sensor and conditioning	15.8	0.4	0.4	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6
Earth sensor assembly	10.6	10.6	10.6	14.9	10.6	10.6	14.9	11.8	10.6	10.6	10.6	10.6	14.9	14.9	14.9	10.6	10.6	10.6	10.6	10.6	10.6	10.6	10.6	10.6
Three-axis inertial reference assembly	44.0	6.7	0.0	44.0	44.0	44.0	44.0	44.0	44.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Digital operational controller	0.0	0.0	0.0	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	0.0	0.0	0.0
Analog backup controller	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	5.5	5.5	3.5
Total	97.9	21.9	11.0	144.1	112.7	111.1	141.1	112.1	110.1	141.8	84.3	81.3	146.1	88.6	85.6	88.5	62.9	62.5	123.0	97.3	96.5	61.1	35.8	33.4

(1) Polaris star sensor activation of sun shutter. Twenty-five watts on sun interference only. Low duty cycle.

(2) APS latching valve requirements - 50 watts upon activation for approximately 100 msec (not shown). Low duty cycle, but needed to accomplish redundancy switching and disconnect malfunctioned APS components.

Table 9.2-3. Controller Physical Data and Reliability

Item	DOC (each)	Primary Plus Standby DOC
Size (cu in.)	785	1570
Weight (lb)	17.0	34.0
Power (watts)	26.9	26.9
Reliability at 2-year point	0.618*	0.915

* $\lambda = 2.7519\%$ per 1000 hours

Weight. - The weight of the digital operational controller is 17 pounds. Since two controllers are required, a primary and a backup unit, the combined weight is 34 pounds.

Size/Packaging. - An outline drawing for the DOC is shown in Figure 9.2-1. In determining the packaging techniques to be used, there were several prime considerations:

- Proven, space-qualifiable design concepts
- Maximum component density
- High reliability
- Structural integrity in space flight environments
- Functional subassembly placement
- Repairability
- Light weight
- Minimum schedule risk

The package features a lightweight structural housing using fabricated electron beam welded panels. This design technique has been proven on several devices for the Apollo and MOL programs. After welding, this housing provides a one-piece, high-strength support for the electronics and also provides device mounting at the primary structural supports. A cutaway view is shown in Figure 9.2.2

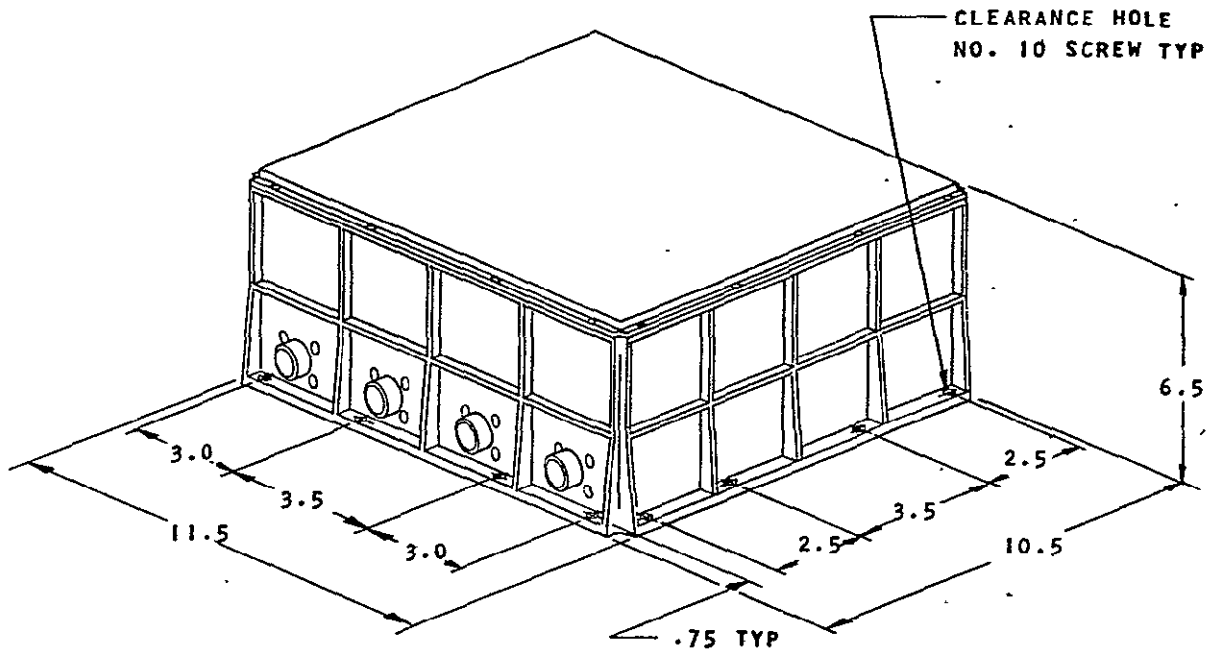


Figure 9.2-1. Digital Operational Controller -- Outline Drawing

Multilayer printed circuit board assemblies were chosen as the best way to provide maximum component density while maintaining high reliability and repairability in a lightweight device. A standardized card size and flat pack mounting array contains all circuitry for interconnection of the flat pack array while providing access for interconnecting the various cards. Flat packs are soldered to the board using machine-controlled re-flow soldering. The frame of the board assemblies is supported by the main structure of the welded housing.

9.2.2.1.2 Central Processor Electronics

Selection Considerations. - The selection of a central processor design involves a five-way tradeoff. These factors are:

- Number of instructions (computing power)
- Instruction execution speed
- Hardware complexity
- Power consumption
- Reliability

Of course, it is desirable to have as many instructions as practical because this makes programming easier and reduces the memory storage capacity requirements; but, on the other hand, hardware complexity and power consumption are increased while reliability is decreased.

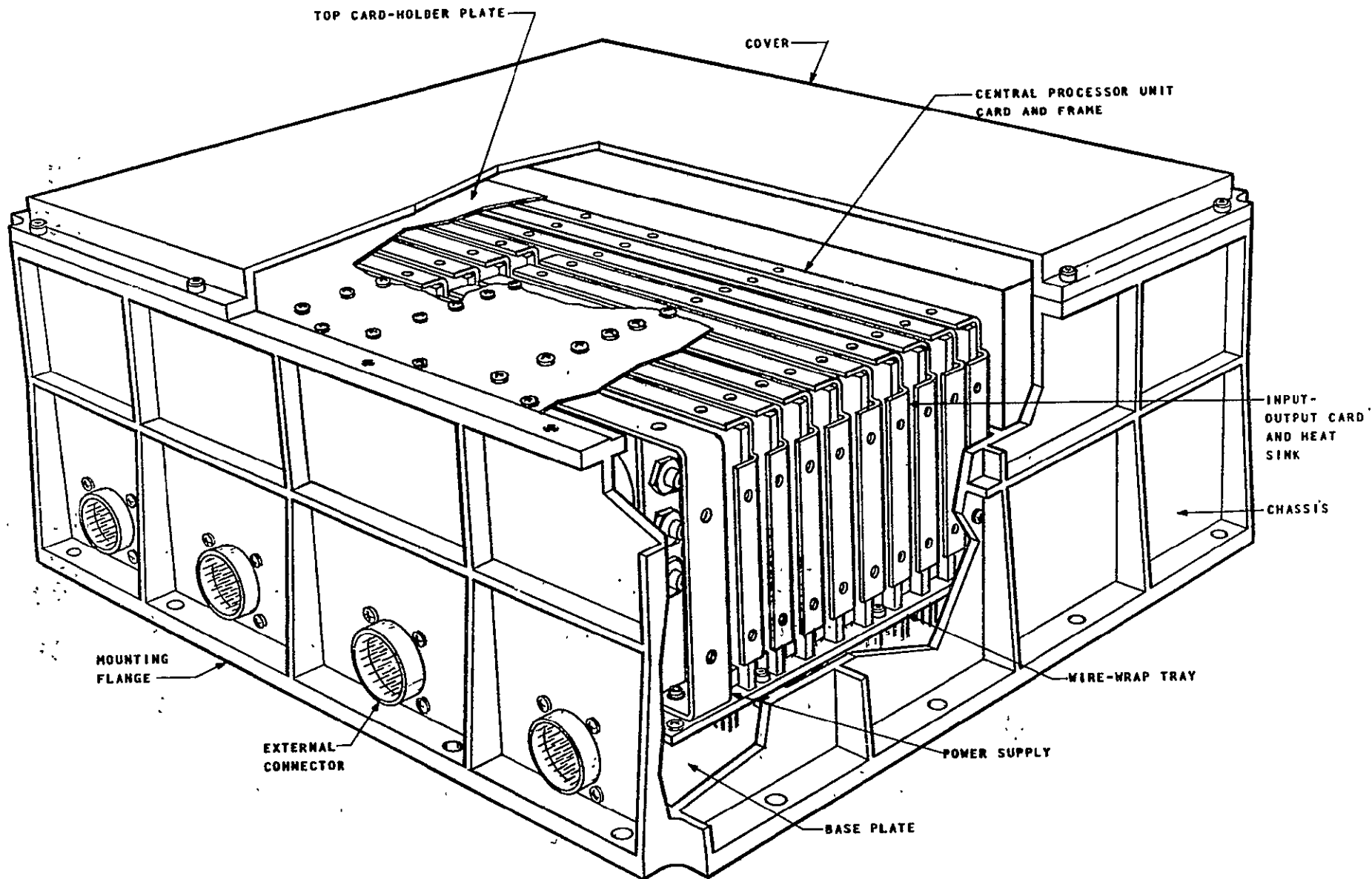


Figure 9.2-2. Digital Operational Controller -- Cutaway View

- e) Double precision add 15 μ sec
- f) Single word I/O transfer 15 μ sec
- 3) Signal Levels
 - a) Logic zero 0 vdc
 - b) Logic one +5 vdc
- 4) Standard input/output lines
 - a) Seven-bit address bus
 - b) Input bus: 16 bits parallel
or
4 bit bytes
 - c) Output bus: 16 bits parallel
or
4 bit bytes

The general format of the instructions is presented in Figure 9.2-4. The instruction repertoire is summarized in Table 9.2-4.

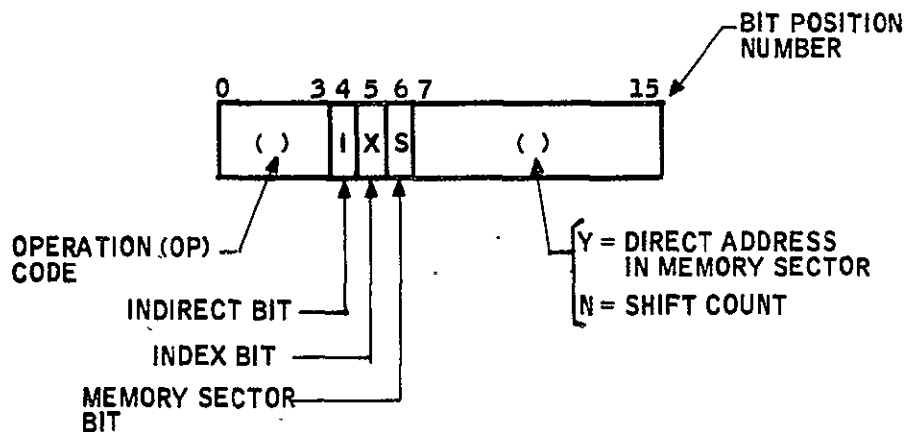


Figure 9.2-4. General Format of Instruction Word

Description of Operation. -

Word Structure -- Data words are stored in binary form using two's complement notation. The central processor unit accepts and processes data words in both single and double precision. Single-precision data words include 15 magnitude bits, plus a sign bit, and represent a data range of $\pm 2^{15}$, or $\pm 32,768$.

Double-precision data words include two data words, resulting in a data range of $\pm 2^{31}$, or $\pm 2,147,483,648$. The first data word (most significant) includes the 15 most significant bits of the number, plus the sign bit. It is identical to the data word used for single-precision operations. The second data word (least significant) includes the 16 least significant bits of the number. Thus, the number is represented by a sign bit and 31 magnitude bits.

Table 9.2-4. Instruction Repertoire Summary

Type	Mnemonic	Time (cycles)	Definition
Load and Store	LDA	2	Load A (single precision)
	LDA	3	Load A (double precision)
	STA	2	Store A (single precision)
	STA	3	Store A (double precision)
Arithmetic	ADD	2	Add (single precision)
	ADD	3	Add (double precision)
	SUB	2	Subtract (single precision)
	SUB	3	Subtract (double precision)
	MPY	18	Multiply
	DIV	32	Divide
	AAS	3	Augment and skip
Logical	AND	2	Logical AND
	ORR	2	Inclusive OR
Shift	ARSA	$1 + 0.20 (N-2)$	Arithmetic right shift
	LRSA	$1 + 0.20 (N-2)$	Logical right shift
	CRSA	$1 + 0.20 (N-2)$	Circular right shift
	LLSA	$1 + 0.20 (N-2)$	Logical left shift
	LRLS	$1 + 0.20 (N-2)$	Long logical right shift
	LLSL	$1 + 0.20 (N-2)$	Long left shift
Input/Output	INA	3	Input data to A
	IST	3	Input status to A
	OTA	3	Output from A
	SKS	3	Skip if ready line set
	IOC	3	Input/output control
	SMK	3	Set mask
Control	JMP	1	Unconditional jump
	JSP	2	Return jump
	SKP	2	Skip if condition is met
	ENB	1	Enable program interrupt
	INH	1	Inhibit program interrupt
	IDL	1	Computer idle
	DBL	1	Enable double-precision mode
	SGL	1	Enable single-precision mode
	SRF	1	Set-reset flag
	CAS	3	Compare
Register exchange	CMA	2	One's complement of A (single precision)
	NEG	2	Two's complement of A (single precision)
	CLA	2	Clear A (single precision)
	CLA	3	Clear A (double precision)
	CIA	2	Copy DBL flip-flop, flag register, and cycle counter into A
	AIC	2	Copy A into cycle counter
	RSAP	2	Copy A into DBL flip-flop and flag register

Instruction words may be classified as memory reference, generic, input/output, and shift instructions.

For memory reference instructions the four most significant bits (bits 0 through 3) specify the operation code (op code), bit 4 specifies indirect addressing (if the bit is a 1), bit 5 specifies indexing (if the bit is a 1), bit 6 is the memory sector selection bit, and bits 7 through 15 specify the memory address. If the sector bit is zero, sector zero is selected. Conversely, if the sector bit is 1 the sector from which instructions are currently being fetched is selected.

For generic instructions, all 16 bits are used to identify the op code. When an input/output instruction is specified, bits 0 through 6 specify the op code, bits 7 and 8 specify the input/output data format, bits 9 through 12 specify the input/output device address and bits 13 through 15 specify the input/output address. Shift instructions are identified with bits 0 through 10 specifying the op code, and bits 11 through 15 specifying the number of shifts to be performed.

The standard memory reference instruction word format allows 10 bits for memory addressing. The memory is divided (conceptually) into 512-word sectors, requiring nine bits to specify the address of a word in a particular sector. The remaining bit (called the sector bit) is used to specify this particular sector as sector zero, if the bit equals zero, or the sector currently being used by the p counter, if the bit equals 1 (used for the instruction address). Table 9.2-5 shows this scheme for an 8192-word memory, for example.

Table 9.2-5. Memory Sector Organization

Sector	Octal Address	Sector	Octal Address
0	00000-00777	10	10000-10777
1	01000-01777	11	11000-11777
2	02000-02777	12	12000-12777
3	03000-03777	13	13000-13777
4	04000-04777	14	14000-14777
5	05000-05777	15	15000-15777
6	06000-06777	16	16000-16777
7	07000-07777	17	17000-17777

Data Flow -- The data flow scheme for the central processor unit is shown in Figure 9.2-5. Operands fetched from the memory are transferred to the M register. From there, they are sent through the adder to either

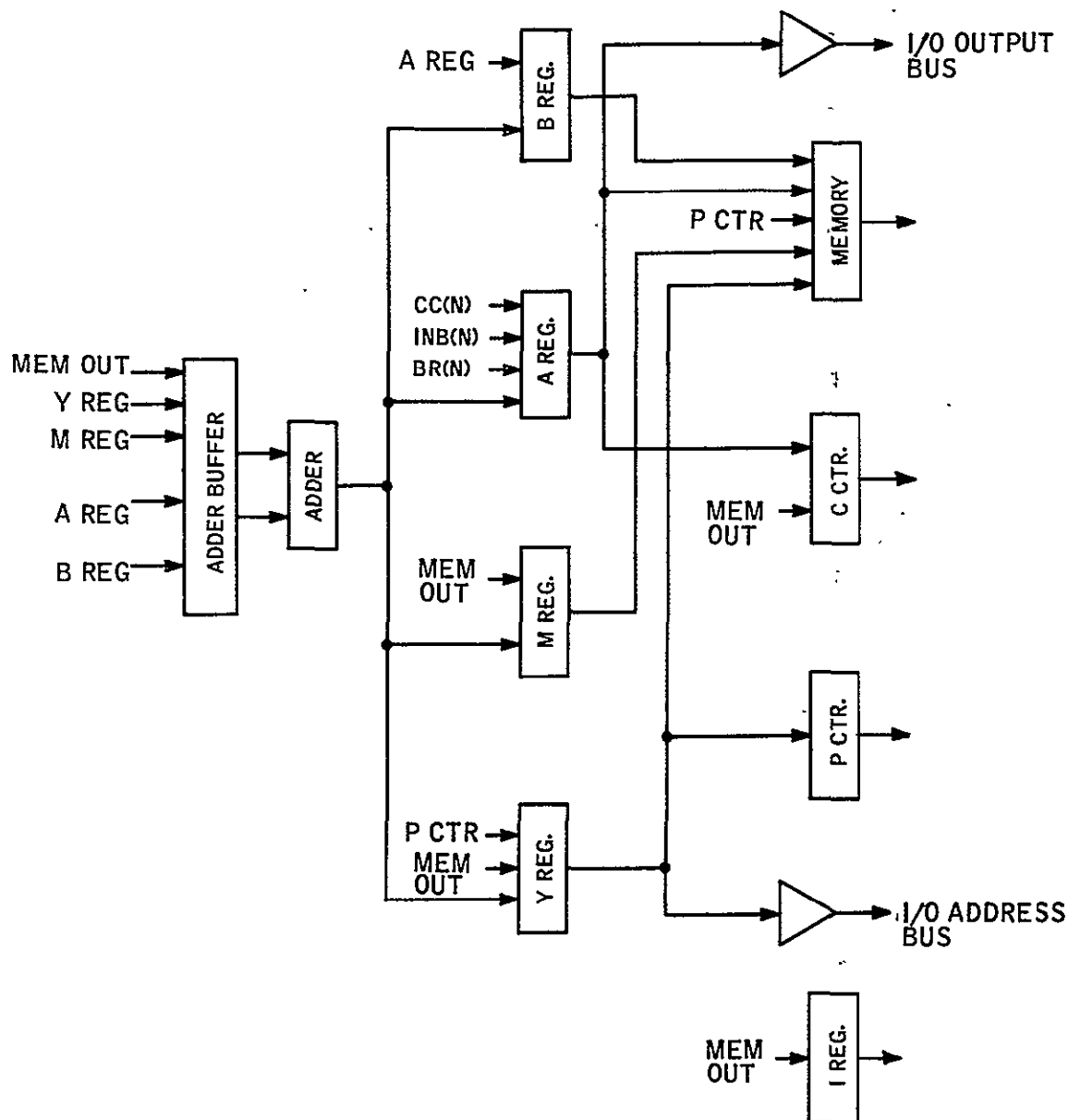


Figure 9.2-5. Data Flow Diagram

the A or B register. Instructions fetched from the memory are transferred to the I and Y registers and to the C counter. They are then decoded by control logic. Arithmetic operations are performed by sending information to the adder from the M, A, or B registers. The results are then transferred to the A or B registers. Indexing is performed by sending outputs from the memory and Y register through the adder to the Y and M registers. Input/output (I/O) operations are performed by transferring information from the input bus (INB) to the A register, or from the A register to the output bus (OTB). The I/O device code, etc., is transferred to the I/O unit on the address bus (ADB). Data to be written into the memory is transferred directly from the register involved. Jump operations are mechanized by transferring the contents of the Y register to the P counter.

I Register: The seven-bit I register stores the four-bit operation code and three-bit address modifier during the execution of each instruction.

Y Register: The Y register stores the operand address during the execution of a memory reference instruction.

P Counter: The P counter stores the address of the next instruction to be executed.

C Counter: The C counter counts the number of shifts during the performance of shift instructions by being initially set with the 1's complement of the desired number of shifts, and then being incremented by 1 during each shift operation.

Adder: Four full adders are used to form a four-bit parallel adder. Arithmetic is performed four bits per byte, four bytes serially. Thus, an addition requires four consecutive operations by the adder. Since the logic used is relatively slow, inputs to the adder must be sent through an adder buffer register to sufficiently reduce adder control logic propagation delays.

From the buffer register, information is fed to the A and B inputs of each of the five adder stages. The first four stages of the adder are used for all the arithmetic processes. The fifth stage is used for multiplication and indexing.

M Register: The M register is used for operand storage in arithmetic and logic instructions.

A Register: The A register is the main arithmetic register in the computer. In arithmetic and logical instructions, it accepts information four bits at a time from the adder and at the same time shifts its contents four bits at a time to the right.

B Register: The B register is used as an extension of the A register, in double precision, MPY, and DIV operation. Input and shifting is accomplished in the same way as in the A register.

I/O Interface: Information from the special input/output (SIO) is sent to the central processor unit via the A register. Conversely, information is transferred from the A register to the SIOs for outputting operations.

9.2.2.1.3 Input/Output Electronics

This section describes the input/output electronics of the digital operational controller. The DOC establishes communication with various external devices through the I/O electronics. Data transferred between the DOC and the external devices are channeled through the I/O electronics. The I/O commands issued by the DOC software are transacted by the I/O electronics. The following subsections describe in detail the system requirements, philosophy, timing and the special interface of the I/O electronics.

Interface Requirements. - A simplified functional block diagram of the digital operational controller is shown in Figure 9.2-6. This figure illustrates the special-purpose input/output electronics circuits and their interface with the I/O control and the external devices. Interface signals are listed in Table 9.2-6. The signals are classified as analog, pulse trains, serial digital, parallel digital, and discretes. Each of the two ATS digital operational controllers has an identical interface with the external devices (sensors, actuators, etc.). There is no signal switching required to use the primary DOC or redundant DOC. The sensors provide separate line drivers for each of the devices that it feeds; therefore, a shorted line driver or receiver will not fail the sensor. The same concept applies to the actuators. A single command from the ground will power-up the selected DOC and place it into operational control of the attitude control loop.

I/O Electronics Design Philosophy. - This section describes the design philosophy of the input/output electronics portion of the digital operational controller. The I/O electronics include the special-purpose input/output (SIO) electronics circuits that provide an interface with the external devices. The external devices are the attitude control sensors, attitude control actuators, experiments, telemetry, and command. In addition to interfacing with the external devices, the SIOs provide a proper interface with the I/O control portion of the central processor unit.

It is understood that system design is evolutionary, and therefore certain portions of that system should be readily capable of responding to new requirements at minimum cost. The areas of concern for the ATS controller are the number of input output connector pins, the design philosophy of the input/output electronics and additional volume available for expansion, and the design philosophy and additional volume available for memory expansion. With adequate concern for these factors, the ATS controller will readily adapt to system changes and experiment additions late in the program.

Standardized module sizes permits the SIOs to be functionally unique while allowing the possibility of changing the location assignments of individual SIOs allowing the addition of I/O functions without affecting the existing mechanization.

Reliability cannot be overemphasized. The failure of a particular SIO should not cause the failure of all SIOs, nor should the external devices be disabled by an SIO failure. The I/O design philosophy has incorporated these simple rules.

A functional block diagram of the input/output electronics is shown in Figure 9.2-7. The figure includes the I/O control section of the central processor unit and the interface with the external devices.

The I/O control section provides the following functions:

- Stores priority interrupt word in mask register
- Enables and disables interrupt system

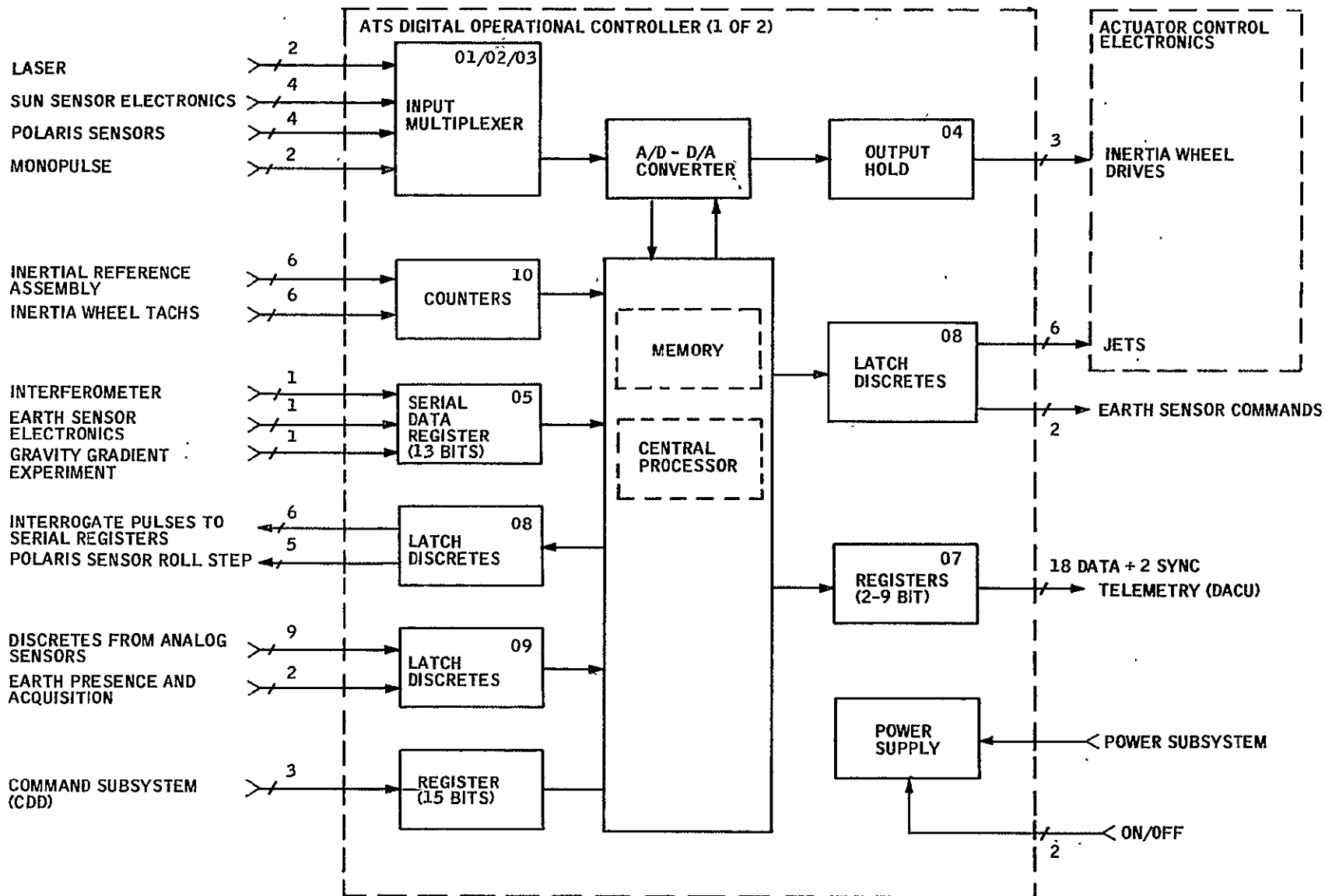


Figure 9.2-6. Digital Operational Controller Interfaces

Table 9.2-6. Digital Operational Controller Interface Signals

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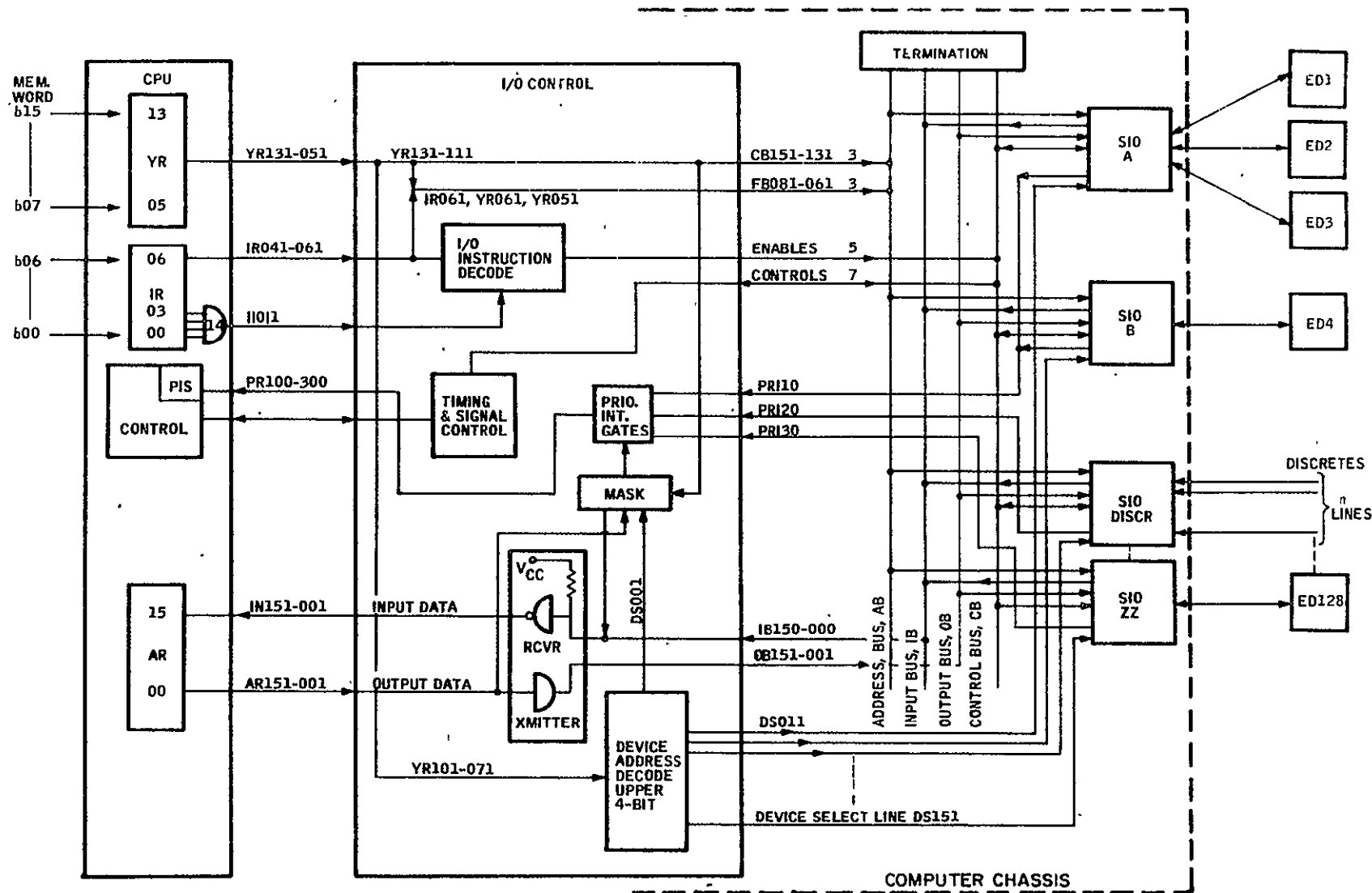


Figure 9.2-7. Digital Operational Controller Input/Output Electronics Functional Block Diagram

- Provides synchronization pulses to SIOs for data transfer
- Provides address decoding to select a particular SIO
- Decodes I/O instruction
- Interrogates and resets SIO ready lines
- Provides line drivers and receivers to assume adequate fan-out to, and restore signals from the SIOs.
- Provides proper program branch upon a priority interrupt.

The I/O operation can be described as follows. On a memory read cycle, an instruction is loaded into the operand address register within the CPU. If the instruction is an I/O instruction, decoding is initiated to uniquely select a specific SIO device. In addition, the type of operation, subdevice information, and data transfer format are specified by the address and control bus codes.

The CPU waits a fixed time interval for a response from the addressed SIO. The addressed SIO shall decide if the current instruction can be accepted or not. If it can, the SIO shall respond on the ready/busy line. Only the addressed SIO can issue a response.

If the SIO accepts the command, the CPU sends strobe pulses to perform the desired I/O operation. The program skips the next instruction when the current I/O operation is successful. Should the device reject the command by answering "busy," the I/O instruction is terminated and the program executes the next instruction in sequence.

Three external priority interrupt lines can be connected to those SIOs which use the interrupt system for I/O communication. These lines are generally unique to a device, although as many as 16 devices may share the same priority interrupt line.

The mask register provides the masking of the priority interrupt. It is this mask which allows the program to modify the relative priority level. A correct mask shall be loaded to correspond to the interrupt state.

Special-Purpose Electronics. - The I/O control section of the DOC communicates with the sensors via the special-purpose electronics associated with the particular sensor. These electronics (hereafter referred to as SIO) differ considerably from one sensor to the other. It is the SIO which performs, as appropriate, the following items in order to serve as an intermediary between the CPU and the sensors: analog-to-digital conversion, digital-to-analog conversion, serial-to-parallel or parallel-to-serial conversions, discrete input or output signal conditionings, accumulation of pulses, and the processing of I/O commands.

The following subsections contain brief descriptions of the SIOs. Table 9.2-7 lists all of the SIOs and their address designation.

Table 9.2-7. Special I/O Electronics

SIO Designation	SIO Description
00	Mask register
01/02/03	A/D converter
04	D/A converter
05	Serial data register
06	Command (CDD)
07	Telemetry (DACU)
08	Discrete outputs
09	Discrete inputs
10	Up-down accumulators and wheel speed counters
11 - 15	Spares

Mask Register (SIO-00) -- The interrupt mask register stores interrupt mask words for the priority interrupt system.

A/D-D/A Converters (SIO-01, -04) -- The A/D-D/A converter converts analog voltages, such as those originating in the star tracker, sun sensor, and monopulse, to digital signals; also, it develops analog signals for devices such as the inertia wheels. The basic converter hardware is time-shared to reduce parts count. Up to 24 input and 8 output signals can be provided by the present addressing allocations. A block diagram of the A/D-D/A converter SIO is shown in Figure 9.2-8.

A/D-D/A Characteristics are:

1) Digital output

- Voltage range -10 to +10vdc
- Input impedance 50K ohms minimum
- Conversion time 750 microseconds
- Resolution 5.0 millivolts
- Accuracy ± 0.1 percent of full scale $\pm 1/2$ LSB

2) Analog output

- Voltage range -5 to +5 vdc
- Output drive ± 2.0 milliamperes maximum
- Conversion time 500 microseconds
- Resolution 8.0 millivolts
- Accuracy ± 1 percent of full scale

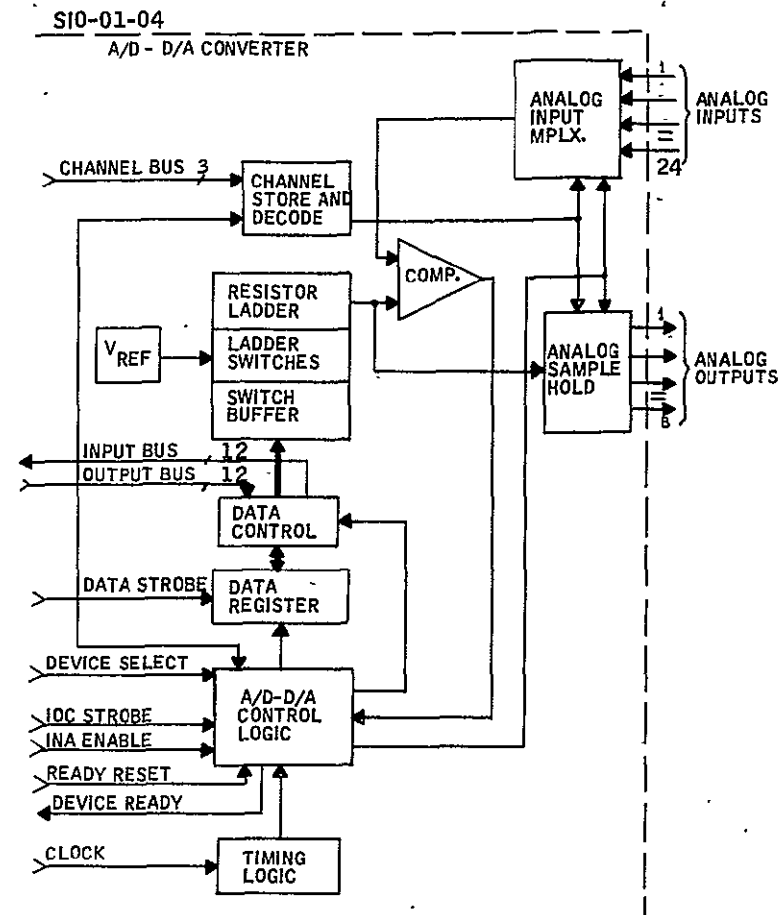


Figure 9.2-8.. A/D - D/A Converter SIO Functional Block Diagram

Serial Data Register (SIO-05) -- The serial data register SIO, Figure 9.2-9, can multiplex up to eight digital input signals, including data received from the earth sensor, interferometer, and gravity gradient experiment. The sensor to be interrogated is selected under program control by setting the appropriate interrogate latch (output discrete SIO) and commanding an input data transfer via the device select line (DS051).

Discrete Outputs (SIO-08) and Discrete Inputs (SIO-09) -- The SIO-12 and SIO-13 interface respectively with all discrete outputs and discrete inputs of various sensors. A specific sensor can be chosen with an appropriate channel bit in an I/O instruction. A discrete line (whether it be input or output) is allocated a unique bit position in the memory word. Individual discretes can be set or reset under program control.

Pulse Accumulators (SIO-10) -- The pulse accumulator SIO provides three up-down counters for interface with a pulse rebalance gyro assembly and three up counters for interface with the inertia wheel tachometers. Each counter is nine bits (eight bits plus sign) in length and will accept input pulse rates up to 25 KHz.

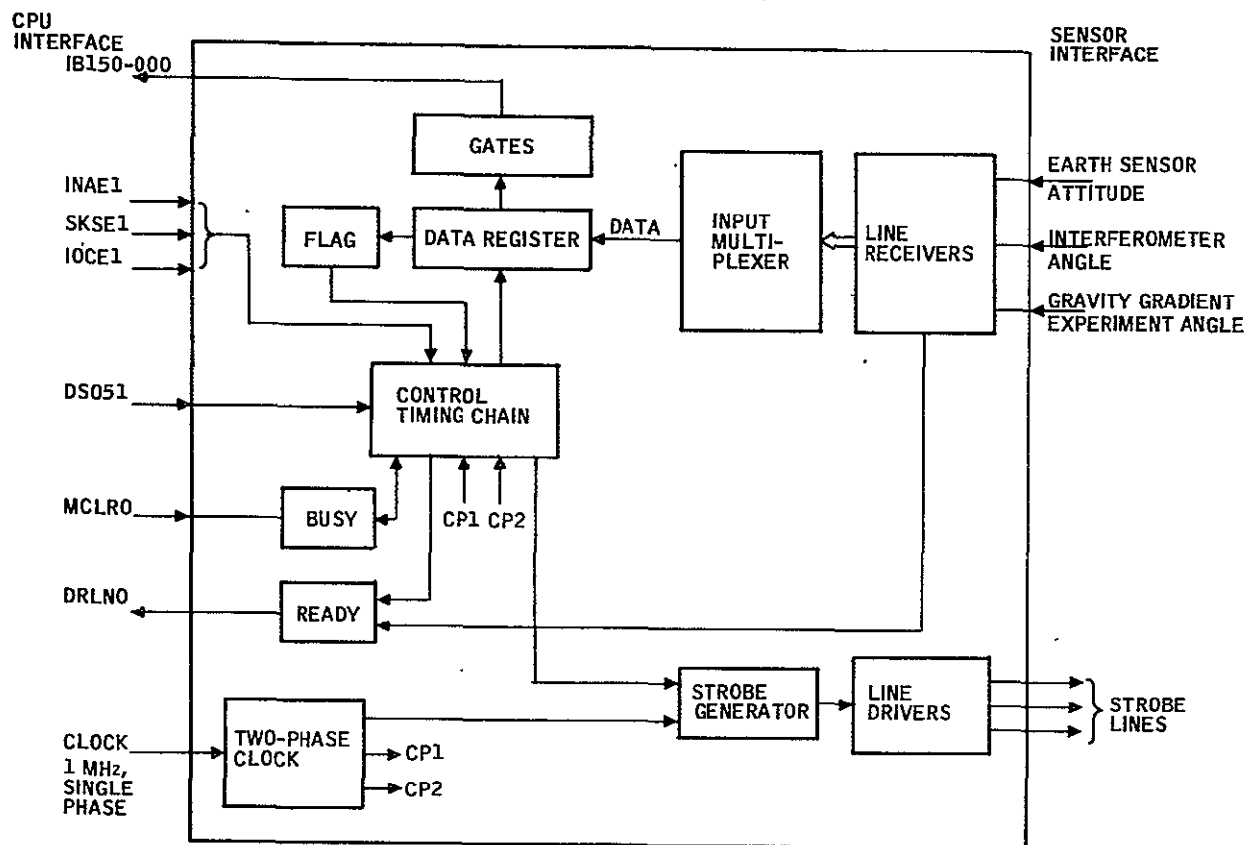


Figure 9.2-9. Serial Data Register SIO

A functional diagram of the pulse accumulator (gyro interface) is shown in Figure 9.2-10.

Command Decode and Distribution (CDD SIO, -06) -- Figure 9.2-11 shows the CDD SIO block diagram. An input register (IR) of 15 bits is used to store the serial word being transmitted by the CDD subsystem. A train of pulses representing six-bit address words and nine-bit data words is transmitted serially in the input data line. As it is presently defined, these pulses are sent to the DOC CDD SIO only when these pulses are meant for the DOC.

The strobe pulses are sent serially on the strobe line. A burst of 15 pulses is sent at a time. The CDD SIO uses these pulses to sample the input data line.

The DOC command and execute line informs the CDD SIO that a 15-bit word has been transmitted and that the SIO will transmit this word into the DOC as soon as possible.

Upon the receipt of the DOC command and execute signal, the control unit of the CDD SIO becomes ready for an INA (input to A register) instruction.

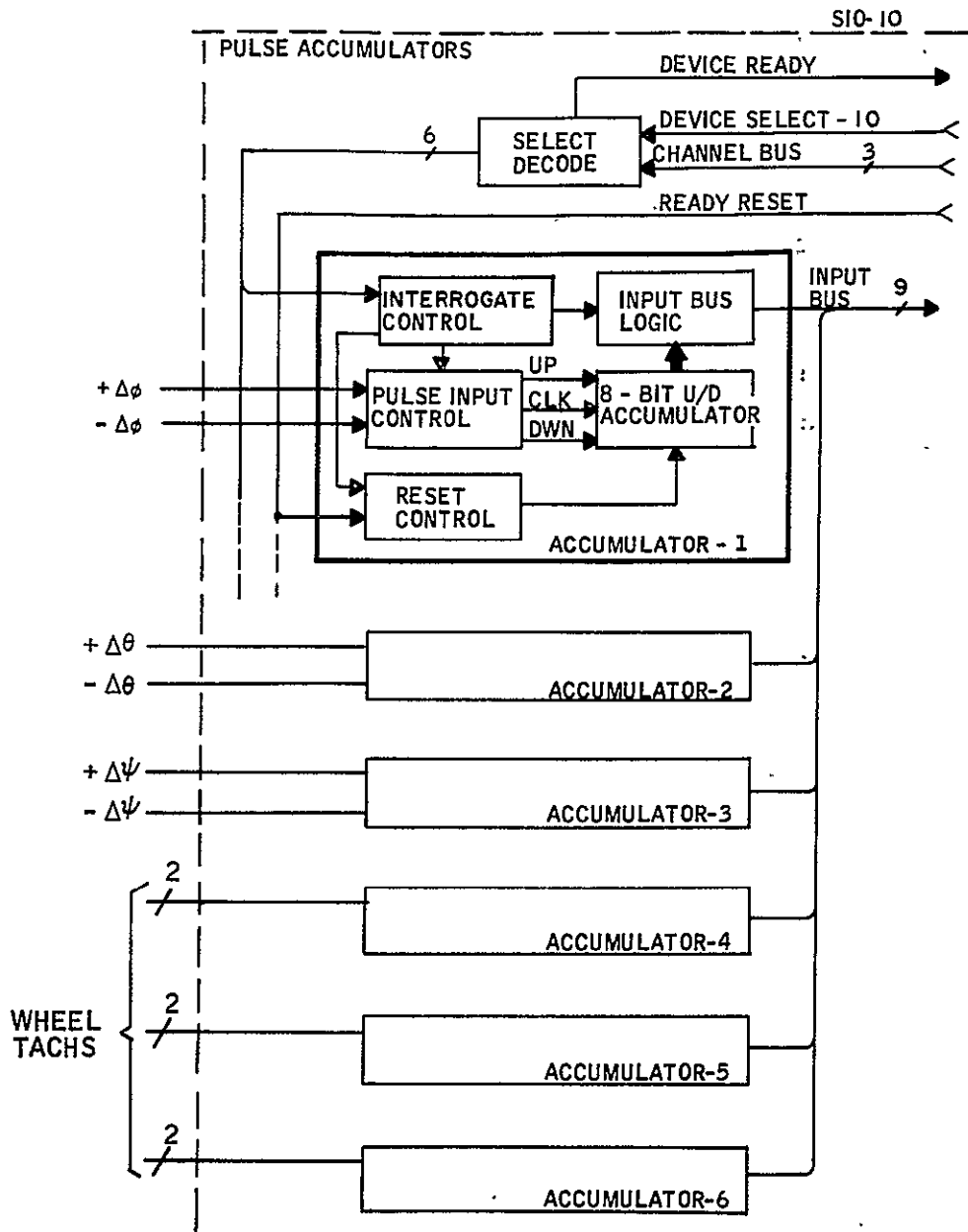


Figure 9.2-10. Pulse Accumulator SIO Functional Block Diagram

Telemetry (DACU) SIO (SIO-07) -- Figure 9.2-12 shows the block diagram of the telemetry (DACU) SIO. Two output registers, OR1 and OR2, of nine bits each are used to store the nine-bit word transmitted from the DOC. The DACU continuously samples the telemetry SIO when it is on dwell mode. The OR1 and OR2 registers are connected in parallel via associated gates to the output bus, OB, of the DOC. These registers are alternately loaded under the control of the control unit and the end-of-read signals associated with the register.

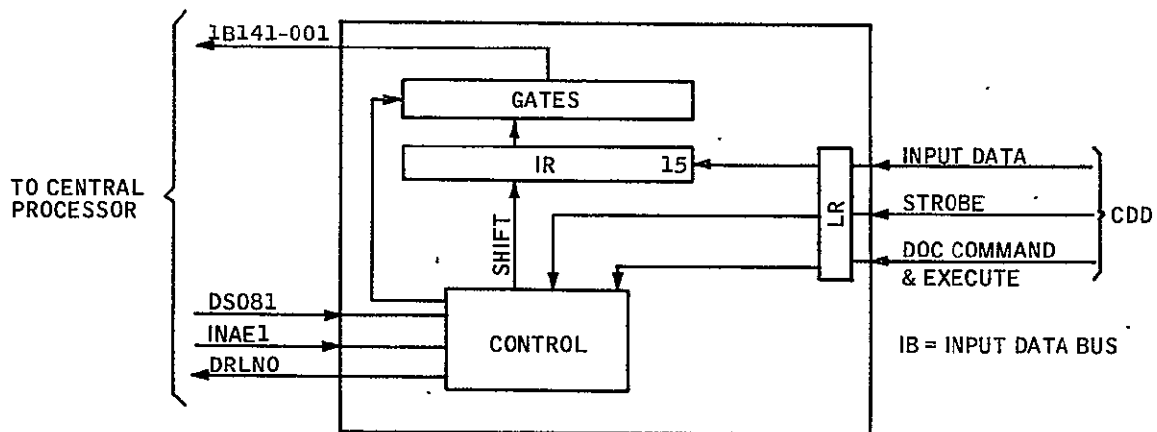


Figure 9.2-11. CDD - SIO Block Diagram

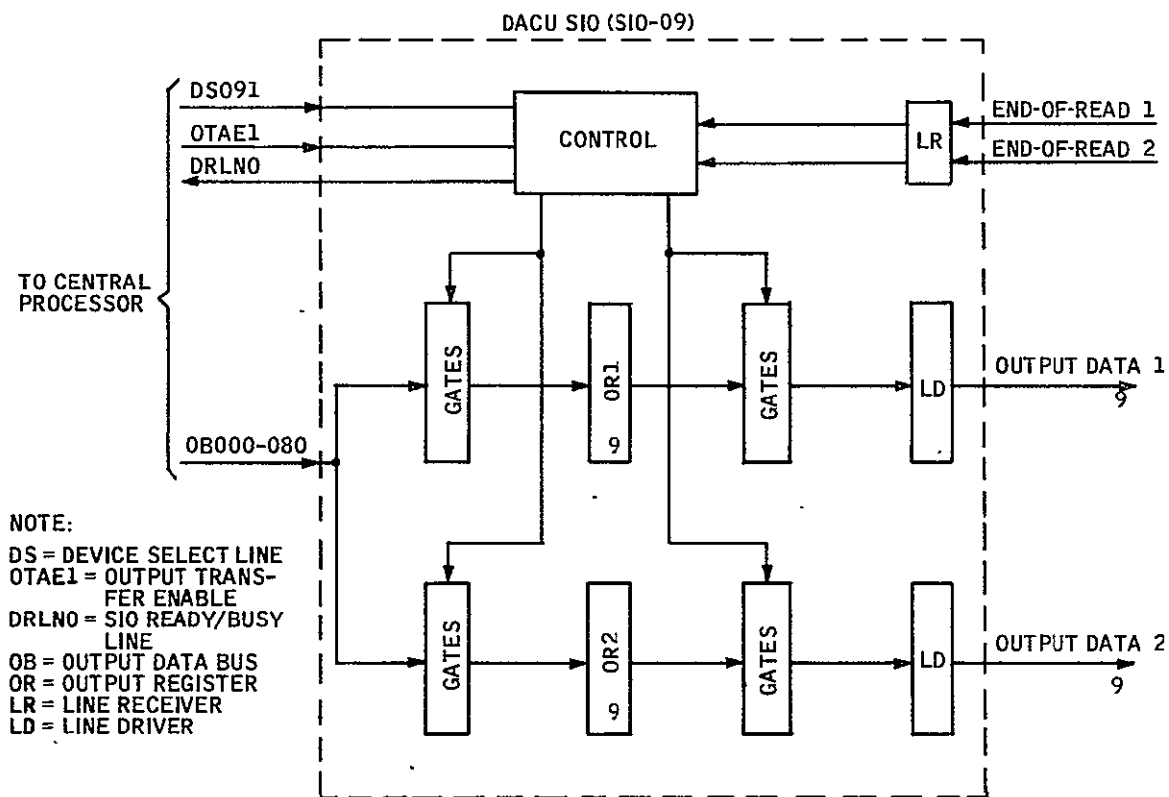


Figure 9.2-12. Telemetry (DACU) SIO Block Diagram

9.2.2.1.4 Memory Electronics

Memory Requirements. - The selection of a memory for use with the DOC required a detailed study of the various types of memories and their individual characteristics. All volatile forms of memory storage were eliminated for reasons of program safety. The remaining memory systems were then compared on the basis of reliability, power consumption, weight and volume. Additional design features such as nondestructive readout and read-only capability were included to complete the definition of memory requirements.

The desirability for nondestructive readout is based on reasons of program safety and power savings. In a destructive readout memory, if the stored content is read out incorrectly, it is also restored incorrectly, thereby changing the data or the computer program. Elimination of the restore operation further eliminates the high-power write cycle, reducing the total memory power consumption by as much as a factor of five.

The desirability for read-only storage is based on program safety. A combination memory (read-only program storage with alterable scratch pad) provides program protection from addressing errors, power interruptions, etc., while permitting changes in constants and microprograms in the alterable section.

The result of the memory study was to select a plated-wire memory for use with the DOC.

The capacity of the memory has been set at 4096 words (of 16 bits each), since programming studies have shown a need for slightly less than this capacity. Recognizing that software requirements may increase, the actual memory design and organization will be based on an 8192-word capacity. Honeywell's experience in the design and construction of these memories has shown that the volume of an 8192-word memory is approximately 25 percent larger than for 4096-word memory, and the reliability is only slightly affected. If the 4096-word capacity proves sufficient, the additional circuitry for an 8192-word memory will simply be omitted. The package size will however, remain at that of an 8192-word memory.

Plated-Wire Memory Features. - A plated-wire memory has been selected for use with the DOC because it offers the following advantages:

- Nondestructive readout
- Low-power operation
- High reliability
- Read-only capability
- Non-volatile storage
- Proven technology

Plated-wire memories inherently provide low-power system operation largely because of the physical makeup of the storage element. The basic plated-wire element and a schematic of its use in bit formation are shown in Figures 9.2-13 and 9.2-14, respectively. The closed magnetic flux path of the wire plating combined with the use of the copper substrate as a digit line yields digit current requirements that are approximately one-tenth those necessary in a typical coincident-current core memory. Additionally, the high-speed capability of the memory permits the system to complete a cycle and return to the low-power standby condition for a majority (minimum of 80 percent) of the time. The total worst-case power consumption for the plated-wire memory is less than one-fourth the power consumption of a coincident-current core memory configured for the same application.

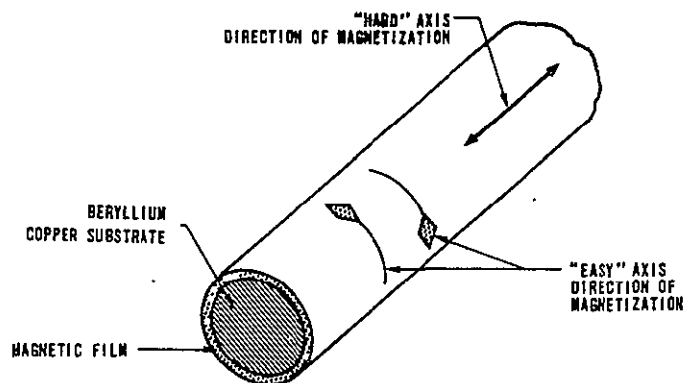


Figure 9.2-13. Basic Plated-Wire Element

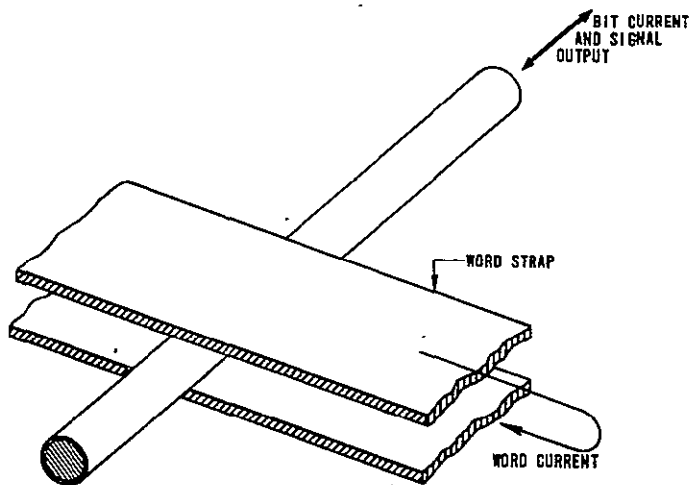


Figure 9.2-14. Basic Plated-Wire Bit

The reliability of the plated-wire memory has been maximized by careful component selection and by maximum use of integrated circuitry. The nature of the memory element (a long continuous copper wire plated with permalloy) minimizes the number of interconnects necessary for the memory stack, further improving system reliability.

The NDRO nature of the memory permits the memory to be divided into alterable and read-only sections. Furthermore, the sectioning can be controlled by changing the wiring in an exterior connector to enable or disable the memory write electronics for a given section of the memory. This unique capability allows program changes to be readily made, or the complete system may be operated as an alterable memory with final program protection provided by operating portion of the memory in the read-only mode once the program has been finalized.

Plated-wire memories are typical of magnetic memories in that loss of power does not cause loss of memory content (nonvolatile storage). This feature is a necessity to preserve the computer program.

Honeywell has been developing and refining plated-wire memories since 1963. Present designs incorporate many parts that have been qualified or are presently being qualified for programs requiring high-reliability components. This development has been supported by several government agencies and by Honeywell because of the inherent advantages of plated-wire memory systems.

The electrical and physical characteristics of the plated-wire memory are summarized as follows:

● Storage capacity	Up to 8192 words maximum (3200 words estimated requirement)
● Number of bits per word	16
● Mode of addressing	Random access
● Read cycle time	1 microsecond
● Read access time	750 nanoseconds
● Write cycle time	1 microsecond
● Continuous read power consumption (200 KHz)	6 watts
● Continuous write power consumption (200 KHz)	7 watts
● Standby power	4 watts
● Physical size	225 cubic inches
● Weight of assembly	7.5 pounds

The function of the power supply electronics is to convert the spacecraft primary power into the various appropriately regulated voltages required internally by the digital operational controller. In addition to generating the necessary voltages, the power supply must automatically turn these voltages on and off in the proper sequence on ground command.

Power Supply Design Criteria. - The following items were considered of primary importance for the power supply design:

- Input characteristics and protection
- Output characteristics and protection
- Regulation, ripple, and efficiency
- Internal controller power distribution network
- Power sequencing

The power supply must, of course, satisfactorily perform over the range of expected line voltage variations. Primary power will be nominally 30.5 volts at the spacecraft power source output. As received by the power supply, the primary power feed may contain induced and coupled transients and other noise. Line losses and the effects of pulsed loads may further degrade the input power as received by the load. Careful design must consider and suppress these undesirable affects. The power supply includes protection to prevent damage should the power supply lead polarities be accidentally reversed.

The power source must be protected from adverse conditions caused by the load. For instance, the power supply shall not draw excessive current during turn-on, nor shall an internal short be permitted to appear across the primary power feed. The amount of noise conducted back into the power feed will be adequately controlled by internal EMI filtering. The EMI filter has low-pass elements in both the high and low side of the power line.

The digital operational controller's power supply must provide a majority (>50 percent) of its power at low voltage (+5 vdc), since the load is mostly integrated circuits. High efficiency and low output voltage are basically incompatible. The losses inherent in rectifiers and transistors cause reduced efficiency as the output voltage is lowered. Special design techniques must be used to maintain acceptable overall operating efficiency.

A power supply providing the properly regulated output voltages is only part of the problem. With an integrated circuit load considerable noise and transients are present on the internal power distribution network. It is essential that the low output impedance of the power supply be maintained all the way to the modules and to the actual integrated circuit chips. This problem can only be solved by incorporating a low-impedance strip line distribution network. The design criteria is high shunt capacity

between power and ground and very low inductance in series with power and ground leads. The network is formed by laminating alternate strips of copper separated by a thin, high-dielectric-constant insulator. The flat conductors thus formed have the desired characteristics. Power and ground layers are maintained on the circuit cards, as well, to suppress the conducted power line noise to a known, controlled level.

Circuit Description. - A feedback regulator scheme was selected for the d-c/d-c converter, since it provides the highest overall operating efficiency and simplest circuitry. Feedback is taken from the +5 vdc output and is used to pulse-width modulate a series switch preceding the converter. The output voltage is monitored through a comparator; and a high- or low-voltage indication out of the comparator modulates the width of a pulse controlling the series switch "on" time. The design is simple, but effective, and has been successfully proven on other high-reliability programs.

The power supply organization is shown in Figure 9.2-15, with characteristics summarized in Table 9.2-8.

9.2.2.1.6 Program Organization and Implementation

Executive Program. - From consideration of DOC requirements and software design ground rules an overall program organization as shown in Figure 9.2-16 has resulted. An executive routine will serve to command DOC functional operation by controlling execution of subprogram blocks. The executive shall react to and implement commands from the command subsystem. These commands will be used to select operational modes, select sensors and actuators for a given mode, transmit data to the DOC, and select output data blocks. The executive routine will also control data transmission from DOC memory to the ground.

Subprograms under executive control are categorized as mode execution, auxiliary functions and controller self-test. Mode execution subprograms include all routines required to implement the operational modes. These routines include sensor/actuator and input/output control and processing plus associated control law computation and logic. The auxiliary functions include routines to accomplish gyro drift trim, orbital bias computation for static pointing mode correction, and satellite tracking command generation. Pointing and tracking command generation are designated target vector update in Figure 9.2-16, since one computational procedure suffices to handle both functions. The controller self-test subprogram includes routines that effect checks on the DOC central processor (instruction test), memory, and input/output system.

In the proposed DOC program organization shown in Figure 9.2-16, an executive routine provides overall operational software control. Need for a capable executive routine results from the multi-task requirements imposed on DOC software by ACS requirements. The DOC will have to be programmed to execute several different control modes at different iteration frequencies. Concurrent with mode execution, auxiliary functions such as

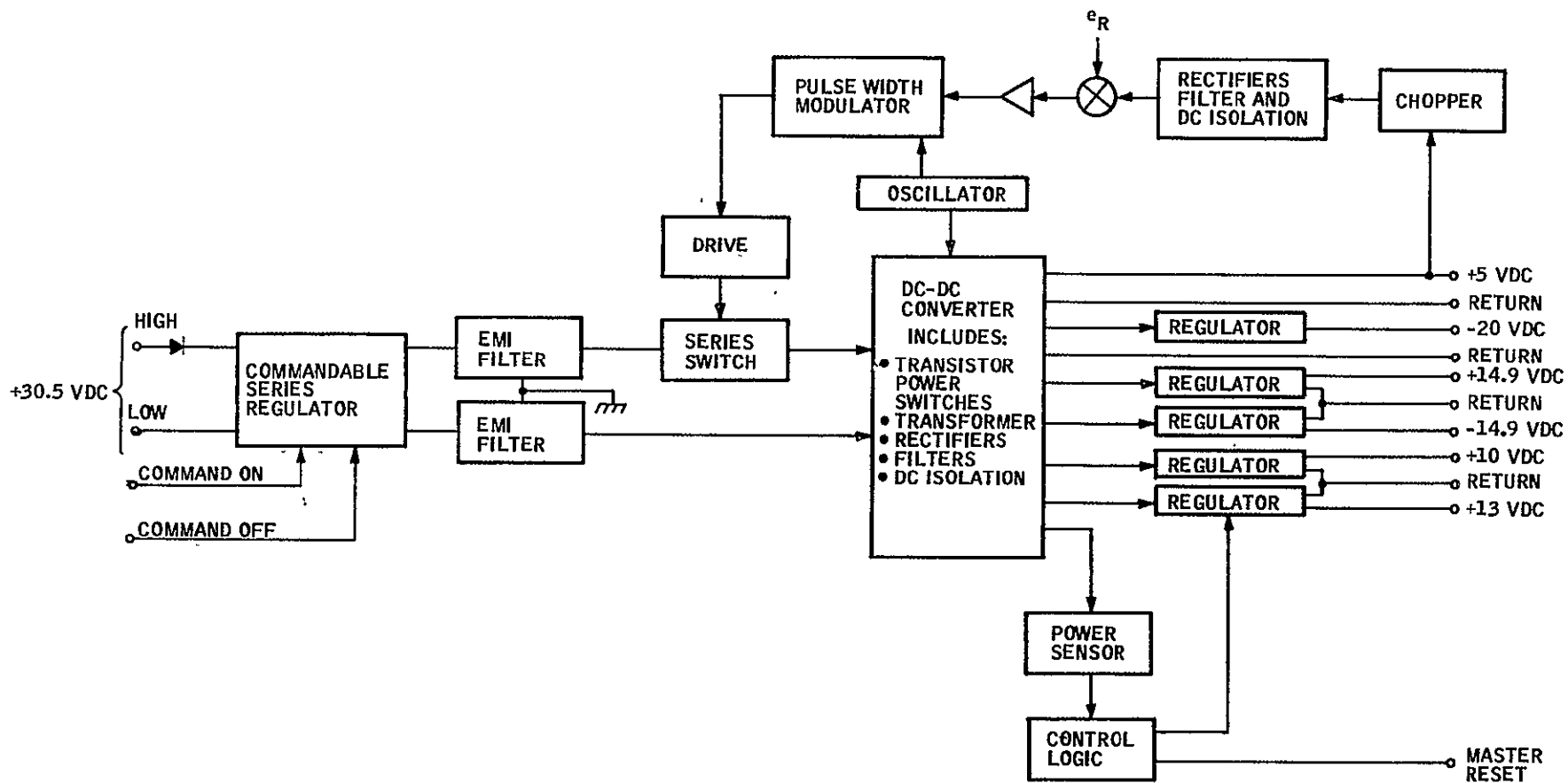


Figure 9.2-15. Digital Operational Controller Power Supply Organization

Table 9.2-8. Power Supply Characteristics

Output Voltage (vdc)	Tolerance* %	Load Description
+5	±5	Integrated circuits
+10	±2	Bias power for memory
+13	±2	Memory current source
+14.9	±1	Analog operational amplifiers
+14.9	±1	
-20	±2	Multiplexing switches bias

*Includes line, load, and ripple contributions.

point command generation and satellite tracking calculations will be iterated at a lower frequency. Provision will also have to be made to control DOC self-test routine execution when time is available; i.e., self-test is treated as a background function having no time constraint.

An auxiliary task status table is maintained which contains task number, status, starting address, and relative priority number. Task status can be either active, inactive and complete, or inactive and incomplete. An active task is one that was active in the previous time slice; an inactive partially-complete task is one that was initiated in some previous time slice but then had to be put aside uncompleted because a higher priority task appeared. An inactive complete task is one waiting to be restarted as soon as its priority permits. Provision is also made to put a task in abeyance indefinitely by assigning it zero priority. Other priority level changes are made within the executive as a function of time and/or ground command or by a control mode subprogram.

Flow through the executive, as shown in Figure 9.2-17, begins with saving all registers to enable returning to the interrupted instruction. Operational mode calculations are then commanded if update time has occurred.

Following control mode computation, a word will be transferred to the TM output register if data block output mode has previously been commanded. The command-interface will then be interrotated to look for an incoming command or data word. If a data word is present, it is assembled into an N-bit DOC memory word and stored. If a command word is present, its legality is checked and the implied operation is initiated. In the case of mode execution commands, control is transferred to the mode initialization subprogram. Following input of either a data word, command word, or no word, all primary tasks have been completed. It is assumed, however, that the next interrupt could occur while executing the command implied operation.

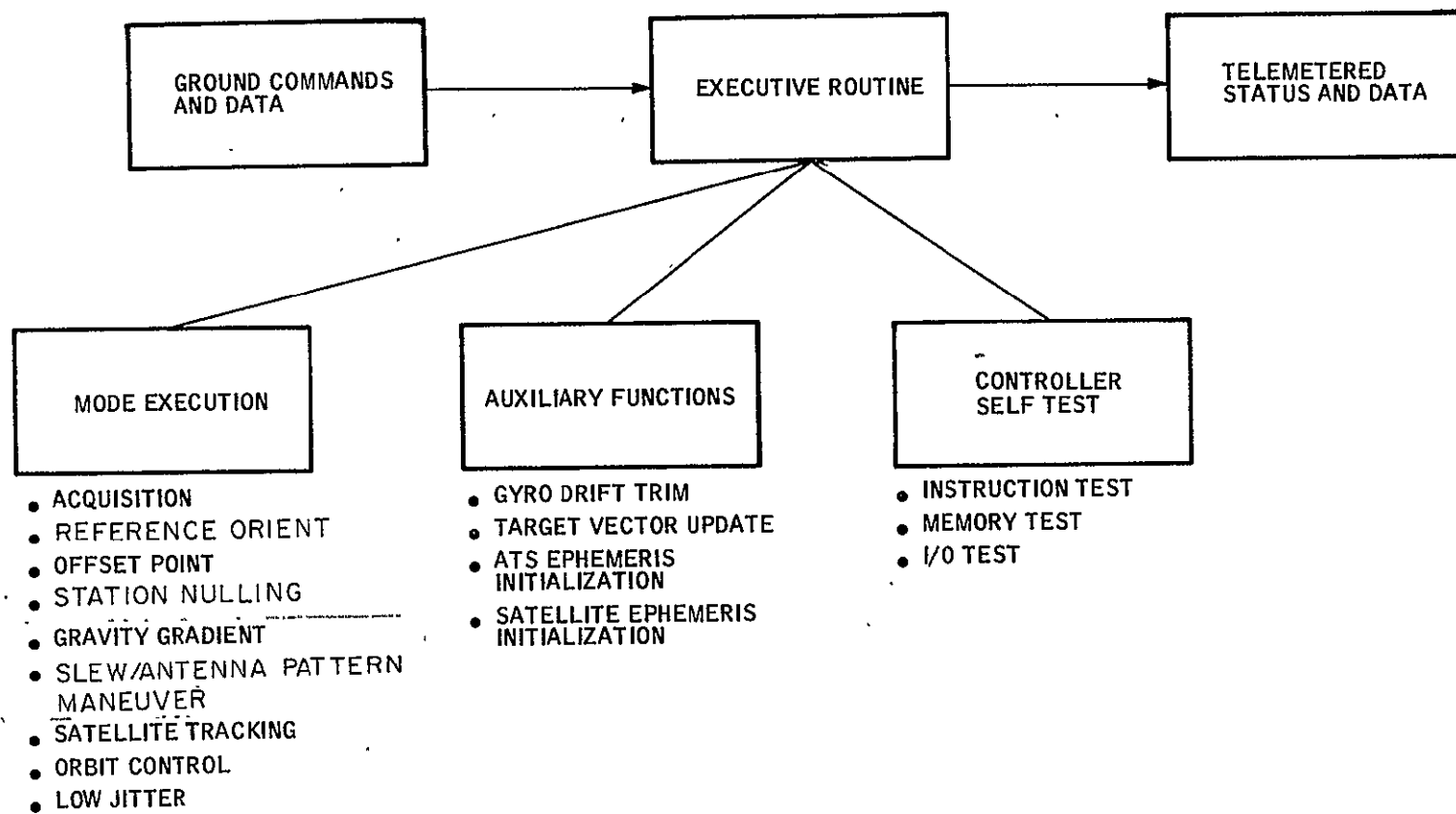


Figure 9.2-16. Digital Operational Controller Program Organization

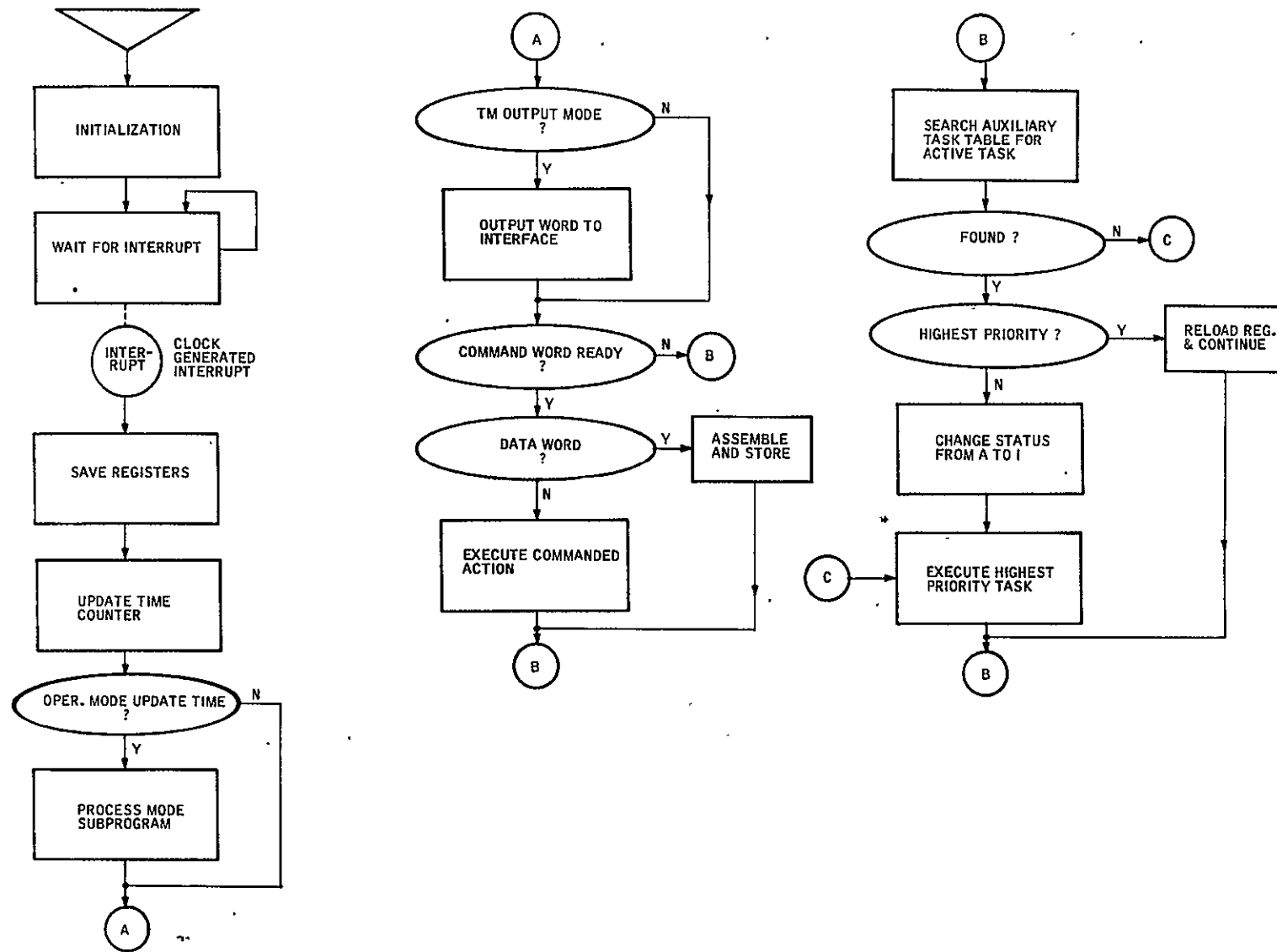


Figure 9.2-17. Simplified Digital Operational Controller Executive Routine Flow Chart

The executive now proceeds to locate the highest priority task in the auxiliary task status table, and commands it either to continue from where it was interrupted or to begin. If the task is completed before the next interrupt, control is returned to the executive and the next priority level task is initiated.

In addition to controlling mode and function execution, the executive must input, decode, and initiate commands and/or input data at rates up to two words per second. At the same time, the DOC can be outputting data to the DACU system at a maximum rate of one word every 22.5 milliseconds. Thus, the executive must provide capability to concurrently control mode execution, auxiliary function execution, self-test routine scheduling, input from the CDD, and output to the DACU, all at different rates which are not synchronized.

9.2.2.1.7 Digital Operational Controller Physical Data

Location/Mounting/Orientation/Alignment. - The DOC units are mounted with the mounting surface against the thermally-controlled spacecraft structure. Mounting holes in the baseplate are provided for this purpose. There are no special orientation and alignment requirements.

Power. - Digital operational controller power requirements are summarized below. Note that the numbers are for a single DOC, since the backup unit is in the power-off condition.

<u>Assembly</u>	<u>Power</u>
Central processor and I/O Control	4.8
Memory	6.0
Special I/O	5.0
Power supply (at 65 % efficiency)	8.7
Load interface losses	<u>2.4</u>
Total	26.9 watts

Thermal. - There are expected to be no adverse problems with respect to thermal environments for the DOC. Areas of concern are:

- A/D - D/A converter
- Memory
- Frequency-base crystal

The above items limit the temperature range of the controller. The A/D-D/A converter is designed to perform within specification over a 50°C temperature range. Outside of this range, performance is degraded.

The memory is capable of performing satisfactorily at temperatures to 100°C for an estimated 10^7 hours, but compensation is required for temperatures in excess of 50°C. Likewise, experience has indicated that temperatures in excess of 100°C will permanently damage a piezoelectric crystal.

From the above considerations, the baseplate temperatures have been determined that will ensure in-specification performance. These are:

	<u>Min (°F)</u>	<u>Max (°F)</u>
• Operational	+40	+100
• Qualification	+13	+127
• Survival (nonoperating)	-22	+140

The temperature rise from baseplate to pieceparts is controlled by careful thermal design. There are no hot spots within the controller; heat is uniformly distributed making conduction cooling more effective. Short thermal conduction paths are provided from the pieceparts to the frame and ultimately to the thermally-controlled spacecraft surface.

Vibration/Shock. - The unit has the capability of meeting vibration/shock requirements imposed by the ATS structure and launch vehicle characteristics.

The structural concept of the unit is based on the use of a stiff chassis and reinforced modules and component boards. A structure of this type has low development risk for hard-mounted applications. Stiffness implies small deformations, low stress levels, and high fundamental resonant frequencies. The proposed structure is composed of a rigid baseplate containing the mounting feet and attached truss-type structure to provide load paths and rigidity at minimum weight. The unit is base-mounted and provides a single-plane flange for a gasket seal.

The design and development experience gained on Apollo Block I and Block II and MOL space projects is directly applicable, since the structural concept and components are similar. For these devices, tests for random vibration response at the piece-part locations indicate that amplification will be limited to a maximum of three to four times the input level to the unit.

EMC/EMI/RFI. - An EMC/EMI/RFI program forms an integral part of the Phase D effort.

Careful execution of standard design practices is necessary to avoid problems. Some of these design considerations as applied to the DOC are listed below.

Bonding --

1) Provisions are included for attaching a metallic bonding strap to the device at a conductive surface.

2) The cover is bonded by controlling the mating surface finishes and use of a conductive gasket.

3) Internal metallic parts are connected through an electrically-conductive surface.

4) Connector shells use electrically-conductive finishes to assure a bond to the case.

Shielding --

1) Case - The external cover of the device is jointed to provide electrical conductivity across all joints in order to contain all internally-generated electromagnetic energy and prevent unwanted external energy from entering the device.

2) Internal - Circuit boards are grouped functionally with metallic dividers between functions to reduce coupling from one energy source to a susceptible circuit.

3) Wires - Digital-type signals are shielded to meet radiated interference requirements of the EMI control specification. Provisions are made to carry through the shield pigtail and ground it to the chassis at a point not to exceed three inches from the connector. Multiple-point grounding is applicable for the digital circuits.

Multilayer PC Boards. Decoupling techniques are used on PC boards to contain the interference energy at the board level and prevent coupling and cross-talk on the interconnecting wiring to the extent necessary for adequate functional performance.

Return Interconnection. All signal power is developed as isolated secondary power to provide isolation between primary power and secondary power returns. The secondary power returns are connected within the device and an isolated ground wire will be brought out on a connector pin for referencing to system ground.

Chassis Return. The structural tie to the device is by interconnection of the customer-provided bond strap to a suitable point which is electrically continuous with the entire device case. This connection is dc-isolated from the primary and secondary power return.

Prime Power Return. The prime power return wire is dc-isolated from the device case and secondary power return and provides the only return path for the primary input power. Provisions are included to control the interference currents generated within the converter from flowing on this wire within the limits defined in the EMI control specification.

9.2.2.2 Analog Backup Controller (ABC)

9.2.2.2.1 Physical Data and Packaging Features

ABC Weight. - Figure 9.2-18 is an outline drawing of the ABC. The device weighs 5 pounds, and its center of gravity is essentially at the geometric center of the assembly.

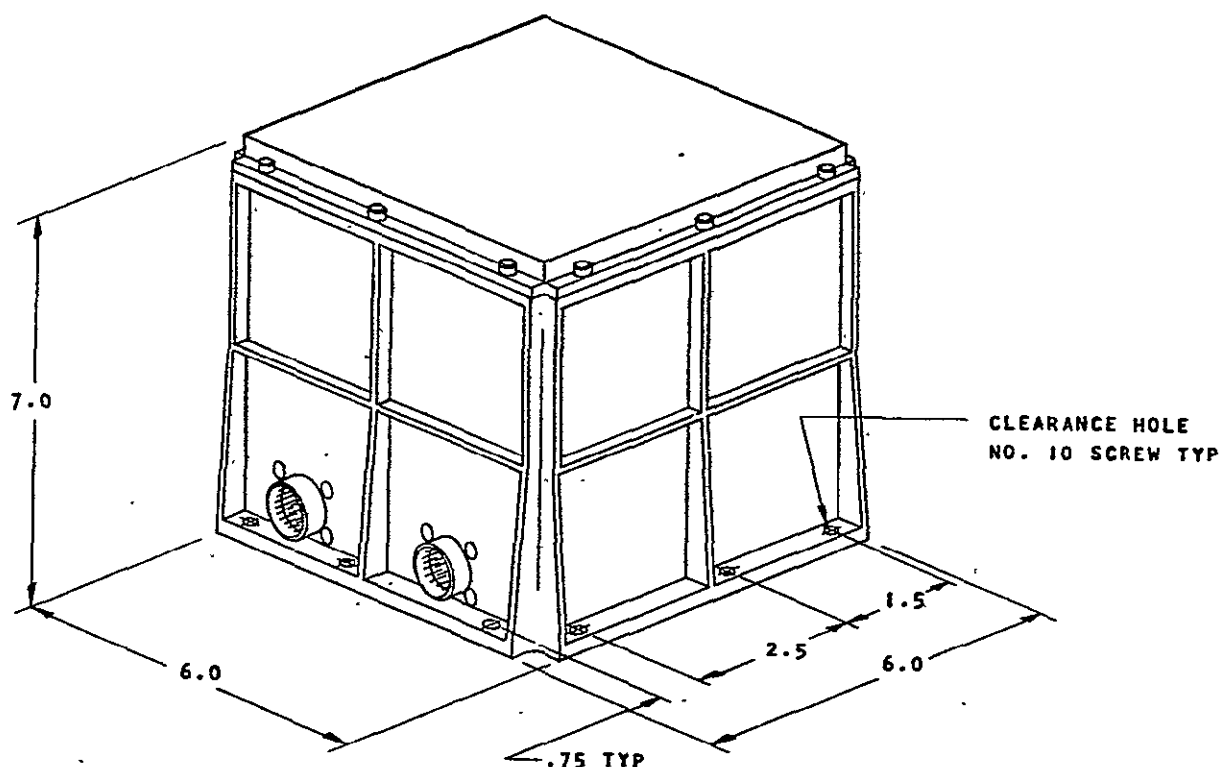


Figure 9.2-18. Analog Backup Controllers -- Outline Drawing

Packaging Features. - The ABC contains integrated circuits mounted and electrically connected to printed circuit paths on the various card assemblies. Discrete components used in the signal processing (loop shaping, filtering, gain trim, etc.) will be attached to a card either through use of standoff terminals or encapsulated modules using cordwood construction. The detailed circuitry requirements will dictate whichever method is appropriate for each piece part under consideration. Suitable encapsulation will be applied to each discrete part to provide assurance of satisfactory operation during and after vibration and thermal shock, and to isolate effects of deep-vacuum environmental conditions.

These methods were proved to be satisfactory as applied to the equipment supplied by Honeywell Inc. to Jet Propulsion Laboratory for the successful Mariner Mars '69 project. A cutaway view of the ABC showing the card assemblies is shown in Figure 9.2-19. The ABC provides access to circuit elements for changes of control parameters as needed.

The circuit design follows the pattern used in earth orbital and deep-vacuum spacecraft flights of the Gemini and Mariner Mars '69 programs. These projects required the use of high-reliability piece parts as determined by low-failure rates when subjected to critical environmental qualification tests. The circuits used in these projects also used deadband, saturating, and delayed on-off type of circuitry for jet thruster control using spacecraft position/rate sensors.

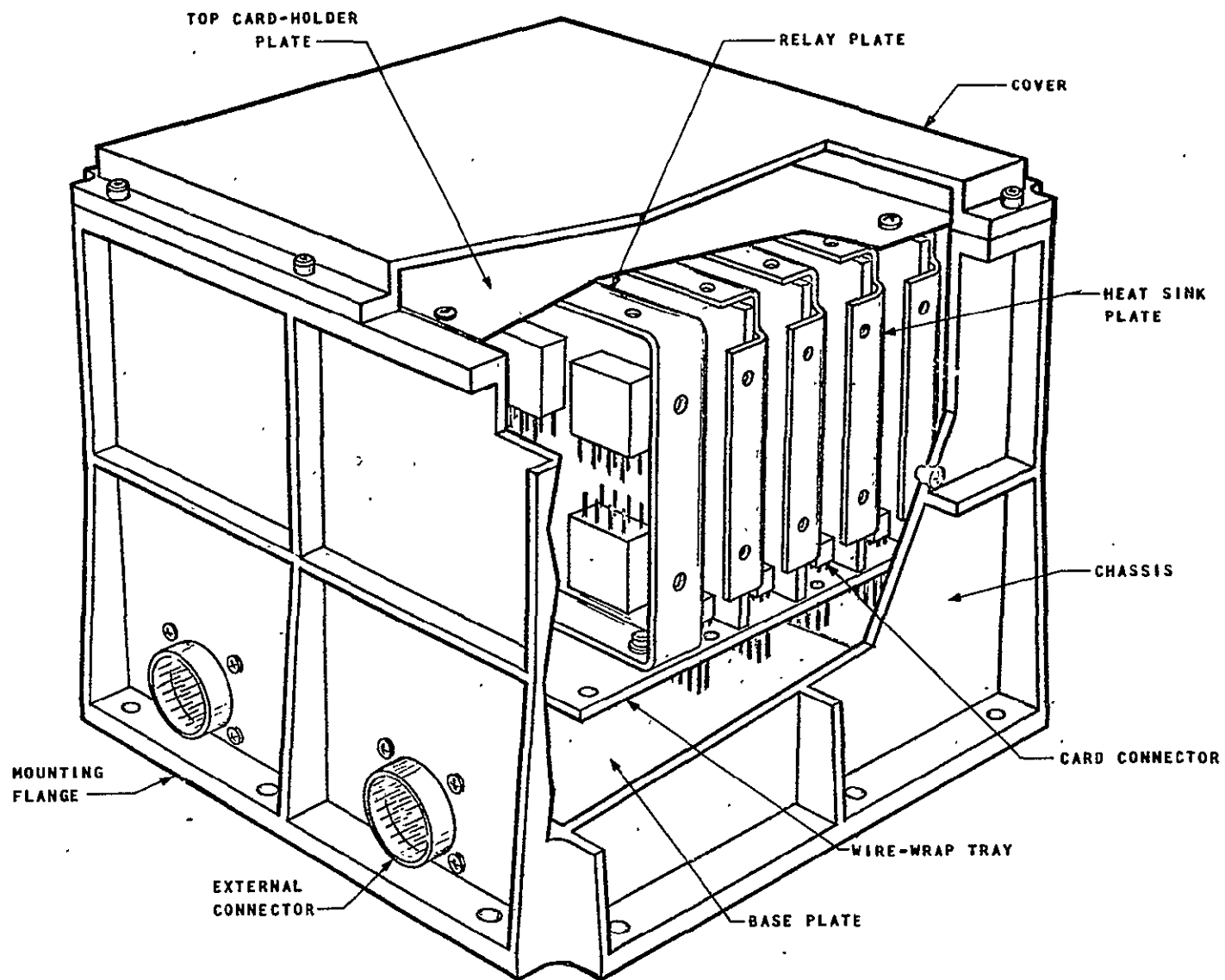


Figure 9.2.19. Analog Backup Controller -- Cutaway View

9.2.2.2.2 ABC Circuitry

The ABC contains circuitry for onboard three-axis backup control for selected acquisition and operation modes using primary and secondary analog sensors. These functions are achieved using minimum hardware consistent with backup control operation.

Circuitry to enable backup control of the spacecraft at reduced power levels is listed below:

- Deadband/Minimum Impulse Jet Control Amplifier
- Pseudo Rate Damping Control Circuitry
- Sensor Source Summing Amplifier
- Rate Command Network Logic
- Mode Selection Logic
- Power Supply
- Electromagnetic Interference Filtering

Figure 9.2-20 presents the ABC functional block diagram. This diagram shows the three-axis loops, the mode logic, and the rate logic derivations.

9.2.2.2.3 Thermal

There are no problem areas with respect to the thermal environments for the ABC. The design is relatively simple and straightforward as has been proven by past experience on previous spacecraft programs.

9.2.2.2.4 Vibration/Shock

The same packaging concept as that planned for the DOC will be used for the ABC, and, based on past experience, no problems are anticipated.

9.2.2.2.5 EMC/EMI/RFI

Past experience on other space programs will be used in designing the EMI rejection circuitry necessary for the ABC. The considerations discussed previously for the DOC will also be observed in the design of the ABC.

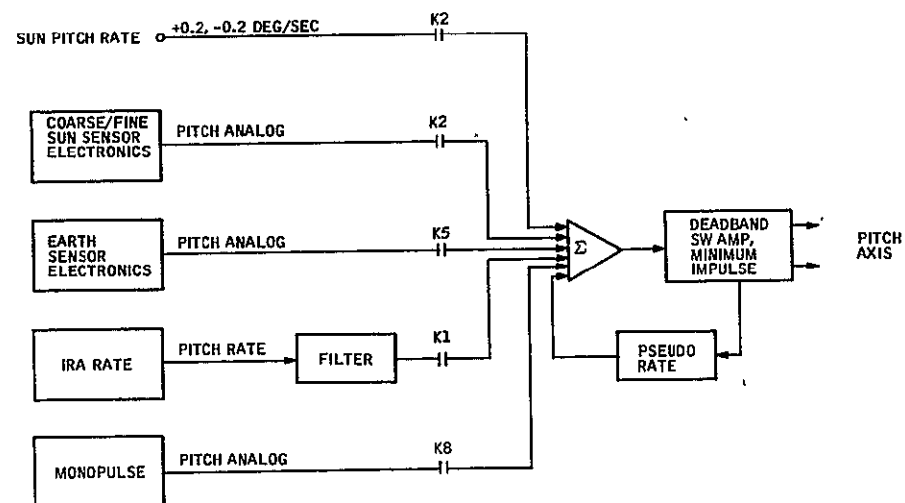
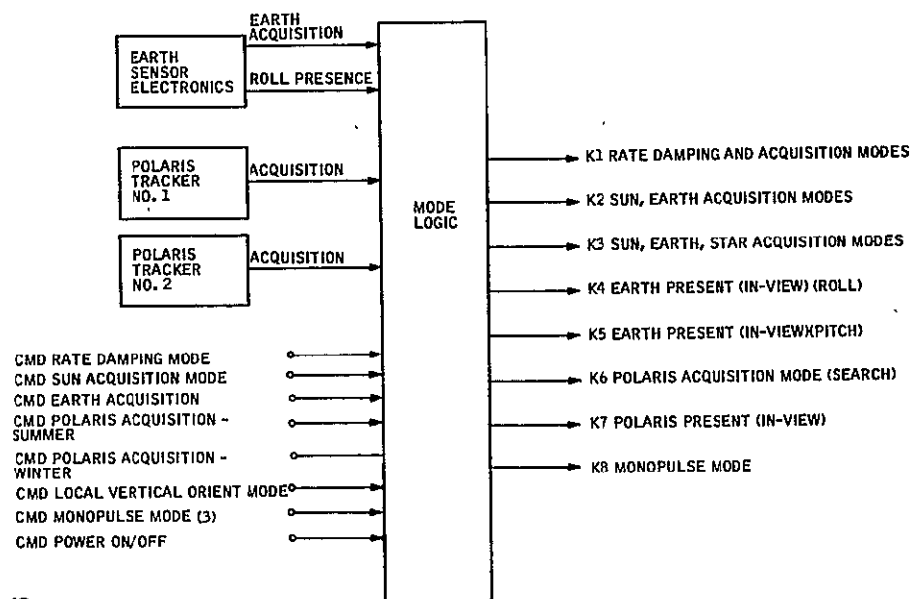
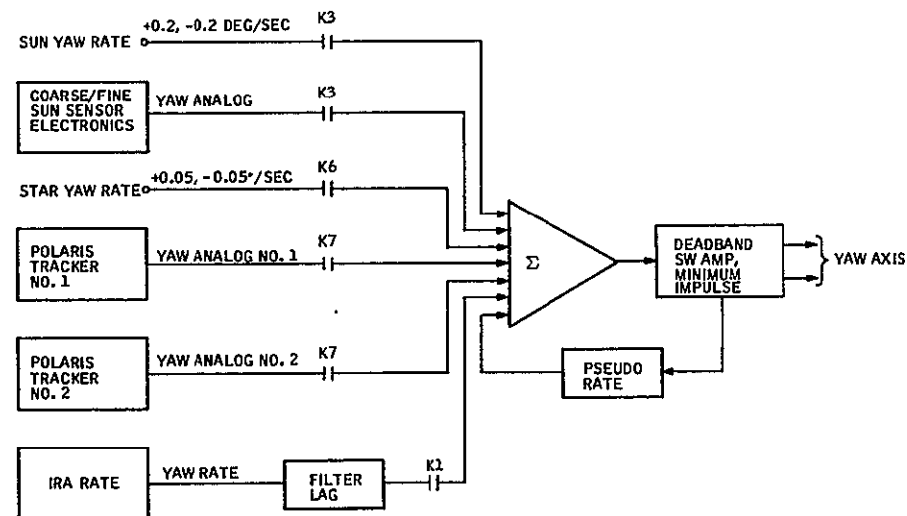
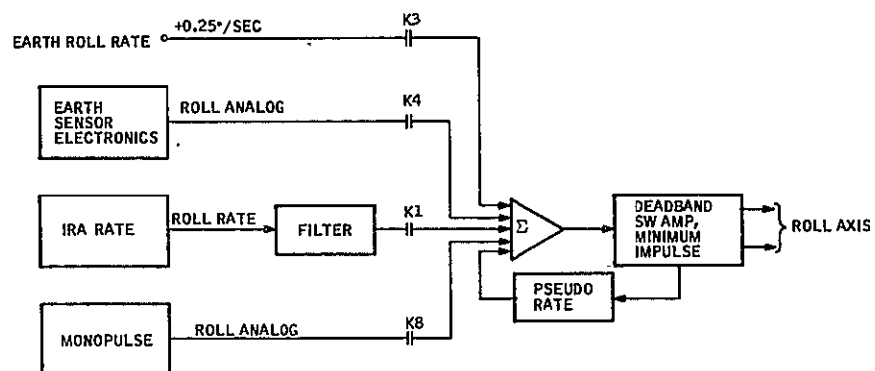


Figure 9.2-20. Analog Backup Controller Functional Block Diagram

9.2.2.3 Earth Sensor

9.2.2.3.1 Weight/Dimensions

Sensor Head. - The sensor heads are identical and weigh 3.7 lbs each. The head is machined from a solid block of aluminum, and is designed to retain a maximum of structural stability. Figure 9.2-21 is a photograph of the head assembly.

The head dimensions are 5.7 (w) x 6.00 (l) x 4.0 (h) inches, plus mounting feet.

Electronics. - The dual-axis electronics are packaged separately, and measure 7.0 (w) x 11.0 (l) x 5.0 (h) inches. The electronics are built up through layers within separate aluminum frames which are bolted together to form the final package..

The ESA electronics are predicted to weigh 6.10 lb although the assembly remains a prime candidate for Phase D weight reduction.

9.2.2.3.2 Electrical/Optical/Mechanical Description

Electrical Circuitry Description. - Earth sensor circuitry is divided into the following subsections:

- Bolometer bridge
- Radiance processing
 - 1) Gain section
 - 2) Processing section
- Scan generation
- Visor drive
- Scan plane offset drive
- Visor readout
- Digital output signal computation
- Sun sensor
- Sun avoidance circuits
- Power processing

Bolometer Bridge Circuit -- The basic radiance detection circuits consist of two thermistor bolometer flakes connected in a bridge arrangement.

Only one of the two flakes is exposed to the radiant energy collected by the telescope. Thus, the input radiance causes a small resistance change in one flake, unbalancing the bridge and providing an output signal.

The bridge circuit must be biased from well-regulated low-noise supplies to prevent noise signals from causing a signal-to-noise degradation.

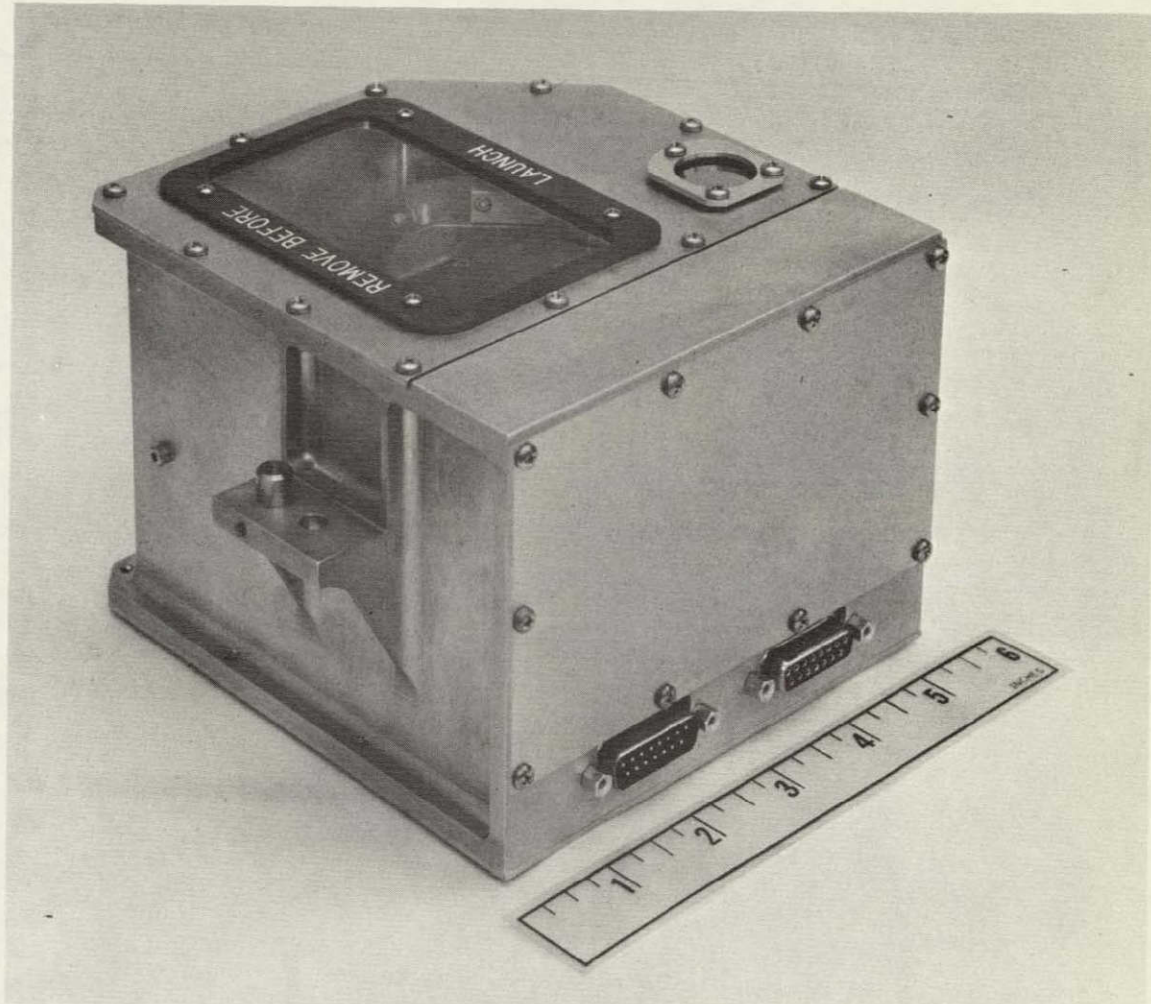


Figure 9.2-21. Flight Head Assembly

Radiance Processing Circuits -- The radiance processing circuits amplify, filter, d-c restore, and threshold detect the earth radiance signal output from the bolometer bridge circuit.

The radiance gain circuit block diagram is shown in Figure 9.2-22.

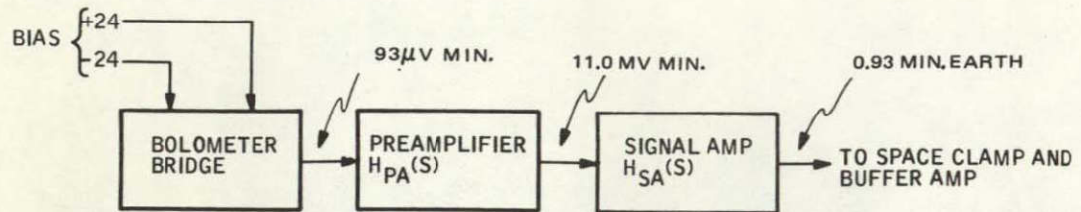


Figure 9.2-22. Radiance Gain Circuits

The processing section is shown in block diagram form in Figure 9.2-23.

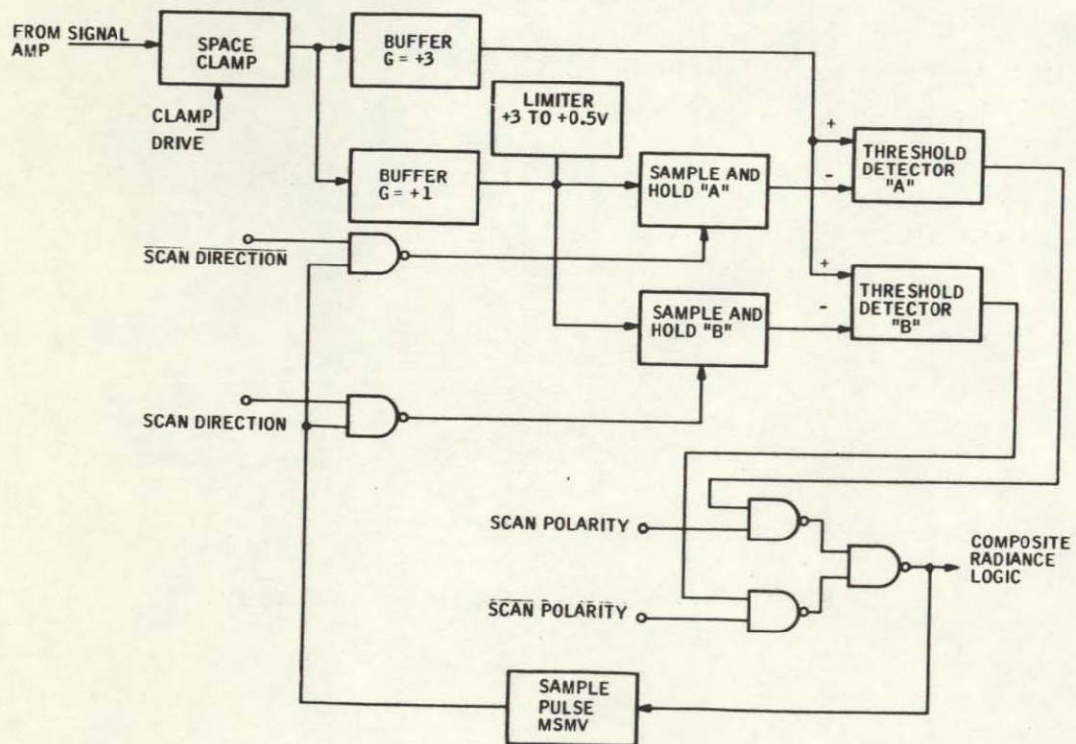


Figure 9.2-23. Radiance Processing Block Diagram

From a keyed clamp which re-establishes a cold space radiance signal reference level, the signal is sent to two separate buffer stages.

One buffer stage with a gain of 3 feeds the noninverting input of each of two threshold detectors. The second buffer stage has unity gain and drives two sample and hold circuits.

The sample and hold circuits each provide inputs to the inverting input of the threshold detector.

Scan Generation Circuits -- The basic function of the scan generation circuits is to provide the 5-Hz sinusoidal scan drive voltages to the scanning mirror of each sensor head. A scan polarity logic waveform is also developed.

The basic oscillator provides both a sine wave and a cosine wave output. By driving the scan mechanism of one head with a cosine wave and the other head scanning mechanism with a sine wave, a more uniform sensor power dissipation is obtained, since one mirror is at its zero rest position whenever the other mirror is at a maximum scan angle.

The basic oscillator consists of two IC operational amplifiers used as integrators. A noninverting integrator is cascaded with an amplitude-limited inverting integrator in a feedback loop to provide the oscillator action.

The basic circuit arrangement is as shown in Figure 9.2-24. The time constant R_1C_1 in the figure is deliberately made slightly larger than RC so that the circuit is slightly unstable.

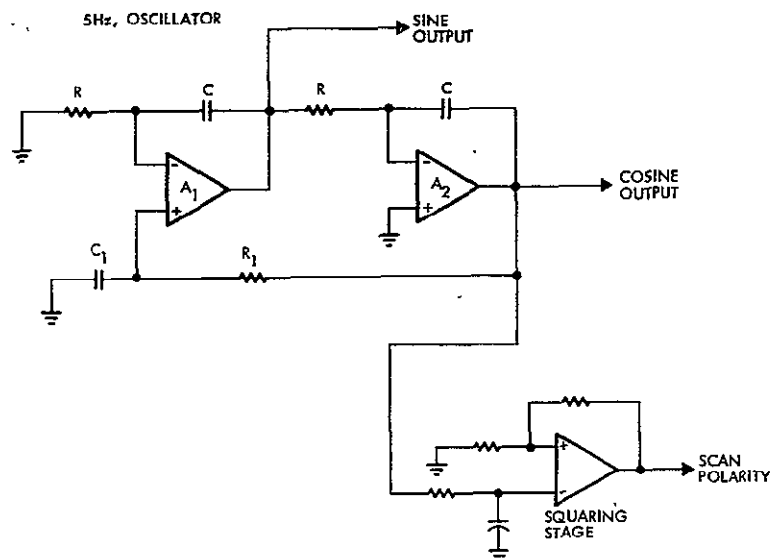


Figure 9.2-24. Scan Generator Simplified Diagram

Visor Drive Circuits -- The visor drive circuits consist of a multi-input current power amplifier and a rate feedback generator to provide rate damping.

The rate feedback increases the mirror damping factor (ξ) from 0.005 undamped to $\xi = 0.6$ with full rate feedback.

An implementation block diagram is shown in Figure 9.2-25.

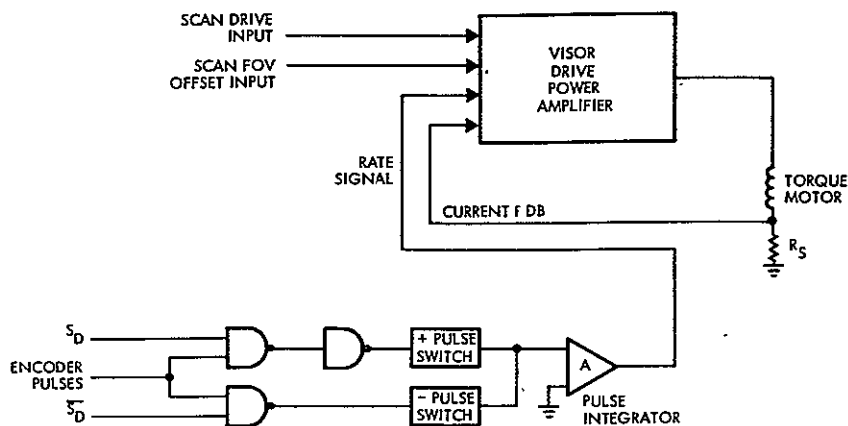


Figure 9.2-25. Visor Drive Circuits

Scan Plane Offset Drive Circuits -- The scan plane offset mirror uses a torque motor identical to the visor torque motor; thus, the drive and rate control requirements are very similar.

Visor Readout Circuits -- The visor readout circuits process the visor scanning mechanism incremental encoder output to provide signals for angle development. These circuits are shown in Figure 9.2-26.

The visor mechanism incremental encoder contains three pickoff photodiodes and three gallium arsenide light diodes positioned in pairs on opposite sides of a reticle pair.

Two of the photodiodes are called "fine track pickoffs" and provide quadrature-phased sinusoidal signals which are amplified and squared about their zero crossings. Each zero crossing transition is then differentiated and the two signals are combined to give the pulses which each represent 0.044 degree of scan angle rotation.

The phase relationship between the two squared pickoff outputs is constantly checked by a flip-flop to establish a measure of scan rotation direction. The flip-flop is either held set or reset depending upon the scan direction as determined by the signal phase relationship.

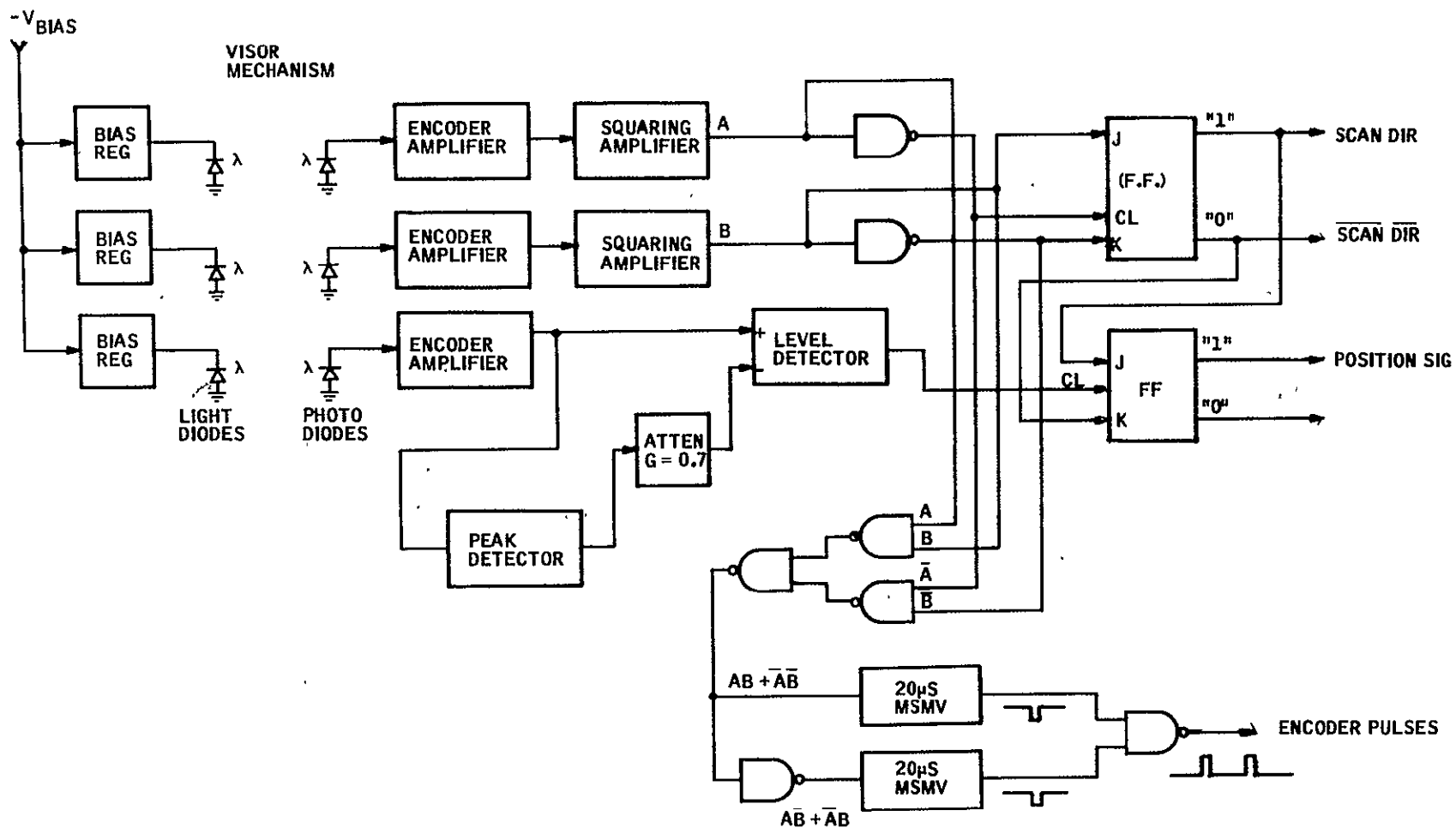


Figure 9.2-26 Visor Readout Electronics

The third pickoff is the null reference pulse outputs. A 9-line geometric correlation pattern is used to provide this output which is a series of pulses generated at the mirror zero position with the center pulse at least 40 percent greater than the adjacent pulses. This technique gives a much larger signal than the single-slit approach.

After amplification, the null reference pickoff photodiode output is peak detected and used to set the trip voltage of a threshold detector at 70 percent of the detected peak. The amplified reference pulse is also fed directly to the other input of the threshold detector.

This technique provides reference pulse detection at the same relative location on the pulse regardless of pulse amplitude.

The threshold detector output is used to trigger a flip-flop providing the scan position logic signal which is used in the digital output signal computation.

Sun Avoidance Circuits -- The function of the sun avoidance circuits is to develop a d-c bias voltage which is used to automatically offset the scan plane to avoid sun interference with the earth radiance signal. An overall block diagram of these circuits is given in Figure 9.2-27.

Digital Output Computation Circuits -- The Phase D earth sensor will contain circuits required to develop the digital output errors.

These circuits develop a digital word representing the angular deviation of the sensor null reference axis from the center of the earth in a particular scan plane. This function is performed by processing incremental angular encoder pulses from the scanning mechanism in each head. Each sensor head, making up the complete earth sensor, has identical and independent computation circuits.

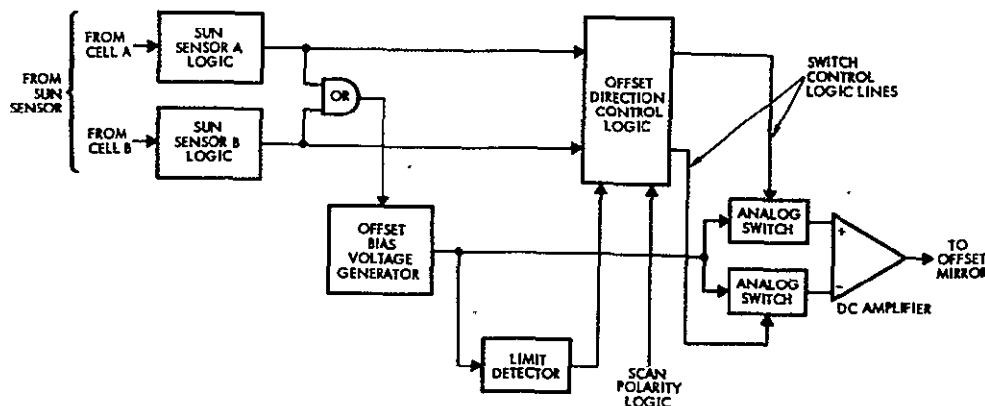


Figure 9.2-27. Sun Avoidance Circuits Block Diagram

Basic error generation is accomplished by counting encoder pulses in an up-down counter in accordance with control logic signals. The resultant output information is then available as a 12-bit plus sign binary number.

The angle computation circuits are shown in block form in Figure 9.2-28.

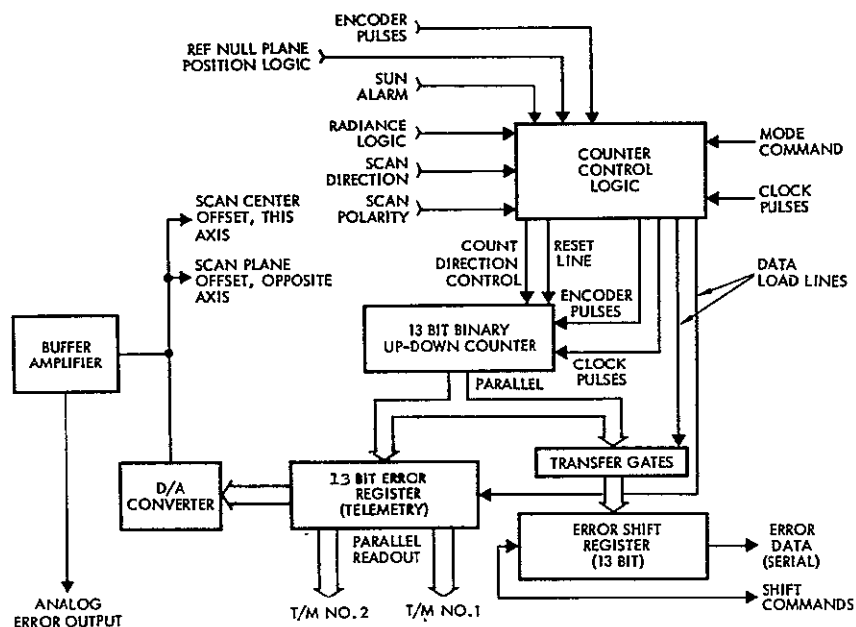


Figure 9.2-28. Angle Computation Circuits (Single-Channel)

Power Distribution Circuits -- The power distribution approach planned for the phase D earth sensor is shown in the block diagram of Figure 9.2-29.

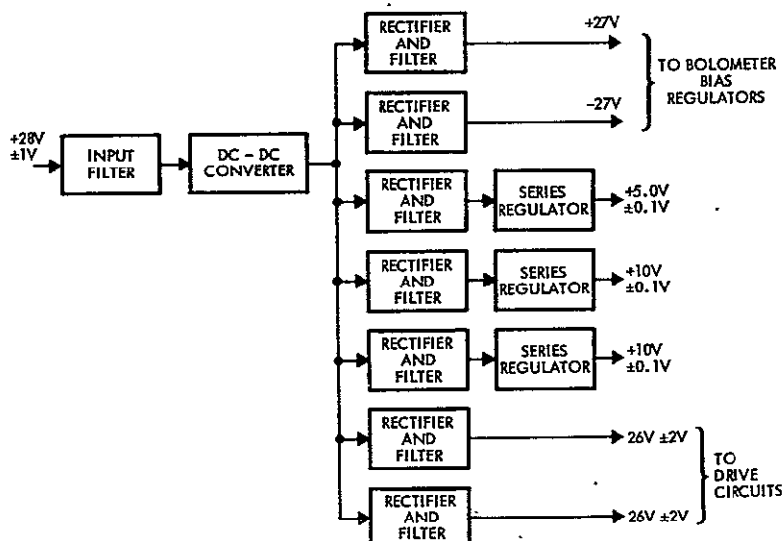


Figure 9.2-29. Power Distribution Circuits

Optical/Mechanical Description. - The sensor assembly consists of individual pitch and roll sensing heads and an electronics package. The following sections describe the optical and mechanical design. All photographs are of actual ATS hardware.

The head contains the following subassemblies:

- Scanning mirror
- Offset mirror
- Bolometer telescope
- Sun sensor
- Bolometer excitation and preamplifier
- Sun sensor and encoder preamplifiers
- Machined head, plus cover plates
- Removable dust cover
- Alignment mirror

Scanning Mechanism -- The basic approach to detecting the infrared radiance involves scanning the instantaneous FOV of the immersed bolometer detector across the earth. The scan mechanism represents a challenging design problem for a long-term space application. The approach chosen for the ATS ESA, shown in Figure 9.2-30, has the following characteristics:

- 1) The mirror is suspended by flexure pivots, not bearings. This technique avoids lubrication problems in the space environment and eliminates bearing wear. The flexure pivots are commercially available and have been space-qualified.
- 2) The mirror drive is provided by a brushless d-c torque motor which is space-qualified. This drive eliminates the need for slip rings without having to resort to an a-c drive and the additional circuit complication for providing a field coil excitation. The mechanism is caged during launch to minimize the effects of vibration.
- 3) Angular position is measured using an incremental digital encoder. This encoder provides electrical signals which are used to determine the angular position of the mirror relative to a "zero crossing" position. The "zero crossing" position indication is provided by the encoder when the mirror passes through the reference null position, normally at the center of the earth sensor FOV. This encoder offers the high accuracy, linearity, and stability over a wide temperature as is required for accurate off-null performance. These features are not available in any existing analog readout mechanisms.

Offset Mirror -- The operation of the ESA during offset maneuvers is made possible by positioning the scan plane in a direction perpendicular to that plane. The attitude deviation from null, measured by the orthogonal head, is utilized to retain the scan plane on the maximum chord length. The scan plane positioning accuracy does not enter into the attitude error calculation; thus the requirement for offset mirror attainment of the commanded position is one degree.

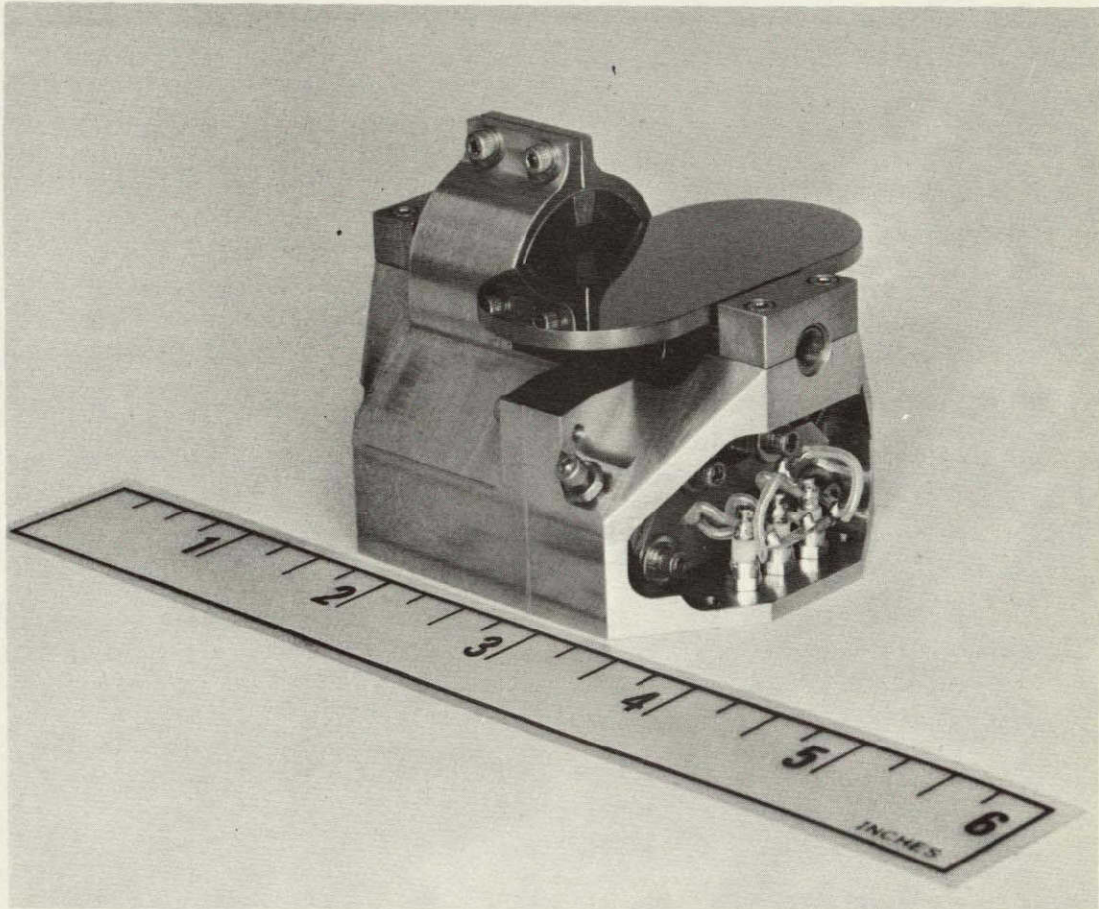


Figure 9.2-30. Scanning Mechanism

The offset mirror size is larger than the scanning mirror in order to properly "reflect" the instantaneous FOV over the 26-degree amplitude scan, plus the biasing of that scan along the scan direction.

The offset mirror assembly is shown in Figure 9.2-31. The mirror assembly consists of the polished beryllium mirror, flex pivot support, and the same d-c torque motor as utilized in the scanning mechanism.

Bolometer Telescope -- The bolometer telescope parts are shown in Figure 9.2-32. The assembly consists of a simple germanium meniscus lens, a spectral cut-on filter, and a thermister bolometer immersed in a hemispherical germanium lens. These components are bonded in place in an aluminum barrel.

The interference filter, deposited on the germanium lens, sets the lower limit on spectral sensitivity; the cut-on wavelength is 13.4 microns. The natural germanium rolloff provides the upper bound on spectral sensitivity at approximately 25 microns. The radiance in this broad band is sensed with half-power points of 14 and 17 microns.

Sun Sensor -- The sun sensor (Figure 9.2-33) is a very simple device consisting of a small reflecting mirror, two circular FOV defining apertures, and a split silicon solar cell.

The mirror is located just below and in front of the telescope assembly for the bolometer. This allows the sun sensor FOV to be aligned with and scanned with the bolometer IFOV by the scanning mirrors.

Flight Head -- Figure 9.2-34 shows the flight head with the front cover plate removed. The mounting arrangement of the scanning mechanism, the offset mirror, and the bolometer telescope is shown in this view.

9.2.2.3.3 ESA Power

The power consumption of the earth sensor is a strong function of the offset angles in the two axes. Estimates of mean power consumption have been made, based on measurements in the Phase C unit and calculations for the additional Phase D circuits. Two different coil resistance torque motors are under consideration for the Phase D scanning mirrors. Depending on motor selection, the expected Phase D unit power dissipation is either:

a) 9.7 watts at zero offset plus 0.35 watt per total (pitch and roll) degree offset (175-ohm motor)

b) 9.6 watts at zero offset plus 0.25 watt per total (pitch and roll) degree offset (100-ohm motor).

9.2.2.3.4 Thermal Considerations

The thermal situation of the earth sensor differs greatly between the electronics and the head assemblies.

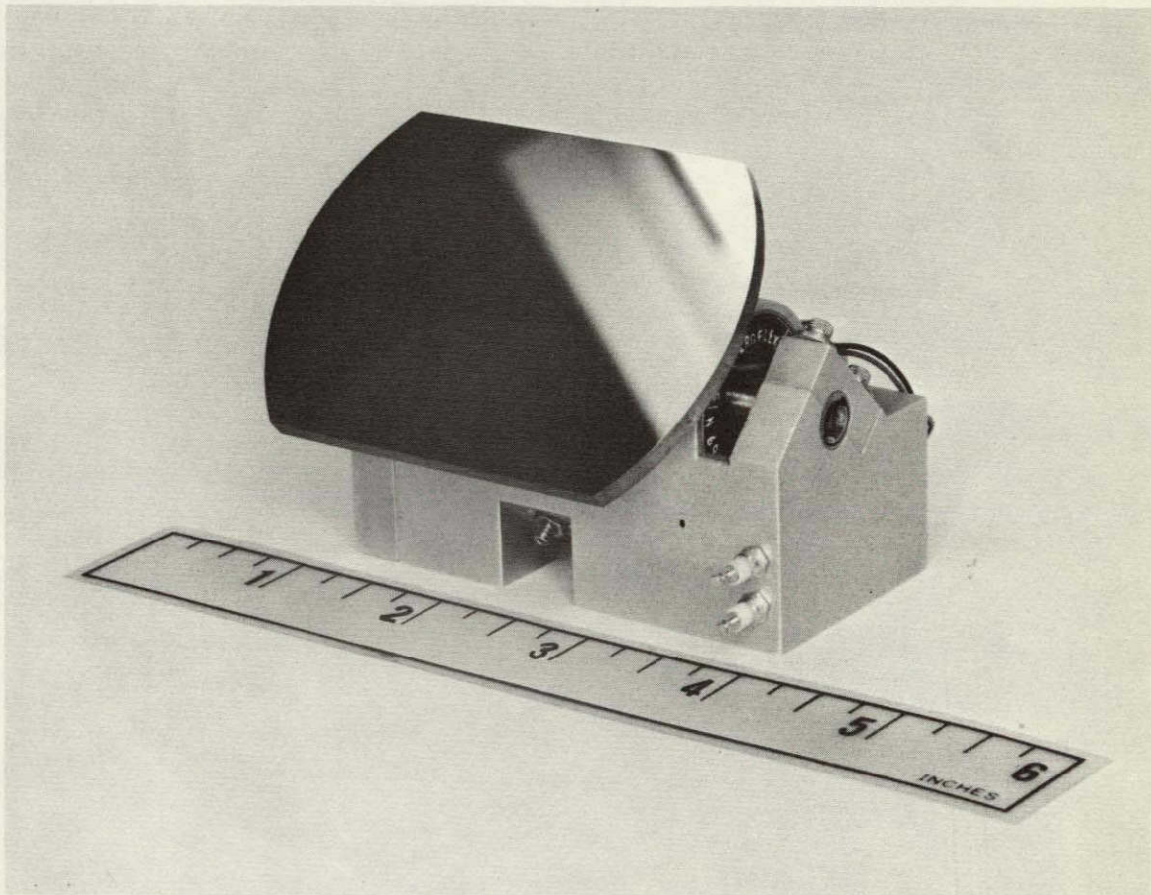


Figure 9.2-31. Offset Mirror Assembly

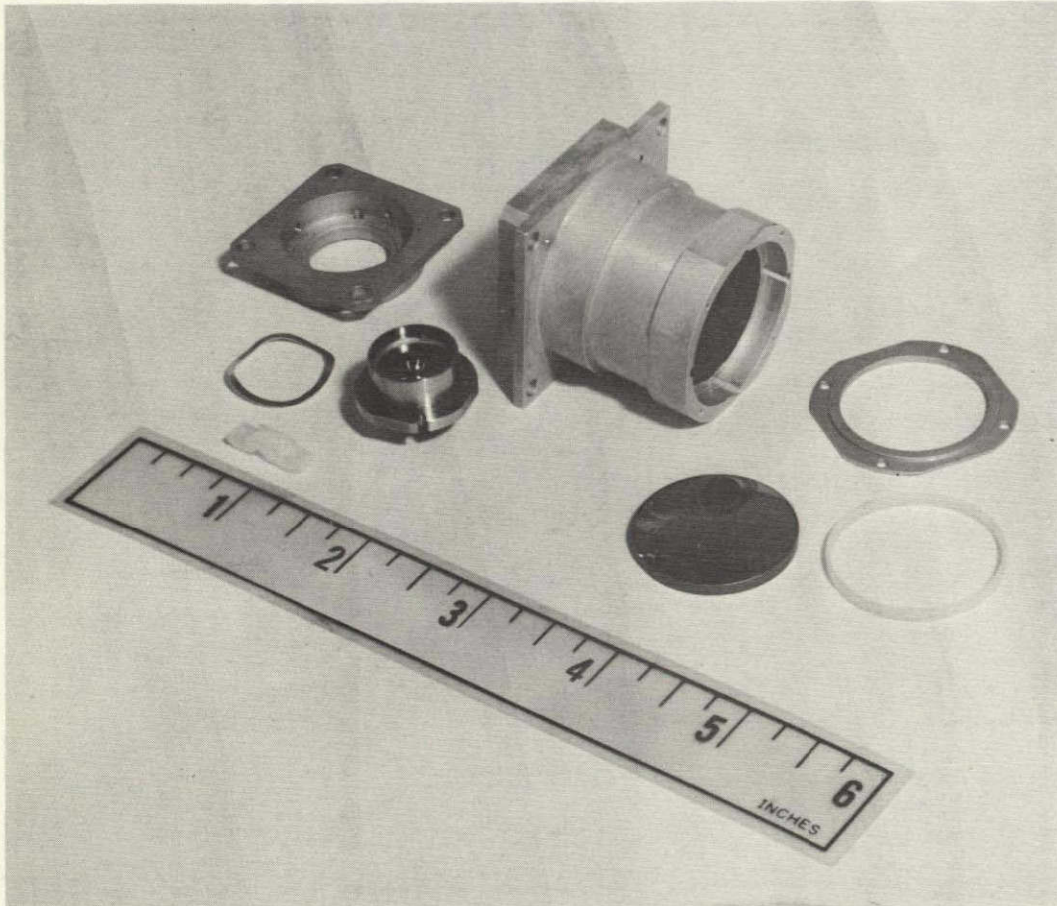


Figure 9.2-32. Bolometer Telescope Parts

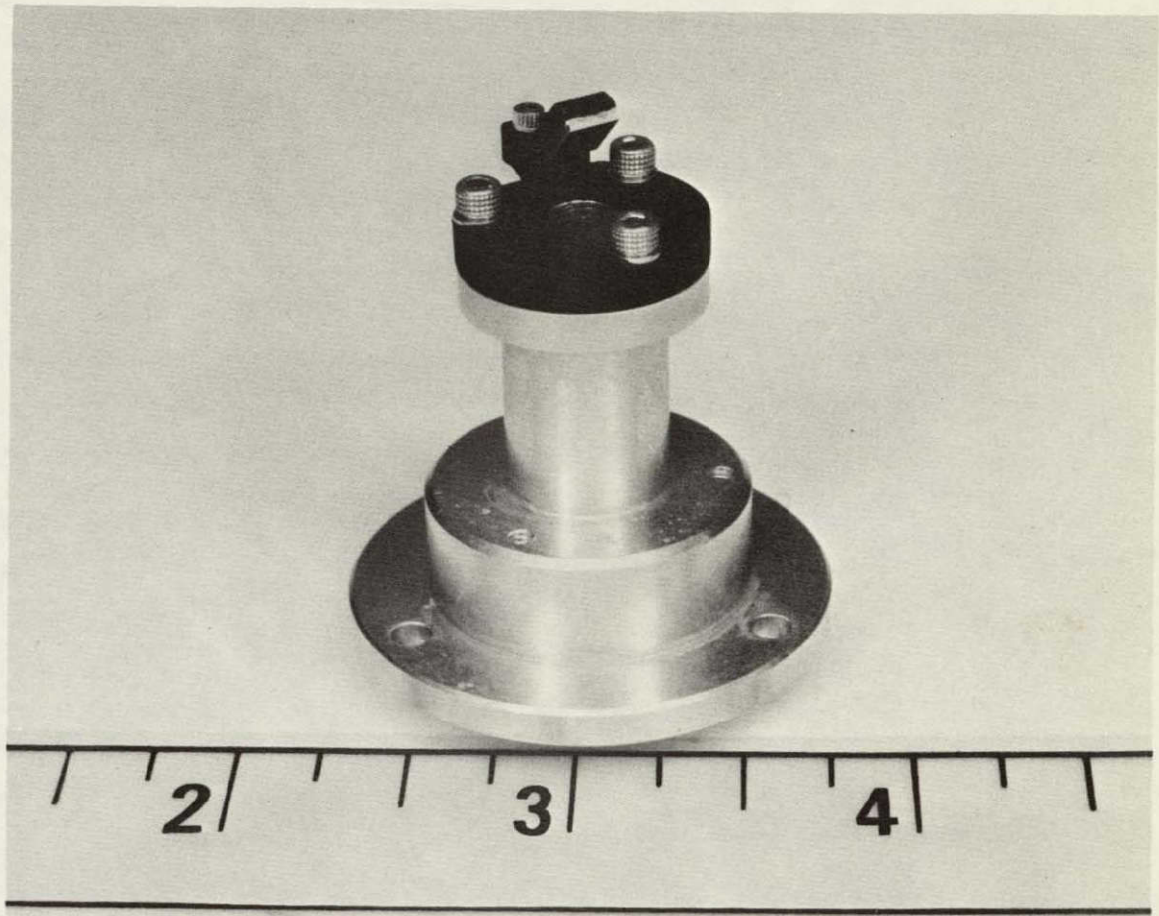


Figure 9.2-33. Sun Sensor Subassembly

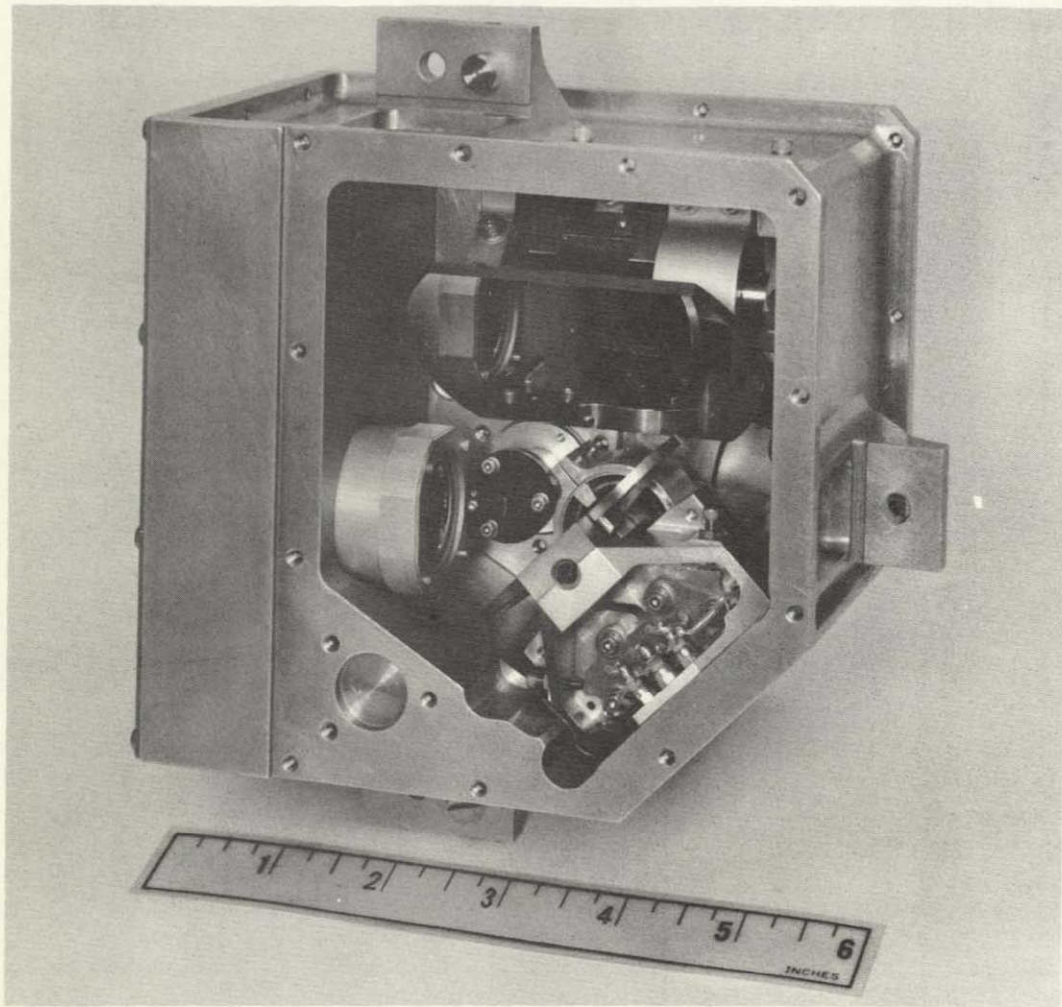


Figure 9.2-34. Flight Head Assembly with Front Cover Plate Removed

The electronics assembly is mounted within the protective environment of the structure. The $20^{\circ}\text{C} \pm 10^{\circ}\text{C}$ environment is mild and allows for derated operation of all components. Higher power components are to be mounted on the inside walls of the assembly for direct thermal transfer.

The thermal situation in the head assembly is not so mild and is more difficult to analyze. A preliminary thermal analysis was performed to obtain "ball park" values for the maximum and minimum temperatures anticipated for the head assembly.

The analysis results show a calculated average chassis temperature as follows:

Treatment of Front Surface Which is Exposed to Space	Chassis Temperature ($^{\circ}\text{F}$)	
	Maximum	Minimum
Black paint	137	34
10 layers of MLRI insulation	109	47
White paint	100	35

The basic assumptions made to obtain the above results are contained in Volume I-C of the ATS F&G Proposal.

9.2.2.3.5 Vibration/Shock

The nature of the mechanical mirror mechanisms requires that provision be made to survive the vibration environment. It is presently planned that a launch mode be designed into the unit. In this mode the mechanisms will be actively caged. The normal drive circuits will be mode-switched to provide a high rate gain and no scan drive, such that the devices are tightly held about their null position. This implies that some or all of the circuits be powered; for simplicity, it is anticipated that all of the earth sensor circuits will be powered in order to eliminate the unreliability of power switches.

9.2.2.3.6 EMC/EMI/RFI

The ESA design has been guided by extensive past experience in solutions of electromagnetic problems. The ESA is particularly sensitive to EMI in the radiance detection and gain circuitry. The handling of low signal leads from the bolometer to the preamplifier requires careful attention. Problems with charge accumulation in these leads have caused undesired output during mechanical inputs such as jet firings.

The program guidelines involve: \

- Control plan for Phase D
- Design practice
- Test program

9.2.2.4 Polaris Sensor Physical Description

9.2.2.4.1 Weight/CG/Inertia

Eight pounds is the expected weight of the Polaris sensor (excluding the stray light baffle and conditioning electronics).

The stray light baffle is expected to contribute 2.0 pounds and the conditioning electronics 3.0 pounds.

9.2.2.4.2 Size/Packaging

The Polaris sensor (excluding the stray light baffle) is a rectangular parallelepiped envelope with dimensions of 12.0 inches long by 5.6 inches wide by 5.2 inches deep. The stray light baffle occupies a volume of 7.0 inches long by 5.4 inches wide by 12.5 inches deep. The conditioning electronics are 4.8 inches long by 3.0 inches wide by 2.0 inches deep.

9.2.2.4.3 Polaris Sensor Primary Structure

A simplified block diagram of the Polaris sensor and conditioning electronics is shown in Figure 9.2-35.

The primary structure of the Polaris sensor, which provides the four mounting feet for attachment to the baseplate within the EVM and which provides the chassis for mounting the lens assembly, the image dissector tube (IDT), electronics, connector, etc., is machined from a solid bar of 6061-T6 aluminum alloy. The geometrical shape is designed to provide a lightweight, highly efficient and stable member (in terms of vibration) that is most favorable for the double kinematic mount of the image dissector tube.

Figure 9.2-36 shows the Polaris sensor chassis. The primary structure is enclosed with sheet aluminum that receives a highly polished finish to present a thermal surface that has high emissivity and low absorptivity. This sheet aluminum is fastened to the primary structure with stainless-steel fasteners.

Stray Light Baffle. - The stray light baffle, which is a modification of the successful MM'69 Canopus baffle, is a separable assembly that attaches to the Polaris sensor and is aligned to the sensor's optical axis. Since the baffle must provide sufficient attenuation to the off-axis and reflected energy inputs from a source (the sun) of very large apparent magnitude ($m_v = -26.7$), the baffle is designed with very sharp edges (0.00003-inch radius). Analysis and experience has shown that reduction of the scattering area improves the scattered light attenuation more than blackening, with its associated paint buildup and increase in radius. The stray light baffle is shown in Figure 9.2-37.

Lens Assembly. - This system uses an 0.84-inch effective focal length refractive lens with a 0.48-inch aperture. This system was computer-designed to accommodate the curved focal plane of the IDT. The glass (which forms the end of this tube) is also part of the lens system. Thus, as contrasted with the MM'69 tracker, no fiber-optic face plate is used in the proposed design.

Adjustments are incorporated to provide alignment with the IDT in all three axes. After alignment, the lens assembly is locked into its permanent orientation with the tube. The lens assembly creates compartments that would trap gas. Porting holes and venting grooves are incorporated to minimize outgassing during evacuation tests. The lens assembly is shown in Figure 9.2-38.

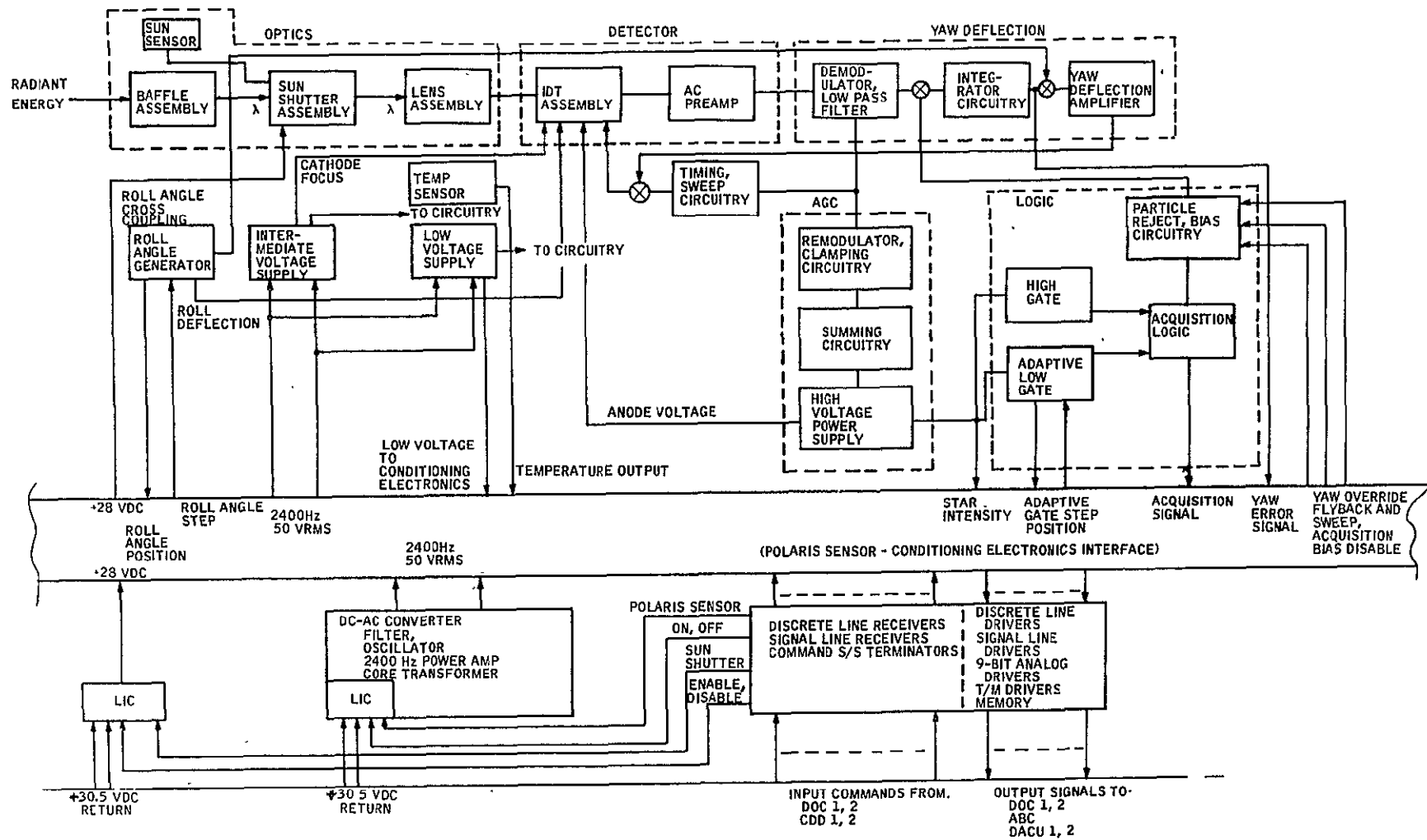


Figure 9.2-35. Polaris Sensor, Conditioning Electronics Simplified Block Diagram

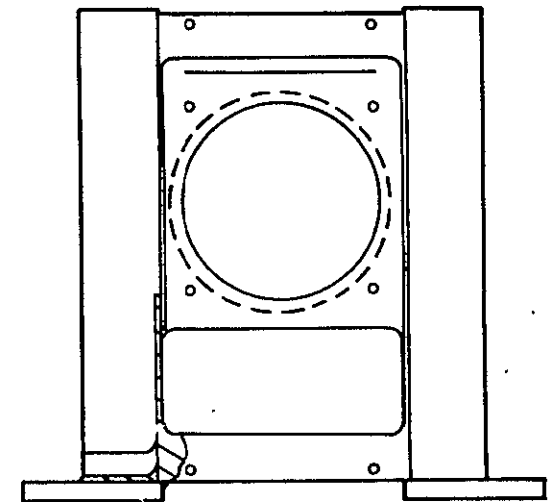
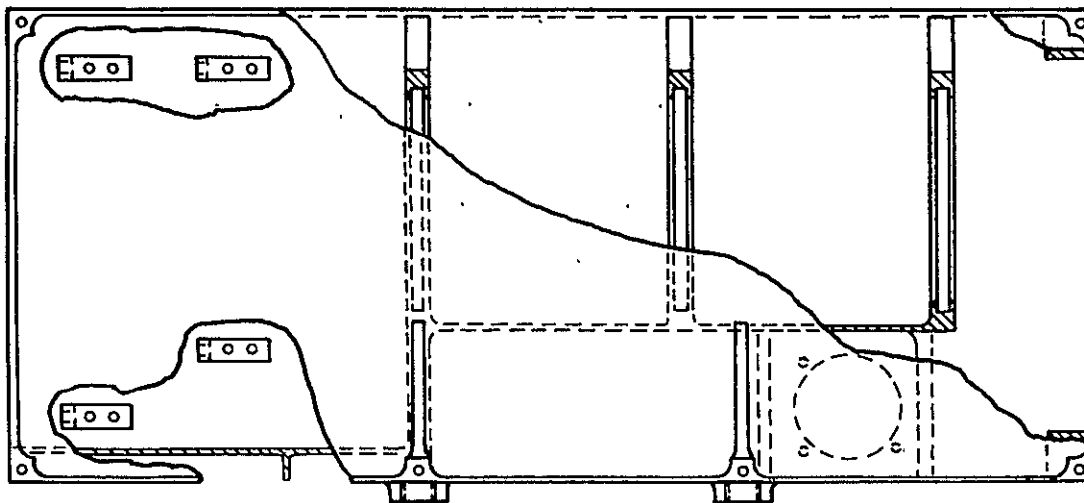
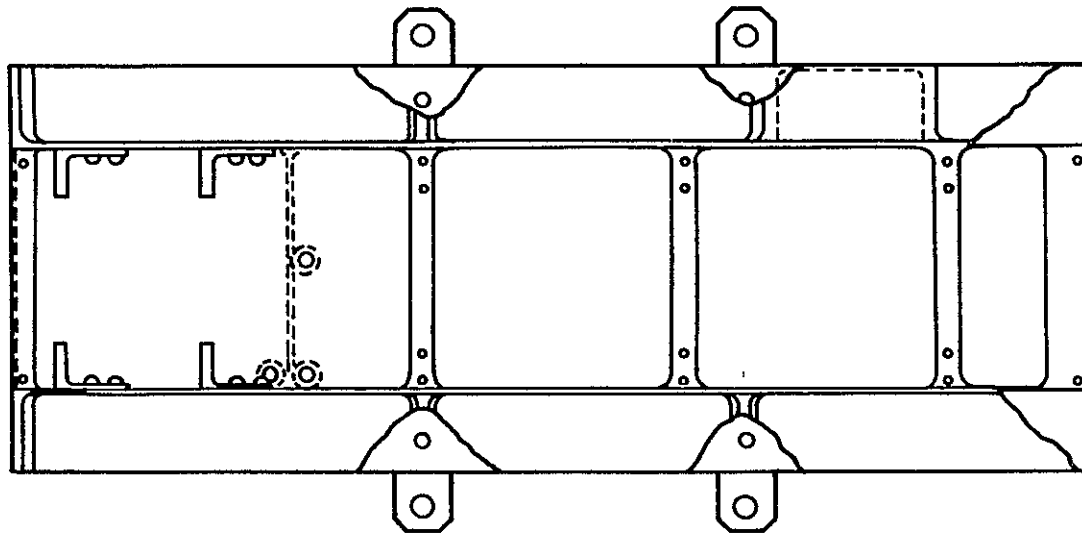


Figure 9.2-36. Polaris Sensor Chassis

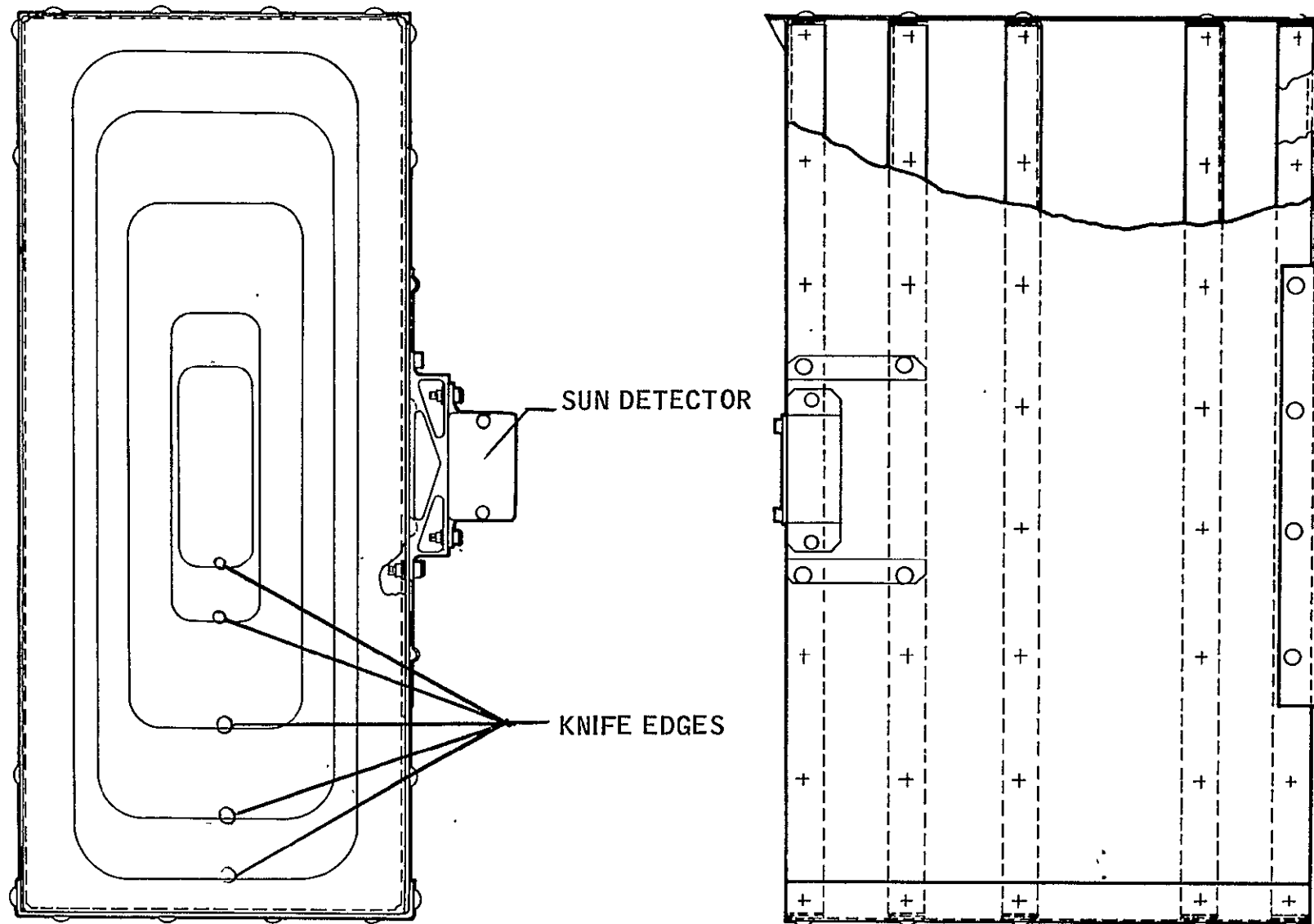


Figure 9.2-37. Stray Light Baffle -- Second Stage

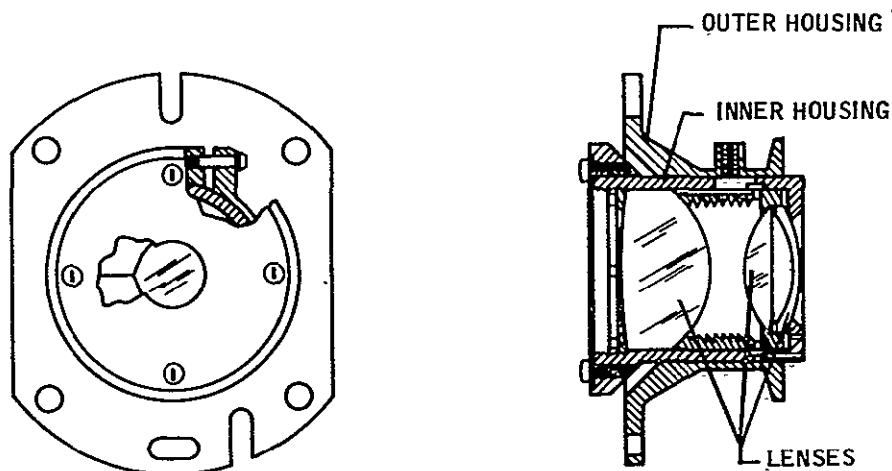


Figure 9.2-38. Lens Assembly

Materials used in this system are the highest grade of synthetic quartz and flint glass. This minimizes fluorescence of the glass under ultraviolet radiation and also prevents transmission losses within the lens system from high-energy particle radiation. The glass surfaces are coated with a special antireflection coating (from Optical Coating Laboratories) that supplies a protective surface to the glass and minimizes system reflections (thus, enhancing transmission). In this manner, glare and hazing effects are prevented.

These same materials were used in the MM'64 design and have operated successfully for two years. An identical system is being used in the MM'69 mission.

Image Dissector Tube (IDT). - The Polaris sensor uses an Image Dissector Tube to provide the required FOV, without the need for mechanical gimbals. Furthermore, the sensor has no moving parts; thus, fatigue and wear problems associated with vibrating or oscillating parts and rotating wedges or reticles are avoided. Also, since only one detector is used, the problems of spectral response and gain uniformity, alignment of the detectors, and drift of the detector and circuitry inherent in multiple-detector systems are avoided.

An IDT functionally consists of a photocathode, a focusing electrode, an electrostatic (Schlesinger) deflection system, an electron aperture, and a photomultiplier section. This tube is the basic IDT that was chosen by JPL for the Mariner Mars program and has proved its stable performance in space. The tube has been strengthened since its use in the MM'64 flight and is presently performing in the MM'69 mission. The tube itself is lightweight (approximately 6 ounces) and has low-power application (in the no-signal mode the tube consumes no power). Because of its electrostatic deflection system, this IDT eliminates the need for heavy, complex

deflection coils. Figure 9.2-39 represents a cutaway view of the sensor showing the orientation of the various components. The IDT is within the magnetic shield. The lens assembly, sun shutter, and first-stage light baffle are all on the optical axis. The electronics are packaged on four sides around the tube.

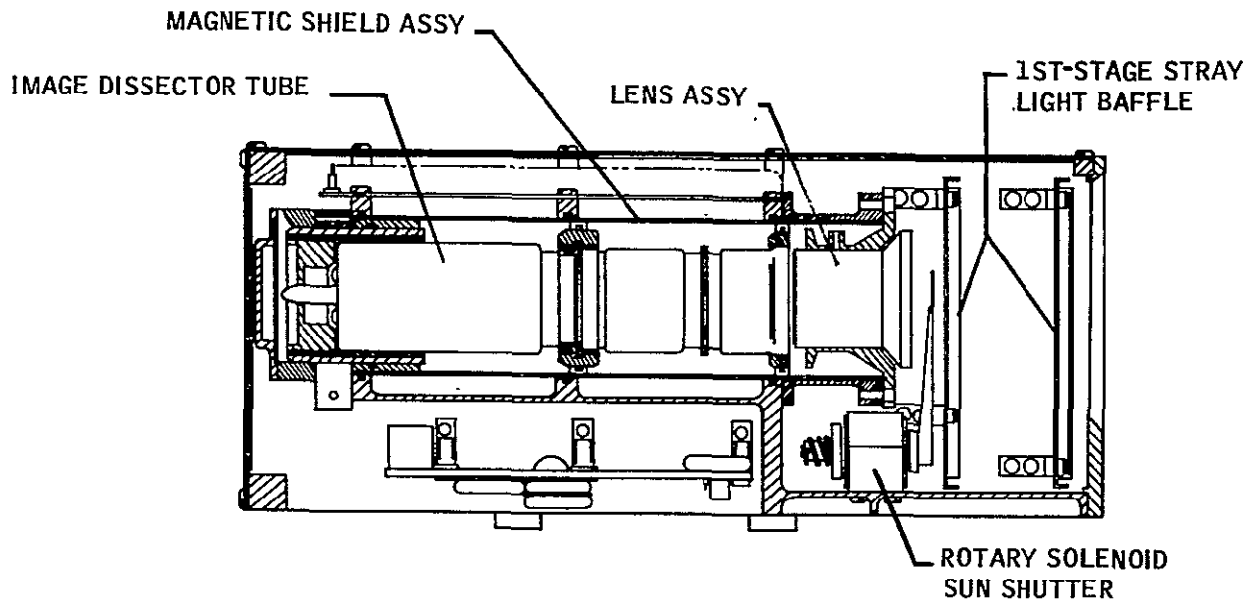


Figure 9.2-39. Polaris Sensor Assembly -- Cutaway View

Sun Detector and Sun Shutter. - Protection to the Polaris sensor IDT photocathode is provided by a sun detector and sun shutter, operating together to detect the presence of the sun in the FOV of the Polaris sensor.

Figure 9.2-40 shows the sun detector assembly. This assembly is a cadmium sulfide photoconductor mounted on an aluminum housing with baffles to define its FOV. This detector senses the sun and gives a signal actuating the rotary solenoid sun shutter. This design is virtually unchanged from the MM'64, and has flown on the Mariner Venus 1967 and MM'69.

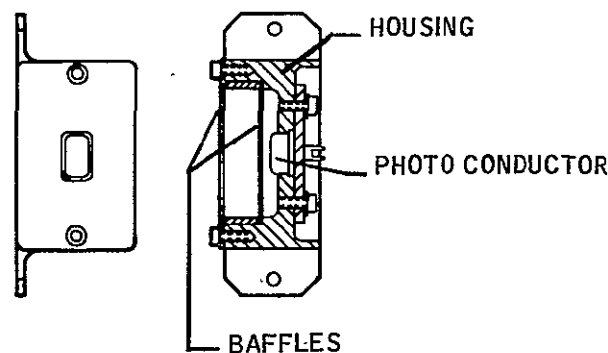


Figure 9.2-40. Sun Detector Assembly

Figure 9.2-41 shows the rotary solenoid sun shutter. The shutter is operated in the system in a spring-loaded, open configuration for safety. This device has undergone thorough testing on the MM'69 program. The thermal vacuum test proved out the stainless steel on Delrin bearings in the hard vacuum of outer space. It is now in life test with millions of actuations completed. With the exception of the blade, this device is the same as the rotary solenoid sun shutter successfully flying on the Mariner Venus 1967 spacecraft and is the same as that being used on MM'69.

Electronics Packaging. - Throughout this component description discussion, packaging features were presented in all areas, with the exception of the electronic components.

Figure 9.2-42 shows the general arrangement of the electronic component packaging. The top board has the sensitive signal circuits. The bottom board contains power circuits, relays, and other components not directly involved in signal processing. Heavy components such as the transformers are mounted in this bottom cavity.

The right-side, left-side, single-sided printed circuit boards are each bonded over their entire surface to the full side of the chassis. This gives them the additional use of providing viscous damping to cut down amplification at resonance during vibration.

This wide-open packaging technique allows the installation of selected components at final assembly. This is necessary because of the individuality of the IDT's and the extremely sensitive circuits.

Primary circuitry for the Polaris sensor is identical to the circuitry used in the MM'69 configuration; hence, the circuitry was adequately analyzed, tested, and proved.

These circuits may be listed as the yaw loop, the intensity loop, and the control logic circuits. The yaw and intensity loops provide the primary functioning feedback loops of the system, while the control logic gives the sensor its automatic capabilities.

These capabilities involve the conditions of automatic acquisition determination, particle rejection, FOV bias and "clamp" control, automatic reacquisition flyback and search sweep within the optical FOV, removal of the FOV from an acquired star, provision of an acquire signal when the final determination of acquisition is made, and disablement of the control logic in total when the automatic features of the tracker are inoperative.

Alignment. - The Polaris sensor is mounted in the earth-viewing module so that it views along the -y (pitch) axis to the North Celestial Pole. The sensor is fixed to the mounting baseplate in the spacecraft by four mounting feet on the sensor chassis.

All alignment measurements are made relative to the plane defined by the mounting surface of the four sensor feet and a line in that plane defined by the front edges of the two rear feet of the sensor. These alignment references are shown in Figure 9.2-43.

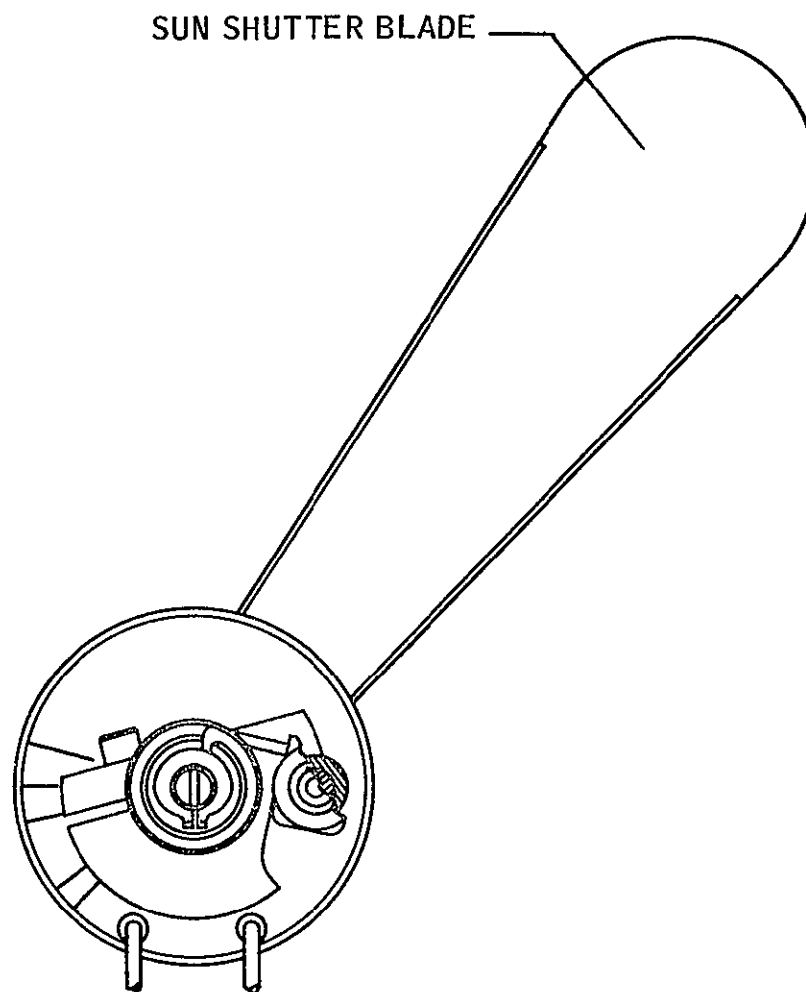
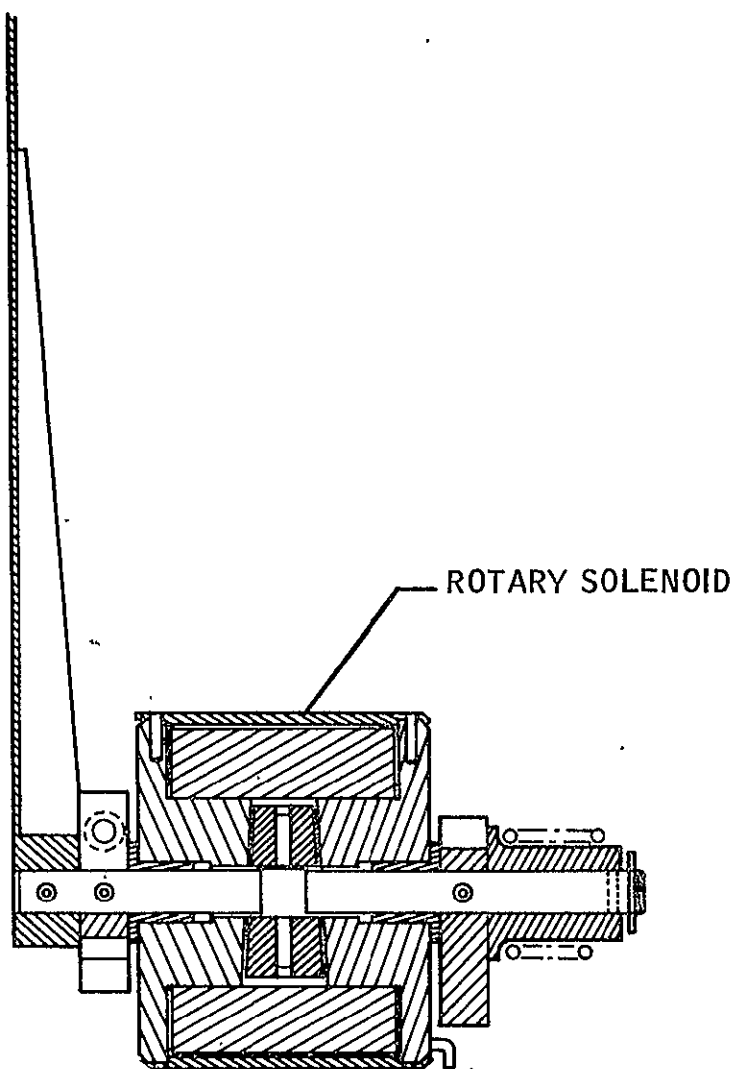


Figure 9.2-41. Rotary Solenoid Sun Shutter

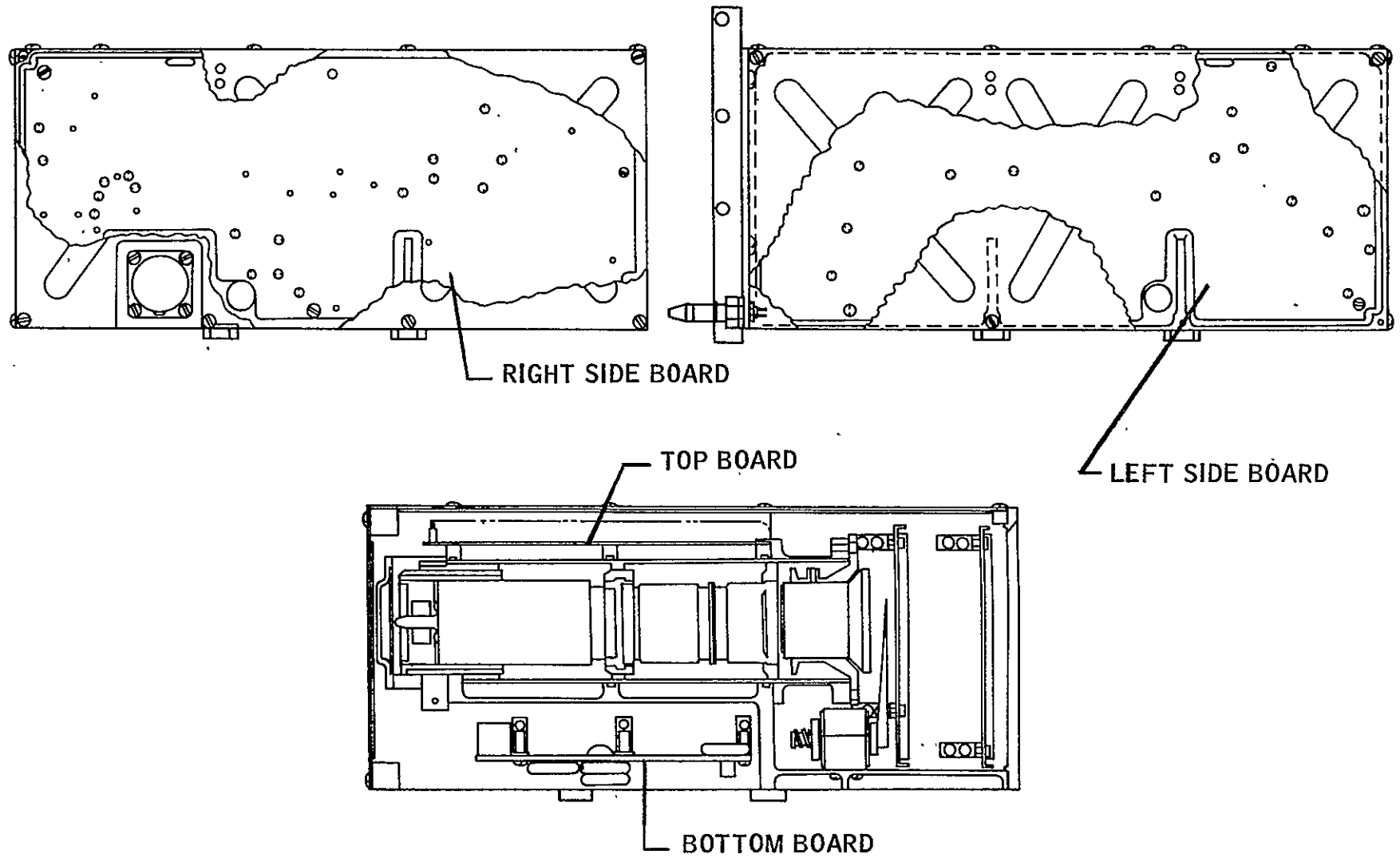


Figure 9.2-42. Electronic Components Packaging

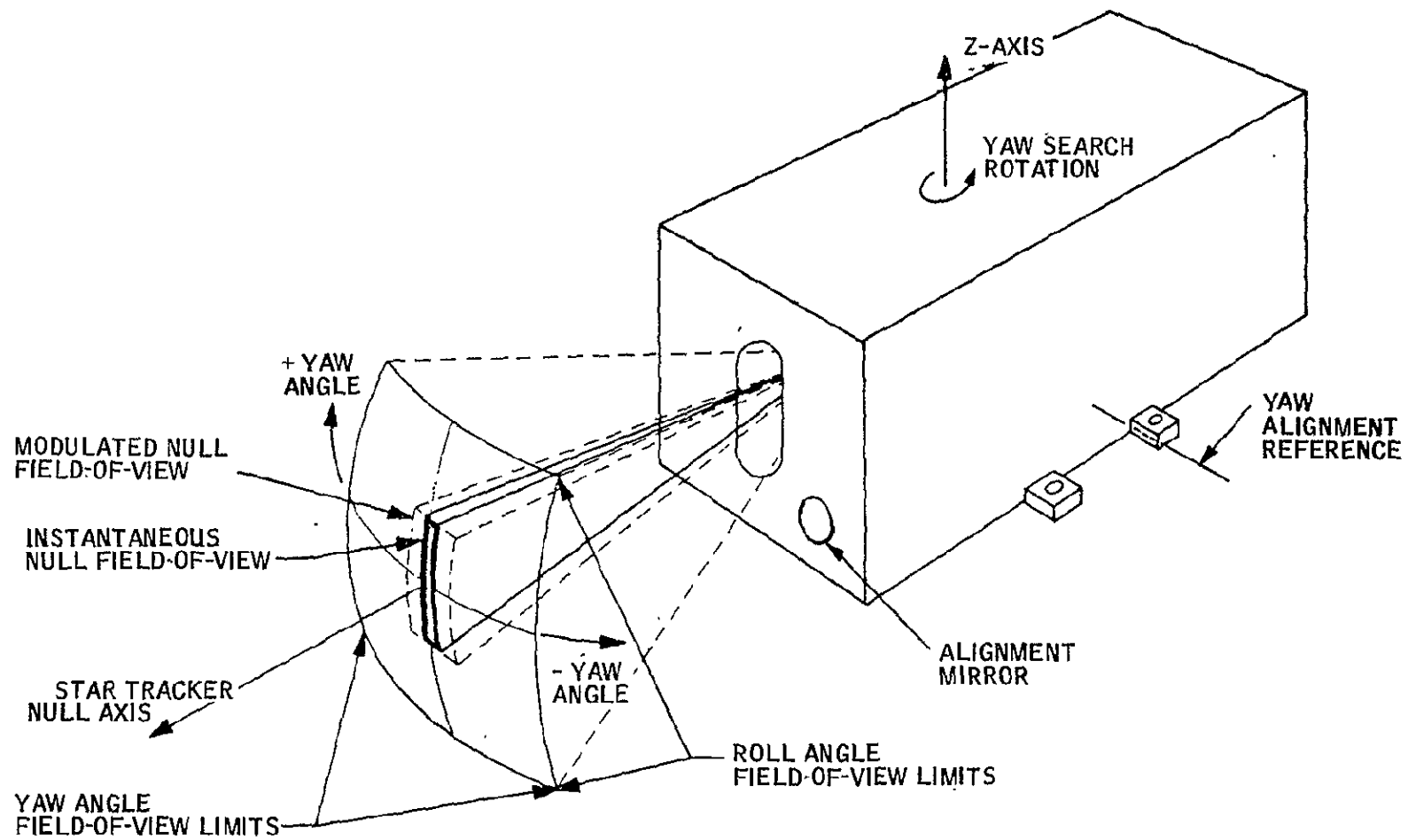


Figure 9.2-43. Polaris Sensor Alignment References

9.2.2.4.4 Power

The Polaris sensor in all of its operational modes consumes an average power of 9.2 watts.

Activation of the sun shutter requires a peak value power consumption of 7.9 watts.

9.2.2.4.5 Thermal

The thermal design for the Polaris sensor, based on the design incorporated in the MM'69 Canopus tracker, has proven to be a capable and satisfactory one since the Canopus system was successfully functionally operated and tested during temperature extremes of +125°F and -4°F.

For the ATS system, the normal operating range of base-plate temperature is +50° to 86°F. The qualification temperatures are +23°F and +113°F.

Studies have been completed to define the accuracy versus temperature dependence of the MM'69 Canopus trackers used throughout the MM'69 program. In all cases, total changes in star intensity and tracking accuracy were well within tolerance over a temperature range of +6°F to 115°F, and if the null change versus temperature for the four Canopus trackers involved in the study is normalized and averaged, the change with temperature is negligible.

9.2.2.4.6 Vibration/Shock

The most critical item of the Polaris sensor, the IDT, is the most susceptible item to effects of vibration. The "double kinematic mount" configuration for the IDT, used on the MM'69 Canopus tracker, has proved to be a very successful means of protecting the tube.

The Polaris sensor for the ATS can be qualified by similarity to the MM'69 Canopus star tracker for the thermal-vacuum environment but should be requalified in vibration for the following reasons:

- 1) The design techniques are the same.
- 2) The qualification vibration levels for the Polaris tracker are slightly higher.
- 3) The qualification thermal-vacuum levels for the Polaris tracker are slightly higher.
- 4) The historical confidence obtained from Canopus star tracker testing.

The MM'69 Canopus tracker design was reviewed to determine the extent of design modifications required to adapt to the Polaris tracking application. Electronics circuit modifications will be made in the pre-amplifier and in the logic circuitry. These changes will be variations in the components values and in the printed wiring circuits. The same techniques

of printed circuit board construction and component assembly between bifurcated terminals will be used in the modified design. All circuits will be thoroughly checked through temperature cycling during the breadboarding stages. Therefore, these changes will not require requalification of the assembly.

The mechanical design will be modified only in the area of the lens assembly. The difference in the irradiance produced by Canopus and Polaris will require an increase in the gain of the system. Part of this required increase will be accomplished in the preamplifier and part in the lens assembly by increasing the aperture.

9.2.2.4.7 EMC/EMI/RFI

The Polaris sensor baseline unit of the MM'69 Canopus tracker was tested in accordance with the Interference Control Requirements document (MIL-I-6181D). Because of this, no EMI problems are anticipated. However, it will be included in the ACS when subjecting the subsystem to the EMI test program.

9.2.2.5 Sun Sensor Assemblies

9.2.2.5.1 Weight, Center of Gravity, Moment of Inertia

Sun Sensors. - The weights of the sensors in their mounting blocks are as follows:

- Fine sun sensors 0.4 lb
- Coarse sun sensors (EVM) 0.6 lb
- Coarse sun sensors (hub) 0.5 lb

Sun Sensor Electronics. - The weights of the three sun sensor electronics packages are as follows:

- EVM electronics 1 0.8 lb
- EVM electronics 2 0.8 lb
- Hub electronics 0.3 lb

9.2.2.5.2 Size, Packaging

Dimensions of the sun sensors and the sun sensor electronics in inches are as follows:

- Fine sun sensor: 2.8 x 1.7 x 1.9
- Coarse sun sensor (EVM): 2.8 x 2.4 x 1.7
- Coarse sun sensor (hub): 2.4 x 2.0 x 1.7
- EVM electronics 1: 6.0 x 6.0 x 2.0
- EVM electronics 2: 6.0 x 6.0 x 2.0
- Hub electronics: 6.0 x 6.0 x 2.0

The coarse sun sensor configurations are shown in Figures 9.2-44 and 9.2-45, while Figure 9.2-46 illustrates the fine sun sensor.

The sun sensor electronics packages are as shown in Figure 9.2-47 and 9.2-48.

9.2.2.5.3 Characteristics

Mechanical, Optical, Electrical. -

Coarse Sensor Eye -- This eye consists of a black anodized aluminum cartridge containing a silicon solar cell as the sensing element, a deep red glass lens that passes radiation in the 0.8-micron region, and an assembly retainer also of black anodized aluminum. Figure 9.2-49 shows the coarse eye assembly.

Fine Sensor Eye -- The major components of this eye are an objective lens, a reticle, a deep red filter, a back aperture, and a silicon solar cell as the sensing element. The container for these components is of chromicoated aluminum. The lens is of radiation-protected borosilicate crown glass. The fine eye assembly is shown in Figure 9.2-50.

Target Detector -- This detector consists of an aluminum (chromicoated) container, a lens of radiation-protected borosilicate crown glass, a reticle, and a silicon cell as the sensing element. (See Figure 9.50).

In all three types of sensors or detectors (coarse sensor, fine sensor, and target sensor), photons passing through filters or lenses strike the active area of a photovoltaic cell and release electrons. These electrons constitute the output current of the sensor, and they are proportional to the total light flux striking the cell.

The detectors do not experience any significant degradation of output current due to shelf or operating time.

Location, Mounting, Orientation, Alignment. - The coarse/fine sun sensor configuration consists of six aluminum blocks which contain the individual sensor eyes. The blocks mount on the spacecraft so that they provide the capability for sunline pitch and yaw orientation along the +X and -X spacecraft axes.

A reference plane for mounting and aligning each block consists of three lapped pads on the block mounting surface. This same type surface must be defined on the spacecraft.

Alignment of the coarse sun sensor blocks is to be within ± 0.5 degree and for the fine sun sensor block to within ± 0.1 degree.

Figures 9.2-51, 9.2-52, 9.2-53 and 9.2-54 show locations and orientations of the sensor blocks on the spacecraft.

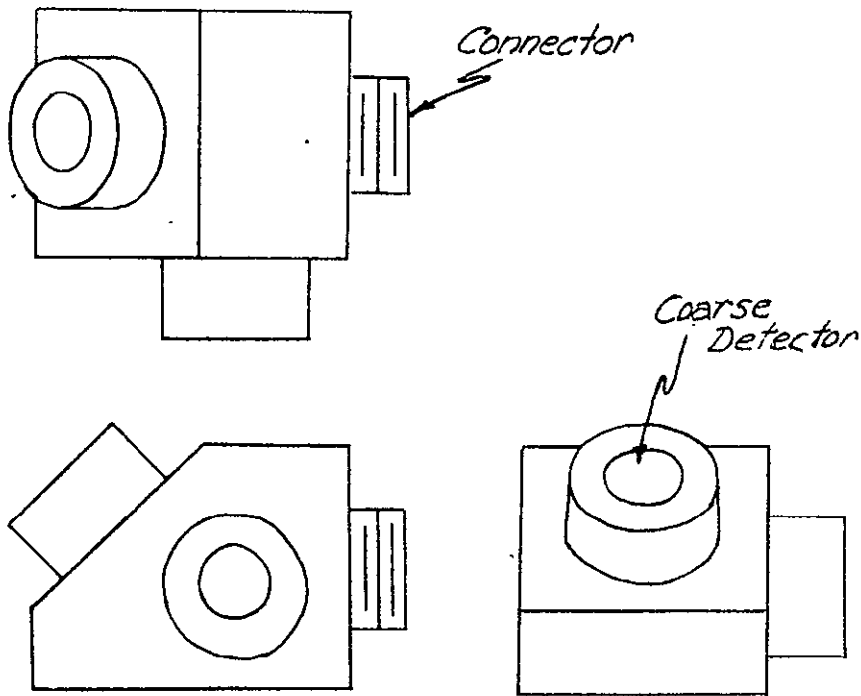


Figure 9.2-44. Hub Coarse Sun Sensor

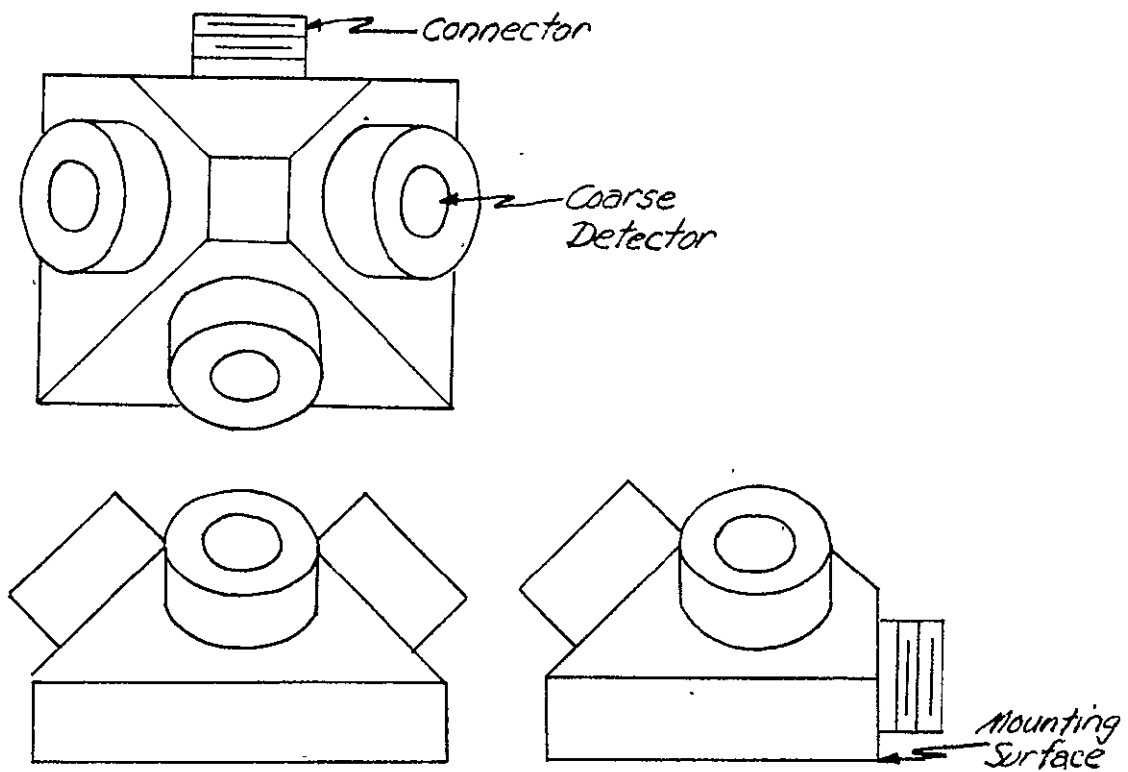


Figure 9.2-45. EVM Coarse Sun Sensor

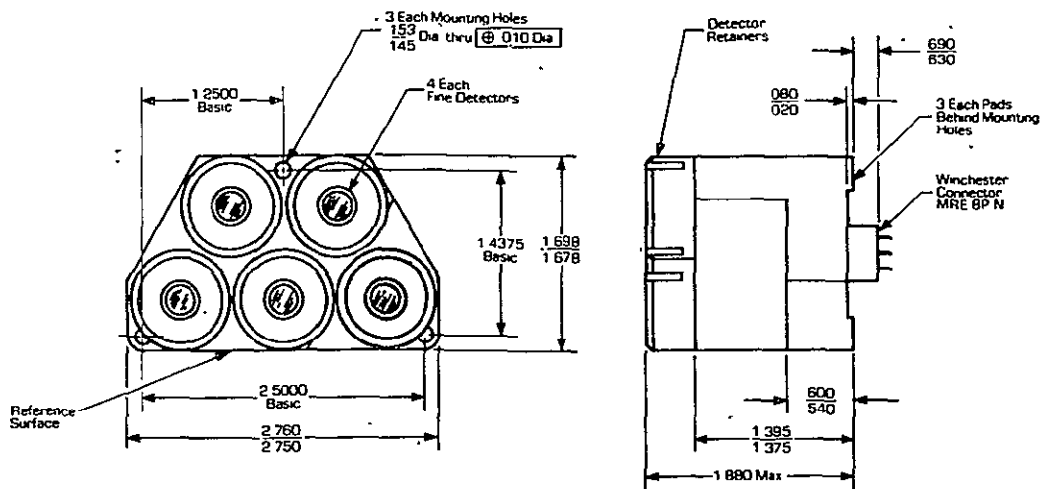


Figure 9.2-46. Fine/Target Sensor Block Assembly

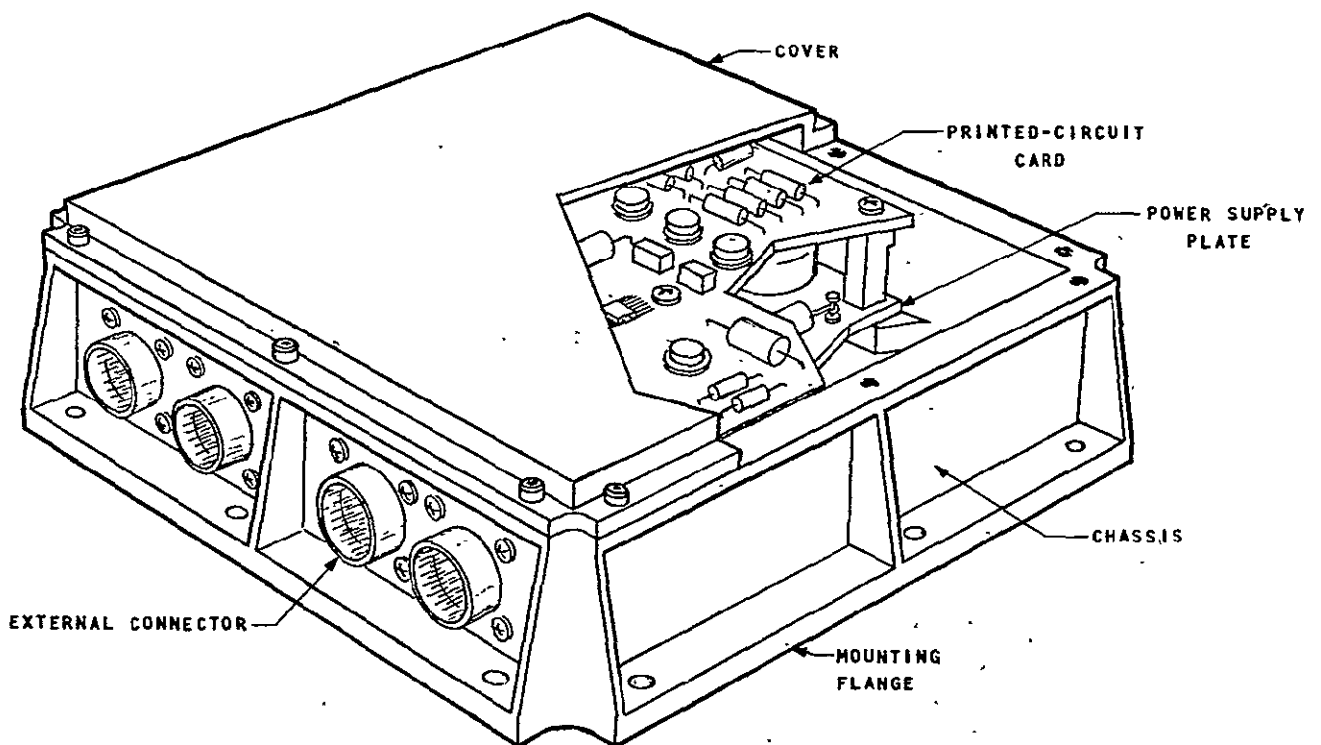


Figure 9.2-47. EVM Electronics 1 and 2

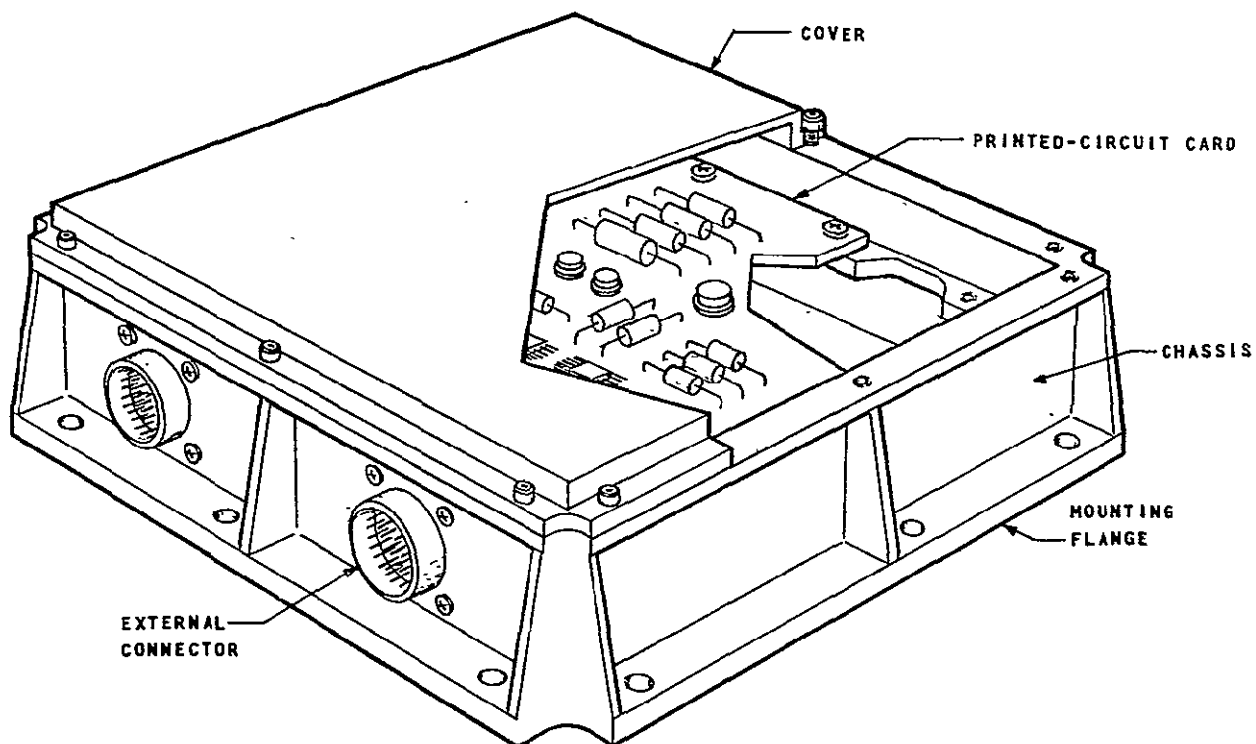


Figure 9.2-48. Hub Electronics

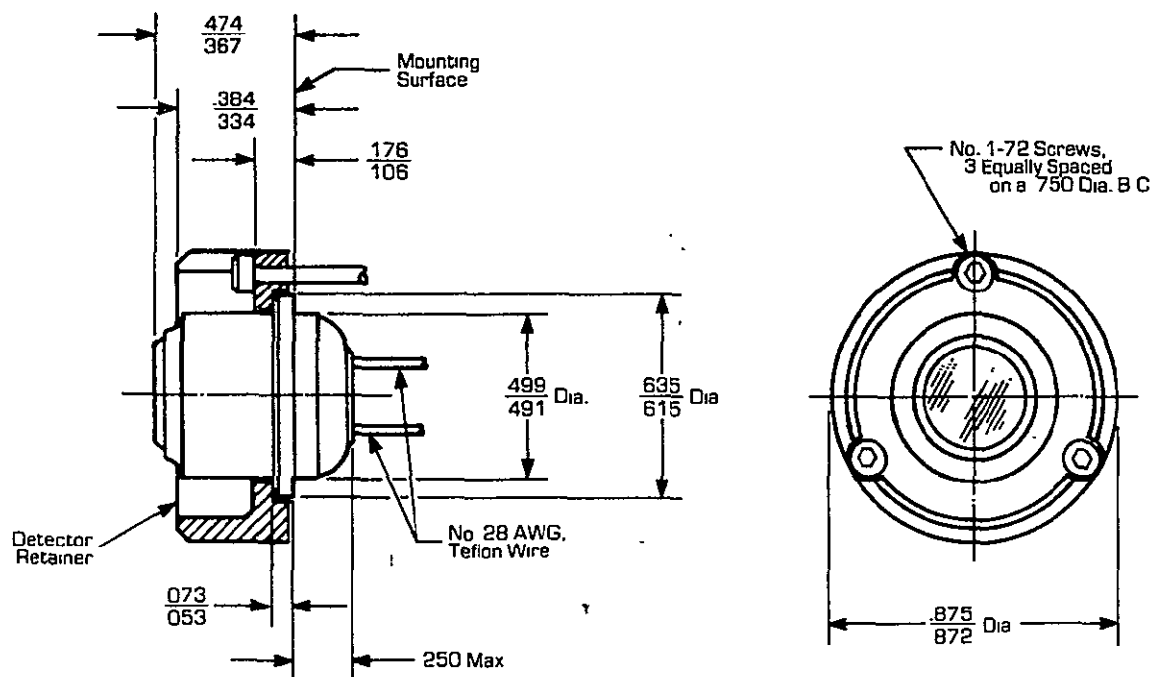


Figure 9.2-49. Coarse Eye Assembly

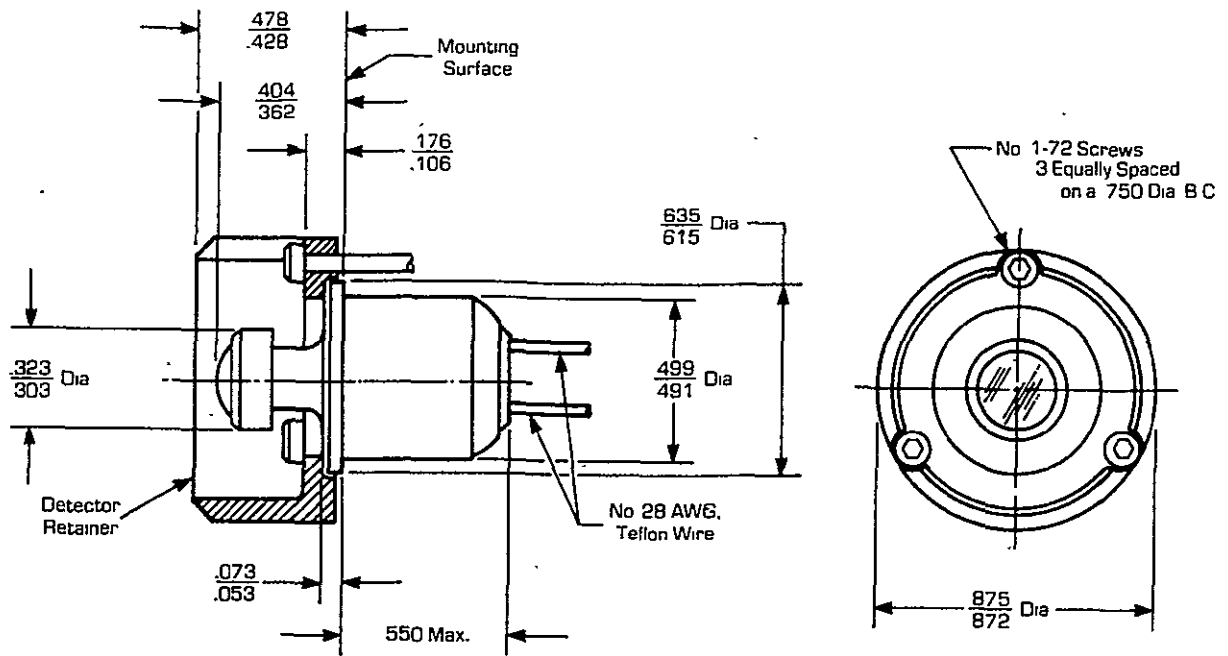


Figure 9.2-50. Fine Eye/Target Detector Assembly

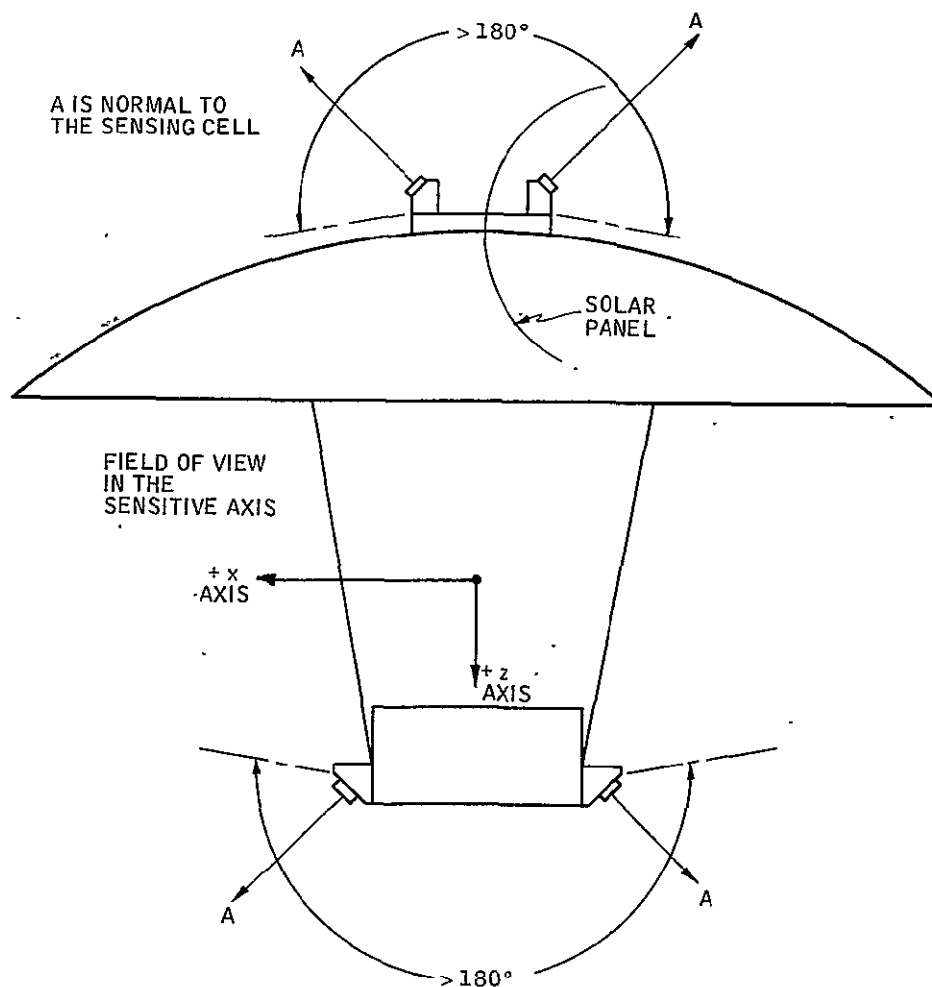


Figure 9.2-51. Pitch Coarse Sun Sensor Location, Orientation

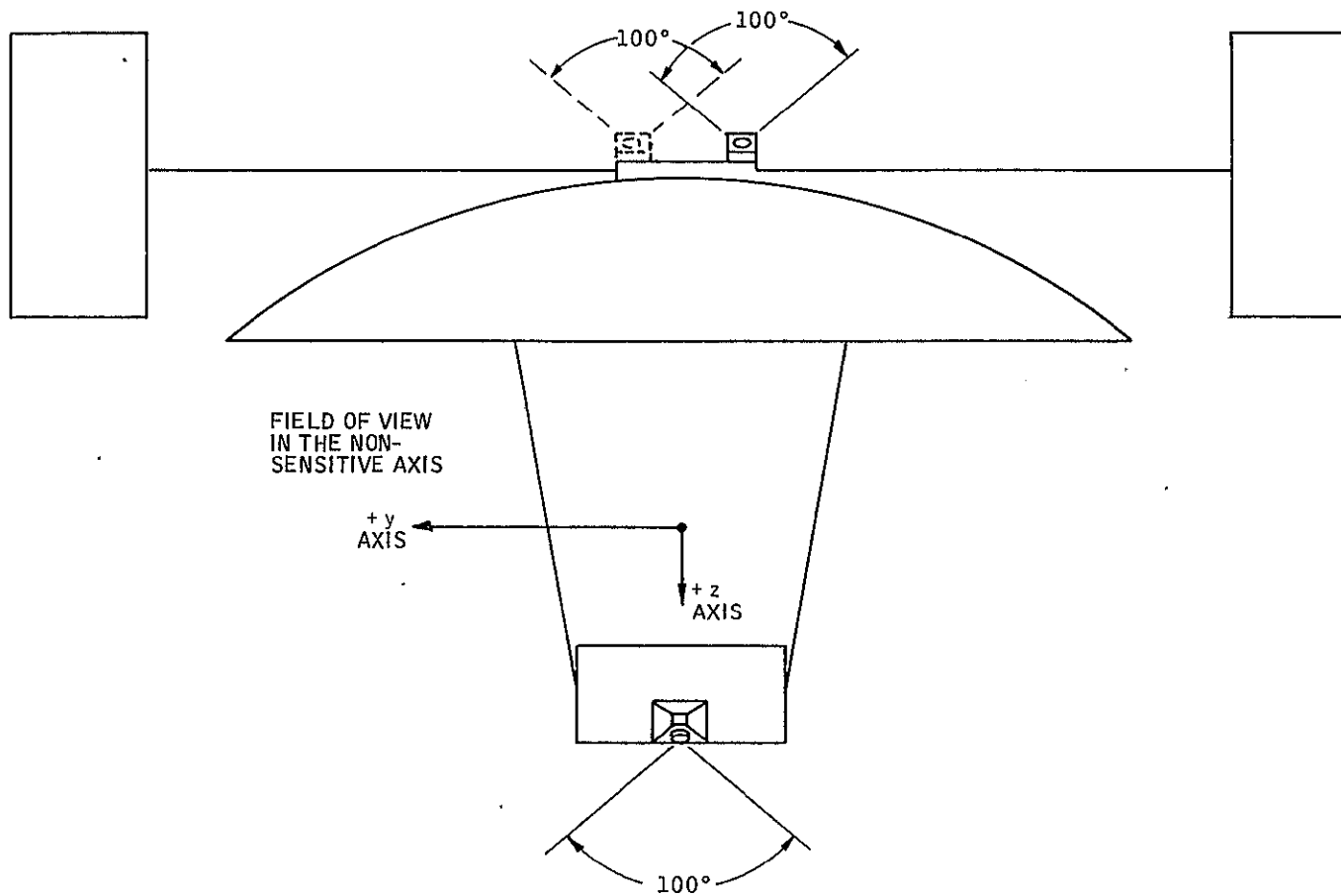


Figure 9.2-52. Pitch Coarse Sun Sensor Location, Orientation

Figure 9.2-53. Yaw Coarse Sun Sensor Location, Orientation

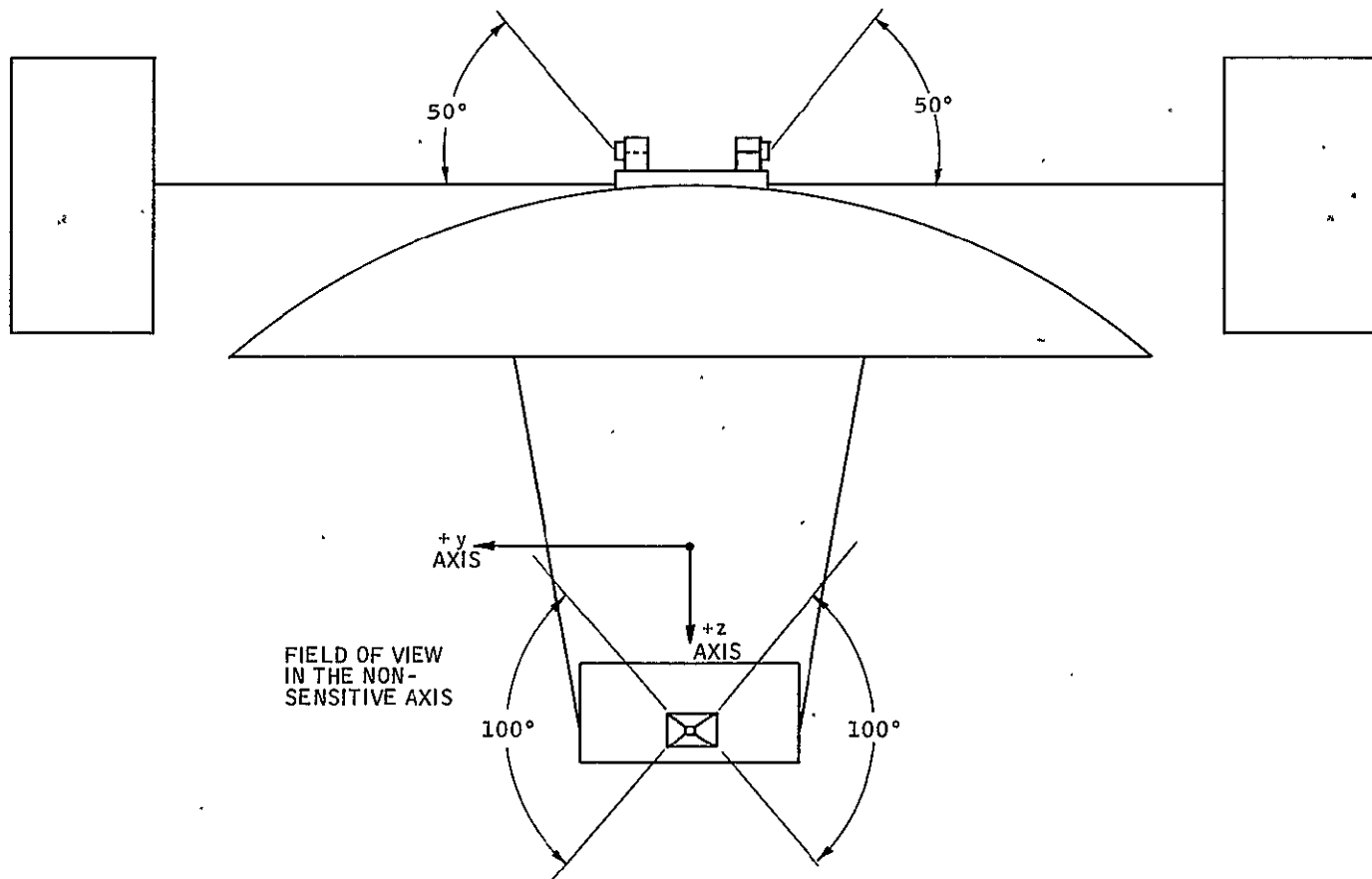


Figure 9.2-54. Yaw Coarse Sun Sensor Location, Orientation

9.2.2.5.4 Power

The sensor eyes themselves consume no power during operation because of their photovoltaic nature.

The sun sensor electronics, operating off the +28-volt d-c input from the spacecraft, require 3.6 watts of power. The power is consumed in the two EVM electronics (sun sensor electronics assembly) packages by the LIC's power supplies, amplifiers, and level detectors. The hub electronics (sun sensor signal amplifier) package contains only amplifiers and level detectors and operates off the ± 5 and ± 15 -volt input from the EVM electronics power supply.

9.2.2.5.5 Thermal

The sun sensor configurations proposed have been completely space-qualified and, in particular, have been operated under temperatures ranging from -20°C to $+85^{\circ}\text{C}$ in a standard atmosphere and in a vacuum environment of 10^{-6} torr and less.

The mounting block surfaces exposed to solar radiation will be treated as required to provide acceptable values of absorptivity and emissivity to minimize thermal problems.

The normal operating temperatures (mounting surface) for the sun sensors and the sun sensor signal amplifier which is mounted on the hub is expected to be -9°C to $+50^{\circ}\text{C}$. The qualification test temperatures are -24°C to 65°C .

The sun sensor electronics packages which are located in the EVM will be subjected to normal operating temperatures of $+4^{\circ}\text{C}$ to $+38^{\circ}\text{C}$. The electronics package makes use of the packaging technique used for DOC, ABC and ACE. No problems are anticipated in meeting the requirements with the thermal environment.

9.2.2.5.6 Vibration/Shock

The sun sensor eyes have been completely space-qualified to vibration and shock levels comparable to those expected on ATS. However, new mounting configurations and the sun sensor electronics must all be qualification tested as required on the ATS program.

9.2.2.5.7 EMC/EMI/RFI

The sun sensor electronics are to be tested as required for EMC, EMI, RFI susceptibility. The precautions indicated previously for the DOC will be observed in the design of the electronics.

The sun sensors have been exposed to and tested under condition called for in applicable sections of MIL-I-6181D.

The sun sensor and electronics will be included in the system when performing the EMI tests.

9.2.2.6 Inertial Reference Assembly

9.2.2.6.1 Weight, Center of Gravity, Inertia

The outline of the IRA is shown in Figure 9.2-55. The package measures approximately 9 inches by 9 inches by 6.5 inches. It weighs approximately 18 pounds. The center-of-gravity location is indicated on the drawing with respect to the reference system located in the plane of the mounting feet.

9.2.2.6.2 Size, Packaging

The IRA is housed in an aluminum casting with the gyros mounted close to the mounting feet to reduce gyro deflections and thus vibration amplification from the mounting points to the gyros. The casting is designed to minimize vibration amplification at critical points. The electronics are mounted in a vibration-isolated tray to minimize the vibration environment that the electronic components must withstand. The functional subassemblies are encapsulated in styrofoam and seal-coated. This protects the microminiature electronics from foreign material and also provides rigidity under vibration. There are three connectors, one for test inputs and outputs, one for signals, and one for power.

9.2.2.6.3 Electrical, Location Alignment

Circuit Description and Operation. - The block diagram shown in Figure 9.2-56 indicates the functional relationships within the IRA. The electrical interfaces are also delineated. Each element of the block diagram is discussed in the following paragraphs.

Power Supply --

1) Switching Regulator: The switching regulator consists of input filters, TCA power supply, and +15 vdc switching regulator.

The input filters attenuate ripple and noise coming in and going out of the IRA. The biggest filter problem is the 800-Hz spinmotor current.

The switching regulator generates a constant +15 vdc from the 24 to 33 vdc input power. The square-wave operation is efficient but electrically noisy and requires careful design and packaging to meet EMI requirements.

The TCA power supply generates +15 vdc with respect to the heater returns to operate the IRA heater control circuits. This eliminates ground problems.

The precision reference circuit generates a very accurate and constant 6.2 vdc to reference the current amplifier that feeds the switching bridge.

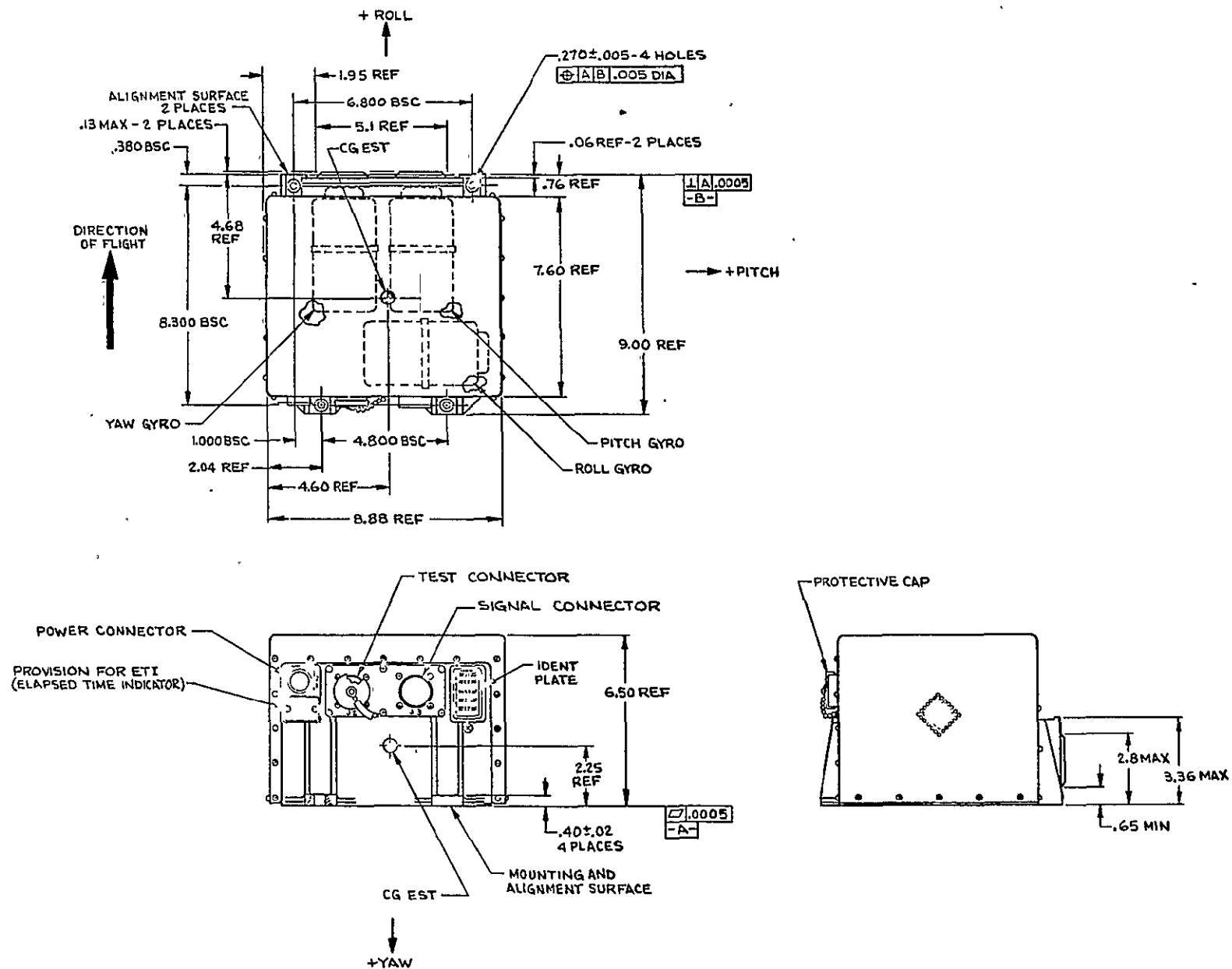


Figure 9.2-55. Inertial Reference Assembly Outline Drawing

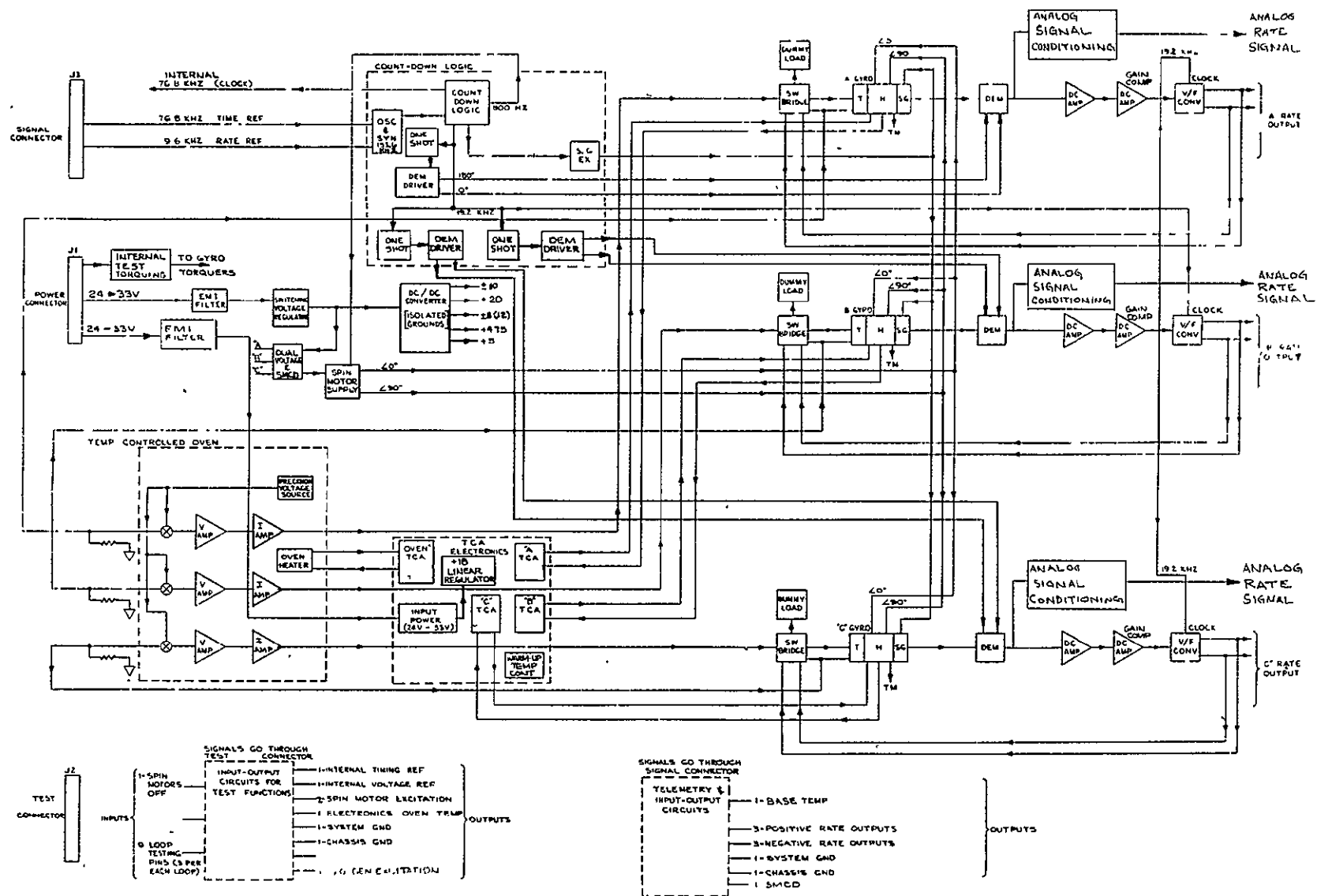


Figure 9.2-56. Inertial Reference Assembly Functional Block Diagram

2) DC-to-DC Converter. The dc-to-dc converter consists of a saturating transformer inverter, rectifiers, and filters to provide the various voltages required by the system, and linear regulators to provide low-impedance outputs to drive critical circuits. Included are three "slave" transformers, and rectifier and filter circuits which are packaged with the loop electronics for each axis to minimize interconnections.

Oscillator - Countdown Logic and Demodulator Drive Circuitry -- The oscillator-countdown logic and demodulator drive circuitry receives external timing (76.8 KHz) and rate synchronization (0.6 KHz) signals which are used to synchronize the internal functions and rate output signals of the IRA. An internal crystal-controlled oscillator (153.6 KHz) automatically generates the necessary IRA signals whenever the timing reference signal (76.8 KHz) is not present. The internal crystal-controlled oscillator will be used at all times when the IRA is in use for the ATS.

Gyro Signal Generator Excitation -- The signal generator excitation converts the square-wave output of the countdown logic to a sinusoidal signal to drive the three gyro signal generator primaries in parallel. This function is performed by a second-order bandpass filter with a Q of 5 that attenuates the harmonics in the square wave, leaving only the fundamental. An active RC filter is used because the necessary gain is readily obtained. A complementary emitter-follower output stage provides the current gain required to drive the load.

Spinmotor Supply -- The spinmotor supply consists of a two-stage, transformer-coupled, push-pull switching amplifier. Input signals from the logic are amplified and then transformer-coupled to the Darlington output stage, which is referenced to the input power return. The output stage is transformer-coupled to the spinmotors. This mechanization allows dual-voltage spinmotor excitation to be accomplished by variation of the d-c voltage supplied to the output state.

Oven TCA -- The oven TCA is mechanized using a bridge consisting of three precision resistors and the oven sensor resistor, to develop an error signal. This signal is proportional to the bridge imbalance and thus to the temperature error, since the sensor resistance is identical to the other bridge resistance at the control temperature. The amplified error signal controls a power Darlington transistor. The bridge and error amplifier operate from regulated +18 vdc, and the power Darlington stages operate from unregulated power to maintain temperature control.

Loop Electronics -- The pulse rebalance loop electronics amplify the gyro signal generator output; demodulates, amplifies, converts from analog to frequency; and gates precision current pulses through the gyro torquer to rebalance it. The loop electronics consist of the demodulator amplifier, temperature compensation amplifier, voltage-to-frequency converter, switching bridge, and the constant-current supply.

Input/Output and Telemetry -- The input/output circuitry buffers both the input and output signals of the IRA. The input signals are buffered for signal level shifting and signal variations. The output signals are buffered for amplitude scaling, output impedance, rise and fall times, and noise filtering. The telemetry circuitry monitors internal IRA functions and generates scaled output signals. Interfaces are as follows:

Analog Rate Signal Output -- The analog rate signal output is obtained by tapping the rebalance loop at the input to the voltage-to-frequency converter. The signal is conditioned with a first-order lag network filter with a break frequency at 10 Hz.

Location Alignment. - The IRA is located in the upper equipment bay of the earth viewing module. The ACS error budget calls for alignment of the IRA axes to within 0.1 degree of the unit alignment surface. This will be no problem, since the IRA is currently aligned to within 4 arc minutes. The reference alignment surface is the plane of the mounting pads.

9.2.2.6.4 Power

During normal operation, the unit draws approximately 40 watts over the input voltage range of 24 to 33 vdc. Total power consumption with the load interface circuit is 44 watts.

The power required by the gyro spinmotors is minimized by applying the two levels of voltage to the spinmotor. By over-magnetizing the motor; i. e., by applying a greater than normal voltage to the motor when the motor reaches synchronism, magnetizing current is no longer needed and the power factor is improved, thus reducing the power required. The use of gyro heaters and their attendant power consumption is eliminated by using electronic compensation within the rebalance loop to reduce gyro torquer scale factor change and loop bandwidth change with temperature.

9.2.2.6.5 Thermal

The thermal behavior of the IRA has been studied with mathematical analysis and verification of the analysis by test. The thermal analysis was conducted with the aid of Honeywell's TB Heat computer program. This program has been used to analyze both the steady-state and transient behavior of the IRA. A math model representing the thermal properties of the IRA has been constructed. This analysis technique was used in the early stages of the P-95 program to ensure a proper thermal design. The unit has been qualified successfully in a thermal-vacuum test.

The critical clock crystal and precision voltage reference are maintained in a nearly constant thermal environment by placing them within a temperature-controlled oven. With the temperature environment maintained by the ATS spacecraft, it is expected that the oven heaters will not be required. This maintains the rebalance pulse weight and clock frequency constant. Gyro heaters are not needed because electronic compensation is used to compensate for torquer scale factor change with temperature, and damping constant change with temperature.

9.2.2.6.6

Vibration, Shock

During the early phases of the P-95 program, a model of the IRA was constructed consisting of a base with dummy loads for components. This unit was subjected to vibration and the results analyzed. The vibration levels within the package, and especially at the mounting points of the gyros, were found to be satisfactory during input vibration of 17 g's rms random and 5 g's sinusoidal. Subsequent to model testing, the actual qualification unit successfully passed both random and sinusoidal vibration.

The same model used for vibration testing was also used for shock. The design was found to be satisfactory for a shock input of 30 g's for 8 milliseconds duration. The qualification unit has successfully passed the shock test.

9.2.2.6.7

EMI

The IRA as a device has met all EMI requirements of the qualification program. Prior to the device test, all noisy or susceptible circuits within the IRA were tested individually to the requirements of MIL-STD-826. The IRA will be included as a part of the ACS tests for EMI.

9.2.2.7

Interferometer Subsystem

Reference should be made to the Interferometer Subsystem Section for a detailed physical description of the interferometer. Weight and power data for the major elements of the interferometer subsystem is summarized in Table 9.2-9.

Table 9.2-9. Interferometer Subsystem Weight and Power Summary.

Assembly A No.	Component	Weight (lb)	Power (watts)
272	Interferometer electronics receiver processor	11.8	15.0
276A1	Receiver front end (electronics)	9.7	11.5*
276A2	Array (horn elements)	13.1	0
	Total	34.6	15.0

*Included in value for A272

9.2.2.8

Actuator Control Electronics

9.2.2.8.1

Physical Characteristics Location

The ACE consists of printed circuit card assemblies mounted and interwired on a frame for attachment to the vehicle temperature control surface having suitable thermal and electrical properties. No special mounting provisions, such as thermal coatings or greases, or electrical insulating materials are necessary.

The general configuration consists of interconnected printed circuit board assemblies having integrated circuits and discrete circuit elements mounted thereon, and cordwood submodules of discrete components (wherever needed to achieve increased density) also mounted on printed circuit boards. Refer to the actuator control electronics assembly outline drawing, Figure 9.2-57. A cutaway view is shown in Figure 9.2-58. Except for size, weight, and number of connectors, the structure is similar to the ABC assembly. The installation may be made without regard to axis orientation.

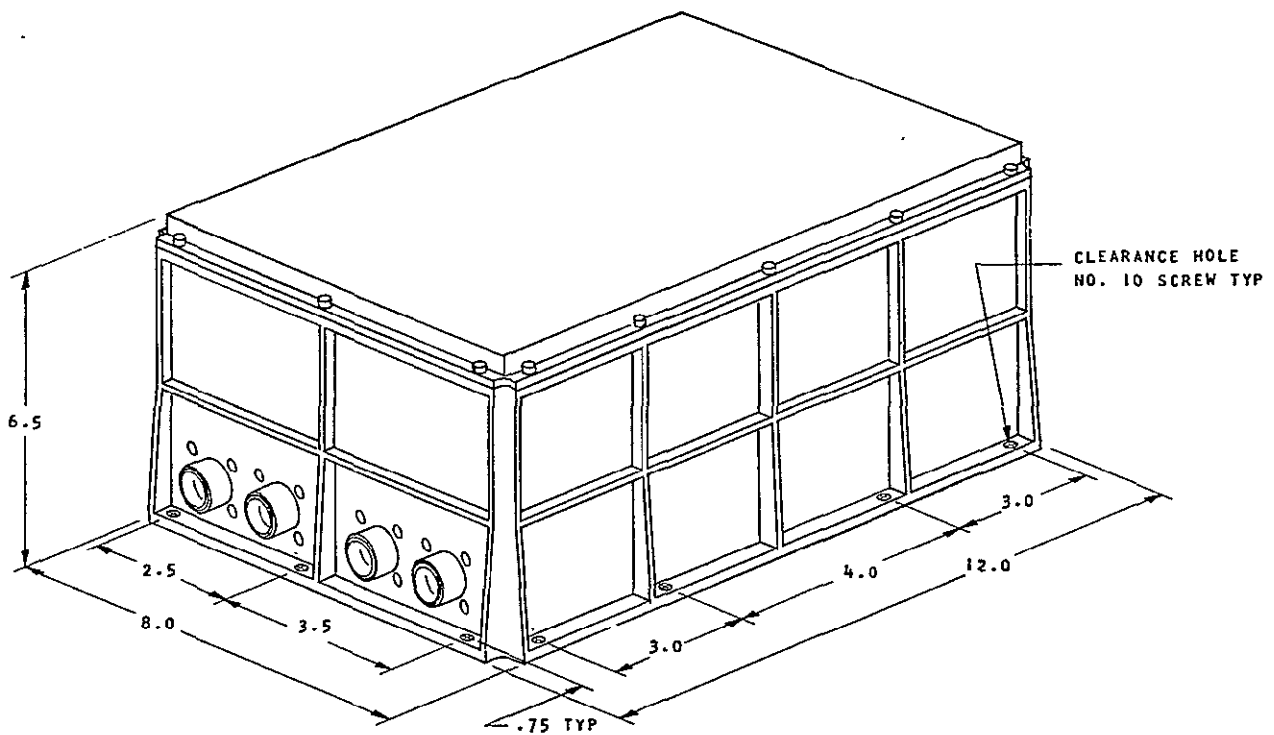


Figure 9.2-57. Actuator Control Electronics Assembly -- Outline Drawing

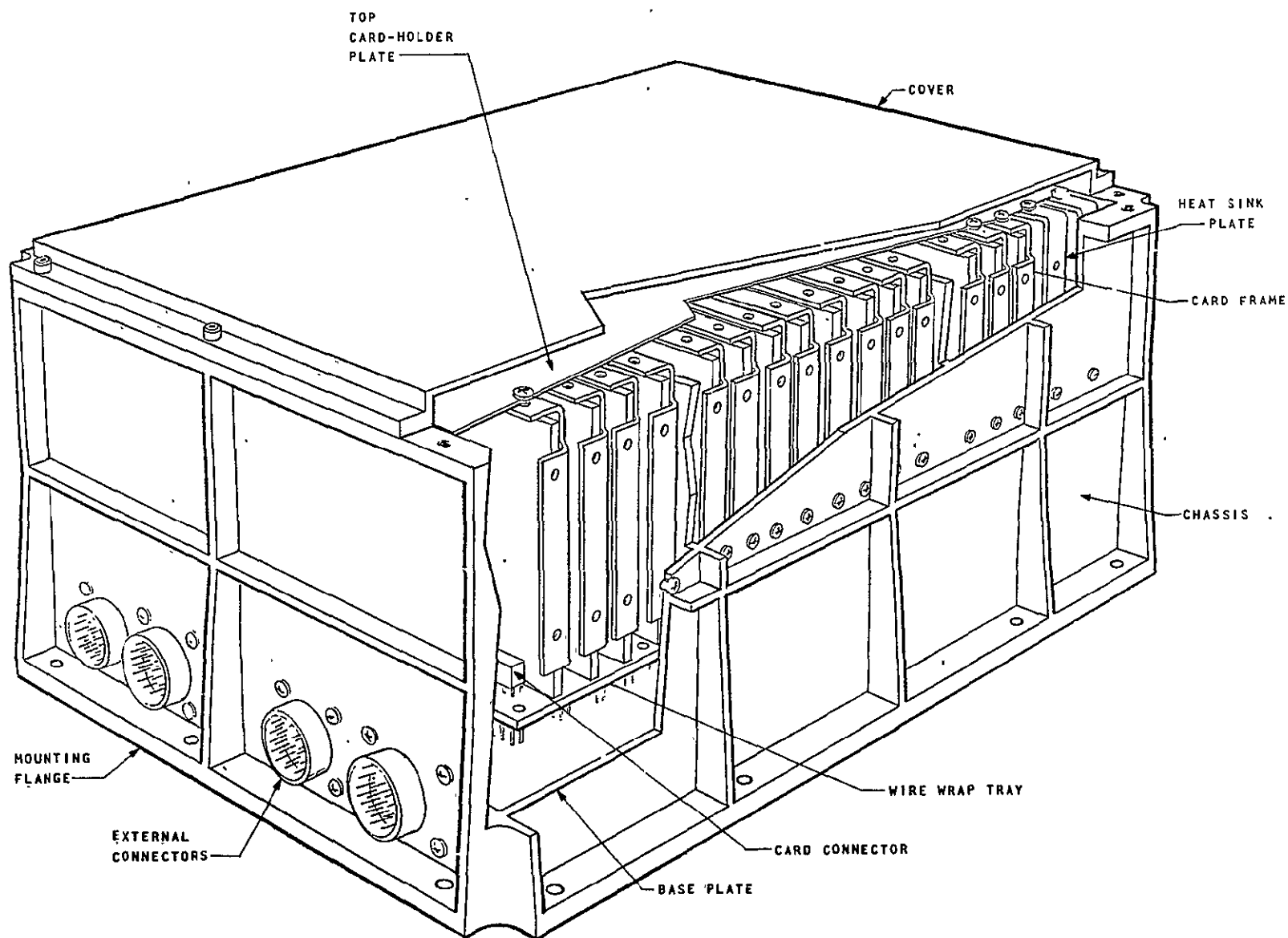


Figure 9.2-58. Actuator Control Electronics Assembly -- Cutaway View

9.2.2.8.2 Environmental Considerations

The ACE is designed to withstand the environmental conditions listed in the FHC Design Specification. No special restrictions on thermal, shock, vacuum, or vibration exposure are in effect for the ACE assembly during shelf life, prelaunch, or post-launch periods.

9.2.2.8.3 ACE Operation Description Summary

The inertia wheel control electronics portion of the actuator control electronics assembly shall perform the following functions in each of three axes:

- a) Amplify the d-c attitude error signal.
- b) Convert amplified error signal to proportional frequency, two-phase signals of a phase sequence responsive to d-c signal polarity.
- c) Power amplify two-phase signal sufficient to drive wheel motor.
- d) Control voltage level of power amplified two-phase signals to control motor flux level at any frequency and error signal command.
- e) Provide d-c voltage supply levels from basic 28-vdc supply as required for the electronics.

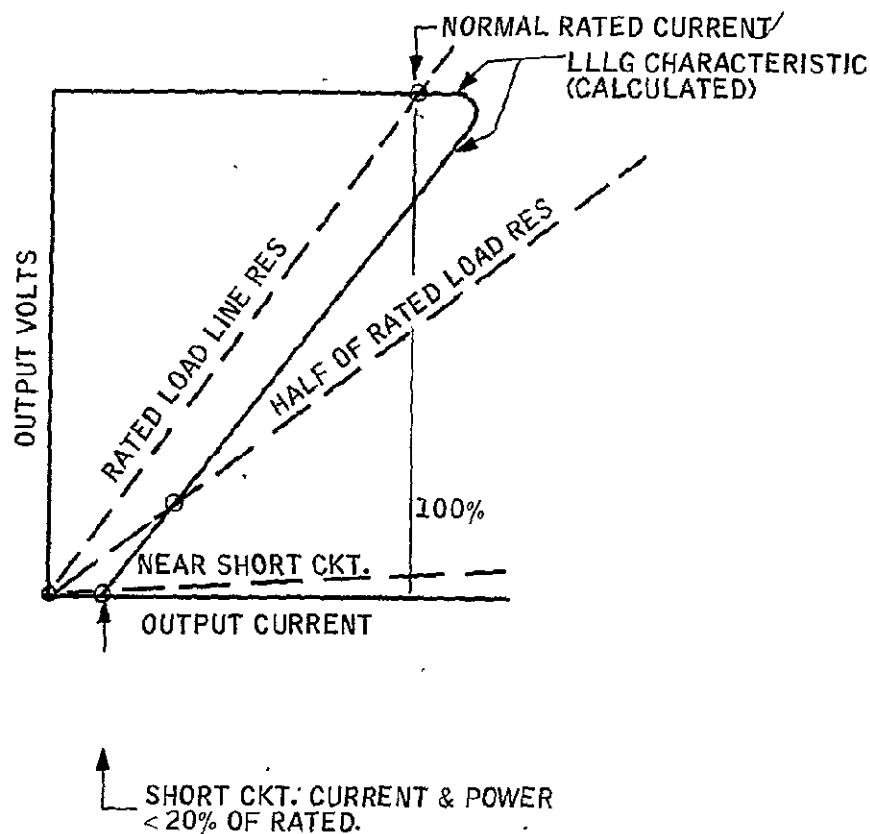
The propulsion control electronics portion of the actuator control electronics assembly shall provide and control the following:

- a) Holding power to appropriate solenoid valves of APS for attitude control.
- b) Pulse power to appropriate solenoid valves for pressure regulating system.
- c) Power to resistojet and circuit heaters.
- d) Alternate redundant path means for controlling cold gas jets and resistojets.
- e) Conditioning for telemetry signals.

9.2.2.8.4 Power Switch Circuit Description Summary

The power switch (or gate) described herein is intended to perform the on-off control of wheel drives and APS loads (valves, solenoids, heaters, etc.) normally powered from an LIC (load interface circuit) - supplied 28 vdc. This power switch approach is intended to replace all power-handling relays and will additionally provide protection against any single load device failure (of insulation or other short-circuit or overload effect) that could otherwise necessitate switching out, for example, an entire bank of jets, etc., normally grouped on bus A or bus B through a common LIC.

Figure 9.2-59 illustrates the basic operating characteristics of the load line limiting (LLLG) gate. The "foldback" characteristic beyond the set maximum current point is set roughly parallel to the rated load line



FEATURES:

1. ACCEPTS I/O LOGIC LEVEL SIGNALS
2. CONTROLS OUTPUT POWER FROM 28V TO ANY LOADS
3. CAN BE CALIBRATED FOR REQUIRED LOAD CURRENT
4. DISSIPATES CONSIDERABLY LESS THAN RATED OUTPUT POWER FOR PARTIAL OR FULL LOAD SHORT CKT
5. PROVIDES FAIL-SAFE LOAD FAILURE
6. PERFORMS POWER RELAY FUNCTION FROM LOGIC SIGNAL

Figure 9.2-59. Load Line Limiting Gate Characteristic

such that if the load line slope reduces beyond the point of intersecting the maximum current limit point, load current will be reduced to a fraction of normal rated current. A sustained short-circuit would then be tolerable to the ACE with respect to both heat dissipation and sustained power used.

Figure 9.2-60 is a block diagram illustrating power flow from the four LICs in the ACE which provides APS power, through a sampling (only eight out of 36 or more) of the LLLGs to the various APS loads. The wheel drive system, which derives its power from a separate pair of LICs, is not included in the block diagram.

Figure 9.2-60. Block Diagram of ACE Power Flow to APS

The essential operational elements of the controlled-frequency reaction wheel drive system are as follows:

- a) A first-order d-c to d-c electronic integrator.
- b) A d-c voltage to two-phase proportional frequency signal generator having a phase sequence dependent on input signal polarity.
- c) Two power amplifiers to drive the two motor phases. These may be of a Class B type, a modified Class B type or a pulse-width-modulated-type amplifier. The output voltage magnitude must be roughly proportional to frequency. This ratio may be maintained by the voltage-to-frequency electronics or by other means, such as a pulse-width-modulated supply for a pair of saturating power amplifiers.
- d) A low-slip, low-cage resistance, induction-type wheel-drive motor, preferably bifilar wound on each of two equal impedance phases.

A block diagram of this approach is shown in Figure 9.2-61.

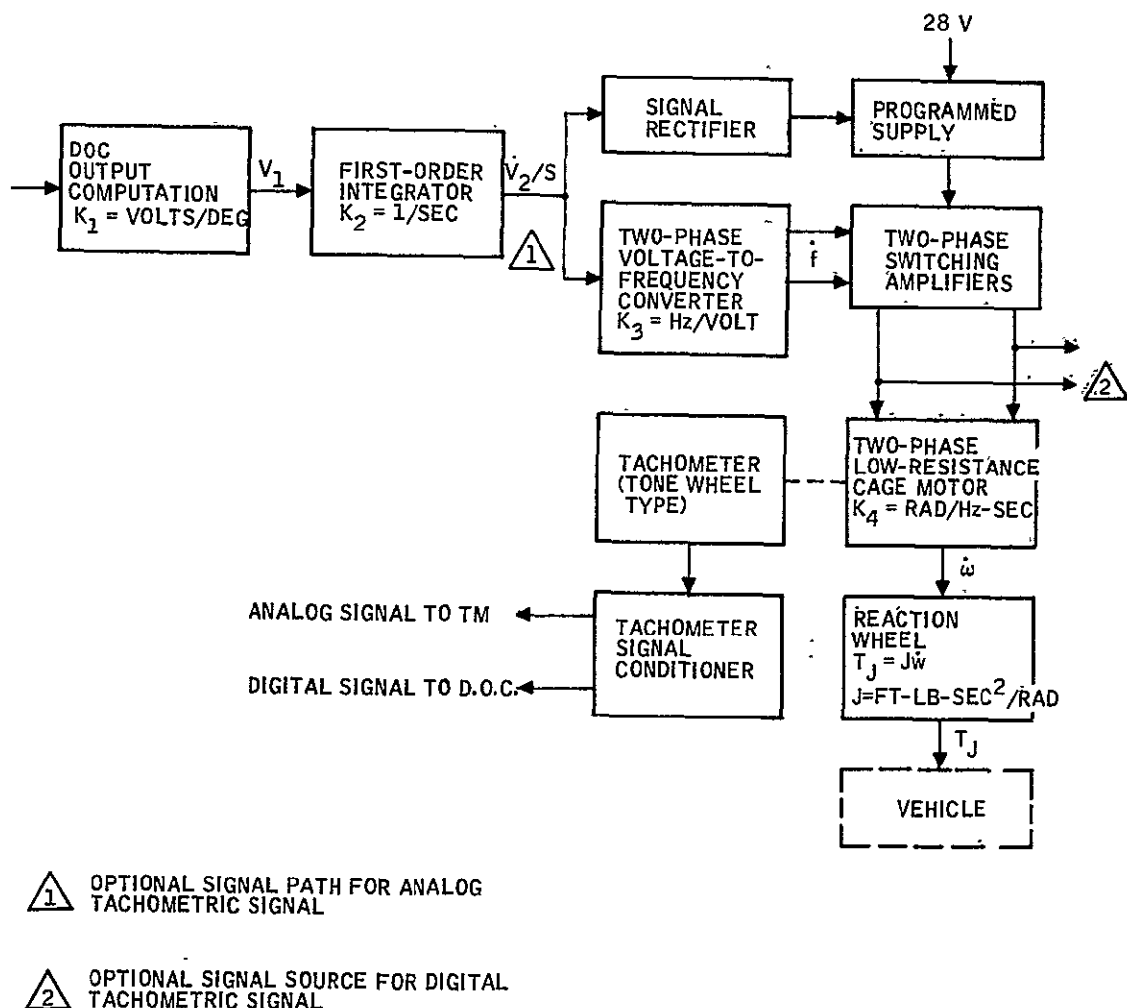


Figure 9.2-61. Controlled-Frequency Reaction Wheel Drive System

The ACE package will include elements for suppressing noise-generating waveforms through such methods of noise immunization as proper isolation of ground currents, by high-frequency parasitic oscillation elimination, by minimizing rate-of-change of current and voltage rise and fall times where feasible in active circuitry, by minimizing the geometric loop area of current carrying line pairs in and out of the package, and by shielding. The suppression elements shall be incorporated nearest to the emission source at the lowest assembly level within the subsystem. Commandable regulators which provide $28\text{v} \pm 2$ percent will give added assurance for associated load electronics to meet compatibility requirements (EMC) with assumed incoming d-c line modulation.

9.2.2.9

Inertia Wheel Physical Description

9.2.2.9.1

Weight, Center of Gravity, Inertia

The weight of the inertia wheels is distributed among the stationary parts and rotating parts. The rotating parts consist mainly of the flywheel and motor rotor. The inertia of the rotating parts is adjusted to provide the required momentum storage at the rotational speed of 1500 rpm which is the unload speed. Table 9.2-10 summarizes weight for the roll, pitch, and yaw wheels. The c.g. and inertia are defined with respect to the reference system shown in Figure 9.2-62.

Table 9.2-10. Weights of Inertia Wheels

Axis	Momentum Storage at 1500 rpm (ft-lb-sec)	Inertia of Rotating Parts About Spin Axis (ft-lb-sec ²)	Weight (lb)
Roll	3.7	2.36×10^{-2}	14.5
Pitch	4.7	2.99×10^{-2}	15.5
Yaw	3.7	2.36×10^{-2}	14.5

9.2.2.9.2

Size, Packaging

The dimensions of the inertia wheels are shown in Figure 9.2-62. The dimensions of the wheel for each axis are the same.

The motor, flywheel, and bearing system are enclosed in a hermetically sealed enclosure. The hermetic seal ensures bearing lubricant retention in a low pressure environment such as outer space and maintains unit cleanliness. The housing is magnesium for strength and light-weight. The magnesium shell is tin plated over copper. After hermetic sealing, the external surfaces are coated with an epoxy compound. The unit has four mounting feet spaced 90 degrees around the circumference of the base.

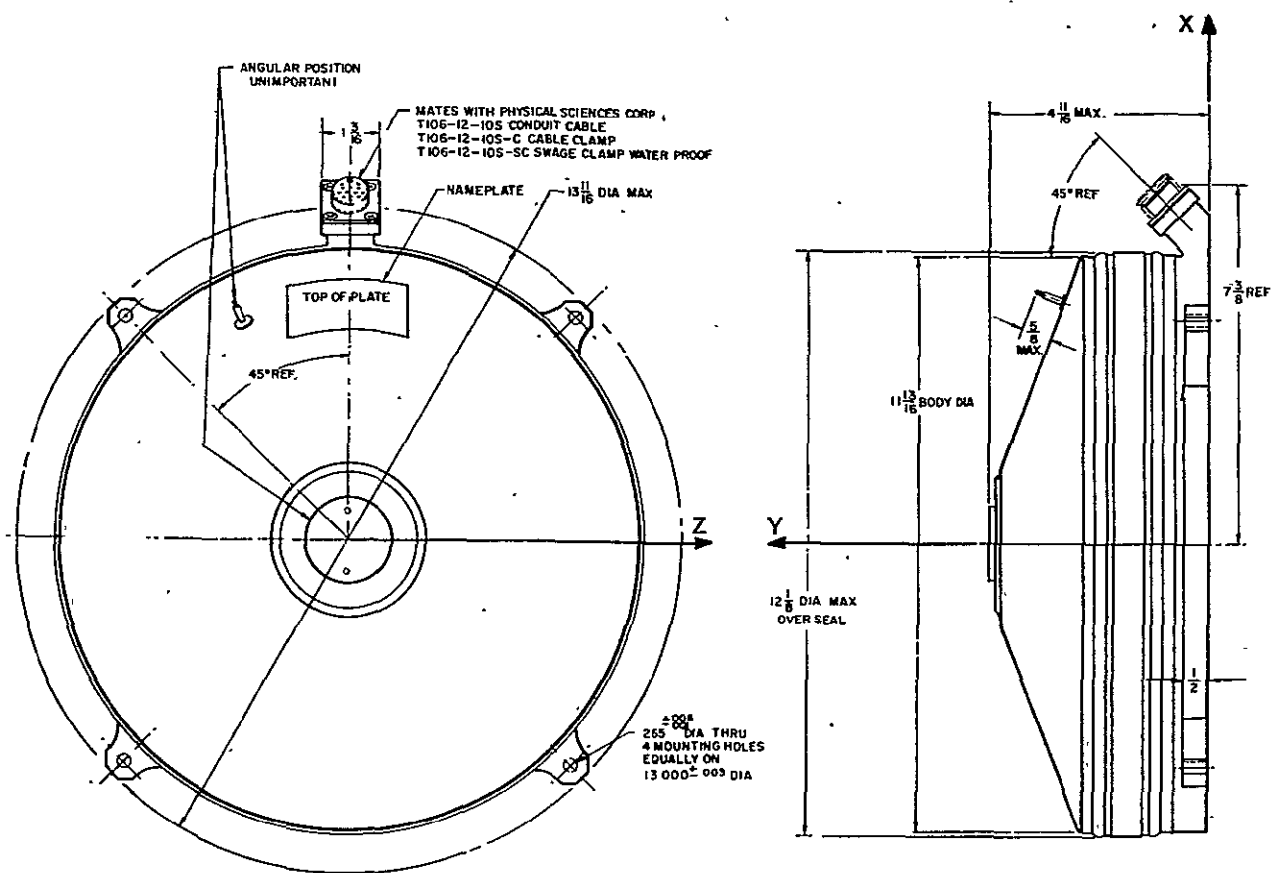


Figure 9.2-62. Basic Inertia Wheel Configuration

9.2.2.9.3 Mechanical Description

Motor. - The motor is a two-phase, a-c; squirrel-cage induction motor. Figure 9.2-63 shows the rotor and stator. The squirrel cage rotor is assembled as part of the flywheel or rotating member and rotates with the flywheel. The directly-excited windings are attached to the stationary parts of the wheel. This inside-out construction places the rotating member further out from the center than could be the case for the usual rotor inside the stator motor construction. This design provides a higher inertia for a given rotor revolution rate. The squirrel cage is made up of low-resistance copper bars so as to produce the desired speed-torque characteristic.

The two control stator windings are assembled as part of the stationary member, and they are insulated to completely isolate one from the other. The stator windings are impregnated for protection against damage due to internal motion and are entirely encapsulated to eliminate all possibility of foreign matter entering which could cause physical damage to the windings.

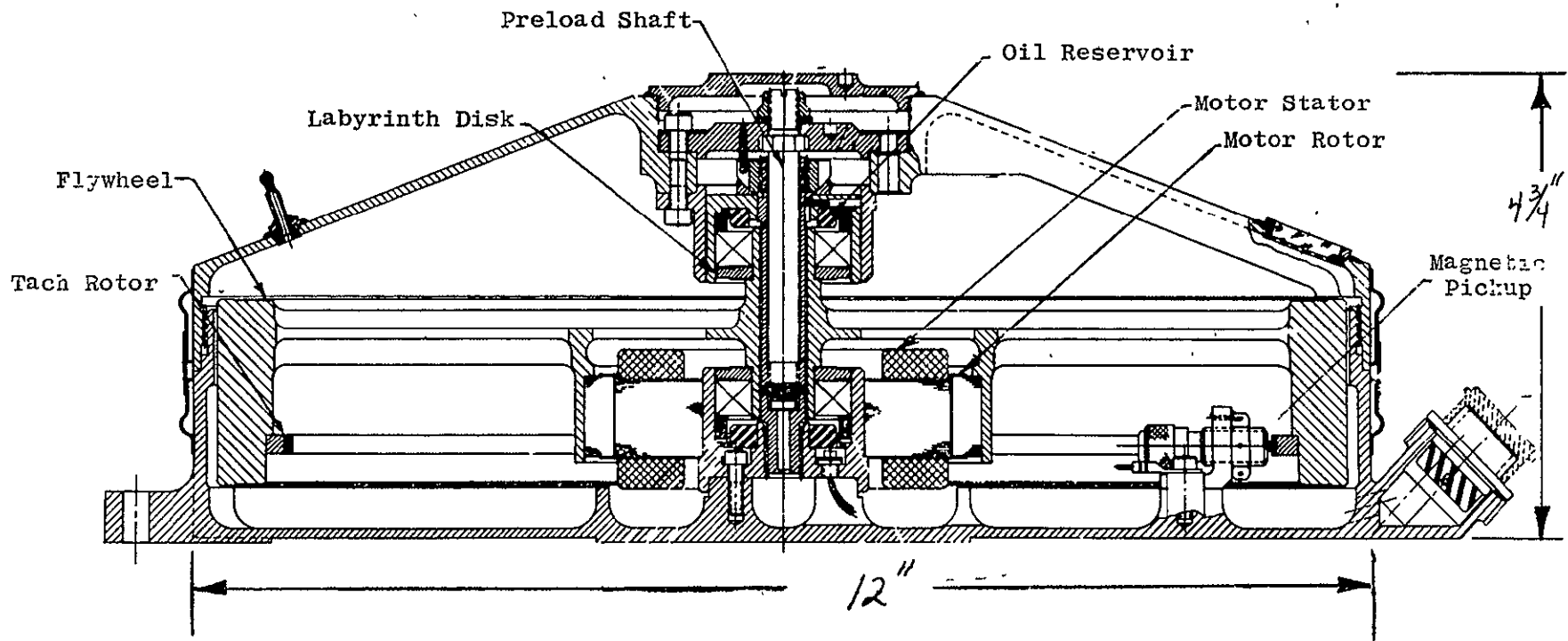


Figure 9.2-63. Cutaway View of Inertia Wheel Rotor and Stator

Rotating Parts Construction. - The rotating element is machined from a single piece of stock, including the shaft, bearing journals, hub, and rim. The single-piece construction of the flywheel rotor was chosen in preference to a rim of more dense material than the hub because the slight increase in inertia for the same flywheel rotor weight produced by the latter technique was overshadowed by the stability of balance and bearing alignment gained by the former.

Stationary Parts Construction. - The stationary parts are as lightweight as possible consistent with their requirements to:

- a) Support the wheel and stator assembly under all specified conditions.
- b) Provide a heat sink surface to remove heat from the unit by conduction to the spacecraft mounting structure.
- c) Provide a hermetically-sealed enclosure for bearing lubricant retention in the hard vacuum of space and for maintenance of internal cleanliness.
- d) Maintain perpendicularity of the wheel spin axis to the vehicle reference surface.
- e) Be opened for service and inspection.

The outer housing parts are magnesium-coated with fused tin plated over copper on all external surfaces. Four mounting feet are located around the periphery of the base. The spin axes of the roll, pitch, and yaw wheels are perpendicular to one another, thus giving control torque in each of the spacecraft axes. The inertia wheels are located in the upper equipment bay of the earth viewing module.

Bearings and Lubrication. - Bearings chosen for the wheel design are the deep groove inch series type of the finest grade available. The material of the rings and balls is 52100 alloy steel hardened to Rockwell C 60 minimum. Rc 60 minimum is a higher hardness range than standard stainless steel bearing material (440C). The capacity of the stainless steel bearing is then equivalent to chrome steel. This material has a thermal coefficient closest to the 300 series nonmagnetic stainless steel flywheel and bearing liners. Stainless steel is used for bearing material because of its high corrosion resistance and good dimensional stability with temperature variation.

9.2.2.9.4 Power

The inertia wheels essentially operate in two modes, one in which small reaction torques are supplied to offset disturbance torques acting on the spacecraft, and one in which up to six ounce-inches of reaction torque are supplied to maneuver the vehicle. Figure 9.2-64 shows the estimated input power to the motor of one wheel for the two modes. The maximum power drawn is about 14 watts per wheel. The maximum power dissipated as heat in the motor is estimated to be eight watts per wheel.

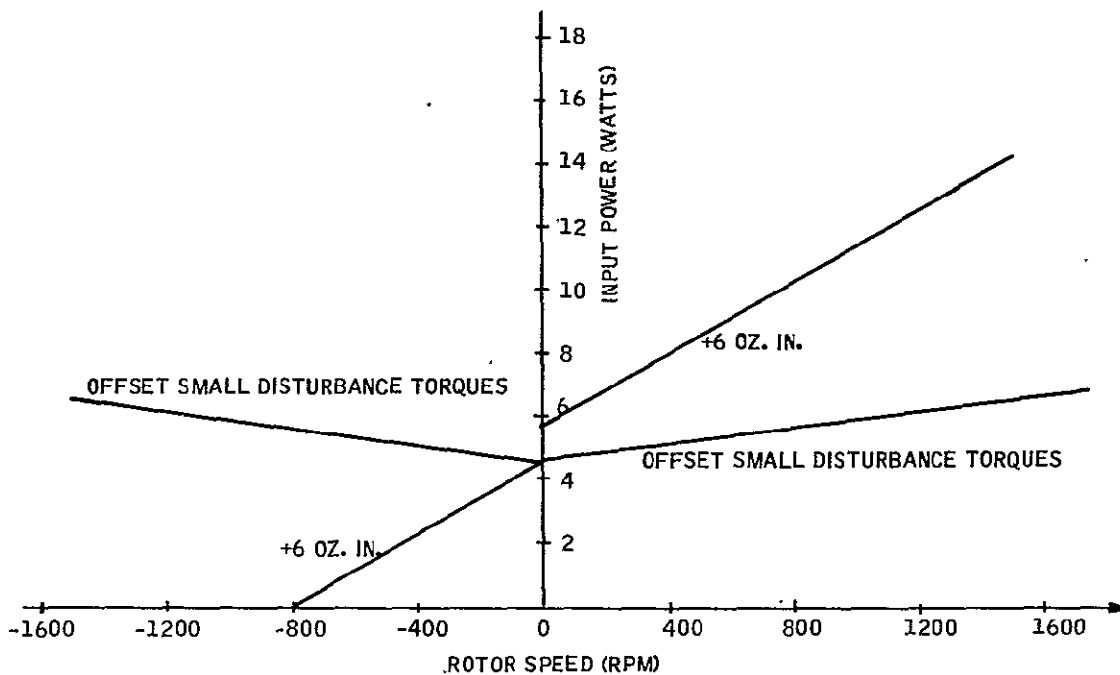


Figure 9.2-64. Motor Input Power

9.2.2.9.5 Thermal

The internal source of heat of the reaction wheel assembly is the motor. The design incorporates features to aid in the rejection of heat to the vehicle frame. The internal surfaces of the unit are dull black to aid radiation from the rotating parts to the frame and stator assembly. The object is to minimize the amount of rotor heat that must transfer through the bearings, thus increasing bearing life. The mounting pads are flush with a flat bottom surface of the unit, so that an entire side of the unit is in contact with vehicle structure. This construction also saves unit weight, because vehicle structure adds to the structural rigidity of the unit.

The power dissipated under the worst-case combination of speed and torque is approximately eight watts per wheel. The thermal resistance of the interface is designed to conduct this heat with a safe temperature.

9.2.2.9.6 Vibration and Shock

High bearing loads during vibration, acceleration, or shock environments are prevented by providing rim stops on the flywheel. The upper and lower face of the rim provides impacting surfaces against the case stops. Under environmental loading, the flywheel excursion is limited by the grease lubricated stops. This distributes the g-loading between the bearings and case and allows the bearing size and therefore bearing friction to be minimized.

The hermetic seal solder joints are designed so that the solder does not carry structural loads, thus reducing the possibility of a seal failure caused by environmental stress.

The bearing lubrication system is so designed that lubrication is maintained within the bearing for the mission lifetime even if a gross hermetic seal leak should occur. The vapor pressure of the oil lubricant in the bearing chamber is maintained by a labyrinth seal adjacent to each bearing. Oil lost through leakage through the labyrinth seal into the hermetically-sealed case is replaced by sacrificial evaporation of oil-impregnated sintered nylon oil reservoirs which are positioned adjacent to each bearing.

The motor windings are impregnated to prevent movement under vibration which could cause shocks or open circuits.

9.2.2.9.7 EMI

There is very little EMI generated within the wheel itself, so this will not cause any problems. The main source of possible EMI interference is in the power leads coming into the wheels. Standard precautions such as filtering will be used to prevent excessive EMI radiation from these lines.

9.2.2.10 Auxiliary Propulsion Subsystem

The auxiliary propulsion subsystem as a part of the attitude control subsystem is designed to provide thrusts of suitable magnitude, duration, direction, and response time for precise three-axis attitude control, to provide momentum unloading of inertia wheels, and to provide translation thrusts for east-west and north-south orbit control of the ATS spacecraft in a nominal synchronous, equatorial, circular orbit.

The auxiliary propulsion subsystem is to provide these functions while operating in the environments encountered in the spacecraft for a minimum two-year mission, with a five-year mission as a design goal.

The spacecraft auxiliary propulsion subsystem selected for this requirement is an ammonia system using cold gas jets operating at 50 to 86°F for attitude control and high-temperature resistojets for orbit control. It consists of an ammonia propulsion assembly containing two regulated pressure feed assemblies in standby redundancy; 12 unheated attitude control thruster nozzles providing redundant three-axis control; and two, heated, dual-nozzle resistojet orbit control thruster modules.

This assembly is controlled by the propulsion control electronics portion of the actuator control electronics assembly containing signal logic, power converter, and power amplifier circuits. The auxiliary propulsion subsystem is integrated with the center section of the earth viewing module of the spacecraft.

The regulated pressure feed assemblies contain tankage for propellant storage, an evaporator or preplenum for conversion of propellant from liquid to gas, a plenum chamber for gas storage which is approximately 20 times the volume of the preplenum chamber, a regulating solenoid valve, a latching on-off solenoid selector valve, pressure sensor transducer, an electrical control with feedback to maintain the regulated set feed pressure, and distribution tubing with latching valves for controlling propellant supply to several groups of thrusters. These assemblies are located in the center section of the EVM.

The 12 unheated attitude control (A/C) thruster assemblies, are mounted on the perimeter at the bottom of the center section of the EVM, with the controlling solenoid valves located inboard on the feed and tank assembly. The two heated resistojet orbit correction (O/C) thruster assemblies, located on the truss in the plane of the spacecraft center of gravity, each contain two jet nozzles with solenoid valves located in the feed assemblies and with electrical heaters immediately upstream of each jet nozzle. Strip heaters and insulation are provided on the propellant lines to the O/C thrusters.

Further details of the auxiliary propulsion subsystem may be found in the Auxiliary Propulsion Subsystem, Section 12.0.

9.2.2.11 Commandable Gravity Gradient Subsystem

The COGGS mechanical assembly (boom, gimbal, and tip-mass deployer) and electronics package are part of the ACS. With the mechanical assembly mounted on top of the reflector hub, the tip mass is deployed away from the earth and the boom, gimballed in pitch and roll, and is driven by pitch/roll torquers. Attitude control is achieved by torquing against the inertia of the boom, based on gimbal command signals provided by the DOC. For control in yaw, the DOC provides signals to the yaw inertia wheel. Further details may be found in the COGGS section of this report (Section 11.0).

SECTION X

INTERFEROMETER

10.1 INTRODUCTION

The stated mission objective of the rf interferometer subsystem

"... to demonstrate precision radio frequency interferometer technology for spacecraft attitude sensing"

is readily achievable by the interferometer design which has been developed during the ATS F&G Phase B/C effort. The interferometer, by affording greater 3-axis sensing precision over a wide field of view than any of the other on-board attitude sensors, will measurably enhance achievement of the ATS spacecraft primary missions.

This report describes the interferometer design chosen to meet this stated objective.

The interferometer, which is a precision spacecraft-attitude angle sensor subsystem in the attitude control system, consists (as shown in Figure 10.1-1) of the spacecraft equipment, ground transmitters, and associated data processing equipment, both on-board (Digital Operational Controller—DOC) and via the main ground terminal.

The spacecraft interferometer determines the signal arrival angle from each ground emitter relative to the interferometer axes and presents this information in digital form to the ACS for closed-loop control and also to the ground data processor (via the telemetry link) for calibration, attitude determination, and/or open-loop spacecraft control.

A switched dual channel configuration was chosen for the interferometer implementation to exploit the inherent advantages of this technique. In the switched configuration, the dual channel phase measuring system is sequentially switched between the pairs of antenna elements that make up the orthogonal interferometer baselines. The switched configuration reduces system complexity, as compared to a dual channel receiver for each antenna pair, and allows use of redundant functional modules resulting in enhanced system reliability. Switching also enables use of two auxiliary (redundant) baselines as shown in Figure 10.1-1. The auxiliary baselines provide redundant attitude measurements for calibration and thus only two ground stations are required to obtain the same information otherwise requiring three ground stations.

The significant operating parameters of the interferometer are briefly summarized as follows:

Frequency	8.150 and 8.155 GHz
Sensitivity (NF)	16 dB
Angular Coverage	+17.5° unambiguous, - +20° operating

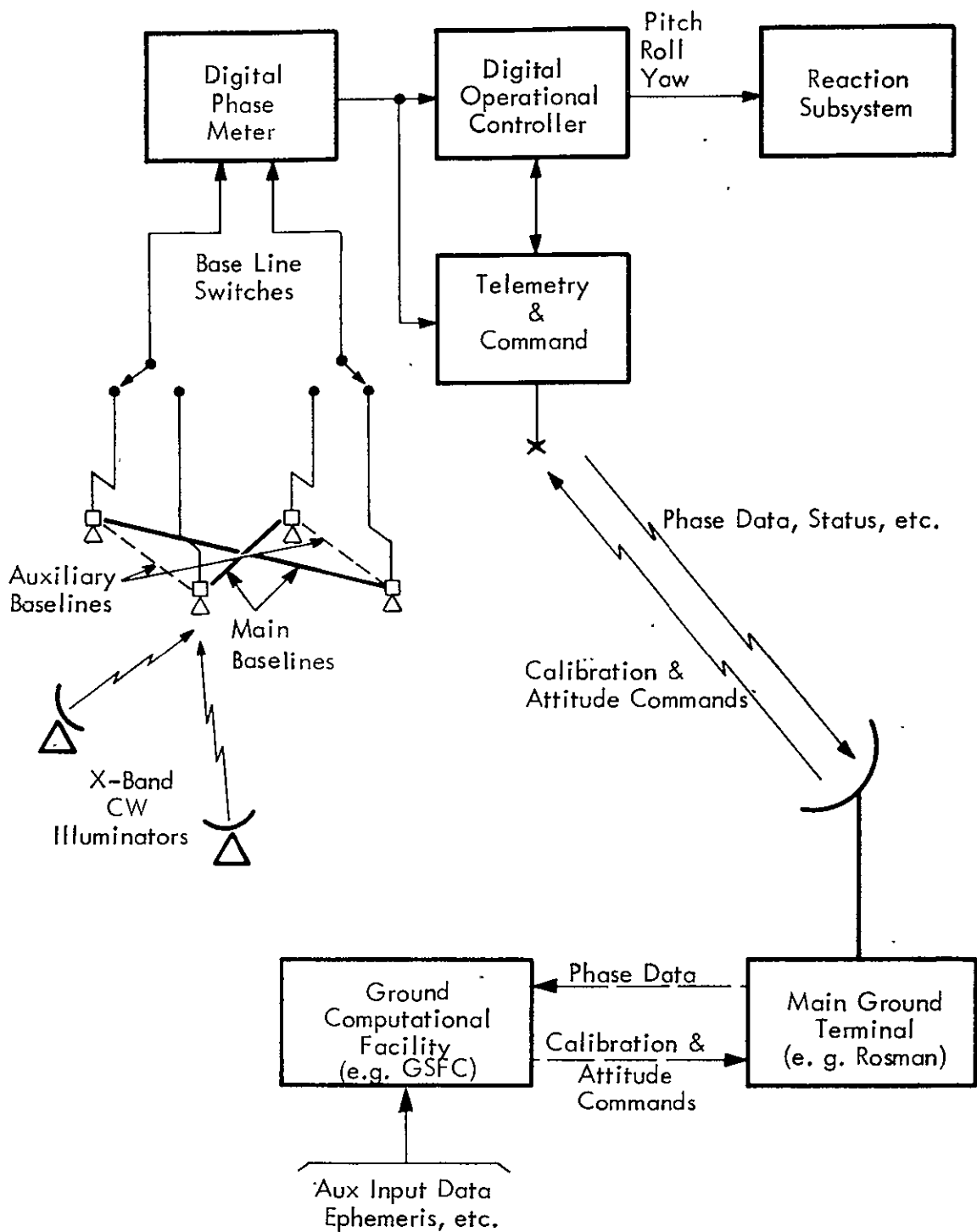


Figure 10.1-1. Interferometer Functional Diagram

Angle Measurement	
Attitude Accuracy	0.03°
Angular Resolution	0.005°
Baselines	
Vernier	26.6 λ (38.5")
Coarse	1.66 λ (2.4")
Required Ground Station ERP	76 dBW
Polarization	Linear
Weight	39.5 lbs
Subsystem Power Requirements	30 watts

As a precision attitude reference sensor for the spacecraft Attitude Control Subsystem (ACS), the interferometer primarily demonstrates techniques of precise attitude sensing and control. In addition, the precision achieved by this sensor allows use for several planned secondary roles:

- Backup earth sensor
- Backup 3-axis sensor
- High-resolution sensor for antenna pattern measurements
- High-resolution sensor for use in the low jitter mode of operation
- Position location.

Along with these planned modes of operation, the interferometer could perform an experiment in 3-axis closed-loop attitude control and could provide a reference for an effective earth coverage raster scan for the RFI experiment. Since the interferometer is operable as soon as the transtage is separated, attitude reference checks could be obtained prior to spacecraft deployment if desired.

10.1.1 FUNCTIONAL DESCRIPTION

The spacecraft equipment functionally consists of an antenna array and a phase measuring receiver. The receiver is divided into two subassemblies: the receiver front end, located adjacent to the antenna array, and the receiver/converter, located in the interferometer electronics unit. The electronics unit, mounted to a bulkhead in the EVM, also contains power supplies and interface circuitry.

Installation of the interferometer in the EVM experiment module is described in section 10.3.4.4. The antenna array consists of two orthogonal vernier baselines, each consisting of a pair of antenna elements mounted to a baseline support structure attached to the EVM. One baseline is parallel to the spacecraft pitch axis, and the other is parallel to the roll axis. An extra antenna on each baseline provides a coarse baseline and is used to resolve the ambiguous phase readings of the vernier channel.

The receiver assembly down-converts the received signals in such a way that accurate phase measurements can be made at a 2 kHz difference frequency. The receiver uses a dual local oscillator (DLO) to provide the 2 kHz offset frequency with the requisite phase stability. A digital converter at the output of the receiver provides a 9-bit digital word which represents the 2 kHz phase difference reading.

Simultaneous signals from two different ground stations are distinguished on the basis of a small frequency offset.

The difference between desired and actual attitude angles (phase readings) can be computed on the ground. The data needed in the computation includes four sets of phase differences which are measured on two orthogonal antenna pairs, receiving signals from two different ground stations. The ground station location and spacecraft ephemeris are also used in attitude determination.

The interferometer can be used either as a two-axis sensor (pitch and roll) or a three-axis sensor (pitch, roll and yaw). For two-axis sensing, one ground illuminator is required, while for three-axis sensing at least two ground stations are necessary.

10.1.2 PERFORMANCE SUMMARY

The performance of the interferometer as an attitude sensor can be characterized by two parameters: accuracy and resolution. Accuracy determines how precisely the interferometer can measure the attitude angles of the spacecraft with respect to a set of earth-centered inertial coordinates. Resolution determines how precisely the interferometer can measure relative attitude angles; that is, angles with respect to an arbitrary set of reference coordinates. Which of these two parameters is the most meaningful in characterizing interferometer performance depends on the application. An examination of the attitude control requirements of the various ATS experiments shows that precise angular resolution is generally more important than a knowledge of absolute attitude angles. For example, when the interferometer is used to control spacecraft slewing, e.g., for measuring the radiation patterns of the 30 ft. reflector, it is more important to have the requisite angular resolution to define the fine pattern structure than it is to know the precise angle at which the slewing took place. Similarly, the interferometer resolution is more important if it is used for an attitude sensor in the low jitter mode during the high resolution camera experiment.

The angular resolution achievable by the interferometer is much better than the angular accuracy since fewer and smaller error sources determine the resolution. The significant performance capabilities of the interferometer are summarized as follows:

- **Field of View:** the accuracy is maintained over a 35° field of view, and system operation is obtained over a 40° field of view.
- **Modes of Operation:** the use of three or four ground stations for redundant phase readings is unnecessary with the switched receiver, since it will provide two additional baselines through proper switch-sequencing. The three- or four-station mode has therefore been eliminated. A two-station mode with ground computation of attitude and a single-station mode with on-board closed loop operation with the attitude control subsystem are used.
- **Sensor Transfer Characteristic:** linear function between the measured phase delay and the spatial angle within the field of view.

- Frequency: the two ground stations will transmit carriers in the 8.025 to 8.250 GHz band with a frequency separation of 5 mHz.
- Antenna Array: a cross-shaped baseline support structure is used which has three antenna elements on each baseline. In addition auxiliary "diagonal" baselines can be configured by proper sequencing of the switches in the receiver. The mechanical stability of the antenna phase centers will be better than 1 part in 17,000 for baseline planarity type deflections. Baseline length changes will be sensed with transducers, and they can be calibrated out in the x-station mode by using phase data measured by the auxiliary "diagonal" baselines. The vernier baseline is 26.6λ long, and the coarse baseline is 1.66λ long. The ratio of these lengths is 16:1.
- Receivers: a switched dual channel receiver with triple down conversion is used. The second down conversion uses a double-local oscillator. The output frequency is 2 kHz. As compared to a 6-channel receiver hard-wired to the antenna elements, the dual-channel receiver has better phase stability since only two (instead of six) channels must phase-track. Standby powered-down redundant units parallel each channel. This configuration provides much higher reliability, lower cost, and less power consumption than a 6-channel receiver of the same size and weight.
- Spacecraft Data Processor: a 60-bit output word is provided in the 2-station mode. It contains mode identification bits and four phase readings—one vernier and one coarse from ground station 1, and one vernier and one coarse from ground station 2. The vernier phase word is 9 bits long, providing 0.0042° attitude angle per count. A complete output word is stored in the buffer every 200 milliseconds but the telemetry reads this word once every 3 seconds.
- Calibration: an on-board calibration oscillator is provided to inject equiphased calibration signals into the system. The calibration injection points are located directly in front of the switches, a location that the error analysis showed to be optimum. In the 2-station mode, scale factor errors due to baseline length or baseline orthogonality changes can be calibrated out by using phase readings from the auxiliary baselines. In the single-station mode, baseline changes can be sensed by using data from the strain and temperature transducers attached to the baseline support structure.
- Ground Transmitting Station: to achieve the specified system accuracy the ground station should be capable of generating a 76 dBw nominal ERP. This can be accomplished by a 15-ft-diameter paraboloid with a 500W input, a fairly modest requirement. These requirements are compatible with the ground station requirements for the communications experiments, so the same ground station can be used. The interferometer

system accuracy is degraded by only 7 percent if the ERP drops to +64 dBw and by 60 percent for a +58 dBw ERP. Polarization tracking to within $\pm 5^\circ$ between the spacecraft and ground station linearly polarized antennas is required to guarantee adequate ERP and low phase error at the spacecraft.

- Ground Data Processor: to achieve 3-axis sensing, calculation capability must be provided to solve for the three attitude angles from the four phase readings. To implement the baseline calibration procedure using diagonal baselines, a recursive least-squares filter algorithm must be available. During the Phase B/C contract, programs similar to these have been developed for the error analysis. These programs will be used in identifying and specifying the software and hardware necessary for the operational ground data processor.

Further interferometer capabilities are apparent if a comparison between the Interferometer subsystem and the optical attitude sensor subsystems (earth sensor, Polaris sensor) is made. The interferometer has the following advantages:

- No moving parts
- Self-calibration, using injection calibration or multiple baselines
- Wide field of view - at least 40°
- Rejection of spurious illuminators.

10.1.3

DEVELOPMENT SUMMARY

During the course of the Phase B/C design effort, an extensive mathematical model was developed and computer analysis of this model was used to perform design optimization studies. Statistical error analyses of the system model were then utilized to determine the manner by which various error sources effected the resultant attitude angle accuracy. The results of this error analysis were then employed to guide the component design. In addition, the mathematical model was exercised to determine which error sources were subject to reduction by in-flight calibration of the interferometer and to determine the feasible extent of these reductions.

Computerized numerical techniques were also used in the thermal, vibration, stress and RFI analyses. Detailed requirements for each subassembly and module were defined and these, in turn, determined the spacecraft equipment design. Design tradeoffs on the most promising candidate approaches were performed and an optimum electrical and mechanical design approach selected. The system was modularized for ease of fabrication, accessibility and maintainability. State-of-the-art circuits were used throughout for maximum reliability, efficiency, and cost effectiveness. Following the computer analysis, an interferometer feasibility demonstration breadboard was designed and fabricated and utilized to verify the results obtained from the math model.

This breadboard system is completely operational and demonstrates conclusively that the interferometer is capable of its predicted performance as a high precision, wide field-of-view attitude sensor.

Design assurance for the successful operation of the ATS Interferometer is provided by:

- Extensive error analysis
- Tradeoff analysis to optimize design
- Selection of existing design and components were feasible
- Selection of components from qualified parts list where possible
- Design, construction and extensive testing of a breadboard model, both on a module and subsystem basis
- Development and verification of alignment techniques
- Incorporation of an in-flight calibration technique
- Enhanced reliability through redundant functional modules.

10.2 FUNCTIONAL DESCRIPTION

10.2.1 SUBSYSTEM FUNCTIONAL ARRANGEMENT

As a system, the interferometer consists of the equipment installed in the spacecraft, two ground stations, and an associated computer. The interrelationship of the units in this system is shown in Figure 10.2-1. The function of the Interferometer Subsystem is to measure the three spacecraft attitude angles. It accomplishes this by making certain phase measurements which can then be converted to attitude angle measurements.

The function of the spacecraft equipment is to measure phase differences between pairs of antenna elements installed in the spacecraft. These phase differences are induced in the antenna elements by the ground transmitter. The basic equation (see Figure 10.2-2) relating the phase difference ϕ to the angle θ between the antenna baseline and the line of sight (LOS) to the ground station is:

$$\phi = \frac{2\pi D}{\lambda} \cos \theta \text{ (radians)}$$

θ = Angle between antenna baseline and LOS to ground transmitter

D = Spacing between antenna elements

λ = Operating wavelength of the received signal

For the two ground transmitters (see Figure 10.2-1), two phase differences ϕ_1 , ϕ_2 are obtained corresponding to two direction cosine components, $\cos \theta_1$ and $\cos \theta_2$.

The spacecraft equipment consists of an antenna array and a phase-measuring receiver.

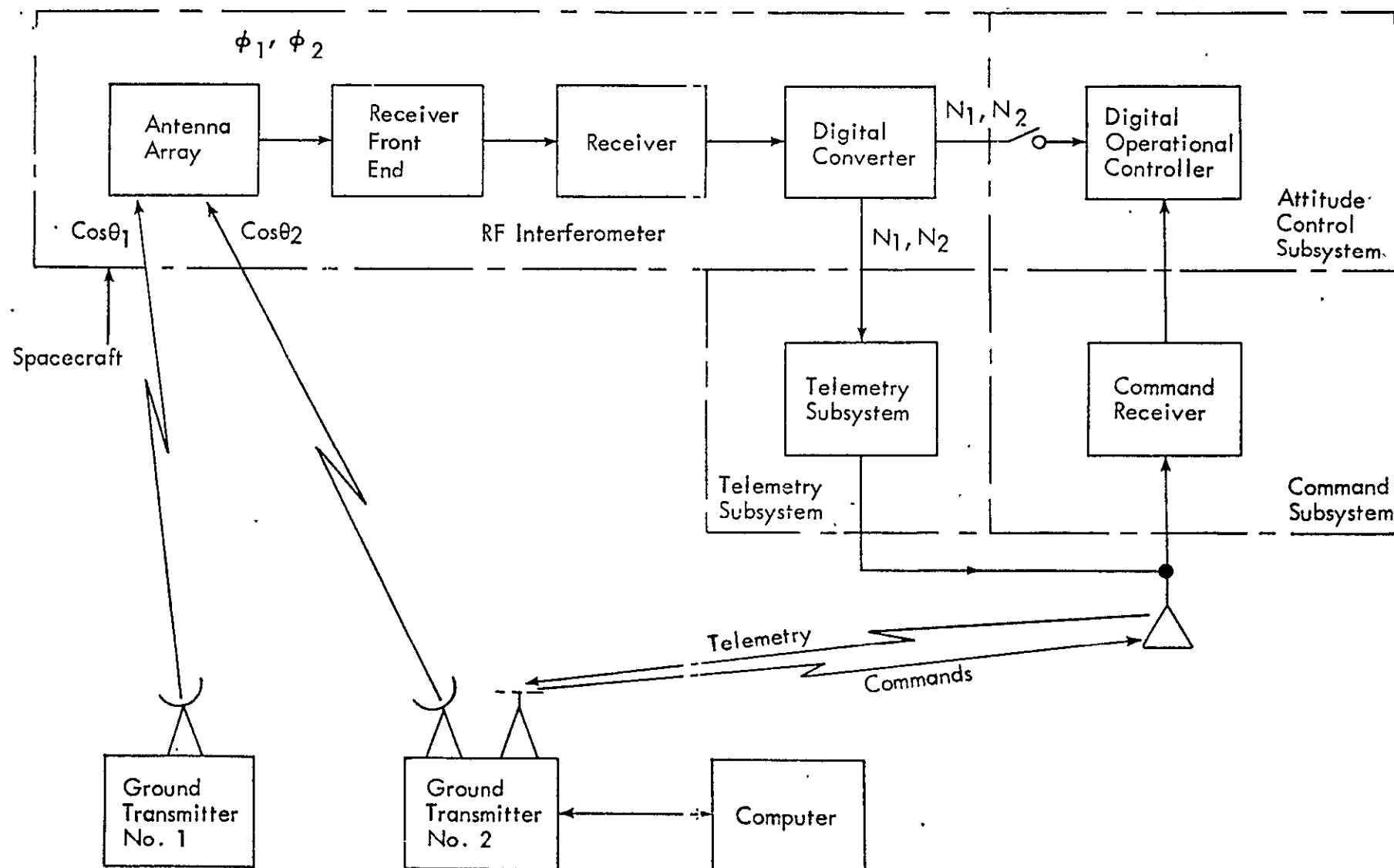
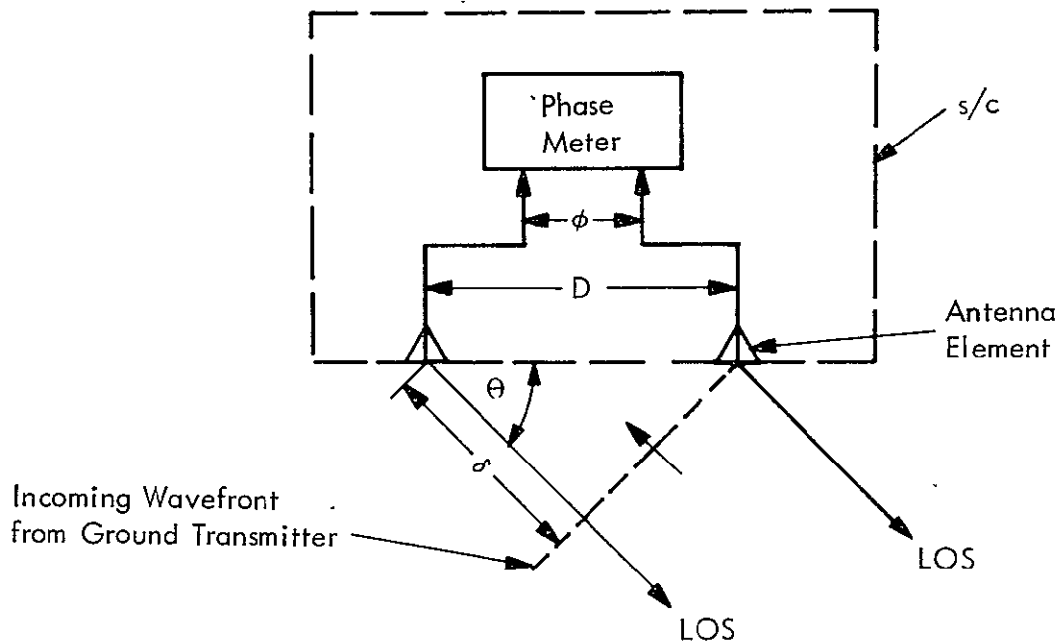


Figure 10.2-1. RF Interferometer System Functional Arrangement

The function of the antenna array is to provide the baselines. Each baseline consists of a pair of antenna elements mounted to a baseline support structure. Two orthogonal vernier baselines are provided by the X-shaped baseline support structure. One baseline is parallel to the spacecraft pitch axis, and the other is parallel to the roll axis. An auxiliary set of short baselines, called the coarse baselines, is also provided. The phase readings from the coarse baselines are used to resolve the ambiguous phase readings of the vernier baselines.



$$\delta = D \cos \theta$$

$$\phi = \text{Phase Diff.} = \frac{2\pi\delta}{\lambda} = \frac{2\pi D}{\lambda} \cos \theta$$

Figure 10.2-2. Basic Equation, Angle Measurement

The function of the receiver is to accurately convert the phase readings (ϕ_1 , ϕ_2 in Figure 10.2-1) which are taken at X-band frequencies into two digital words (N_1 and N_2 in Figure 10.2-1). The receiver consists of the receiver front end, the (main) receiver, and the digital converter. The functions of the front end are to perform a double down-conversion of the incoming signals, to sequentially switch the receiver to different antenna baselines, and to provide for injection calibration. The functions of the (main) receiver are to duplex the IF signal, channeling it to the proper IF amplifier according to its frequency, to provide the necessary IF amplification, and to provide detection for generating the data signal. The functions of the digital converter are to generate the digital words corresponding to the phase readings, to control the sequencing of the receiver, and to format, store, and output the digital words.

The functions of the ground-based equipment are to provide RF illumination of the spacecraft, to compute 3-axis attitude angles from the phase readings, to receive phase readings via the telemetry link, and to command mode changes via the command link.

The ground-based equipment consists of the two ground transmitters, the telemetry and command station equipment, and an associated computer (see Figure 10.2-1).

The function of the ground transmitters is to provide a linearly polarized CW X-band signal of the proper frequency and polarization direction. The function of the computer is to calculate the 3-axis attitude angles from the phase readings. To do this computation, four sets of phase readings are used. These are measured on the two orthogonal vernier baselines receiving signals from the two ground stations, which are at two different locations. The ground station location and the spacecraft ephemeris are also used in the computation.

The function of the telemetry equipment is to enable the phase readings (and also interferometer housekeeping data) to be telemetered to the ground station.

The function of the command equipment is to enable the desired mode of interferometer system operation to be selected. The system operates in either of two basic modes: as a 2-axis attitude sensor with a direct connection to the digital operational controller (DOC) in the Attitude Control Subsystem (ACS), or as a 3-axis sensor with the connection to the ACS closed through the telemetry link, computer, and command link. The on-board closed-loop mode is indicated in Figure 10.2-1 by the switch connecting the digital converter to the DOC. Other interferometer modes (e.g., calibration and redundant unit switchover) are also commandable.

10.2.2 FUNCTIONAL DESCRIPTION, SPACECRAFT EQUIPMENT.

A detailed block diagram of the system is given in Figure 10.2-3. The array assembly, which first receives the interferometer signals, comprises the antenna subassembly, cable subassembly, transducer module, and the baseline structure. The antenna subassembly is a collection of antenna elements which accept the incident energy at the proper radio frequency and over the required spatial field of view. Signals at the X-band frequencies of 8.150 GHz and 8.155 GHz can enter the system through the antenna elements.

The tradeoffs developed in selecting this frequency are given in subsection 6.2 of the technical proposal. There are six linearly polarized antenna elements that are arranged in the form of a cross with arms (baseline) coincident with the spacecraft pitch and roll axes. The polarization tradeoffs are discussed in subsection 6.5 of the technical proposal. The pitch baselines consist of the three antenna elements P_v , P_c , and P_r along the pitch axis, while the roll baselines consist of the R_v , R_c and R_r elements. The P_v and P_r elements form the roll vernier baseline; the P_c and P_r elements form the roll coarse baseline. The phase readings on the vernier baselines are multivalued or ambiguous. By spacing the coarse element R_c close to the reference element R_r , a short baseline is obtained which has an unambiguous count vs angle characteristic. The tradeoffs involved in selecting the coarse baseline method of ambiguity resolution are discussed in subsection 6.4 of the technical proposal.

From the antenna elements, the radio frequency signals are routed to the front end by the cable subassembly which is a configuration of semi-rigid coaxial transmission lines. The baseline structure is the mechanical supporting structure for

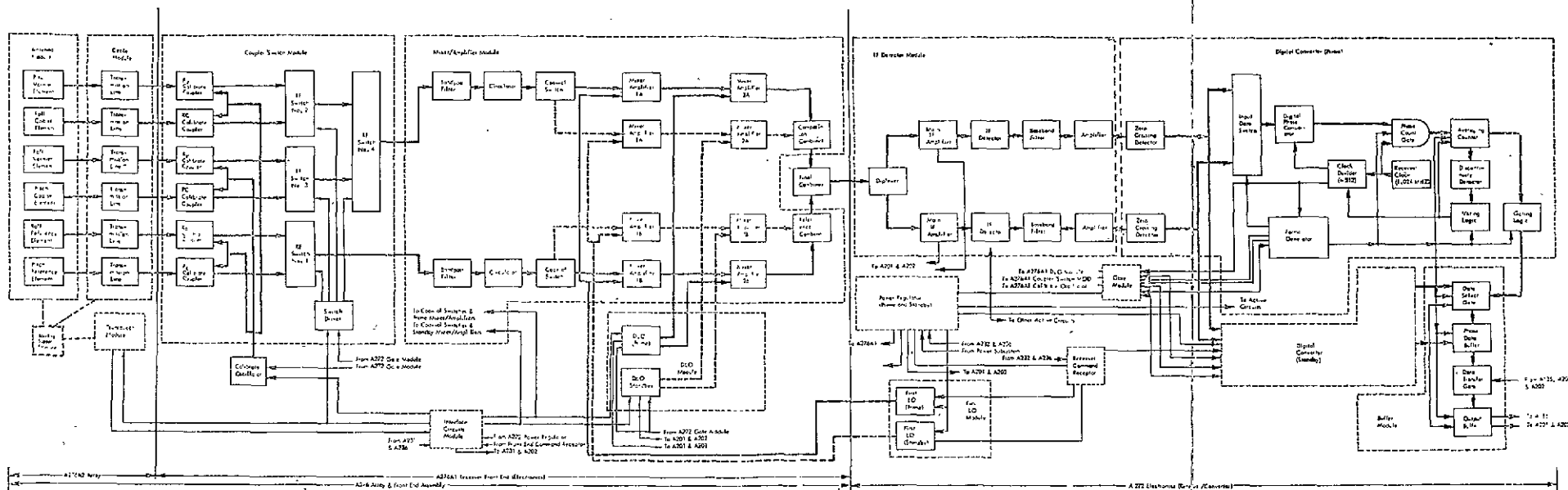


Figure 10.2-3. Interferometer System Functional Block Diagram

FOLDOUT FRAME 7

FOLDOUT FRAME 2 10-11/10-12

Table 10.2-2. Test Summary

No.	Description	Phase	Requirement
1	Passive Checkout	I	Non-illuminated verification of operation, including injection calibration.
2	Active Checkout	I	Illuminated operational verification using ground station.
3	Field of View	II	Determine the maximum field of view over which ambiguities can be resolved.
4	Ambiguity Resolution	II	Determine large angle accuracy limits.
5	Gain and Patterns	II	Combined with coarse and vernier ambiguity cycles to determine launch induced displacements.
6	Polarization Sensitivity	II	Combined with No. 3 for antenna displacements and to ascertain possible use of polarization for coarse yaw information.
7	Dynamic Stability	II	Determine apparent angular errors due to spacecraft rates and station-keeping impulses.
8	Calibration	II	Reduce bias type errors to the maximum extent possible.
9	EMI	II	Determine residual effects of combinations of on-board radiation sources and possible ground interference signals.
10	Boresight Alignment	II	Determine the degree to which the interferometer measured angles can be related to other spacecraft subsystems, especially ACS and Parabolic Reflector.
11	Low SNR Accuracy	II	Determine angle measurement accuracy as a function of signal strength to ascertain minimum ERP requirements for "small-terminal" operation.
12	Angle Accuracy	II	Similar to No. 10 for relatively large angles off boresight.
13	Small Angle Differences	II	Accuracy of small angular differences as a function of angle off boresight.
14	Spacecraft Pointing (open loop)	III	Determine the ability to interpret attitude data from the interferometer in terms of the ARS.
15	Attitude Control P, R, (closed loop)	III	Demonstrate feasibility and usefulness of the interferometer as a closed-loop reference.
16	Attitude Control (P, R, Y)	III	Demonstrate utility of interferometer as a 3-axis sensor.
17	Position Location	III	Employ data from previous tests to locate an "unknown" emitter as a function of angle off boresight and ERP.
18	Tracking	III	Measure performance of interferometer used as a tracking instrument for pointing parabolic dish at moving terminal (like DRS).
19	Drift	II, III	Determine long-term effects of component drift and thermal effects.
20	Propagation	III	Ascertain degree to which angle accuracy is affected by propagation through the atmosphere as a function of time and/or emitter elevation angle to spacecraft.
21	Life	III	Determine long-term degradation effects, if any, and verify anticipated system reliability.

Table 10.2-3. Ground Station to Interferometer Link Parameters

Transmitter ERP (frequency = 8150 MHz)	+76 dBW (500w, 15' dish)
Propagation Path Attenuation	-203 dB
Rainfall Absorption	- 1
Oxygen and Water Vapor Absorption	nil
Receiver Antenna Gain (at edge of field of view)	+12 dB
Received Signal	-92 dBm
Receiver Noise Power Density	-174 dB (mW/Hz)
IF Bandwidth (1.0 MHz)	+ 60 dB
Receiver Noise Figure (incl. 6 dB rcvr. losses)	+ 16
Noise Power in IF	- 98 dBm
Signal-To-Noise Ratio (SNR) in IF	+ 12 dB
Post-Detector SNR (Approx.)	+ 7
Post-Filter SNR (750 Hz BW)	+ 38
Sample Averaging Improvement (64 samples)	18 dB
Data SNR	+ 56 dB
Data SNR Required (to achieve 0.12° gating error)	+ 48 dB (analytical) to + 52 dB (experimental)

10.3 SUBSYSTEM DESCRIPTION

10.3.1 GENERAL SPACECRAFT SUBSYSTEM DESCRIPTION

The interferometer spacecraft subsystem consists of two assemblies:

- The array and front end assembly A276.
- The electronics (receiver/converter) assembly A272.

The array and front end assembly, which is subdivided into the array subassembly A276A2 and the receiver front end A276A1, is mounted to the earth viewing face of the Experiments Module (see Figure 10.3-1). The Experiment Module is the lower section of the Spacecraft Earth Viewing Module (EVM).

The Receiver/Converter Electronics Unit, mounted on an internal thermally controlled support shelf in the experiments section, will interface with the Front End Electronics using three semi-rigid coaxial lines and a conventional wiring harness.

An artist's concept of the antenna array assembly is shown in Figure 10.3-2. Figures 10.3-3 and 10.3-4 show the front end subassembly and the receiver/converter assembly.

10.3.2 MASS CHARACTERISTICS

The mass characteristics of the interferometer are given in Table 10.3-1.

Table 10.3-1. Mass Characteristics

	Weight (lbs)	Volume (cu. in.)
Array Subassembly (A276A2)	11.7	1617
Receiver Front End (A276A1)	13.9	799
Array Assembly (A276)	25.6	
Receiver/Converter (A272)	13.9	750
TOTAL	39.5	

10.3.3 POWER CONSUMPTION

The power consumption of the interferometer system is detailed in Table 10.3-2. The system has the following operating states:

- Normal interferometer operation
- Interferometer with onboard calibration

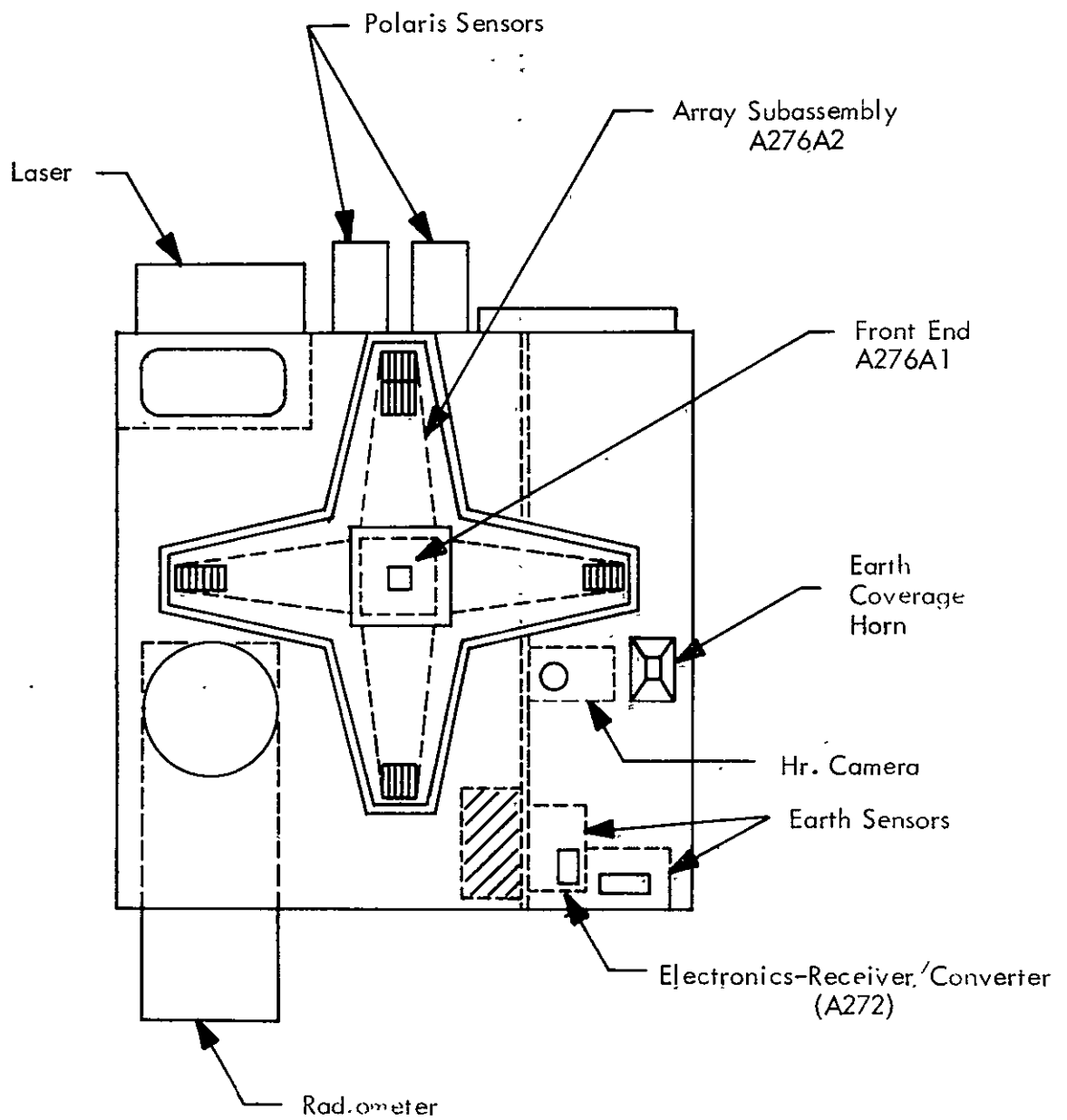


Figure 10.3-1. Interferometer Installation in Experiments Module

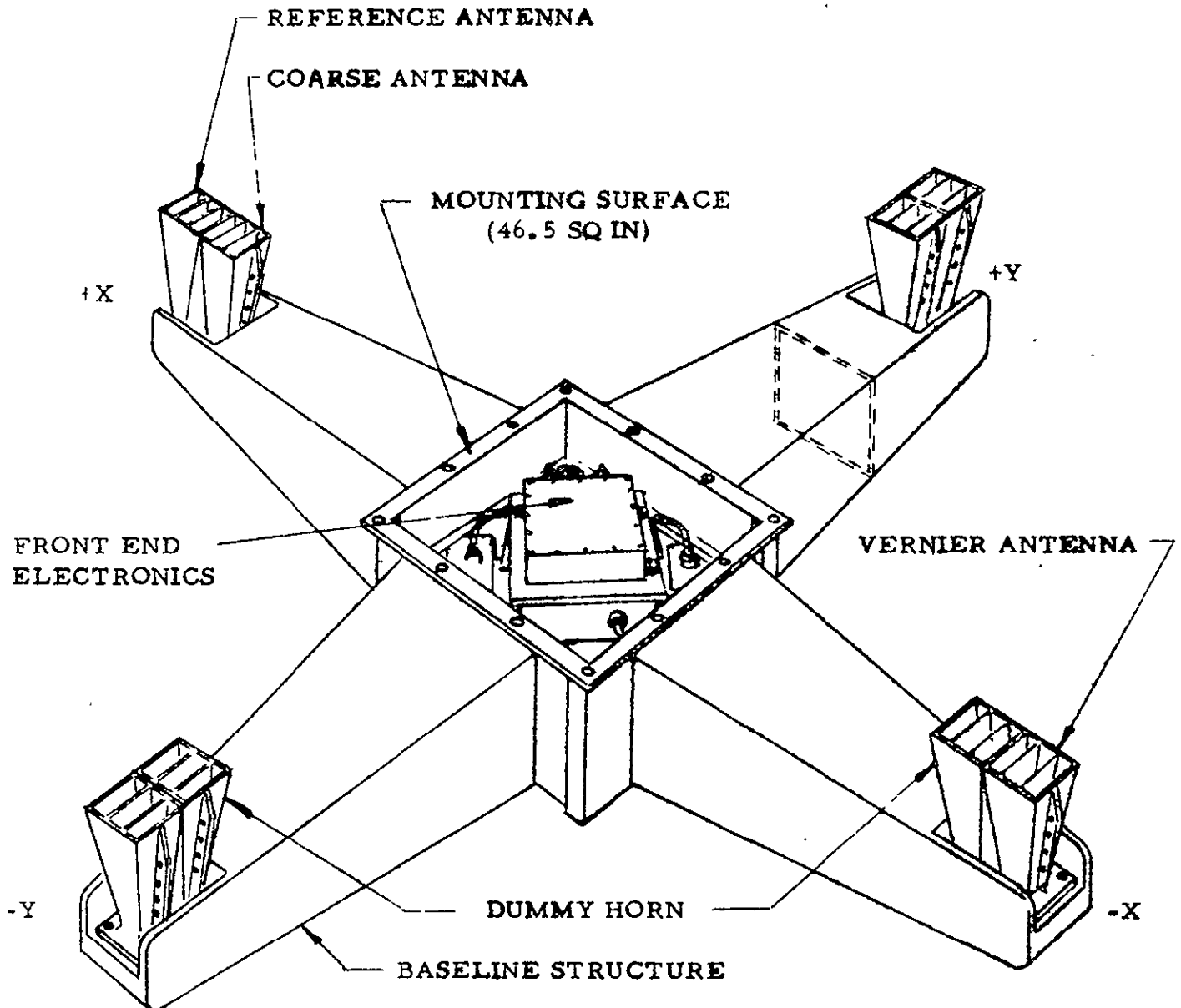


Figure 10.3-2. Interferometer Array Assembly

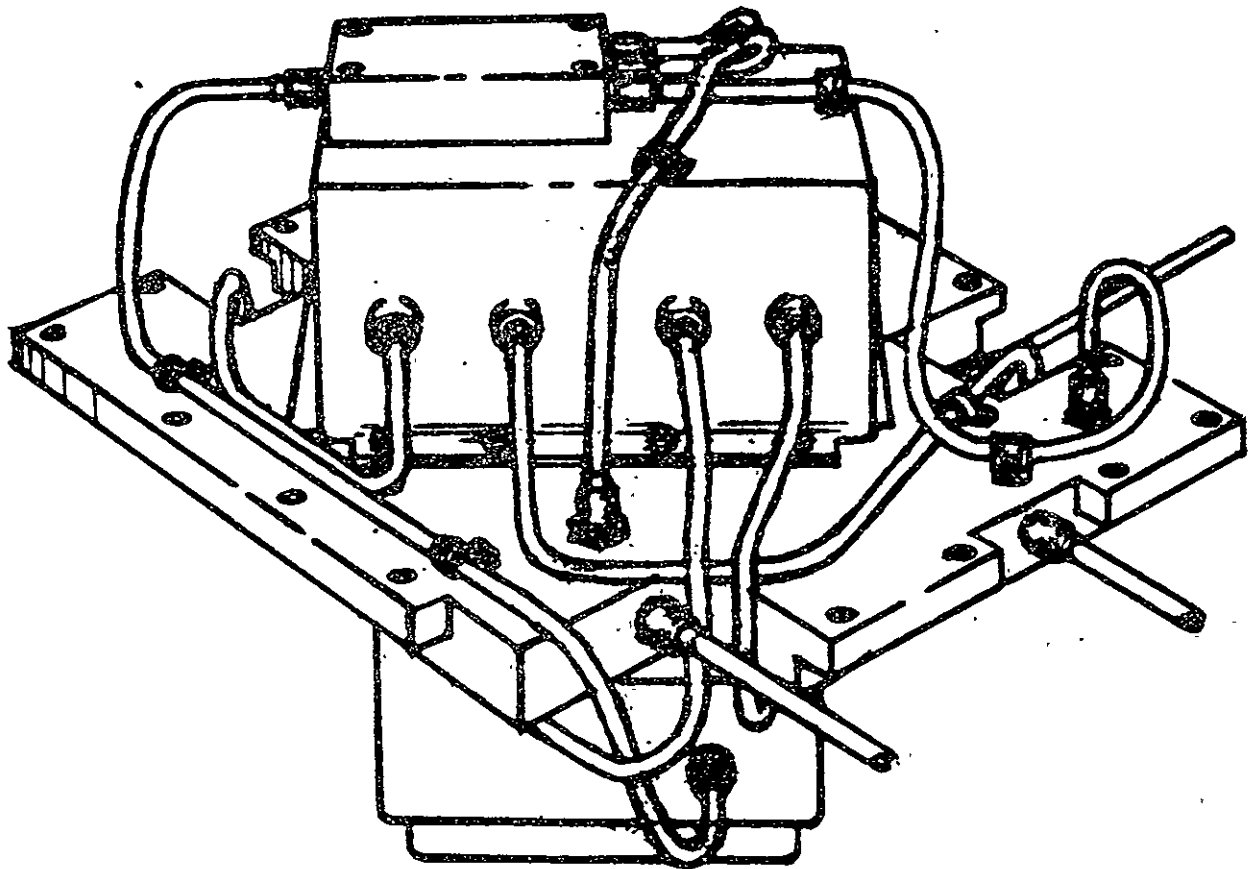


Figure 10.3-3. Receiver Front End

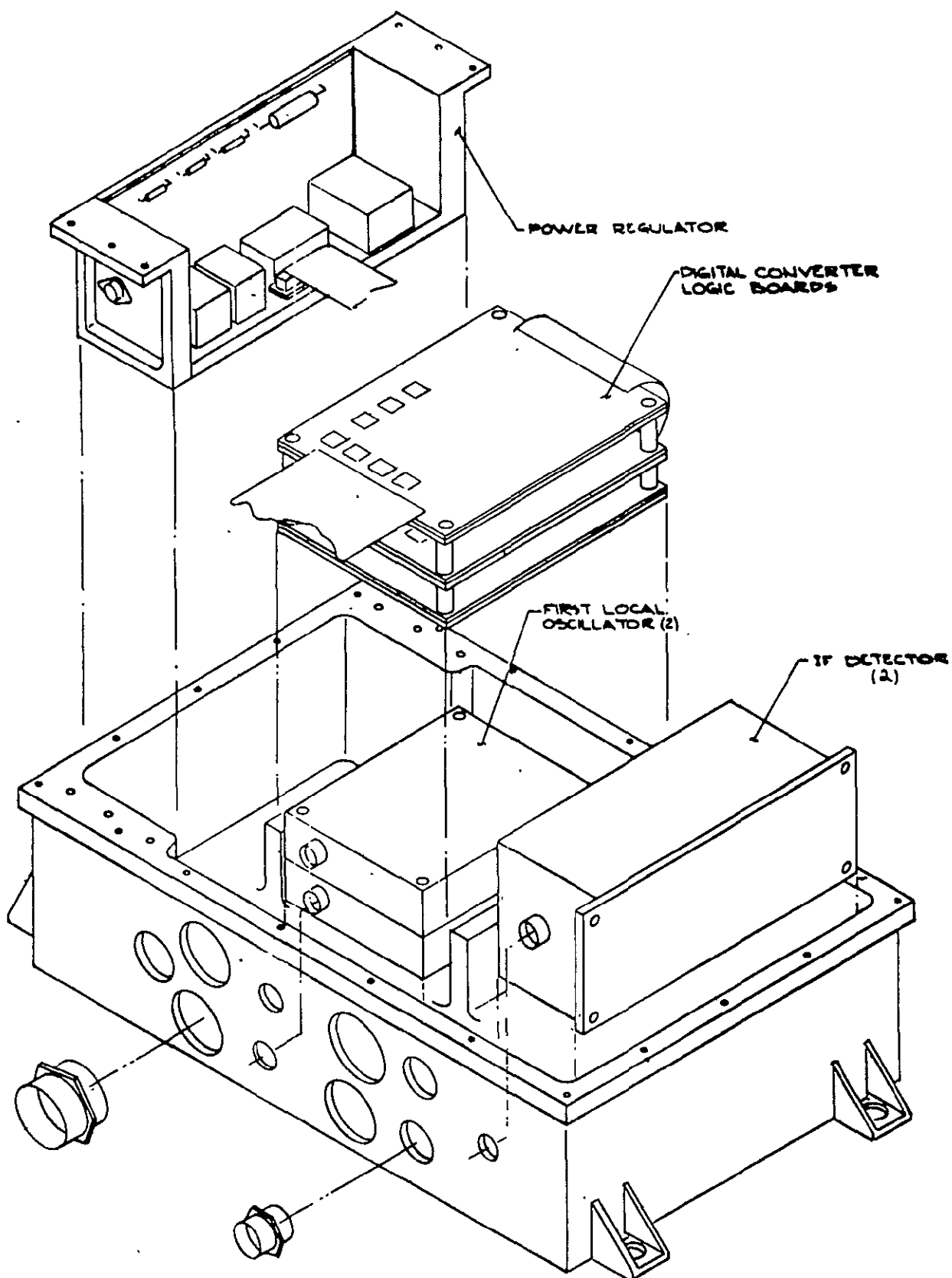


Figure 10.3-4. Receiver/Converter Electronics Unit (RCEU)

- Normal operation plus array transducers operating
- Switching to redundant RF front end (50 msec duration)
- Interferometer OFF

Table 10.3-2. Power Dissipation in Each Assembly*

	State 1	State 2	State 3	State 4	State 5
ELECTRONICS (RECEIVER/CONVERTER) (A272)					
Regulated Power	13.8	13.8	13.8	13.8	0
Regulator Loss	12.1	13.0	12.8	12.1	0.05
Total	25.9	26.8	26.6	25.9	0.05
RECEIVER FRONT END (A276A1)	4.4	5.6	5.4	30.4	0
INTERFEROMETER POWER DISSIPATION	30.3	32.4	32.0	56.3	0.05

*All figures are in watts

10.3.4 ARRAY AND FRONT END ASSEMBLY MECHANICAL/THERMAL DESIGN

The array is designed to sustain stresses introduced during launch and injection without mechanical distortions which will degrade electrical performance. The array also controls the system response to orbital environmental variations to minimize the effect on electrical phase delay. Figure 10.3-5 shows the array assembly.

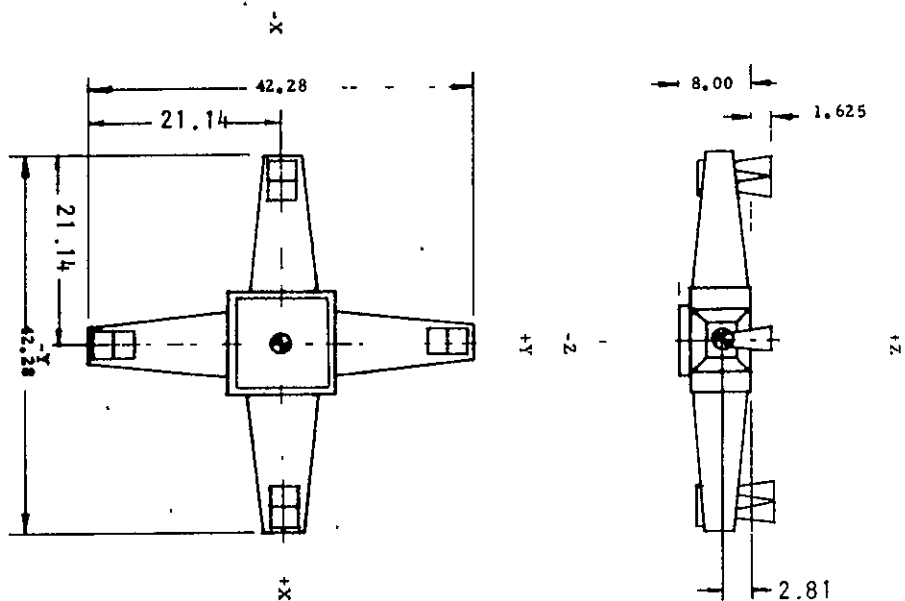
10.3.4.1 Structural Design

The array design concept incorporates "single-point" mounting, a lightweight, rigid, stress-free structure on which six antenna elements are mounted. The structure also supports the front end receiver electronics at its center hub and provides for mechanical support and thermal control of the transmission lines.

10.3.4.2 Vibration Characteristics

The most severe vibration environment is anticipated during launch and results from launch vehicle vibrations transmitted through the spacecraft structure. Preliminary specifications of the vibratory levels expected at the interferometer mounting structure show that during launch, levels as high as 20 G's within a frequency range from 50 to 80 Hz could be experienced for short durations. This range may be extended to above 100 Hz; therefore, the baseline structure is being designed for an input of 20 G's in the range from 50 to 150 Hz along three mutually perpendicular axes, one of which is normal to the mounting surface. At other frequencies it is expected that a maximum of 10 G's will be experienced.

The dynamic requirements of vibration will be met by combining the baseline structure and contained subassemblies into a lightweight integrated assembly with mutually supporting parts. The proposed subassembly designs are logical advances from designs proven in previous space programs, such as the inertial



NOTES

- 1 VOLUME - 1617 CU IN.
- 2 WEIGHT - 11.7 LBS
- 3 MOMENT OF INERTIA
 - I_X - 880 LB IN.²
 - I_Y - 925 LB IN.²
 - I_Z - 1725 LB IN.²

Figure 10.3-5. RF Interferometer Array Assembly

guidance platform mounting bracket used on the Saturn IB and V instrument units. The subassemblies are designed to provide structural integrity when subjected to worst-case vibration, shock, acoustics, and acceleration levels while providing an acceptable "ride" for the contained subassemblies. To accomplish this, the resonant frequencies of the baseline structure and the subassemblies, such as the horns and front end components, are mismatched to prevent coincident resonance frequencies. The proposed array assembly will have a fundamental frequency above 125 Hz.

10.3.4.3 Thermal Design

Thermal design of the interferometer is based on a total subsystem analysis to identify the heat loads throughout the orbital cycle. Steady-state analyses were investigated at the extremes of thermal loading to determine the worst-case temperature distribution throughout the subsystem.

The thermal analyses were performed on an IBM System/360 Model 50 with the use of the NATA II three-dimensional heat transfer program. This program uses numerical methods which consider rectangular parallelepipeds (nodes) to mathematically represent all solid portions of the unit being studied.

Preliminary thermal analyses and tests were performed on the proposed ATS F&G subassembly to verify the design integrity for the thermal environment. Analyses were performed for seven conditions on the entire subassembly. The maximum/minimum temperatures occur at the face of the antenna elements, and are 145°F and 28°F. For more details see Section 3.2.2 of the Technical Proposal.

10.3.4.4 Installation

Figure 10.3-6 is an exploded view of the assembly as it would be installed in the experiments module. The array assembly is mounted to a honeycomb support shelf which is similar in shape to the array structure.

An optical cube, mounted at the junction of the spacecraft coordinates, will provide the alignment reference for the interferometer and all other attitude sensors. The cross-shaped honeycomb support shelf will be pinned to the spacecraft structure, and the optical cube will be affixed with reference to the attitude control reference axes. The shelf will then be removed and given to IBM for interferometer installation. IBM will mount the interferometer array assembly to the plate and align the axes to the optical cube. Since all subassemblies are pinned in place at installation, the interferometer should remain within the alignment tolerances after installation in the spacecraft and closure of all spacecraft sections have been completed.

10.3.5 FRONT END SUBASSEMBLY MECHANICAL/THERMAL DESIGN STRUCTURAL DESIGN

The RF receiver front end consists of seven component modules including: the coupler/switch module (the main structural element), the double local oscillator, the mixer/amplifier module, the combiner module, the calibration oscillator, the calibration power divider, and the interface circuits module. The components or modules are mounted inside the hub of the array support structure.

The coupler/switch module is a stripline module configured to act as the main structural element at the front end. It is designed to mount directly into the hub of the baseline structure. The module will withstand the structural loads

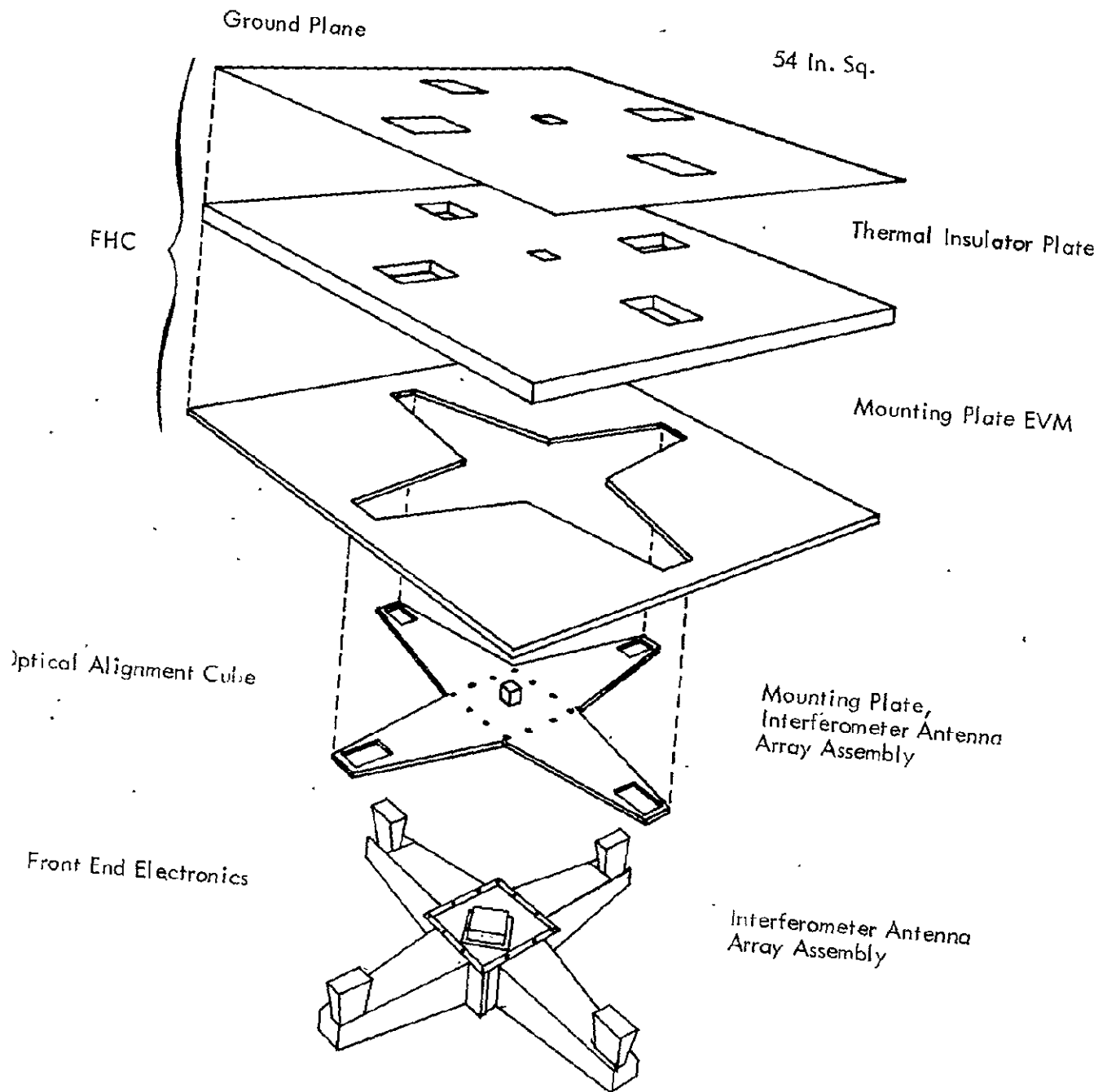


Figure 10.3-6. Installation of Interferometer Antenna Array

produced in the launch environment and will provide a thermal path for conduction of heat to the array subassembly. Figure 10.3-7 shows the front end design layout. The existing structure is used wherever possible in order to minimize weight. Electronic boxes are used as part of the support structure (employing a technique successfully demonstrated in the Surveyor program). The use of additional and separate brackets is unnecessary, permitting a lighter overall assembly.

A further advantage of the configuration is the accessibility and maintainability of individual modules and cables. The unit can be assembled independently of the array structure and can be tested and handled as a separate unit. The only interfaces with the array subassembly are the mechanical mounting on a flange within the hub and the cable connections to the antenna elements.

For the most part, module housings will be fabricated from aluminum, which provides a good combination of thermal capacity, structural strength, and light weight.

10.3.5.1 Thermal Design

The total power dissipation of the RF front end subassembly is 5.6 watts.

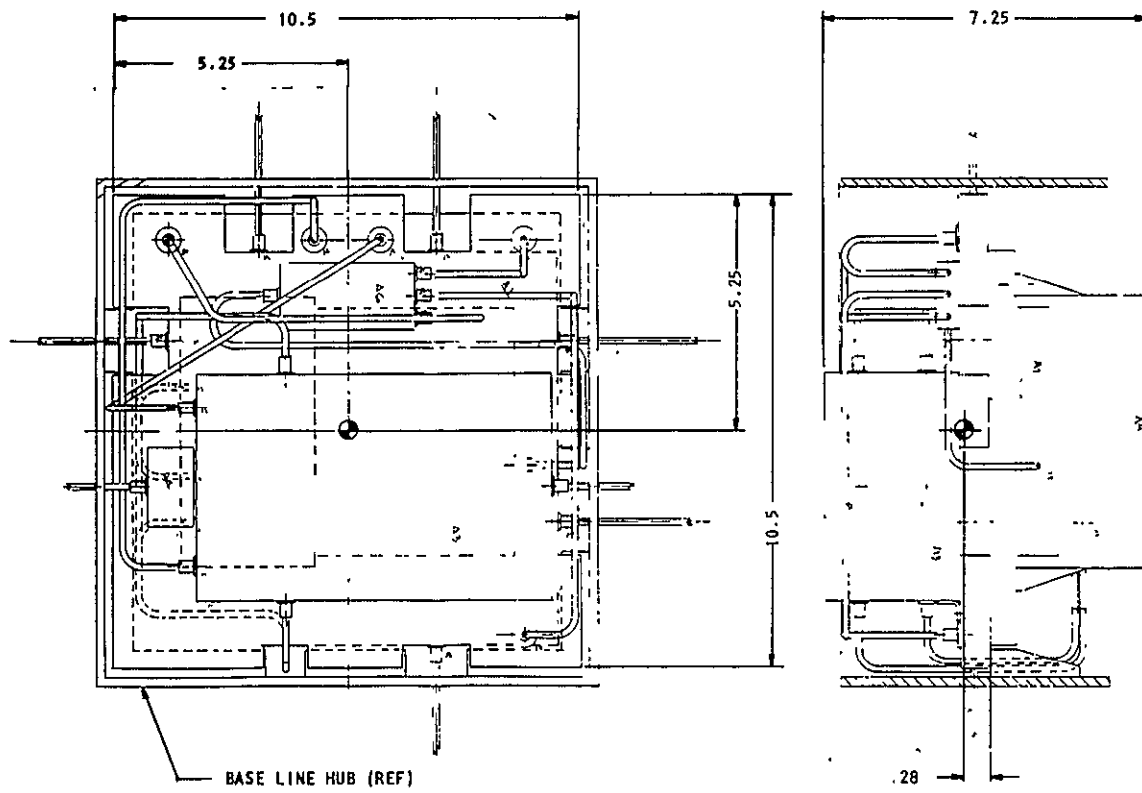
The coupler/switch module is designed to dissipate the heat by conduction to hub walls and array structure. The DLO and MAM will distribute and transfer their heat to the structural portion of the CSM housing, which in turn will conduct the heat array from the electronics to the array hub.

10.3.6 ELECTRONICS RECEIVER/CONVERTER MECHANICAL/THERMAL DESIGN

The receiver/converter electronics unit (RCEU) is located on a thermally controlled mounting plate inside the EVM. The RCEU circuit components consist of six functional units mounted separately in a magnesium housing.

10.3.6.1 Structural Design

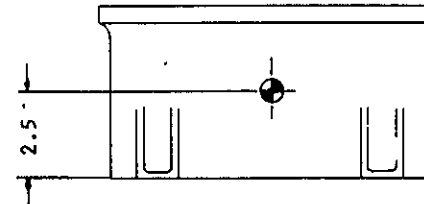
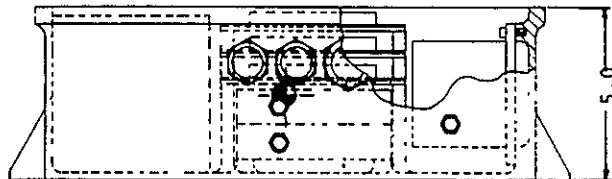
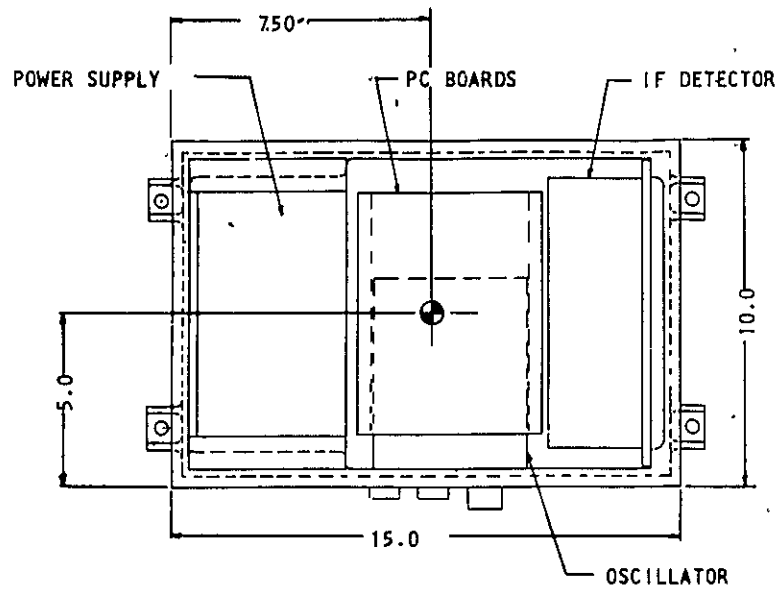
The RCEU housing will be a magnesium alloy casting suitably reinforced to withstand the ATS F&G shock and vibration environment. This reinforcement is principally provided by ribs extending the length of the structure. Machined bosses and/or ribs containing locking type inserts are provided for securing modules. Locking type inserts are also provided along the upper edge of the structure for securing the access cover. Magnesium alloy was chosen for the RCEU structure because of its thermal conductivity, vibration damping properties, high stiffness-to-weight ratio, weight, and excellent machinability. (See Figure 10.3-7A.)



NOTES:

1. VOLUME - 799.3 CU. IN.
2. WEIGHT - 13.9 LBS.
3. MOMENT OF INERTIA
 - $I_x = 101.1 \text{ LB-IN}^2$
 - $I_y = 100.6 \text{ LB-IN}^2$
 - $I_z = 129.4 \text{ LB-IN}^2$

Figure 10.3-7. RF Receiver Front End Electronics



NOTES

- 1 VOLUME - 750 CU. IN.
- 2 WEIGHT - 13.9 LB
- 3 MOMENT OF INERTIA

$$I_X - 144.79 \text{ LB. IN}^2$$

$$I_Y - 289.50 \text{ LB. IN}^2$$

$$I_Z - 376.35 \text{ LB. IN}^2$$

Figure 10.3-7A. Receiver Converter Electronic Unit

10.3.6.2

Functional Packaging

The RCEU will consist of the following functional units:

- The power regulator will be packaged into a lightweight structure capable of dissipating the heat from the power transistors to the outside RCEU structure. Components (resistors, capacitors, indicators, transistors, diodes) will be solder-attached to printed circuited boards mounted in a magnesium alloy frame. High dissipators, such as power transistors, will be "heat-sinked" directly to the structure to ensure operation below the recommended derated levels specified by Reliability. Connections will be made through a cable block and harness assembly.
- The digital section of the RCEU will be located on three multi-layer interconnecting boards (MIBs) interconnected through a cable block assembly and flexible tape cable. Flatpacks and discrete components will be solder-attached to the MIBs which incorporate feedthrough (or plated-through) holes. The MIBs will be stacked and mounted on structural bosses in the RCEU housing. Since the piece parts or components making up the digital converter are all low-power devices, the thermal conduction paths need not be rigorously designed.
- The IF detector module will be a modified off-the-shelf component module purchased to a specification. It will be mounted against a side wall in the RCEU.
- The first local oscillators shown are considered to be two separate units, however, it has been proposed to prospective vendors that the two units be combined into one. Considered as a single unit, it will mount to the base of the RCEU housing. It dissipates 8 watts and, therefore, will be mounted directly to the magnesium housing to provide a conductive path to the thermally controlled mounting surface.
- There will be 10 external connectors on one side of the housing for interface with the balance of the subsystem and spacecraft. Three of the connectors will interface with the RF electronics in the receiver front end, six will be used for power signals, and a single multiple pin connector will be used for test points.

10.3.7

MODULE DESCRIPTIONS

The performance requirements of each module are tabulated in the following subsections.

10.3.7.1

Antenna Element Module

The antenna element to be used is an amplitude-compensated pyramidal horn. The amplitude compensation produces approximately equal field distributions in the E and H-planes, and is achieved by incorporating two plates inside the horn.

Performance Requirements—Electrical

<u>Phase Patterns:</u>	constant within $\pm 0.5^\circ$ over ± 17.5 from main beam peak in either E or H-planes
<u>Mutual Coupling:</u>	less than 40 dB, E and H-planes
<u>Radiation Pattern:</u>	no sidelobes less than 25 dB below main beam
<u>Beamwidth (3 dB):</u>	45° H-plane, 38° E-plane
<u>Gain:</u>	greater than 11 dBI
<u>Polarization:</u>	linear over field-of-view, cross-polarized response greater than 45 dB down
<u>VSWR:</u>	less than 1.1:1, with transition to 50Ω

See Subsection 3.6.1.1, Technical Proposal, for more details.

Performance Requirements—Mechanical

The mechanical design of the antenna elements is based on the need to accurately reproduce the horns and to minimize tolerances within each horn. The most important aspects of fabrication are the maintenance of symmetry about the center, colinearity with the axis through the wave guide section, parallelism of separators, and orthogonality of separator plates and sidewalls.

The design must also minimize thermal distortions caused by localized heating; isolate the baseline from thermal input; have sufficient strength to resist launch, injection acceleration, and vibration distortions; be adaptable to the space environment; and maintain its physical properties during the useful life of the spacecraft. See Subsection 3.6.1.2, Technical Proposal, for more details.

10.3.7.2

Transmission Line Module

The transmission line assembly consists of six transmission lines and waveguide-to-coaxial adapters which interconnect the antenna elements with the RF front end. It is necessary that they maintain the phase relationships introduced at the antennas, with little degradation under all operating conditions. They must have excellent phase stability, low VSWR, low attenuation, and low weight and size. In addition, they must be mechanically stable in order to withstand the environmental variations.

Performance Requirements

<u>VSWR:</u>	less than 1.15:1 (50Ω)
<u>Insertion Loss:</u>	less than 1 dB

<u>Phase Balance:</u>	less than 2° between pairs of lines
<u>Phase Tracking:</u>	phase balance shall vary less than $\pm 0.5^{\circ}$ over temperature and frequency

See Subsection 3.6.2, Technical Proposal, for details.

10.3.7.3 Interface Circuits Module

The interface circuits module contains circuits which perform the following control and monitor functions for the interferometer:

- Filter and distribute the supply voltages
- Apply power to primary or standby components upon command
- Apply power to the calibration oscillator upon command
- Provide the circuitry necessary to accept the array sensor leads (thermistors and strain gauges) and convert to a voltage for telemetry monitor
- Perform on-board calibration of the transducer circuits upon command
- Supply a binary voltage level to digital converter to indicate calibration mode operation, and one bit to telemetry to indicate which redundant front end is powered.

See Section 3.6.4, Technical Proposal, for details.

10.3.7.4 Coupler/Switch Module

The coupler/switch module (Figure 10.3-8) accepts RF input signals from the coarse, vernier, and reference antenna elements through the coaxial cable assembly connected to each antenna element. It also provides the RF switching functions associated with interferometer operation. This unit has provision for injection of RF signals from the calibration oscillator for measurement of the system phase error for calibration. The module provides an RF input for each channel of the dual channel interferometer receiver.

This module will consist of strip transmission line couplers and hybrids with PIN diode switch components imbedded in the boards.

Performance Requirements

Input and Output VSWR, Over Specified Bandwidth:	1.10 max
Insertion Loss:	2.5 dB
Calibration Signal Input Coupling:	30 dB $\pm$ 0.25 dB

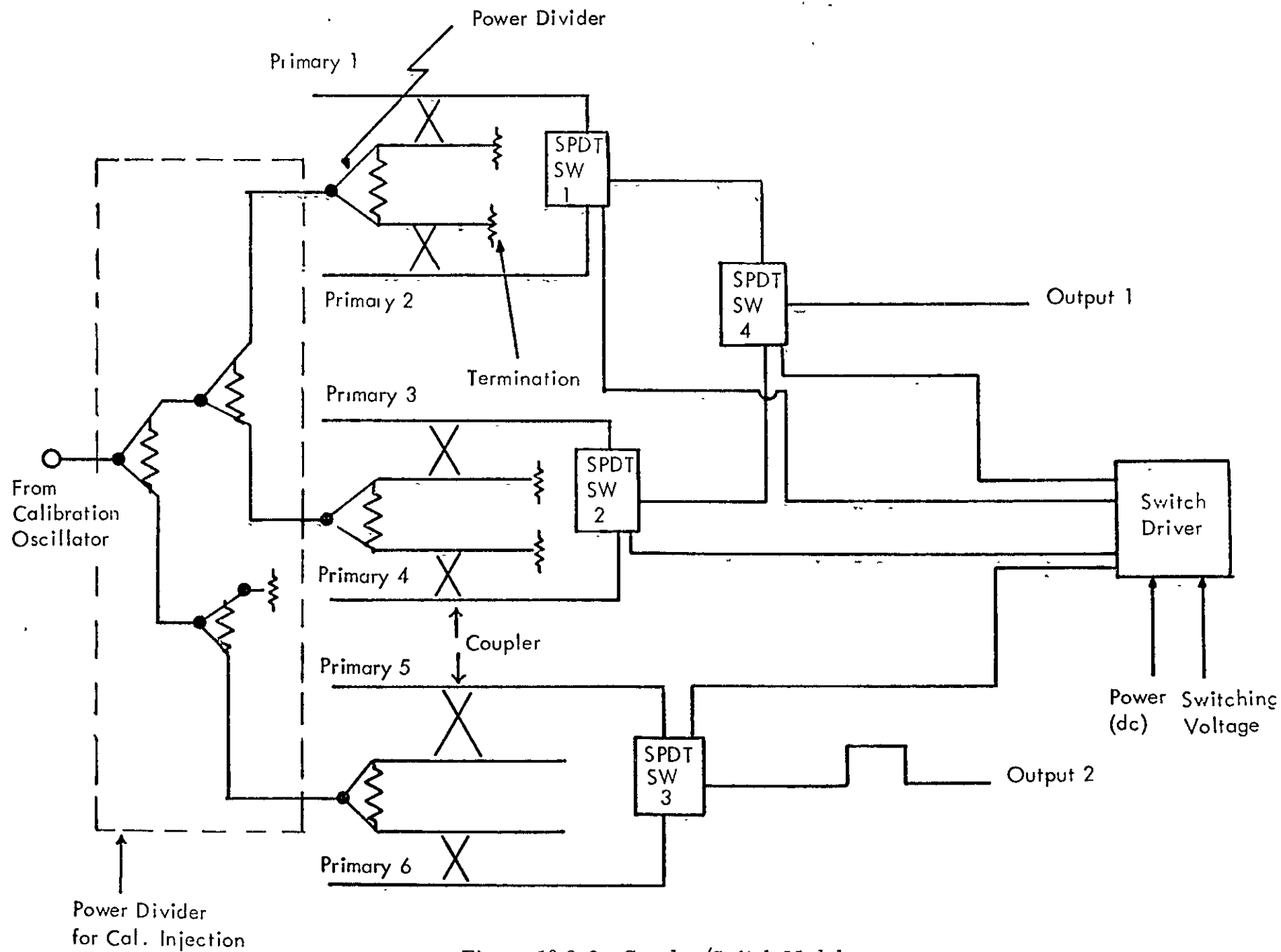


Figure 10.3-8. Coupler/Switch Module

Output Power Unbalance:	± 0.5 dB
Output Phase Unbalance:	± 2 degrees
Isolation:	55 dB min., between all inputs and between the two outputs

See Subsection 3.6.5, Technical Proposal, for more details.

10.3.7.5 Mixer/Amplifier Module

The mixer/amplifier module is a 2-channel dual conversion receiver with two standby redundant channels. This unit, which includes preselector, an electromechanical switch, and summing networks, receives RF signals from the coupler/switch module. One channel receives an RF signal from the reference horn antennas; the other channel receives the output of the coarse or vernier horn antennas, depending on the switch state of the coupler/switch module.

Performance Requirements

Input Center Frequency (signal)	8152.5 MHz
Input VSWR	1.1 max
Input to First Local Oscillator Port	
• Frequency	8000 MHz
• VSWR	1.1 max
• Power Level	4 mW, nominal
• Isolation to Module Signal Input and Output Ports	70 dB
Input to Second Local Oscillator Port	
• Frequency	122.5 MHz
• VSWR	1.1 max
• Power Level	8 mW, nominal
• Isolation to Module Signal Input and Output Ports	70 dB, min
Overall Module Gain	40 dB nominal
1-dB Gain Compression Point Referred to Input Signal Level	-40 dBm or greater
Gain Tracking Between Channels (differential)	0.5 dB max
Phase Tracking Between Channels (differential)	2° max
First IF Center Frequency/Bandwidth	152.5/30 MHz
Second IF Center Frequency	30 MHz
• 0.5 dB bandwidth	10 MHz
• 3.0 dB bandwidth	15 MHz
Impedance at All Ports	50 ohms

Overall Noise Figure	12 dB max
Spurious Responses	60 dB min
RF Bandpass Filter Bandwidth	100 MHz, nominal
RF Bandpass Filter Rejection Bands	70 dB min
• dc to 7850 MHz	
• 8450 MHz to 18 GHz	
Prime Channel to Standby Channel Rejection	70 dB min

See Subsection 3.6.6 of the Technical Proposal for more details.

10.3.7.6 Calibration Oscillator Module

The calibration oscillator provides a low level test signal at 8150 MHz (and 8155 MHz) on command from the telemetry system to the input port of the calibration power divider in the coupler/switch module for calibration of the interferometer system net phase error (exclusive of the antennas and their cable assemblies). A crystal oscillator, step-recovery diode, multiplier circuit will be used, with switched xtal oscillators for the two different output frequencies.

Performance Requirements

Output Frequencies, All Outputs	8.150 and 8.155 GHz (sequentially)
Output Power	-30 dBm
Spurious Content, Including Harmonics	at least 60 dB down
Output VSWR	1.1:1
Output Impedance	50 ohms nominal
Frequency Stability	± 10 ppm over 0 to 50°C
• Long Term	± 2 ppm/month

10.3.7.7 First Local Oscillator Module

The first local oscillator, which provides oscillator power to the first mixer in each channel of the mixer/amplifier module, will consist of a crystal oscillator driving a step-recovery diode multiplier.

Performance Requirements

Output Frequency	8000 MHz
Output Power	8 mW
Output VSWR	1.10 max
Output Impedance	50 ohms nominal

Frequency Stability

- Long Term +10 ppm over 0°C to 50°C
±2 ppm/month
- Short Term 0.1° max rms phase noise
from 60 Hz to 1 kHz

Spurious Content, Including Harmonics at least 60 dB down

See Subsection 3.6.8 of the Technical Proposal for more details.

10.3.7.8 Double Local Oscillator (DLO) Module

The DLO provides local oscillator drive to the 2-second mixers in the mixer/amplifier module. Because the DLO outputs provide a frequency offset of 2 kHz between the two down-converted, phase-bearing signals, this frequency offset must be coherent with the 2-kHz phase reference waveform used in the digital converter. It is also important that the phase relationship between the DLO 2-kHz reference and the DLO output 2-kHz offset remain constant (within ±1°) over the temperature range.

The DLO will consist of a fixed crystal oscillator, a voltage-controlled crystal oscillator, and a phase-locked loop.

Performance Requirements

Number of Outputs	2
Output Frequencies	
● f1 (port no. 1)	122.50 MHz
● f2 (port no. 2)	122.502 MHz (f1 + reference signal frequency)
Output Waveform, Both Ports	Sine wave
Output Power, Each Port	10 mW
Power Outputs Matched to Within	1/2 dB
Isolation Between Output Ports	60 dB min
Output VSWR	1.10, max
Output Impedance	50 ohms, nominal
Rejection of All Spurious	60 dB nominal
Frequency Stability	
● Long Term	+10 ppm 0°C to 50°C ±15 ppm/month

• Short Term	0.1° rms max phase noise within 60-Hz to 1000-Hz bandwidth
Reference Signal Characteristics	2 kHz square wave, 0 to 5V
Telemetry Input	1 bit to indicate PLL locked or unlocked

See Subsection 3.6.9 of the Technical Proposal for more details.

10.3.7.9 IF Detector Module

The IF detector module consists of two nearly identical and parallel chains of circuits. The only difference between the two is that one channel input is tuned to 27.5 MHz and the other to 32.5 MHz. Each channel is designed to receive a pair of CW signals (spaced 2 kHz apart) from the combiner module and to process this pair of signals so that the output is a 2-kHz sine wave.

The diplexer portion of this module consists of two 4-section, LC, m-derived, bandpass filters. Each IF amplifier consists of six stages of conventionally tuned transistor amplifier sections. The phase-stable AGC is provided by three PIN diode attenuators interposed between the first three amplifier sections. The detector is a biased semiconductor diode. The 2-kHz bandpass filter is an LC design, and the 2-kHz amplifier is a conventional feedback, stabilized transistor design. The zero crossing detector is an operational amplifier with diode feedback.

Performance Characteristics

Input Characteristics

• Signal Frequencies	27.5, 27.502, 32.5, and 32.503 MHz
(a) Signal Level	-54 to -24 dBm
(b) Impedance	50 ohms, nominal
(c) VSWR	1.3 max

Output Characteristics at Each Port

• Frequency	2 kHz
• Level	1.2V (p-p) into approximately 1000 ohms over the input signal dynamic range
• Impedance	1000 ohms, nominal
• Rise and Fall Time (when pulsing IF input between max and min levels)	< 14 ms (combination of AGC response time and 2-kHz filter settling time)

● Phase Stability	$\pm 1^\circ$ max change in insertion phase shift over the input signal dynamic range
AGC Response Time	1 ms max
Noise Figure	12 dB
● Bandwidths (at -3 dB)	
(a) Predetection	1 MHz
(b) Post-Detection	750 Hz
● Detector Type	Square law
● Spurious Rejection at 2-kHz output (including harmonics)	60 dB min

See Subsection 3.6.10 of the Technical Proposal for more details.

10.3.7.10 Digital Converter and Buffer Modules

10.3.7.10.1 Performance Requirements

The digital converter will:

- Provide a pair of zero crossing detectors which shall shape the sine wave amplifier output from the IF/detector module into a square wave.
- Digitally phase compare two 2-kHz signals. One of these signals is generated within the digital converter as a reference; the second is obtained from the IF detector module.
- Average the 2-kHz vernier signals over 64 periods to minimize gating errors.
- Detect phase readings near discontinuity and correct averaging of counts.
- Generate a binary output word, normally composed of 60 bits.
- Change mode of operation upon command for either 2-station mode, the 1-station mode, baseline calibration mode, or the calibration injection mode.
- Generate the proper sequencing signals to trigger the RF switches in the coupler/switch module in the various modes of operation.
- Supply a 2-kHz signal coherent with the counter clock to the double local oscillator.

- Switch to a redundant digital converter upon command should a converter failure occur. For all but the single station mode, the digital output is sent to the DACU of the Telemetry Subsystem via the buffer module. The 60-bit output word consists of four phase measurements plus four identification bits in serial form every three seconds. Each phase measurement consists of a 9-bit vernier word and a 5-bit coarse word. There are one vernier and one coarse word for each baseline and for each operating frequency. One of the identification bits identifies frequency, and three identify mode of operation.
- In the single station mode, upon interrogation by the DOC, either of two measurements at one of the frequencies can be provided serially to the DOC. The fifth or least significant bit of coarse data is dropped from the serial data train to the DOC, and the data is clocked out at a 63-kbs rate.

Figure 10.3-9 is a functional diagram of the digital converter and buffer.

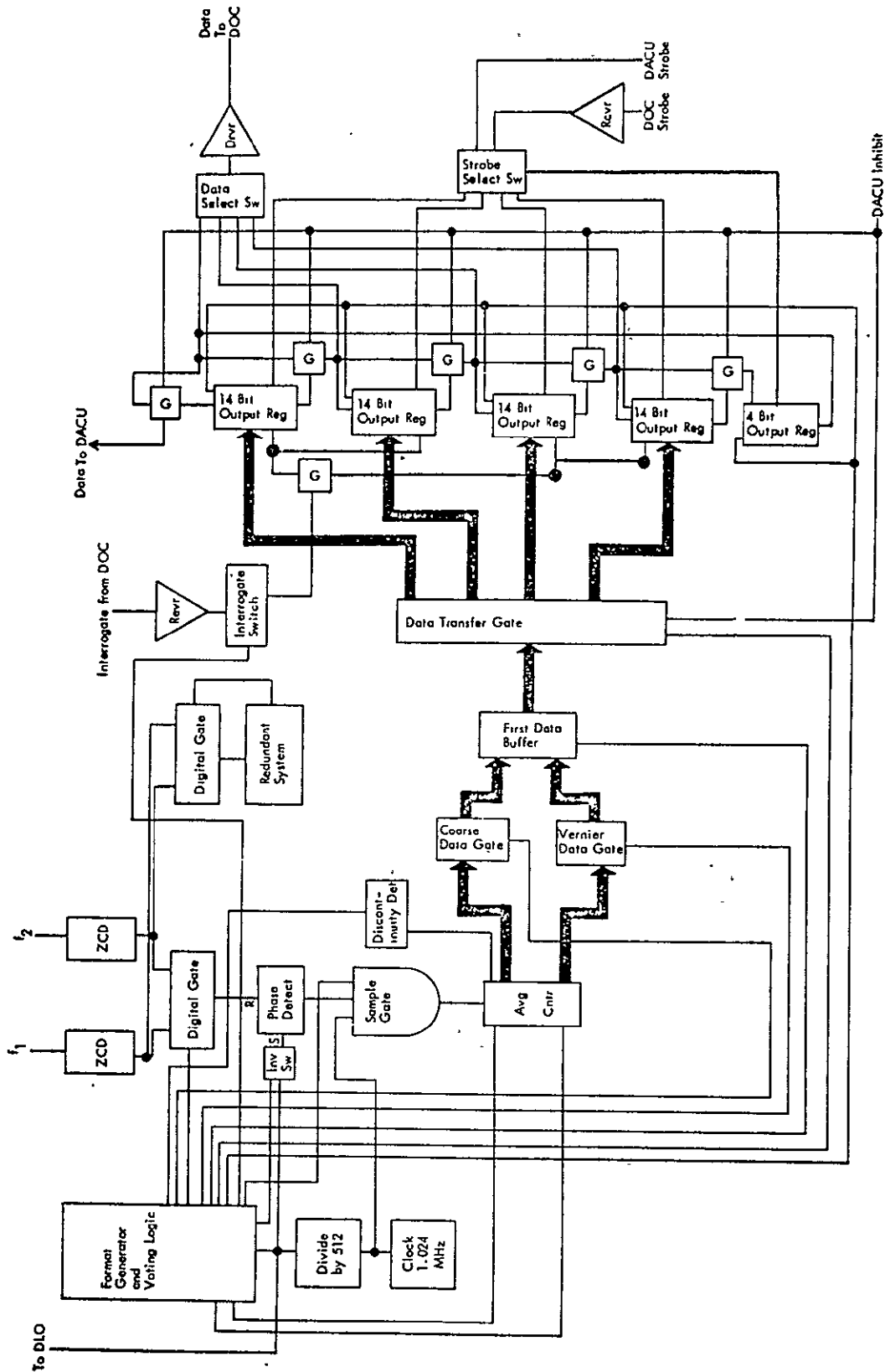
The buffer module shall provide the temporary data buffering for the digital converter data.

Because the digital converter and the units with which it interfaces are asynchronous, data cannot be read directly from the phase data buffer. Once the phase data buffer has completed loading, the data is ready for parallel transfer to the output buffer. Data from the phase data buffer will be transferred via the data transfer gate once loading is complete if neither the DACU nor the DOC are in the process of reading the output buffer. Should the DACU or the DOC be reading, the unit making the readout will provide an inhibit pulse to the data transfer gate for the duration of the readout. After its loading into the phase data buffer is completed, data is retained for a short period during which it may be transferred. At the end of this period, the phase data buffer is cleared, and updated data is sequentially transferred into the buffer. See Subsection 3.6.11 of the Technical Proposal for more details.

10.3.7.11 Gate Module

To fulfill the functional redundancy requirement for the digital converter, it is necessary to or the switching and reference signals from the prime digital converter with the same switching and reference signals from the standby redundant digital converter. The gate module performs this function. Normally the standby redundant converter is powered-down so that it does not drive any signal into the or gate. If the prime converter is powered-down for some reason, the standby converter is powered-up, and it supplies all of the required switching waveforms and the reference signal. Specifically, the following switching and reference signals are supplied:

- 2-kHz reference To DLO A276A1 (prime) and DLO A276A1 (standby)
- Three switching functions To Coupler/Switch Module A276A1
 - One to Roll and Pitch RF Switches
 - One to Pitch/Roll and Reference RF Switches
 - One to Pitch/Roll RF Switch (diagonal measurement)



One switching
function

To Calibrate Oscillator A276A1
(prime) and Calibrate Oscillator
A276A1 (standby).

See Subsection 3.6.12 of the Technical Proposal for more details.

10.3.7.12 Power Regulator Module

The power regulator, which supplies regulated voltages to the various modules, consists of parallel redundant commandable regulators, dc/dc converters, voltage regulators, and switching circuits.

Performance Requirements

Inputs

Power Bus	28 to 33V from each of the two power buses
Command Subsystem (A232, A236)	5V peak, 50-ms pulse on each of two lines (two wires from each CDD) for each command

Outputs

To A272	+12V, -12V, +5, and -5 (and common) to simplex and operationally redundant components, with load regulation at least 1%
	+5V switched upon command to primary or standby digital converter
To A276A1	+12V, -12V, +5V, and -5V (and common) with load regulation at least 1%
To Telemetry (A201, A202)	One bit indicating interferometer power ON
	One bit indicating primary or standby digital converter operational

See Subsection 3.6.13 of the Technical Proposal for more details.

10.3.8 INTERFACE DESCRIPTION

To accomplish its mission objectives, the Interferometer Subsystem must interface with other subsystems in the spacecraft, namely, Power, Telemetry and Command, and Attitude Control.

10.3.8.1 Power Subsystem Interface

The interferometer will receive the unregulated prime power originating in the Power Subsystem from the two spacecraft power buses. The power

regulation section of the Interferometer Subsystem shall satisfy the following design requirements:

- On/Off Requirements—On/Off selection will be supplied external to the equipment by command to the power regulator.
- Redundant Input Power—The input power connector will provide four pins to accommodate redundant input prime electrical power, and the redundant input power buses will be isolated within the equipment.
- Power-On Surge Current, Power Bus Transients, and Fault Protection—The power regulator will furnish these protections in order to maintain the Interferometer Subsystem operational requirements.
- The power regulator will exhibit high efficiency and minimum power drain from the spacecraft power buses during the on and off conditions, respectively.

10.3.8.2 Telemetry Subsystem Interface

For digital phase, angle data output, the digital converter supplies the DACU with 60 bits of serial data in the following format:

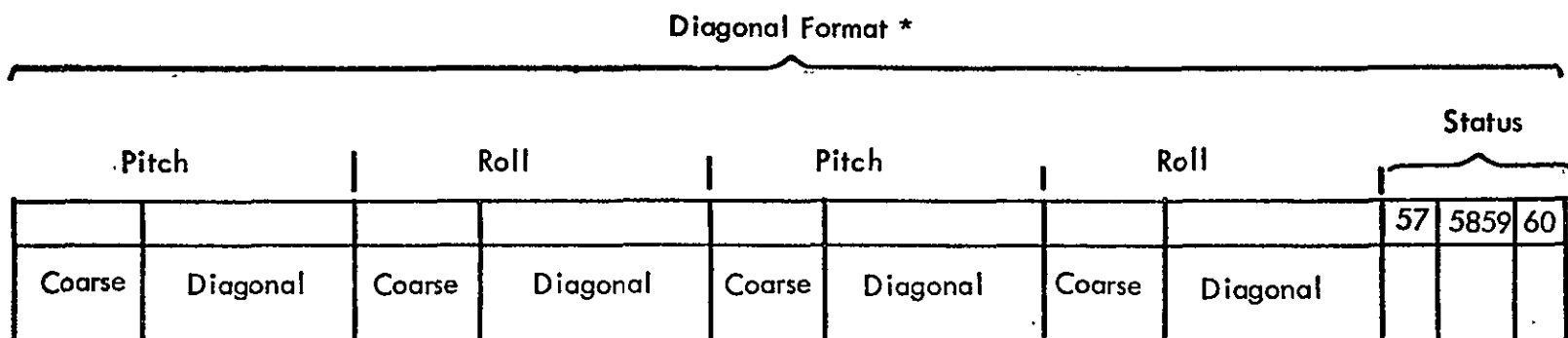
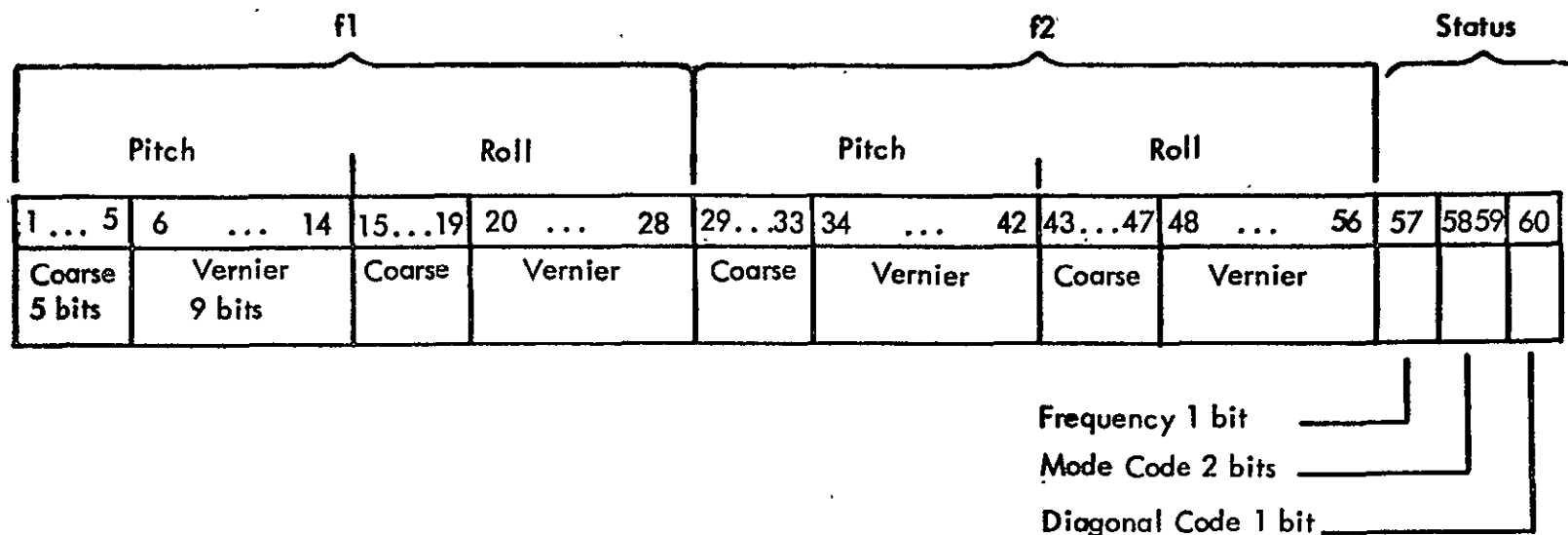
4 mode bits	1 frequency identification 2 mode identification 1 diagonal reading identification
9 bits of vernier data	at f_1 roll at f_1 pitch at f_2 roll at f_2 pitch
5 bits of coarse data	at f_1 roll at f_1 pitch at f_2 roll at f_2 pitch.

The format of the interferometer output record to the DACU is shown in Figure 10.3-10.

The DACU supplies the digital converter with a hold pulse for the duration of data transfer to the DACU. The DACU also supplies a data strobe with which the data is clocked out of the output register in the converter.

Other telemetry requirements of the interferometer are

- Sensor monitoring (temperature and strain gauge)
- Periodic monitoring of AGC voltages in the receiver



Notes: Two - Station Mode: Above is a complete measurement
 Four - Station Mode: Above is 1/2 of a complete measurement

- * The diagonal measurement is commandable and is used only for ground calibration. Pitch and Roll have no meaning in the ordinary sense; the designations "pitch" and "roll" on the format only indicate the sets of antennas being used. Coarse data cannot be used to resolve ambiguities for this measurement.

Figure 10.3-10. Interferometer Telemetry Output Word

- Indication of which digital converter channel is operational and which front end receiver channel is operational
- Indication of status of phase-locked loop in the double local oscillator of the receiver
- Indication of prime power ON or OFF
- Indication of interferometer operational mode.

10.3.8.3

Command Subsystem Interface

The command requirements of the interferometer are:

- Subsystem power ON, OFF (command the redundant regulators separately)
- Calibration system turn ON, turn OFF
- Selection of prime or redundant front end
- Selection of prime or redundant digital converter
- Selection of 1- or 2- station mode
- Selection of mode of coarse channel: 4- or 9-bit readout
- Array transducer circuits turn ON/OFF
- Calibrate array transducers.

10.3.8.4

Digital Operation Controller Interface

The digital converter supplies the DOC with 26-bit words from either the f_1 half or the f_2 half of the output register. The 26-bit words are locked out serially by the DOC strobe. An interrogate pulse from the DOC determines which of the two 26-bit words is required by the DOC. The interrogate pulse will remain up for the duration of the reading. When transferring data to the DOC, the fifth bit of coarse data is eliminated to simplify processing in the DOC.

10.3.8.5

Electromagnetic Interference (EMI) Interface

The basic requirements for the electromagnetic interference interface are that the interferometer receiver performance not be degraded by signals from other spacecraft systems and that it not degrade other systems by signals that it generates.

An analysis was made of the electromagnetic signal environment on the spacecraft. The three most likely modes of interference with the Interferometer Subsystem are:

- Case A—Signals from the spacecraft transmitters mixing in some nonlinear resistance external to the antennas and producing a signal acceptable to the interferometer receiver.
- Case B—Transmitter signals penetrating the receiver front end and mixing in the receiver mixers that produce the receiver IF. The effect on the receiver is the same as the effect from externally generated interferences.
- Case C—Signals from transmitters operating near the first IF frequency penetrating directly into the first IF amplifier.

A computer program was used to evaluate the first two possibilities. It is divided into two major portions: determination of all possible intermodulation products by (1) comparing all transmitter fundamental frequencies taken a selected number at a time with receiver passbands to determine possible external interferences and (2) calculating possible interferences from generation of the intermediate frequency in the receiver mixers.

The results of the analysis for case A (externally generated interferences) showed that there are no interference combinations that exceed the interferometer minimum sensitivity. Since there are no externally generated interferences due to mixing, there will also be no internally generated interferences (case B).

For case C (direct penetration of signals into the first IF), the potential interferences from the TLM transmitters at 136.23 MHz and 137.11 MHz (two watts each) and from beacon 2 at 140.056 MHz (0.25W) were examined. The evaluation consisted of calculating receiver susceptibility to the specific frequencies and estimating the magnitude of EM energy coupled into the receiver by the various paths. Calculation of the receiver susceptibility showed that the telemetry transmitter and beacon signals can produce second- and fourth-order interference signals if allowed to exceed -80 dBm at the first IF amplifier input. Further calculations, however, indicate that the antenna-to-antenna radiated signals will be more than 40 dB below this value, and inspection of the RFI specifications on radiated and conducted emission indicate that these two modes of coupling should present no problem and should lend themselves to effective remedial action, such as shielding and filtering, respectively. In summary, the telemetry transmitters and beacon should present no serious RFI problem to the interferometer. See Subsection 3.7.6 of the Technical Proposal for more details.

10.3.9 GROUND TRANSMITTER STATION

Table 10.3-3 summarizes the ground transmitter requirements and Figure 10.3-11 shows the functional configuration of the ground transmitter. The synchronization capability (time, not frequency synchronization) is required between ground transmitters for an interferometer mode of operation which requires both ground illuminators to transmit at the same frequency. (This is required only upon

Table 10.3-3. Summary of Ground Transmitter Capability Requirements

Frequency of CW Transmission	8150 and 8155 MHz (not simultaneously)
ERP	+76 dBW min +100 dBW max
Antenna Polarization	Linear, rotatable
Spurious Rejection	60 dB, min
Frequency Stability	
Long Term	± 10 ppm max drift
Short Term	0.1° max rms phase noise in 60- to 1000-Hz bandwidth
Accuracy in Time of Transmission Relative to Other Ground Transmitters	1 msec.

failure of one of the operationally redundant IF channels in the interferometer receiver.) The required accuracy of transmission time synchronization is about 1 msec. Except for the synchronization capability, the blocks in Figure 10.3-11 are standard ground station equipment.

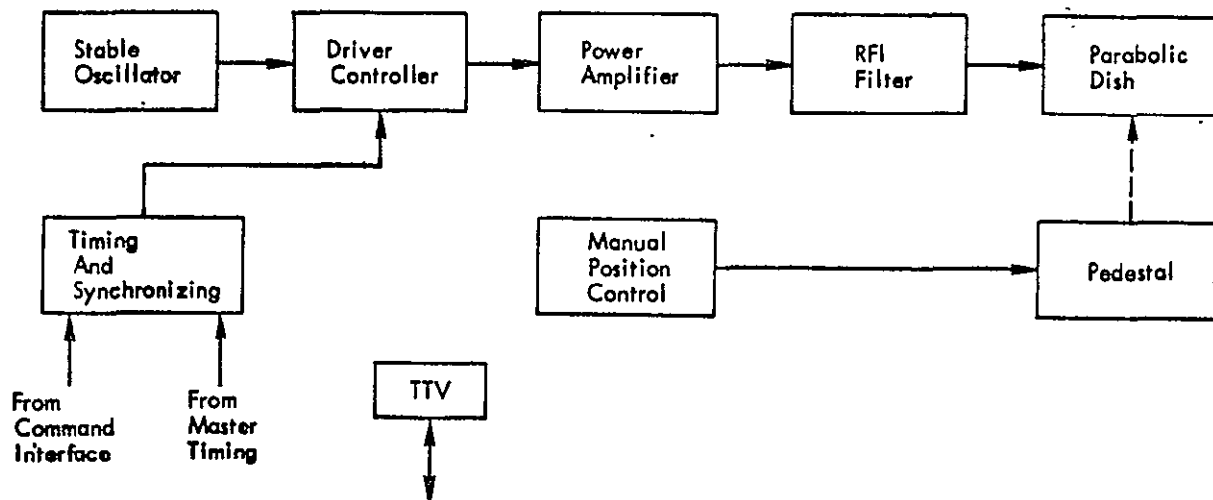


Figure 10.3-11. Interferometer Ground Transmitter

Present information indicates that the modifications planned for Rosman, Mojave, and the transportable ground terminals in support of ATS-F&G will satisfy all requirements for illumination of the interferometer from the U.S., since the frequency of the transmitters can now be tuned in 10-kHz steps. Changing from 8.150 to 8.155 GHz can readily be accomplished. It is expected that the backup mode time synchronization requirement can be met by using the station time code generator which is synchronized to WWV.

10.3.10 GROUND DATA PROCESSOR

The ground data processor interfaces with the following ground systems:

- Command Systems
- ATSOCC
- Interferometer illuminator stations.

Figure 10.3-12 shows these interfaces. The ground processor is identified as the on-site computation, command, and control subsystem. All heavy lines between blocks may be considered hardwired, while lighter lines may be voice or TWX. The heavy lines denote signals or controls which must transmit data on a real-time basis (transmission delays less than a few seconds). The lighter lines denote signals or controls which do not require real-time transmissions.

The following paragraphs briefly describe the total system operation. Starting at the spacecraft interferometer, illuminations from two stations are received. These illuminations cause all eight channels to be filled as follows: two baselines, two vernier counts, two coarse counts for two stations. Diagonal baseline counts are also periodically substituted for the vernier counts, by command.

The counts are then transmitted to the ground station via the telemetry link, where they are demodulated and put into eight parallel channels in the data reformatter. The data reformatter has several functions; the most important are removal of ambiguities and tagging of data with the proper baseline and illuminator code. The data leaving the reformatter now consists of only four counts (i.e., two baselines and two illuminators). The data is properly tagged to identify the counts with the baseline and illuminator.

The purpose of the next block, a data editor, is to remove wild data points and to indicate possible interferometer, telemetry, or data reformatter problems. The data editor takes the difference between the corrected measured count and the predicted count. (The origin of the predicted count will be discussed shortly.) The editor removes wild data (data exceeding 3σ) and replaces it with the predicted data, as long as the number of wild data points does not exceed a certain percentage (e.g., 10 percent of the data during a 10-second span). This percentage is displayed to the control center. If the control center believes the interferometer data is bad, the commands generated by the interferometer can be inhibited.

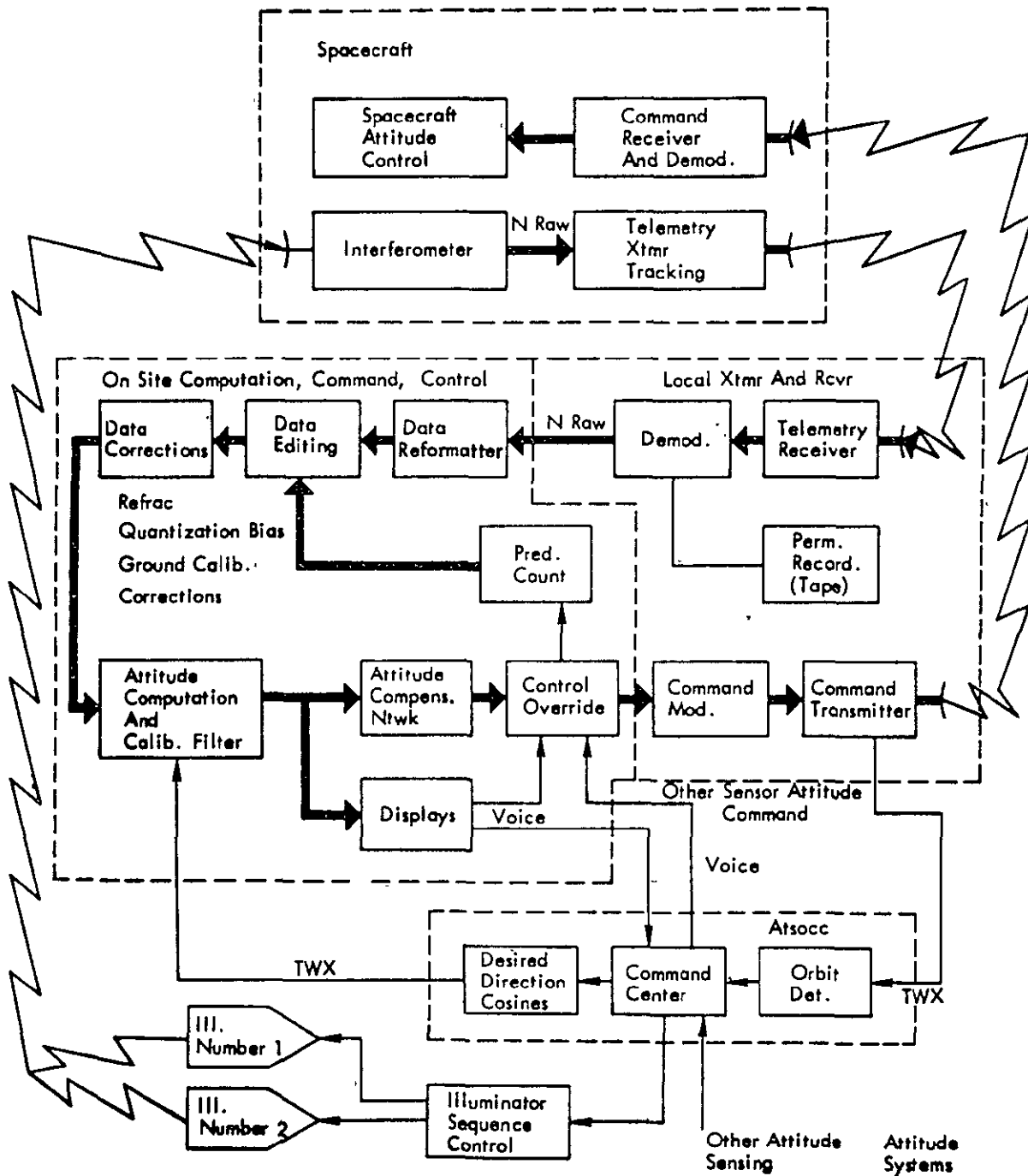


Figure 10.3-12. System Block Diagram

After editing, the data is corrected for the known biases as measured by on-board calibration. It is then passed on to the altitude computations. A recursive least squares implementation is recommended. This approach has the following desirable features:

- The good data can be processed one reading at a time. Thus, wild data points edited out do not cause any interruptions in the generation of altitude commands. The identifying tags on the good data are used to choose the proper i 's and j 's.
- The data sequence is of no concern to the attitude solution; the controller is free to use any sequence at any time.
- The filtering provides a memory with a built-in weighting, which exponentially, for example, decreases with time. Thus if there is no data for a long period of time, the memory confidence decreases. The filter automatically produces a confidence level on its outputs; this level can be displayed to the controller.
- The filtering makes maximum use of redundant data and can be used to estimate certain systematic errors, such as baseline length, baseline orthogonality, and RF phase errors. These estimates and their confidence levels may also be displayed.

The desired direction cosines are entered manually (e.g., punched cards) after their transcription from a TWX sent from ATSOCC.

The control override is a switch with which the controller can switch to any other attitude sensors for generating attitude commands. This does not necessarily prevent the interferometer readings from being processed. In fact, the control can use another attitude sensor to predict what the interferometer counts should be. When the data editing shows a sufficiently low rejection rate, the controller can switch the attitude control to the interferometer.

The commanded attitude is now coded, modulated, and transmitted via the command system to the satellite, where it is demodulated, decoded, and fed into the control systems.

The desired direction cosines, as presently envisioned, will be computed at ATSOCC, where the orbit determination is performed. The desired line-of-sight from the satellite to any ground station is nominally a constant. However, the satellite will not remain truly stationary due to small deviations in energy and angular momentum from the desired amounts. If the satellite's relative motion is too great, a less manually dependent link may have to replace the TWX link between ATSOCC and the on-site computation, command, and control site. However, since the relative motion of the satellite with respect to any ground station is very predictable and a rather smooth regular function, the on-site computer may be able to interpolate (extrapolate) the direction cosines to make the TWX link acceptable.

11.0

COMMANDABLE GRAVITY GRADIENT SUBSYSTEM

11.1

FUNCTIONAL DESCRIPTION

The commandable gravity gradient subsystem (COGGS), through its interface with the attitude control subsystem (ACS), is designed to demonstrate an active control capability on the ATS-F spacecraft using the gravity gradient. Previous applications of the gravity gradient for attitude control have been generally passive attitude control applications which are lacking in the pointing accuracies and attitude maneuver capability required for the ATS mission.

The COGGS is an experiment designed to demonstrate attitude control of a spacecraft via active gravity gradient means to a pointing accuracy of 0.1 degree and with a satellite tracking and slew capability. The normal mode of attitude control uses reaction wheels and reaction jets to provide control torques.

The various studies and simulations performed to date have shown that the design and operation of the COGGS are intimately associated with the spacecraft configuration. Integration of the COGGS into the spacecraft ACS has shown that considerable additional equipment complexity will be required if the COGGS is configured to be largely independent of the primary spacecraft ACS. A much more efficient and workable system will result with the COGGS integrated into the ACS, making use of its digital operational controller (DOC), inertia wheels, and its actuation and control paths.

The COGGS will establish the performance of an active gravity gradient using a single boom attached to a two-axis gimballed torquer. The boom remains in a retracted position until the performance of COGGS is to be evaluated.

The extended boom (Figure 11.1-1) provides a source of reaction torque for roll/pitch attitude maneuvers and a source of external torque to prevent momentum saturation. In the roll and pitch axes, satellite attitude control is achieved by torquing against the inertia of the boom. The average position of the boom in the gravity gradient field should be such that there is no momentum buildup on the spacecraft. The basic function of the momentum wheels in the COGGS control mode is to damp the oscillations of the boom in the gravity field and to provide yaw control.

The auxiliary propulsion subsystem (APS) will be deenergized during this mode of operation.

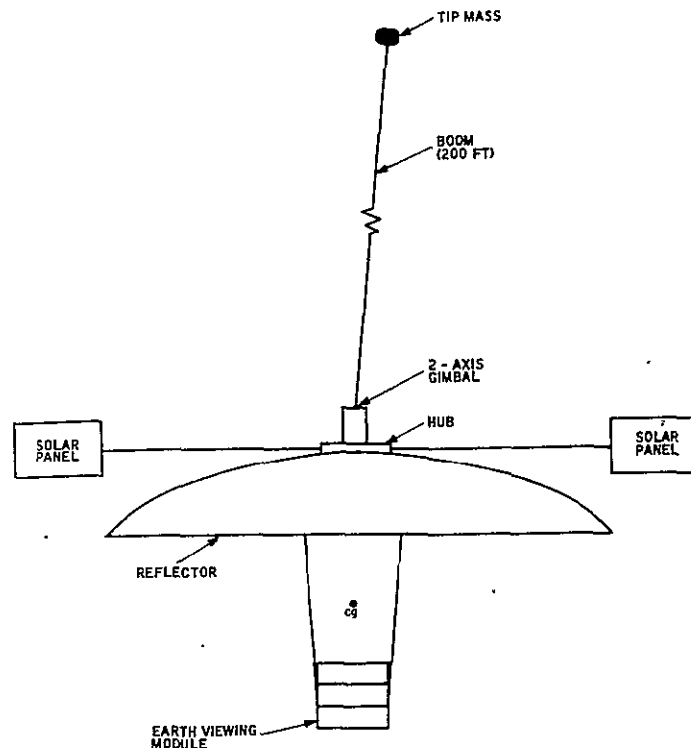


Figure 11.1-1. Boom Spacecraft Configuration

11.1.1 FUNCTIONAL PERFORMANCE REQUIREMENTS

Table 11.1-1 summarizes the basic COGGS requirements and corresponding response to the requirements as detailed by GSFC Specification No. S-732-P-3 and as modified by United States Government Memorandum No. 732-403 dated 17 October 1968.

In Table 11.1-1 any differences between the indicated performance and the requirement are attributed to:

- 1) Revision of the spacecraft configuration
- 2) Simplifications in the COGGS and the interface
- 3) Use of space-proven components

11.1.2 FUNCTIONAL OPERATION AND PERFORMANCE

11.1.2.1 Operation

The COGGS is a combination of active control techniques coupled with the passive gravity gradient technique. It uses active attitude sensors in the control loop to provide high accuracy. The actual torquing effort for slewing and pointing is supplied by a single gimballed gravity gradient boom. The boom is gimballed in roll and pitch to provide torques in these two axes. Control of the yaw axis is maintained on the inertia wheel normally used.

Table 11.1-1. Basic COGGS Requirements and Corresponding Response

Parameter	Requirement	Performance
Roll and pitch attitude control		
a) Static pointing accuracy	± 0.1 degree	± 0.1 degrees
b) Slewing maneuver		
• Angular change	17 degrees	17 degrees
• Settling time to 0.1 degree	Less than 30 minutes after maneuver is initiated	Less than 27 minutes
c) Tracking maneuver		
• Inputs	$10^\circ \sin(0.07t)$, t in minutes 1 degree/minute ramp	$10^\circ \sin(0.07t)$ 1°/min
• Maximum error	Less than 0.5 degree	Less than 0.4 degree
• Settling time to 0.1 degree	10 minutes after termination of command	Less than 10 minutes
Optical digital shaft encoder	13 bit, ± 45 degree range	13 bit, ± 60 degree range
Torquer	D'Arsonval torquer 6 oz-in. each axis	Brushless DC torquer 6 oz-in. roll, 3.5 oz-in. pitch
	Less than 3 watts	1.4 watts
Weight		
• Mechanical assembly	30 pounds	35.7 pounds
• Electronics assembly	5 pounds	3.3 pounds
Power	Average - 4 watts Peak - 10 watts	5.5 watts 15.2 watts
Experiment electronics	- Determine angular rate and position - Accept digital roll and pitch error signals - Generate u_1 and supply voltage to gimbal torquers - Generate u_2 and supply control voltage to roll and pitch wheels - Provide telemetry and command interface	Determined by DOC Accepts torquer commands from DOC. Digital roll and pitch error supplied to DOC u_1 generated in DOC u_2 generated in DOC and supplied directly to the wheels Discretes provided by electronics, gimbal angle provided by DOC
ATS digital controller	- Determines angular attitude error ($\theta_e = \theta_c - \theta$) - Determines gimbal command ($\beta_c = -\theta_e$)	DOC determines DOC determines
Deployment	Design for delayed deployment (1 year)	Delayed deployment
Separation	A means of separating the boom from the spacecraft shall be provided	Explosive bolt employed
Deployer	A tip mass deployer shall be used	Tip mass deployer is used
Volume	18 inches (diam.) by 12 inches (height)	14 inches (diam.) by 18 inches (height)

The COGGS is designed to operate in conjunction with the components of the ACS. These components include the earth sensor, Polaris sensor, digital operational controller, the actuator control electronics for the inertia wheels, and the inertia wheels. The COGGS consists of a deployable boom and tip mass attached to a two-axis gimbal assembly, and the electronics to drive the torquers. The electronics provide an interface with the gimbal angle encoder and other circuitry for telemetry and command. A block diagram of the COGGS is shown in Figure 11.1-2.

The COGGS is in a stowed configuration for most of the ATS mission. The deployer is latched, the boom is retracted, the gimbals are caged, and power is removed from the unit. When it is desired to use the COGGS, ground commands will cause the unit to be energized, except for the gimbal torquer. The deployer will be unlatched and the gimbals uncaged. The boom is deployed to full length by the thermal-battery-powered d-c motor within the tip mass. The gimbal servos are then energized by ground command. The appropriate mode command is transmitted to the DOC to alter the program for the gravity gradient mode.

The DOC supplies pulse-width-modulated signals to the gimbal torquers (via the experiment electronics package) in response to commands generated by the control laws. The signal for driving the gimbal torquers is derived from the angular error between the commanded vehicle angle and the vehicle angle as measured by the earth sensor. A lead-lag compensation on the angular error provides the necessary rate stabilization for the control loop. The DOC reads out the roll and pitch gimbal encoders serially and uses the angle information for driving the roll and pitch inertia wheels to damp the boom librations due to the gravity gradient torques on it.

Sensor attitude measurement is compared with the computed angle command and is used to drive the gimbal torquers. Driving the gimbal torquer results in a reaction torque being applied to the spacecraft at the gimbal. This causes the spacecraft attitude to be directed to offset points, to do slew maneuvers or to track satellites. In addition to supplying a reaction torque for changing the attitude of the spacecraft, the gravity gradient torque developed by the boom can be used to oppose the solar radiation torque acting on the spacecraft and thus maintain inertia wheel momentum storage at a low value.

The inertia wheel is the prime control torquer in the yaw axis, and the control law is the same as used for all other regular modes. Yaw wheel momentum storage due to solar torques can be kept at a low value by the orbital rate coupling of the momentum into the roll axis.

As indicated above, the earth sensor supplies roll and pitch attitude error signals. Roll and pitch attitude signals can also be obtained from the interferometer or the monopulse as alternate sources.

When the COGGS experiment is considered complete, the boom will be released from the spacecraft.

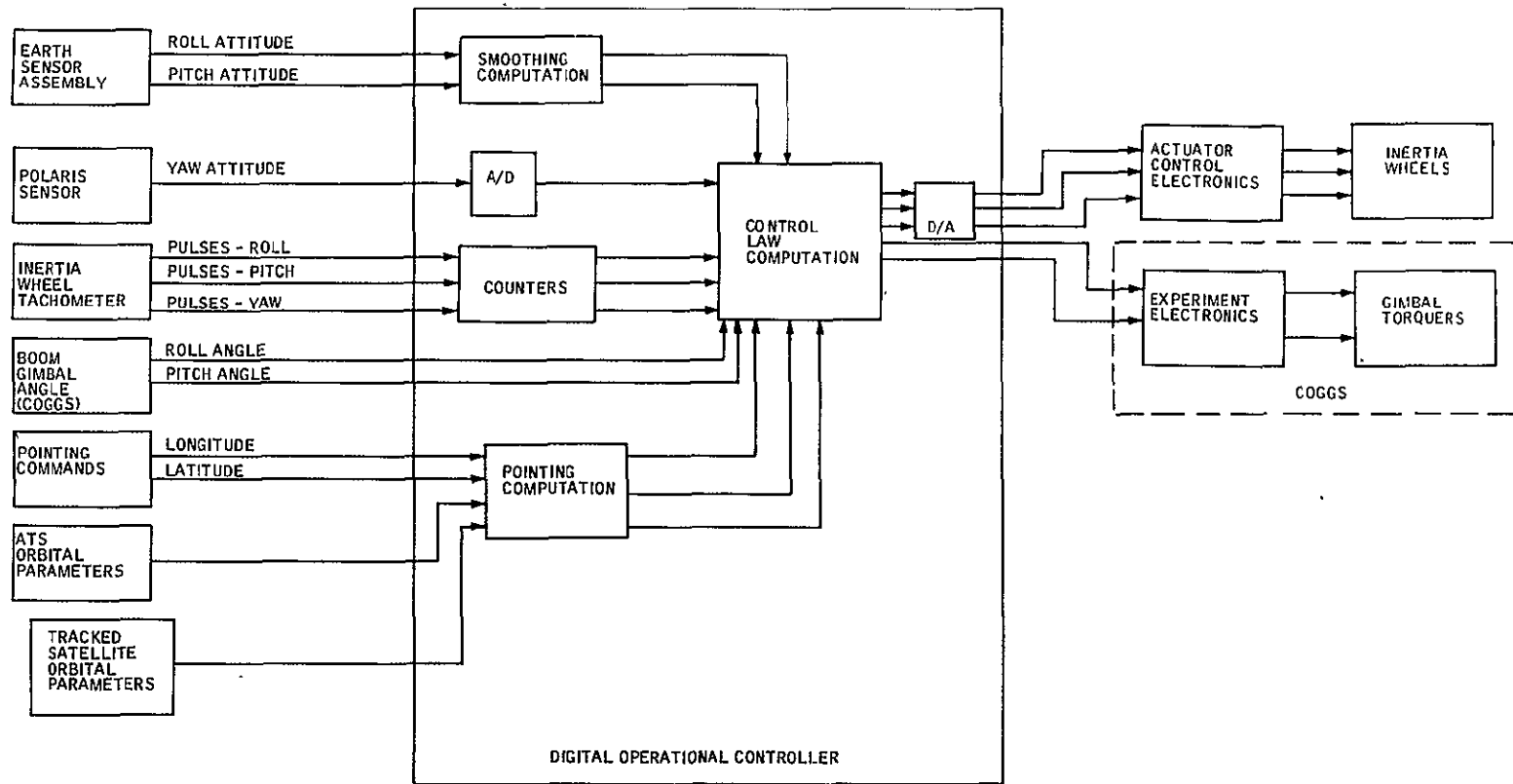


Figure 11.1-2. COGGS/ACS Signal Interface Functional Block Diagram

The performance of the COGGS has been demonstrated by a simulation to verify that the COGGS as configured will achieve vehicle control performance and that there is no vibrational interaction between boom vibration modes and the control dynamics.

A complete set of equations for the nonrigid boom/vehicle structure was derived and then simplified and adapted for the special case of the COGGS control system study.

First, the roll and pitch equations were algebraically solved such that the gimbal encoder and earth sensor outputs appear in explicit form in terms of the control torques and forces and vibrational deflections of the boom.

Second, the simplified equations were programmed on an analog computer to determine vibrational mode frequencies and relative amplitudes of the vibrational modes.

The results of the analog simulation show that the first vibrational mode frequency on the boom is at 0.0626 rad/sec for roll and pitch. The second vibrational mode frequency was determined to be 0.374 rad/sec. The third vibrational mode frequency was determined to be 0.56 rad/sec.

A close scrutiny of the vibrational equations shows that all odd or all even modes have strong mutual coupling. Relatively weak coupling exists between the even and odd modes. However, since vibrational amplitudes of the higher frequency modes are relatively small, the first mode uncoupled frequency does not change noticeably due to higher frequency odd modes. Consequently, for the COGGS control stability analysis and preliminary performance analysis, the boom model with only first vibrational mode modeled is sufficient.

Computer runs were made of response to a step input of 17 degrees (edge-to-edge slew maneuver), satellite tracking maneuver, and a 1-deg/min slew maneuver.

Figure 11.1-3 shows the pitch and roll control response to 17-degree attitude step in pitch and roll. Both attitude errors are reduced to less than 0.1 degree in less than 8 minutes. It should be noted that the amplitude of the boom vibrations is relatively small. Roll (ϕ_E) and pitch (θ_E) attitude errors, roll (ϕ_C) and pitch (θ_C) commands, and roll gimbal output (β_X) are the plotted computer outputs on the figure.

Figure 11.1-4 shows the control performance for simultaneous satellite tracking in pitch and roll. The attitude error is oscillatory, with a maximum amplitude less than 0.4 degree for 10-degree commanded amplitude.

Figure 11.1-5 shows roll control response to sawtooth inputs whose slopes are ± 1 -deg/min ramps. Two observations can be made:

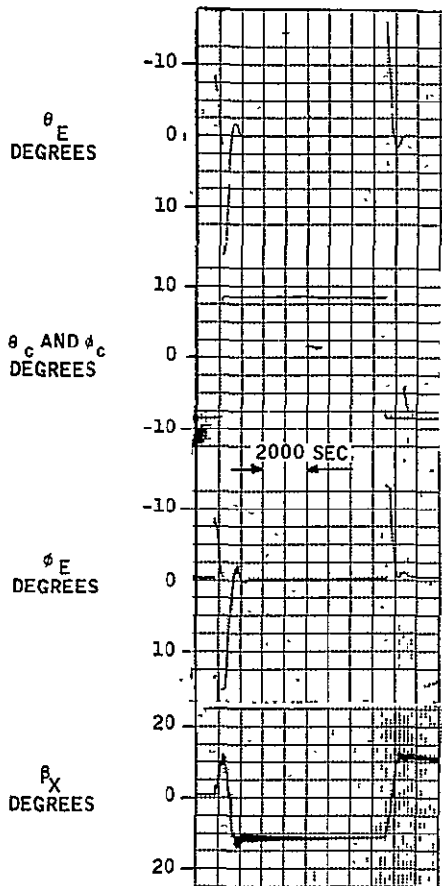


Figure 11.1-3. Pitch and Roll Response to ± 8.5 -Degree Attitude Steps

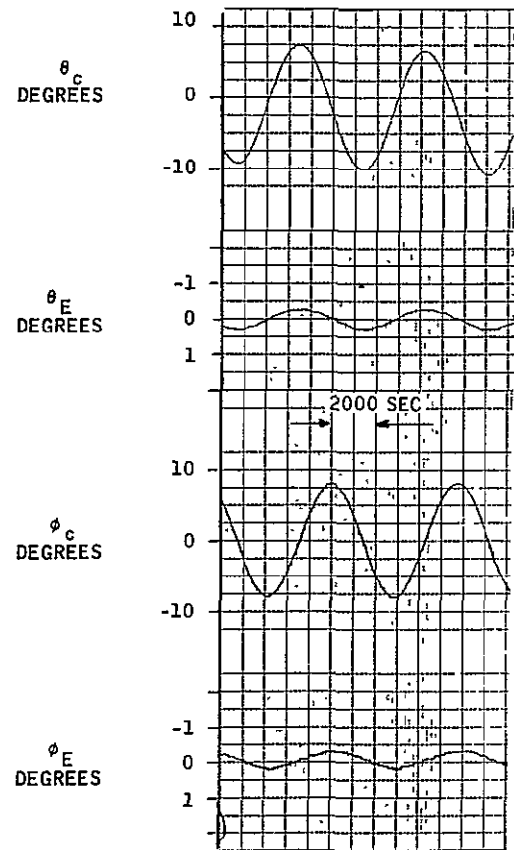


Figure 11.1-4. Pitch and Roll Tracking

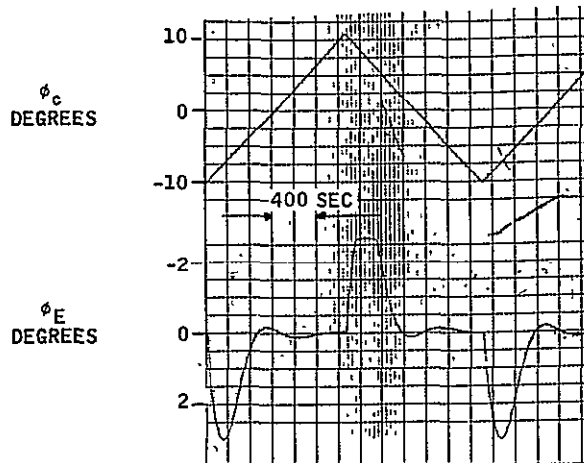


Figure 11.1-5. Roll Response to 1-Deg/Min Ramps

1) the turnaround where attitude error is greater than 0.1 degree lasts about 10 minutes, and 2) attitude error tends to zero as the turnaround transients approach the quiescent state.

11.1.3 INTERFACES

To perform the desired functions, the COGGS must interface with a number of the other subsystems onboard the spacecraft. These interfaces are discussed below.

11.1.3.1 Spacecraft Power Subsystem

The COGGS receives its power for the gimbal readout and torquer electronics directly from the spacecraft power subsystem through a load interface circuit provided in the COGGS.

11.1.3.2 Command and Telemetry

11.1.3.2.1 Command Decoder and Distribution Unit (CDD)

The COGGS will interface with both of the CDDs to receive the ground commands required to put it into operation. The commands include the following:

- Power on-off commands
- Enable torquers

- Unlatch deployer
- Extend boom
- Release boom

11.1.3.2.2 Data Acquisition and Control Unit (DACU)

The COGGS will interface with both of the DACUs to provide ground monitoring capability of:

- Roll torquer current over 30 percent stall-positive
- Roll torquer current over 40 percent stall-negative
- Pitch torquer current over 40 percent stall-positive
- Pitch torquer current over 40 percent stall- negative
- Power on
- Torquer power on
- Deployer unlatched
- Boom extended
- Boom released

11.1.3.3 Attitude Control Subsystem

The interface with the ACS is to the two DOCs. The signals are:

- Roll gimbal angle
- Pitch gimbal angle
- Roll torquer command
- Pitch torquer command

11.1.3.4 Intra-COGGS

The one significant interface within the COGGS system is transmittal of angle information from and torquer commands to the inner gimbal through the flex leads.

11.2 PHYSICAL DESCRIPTION

11.2.1 COGGS LAYOUT/LOCATION/MOUNTING CHARACTERISTICS

The COGGS is a complete electrical and mechanical assembly that is mounted on a truss above the ATS hub, and interfaces with the DOC,

CDD, and CADU in the earth-viewing module. The COGGS consists of a gimbal/torquer assembly and a boom/deployer assembly. Some of the key design features are listed in Table 11.2-1.

Table 11.2-1. Key Design Features for COGGS

Parameter	Value
<u>Gimbal/torquer assembly</u>	
Weight (with electronics)	18.0 pounds
Size	14.0 x 12.8 x 4.4 inches
Power	5.5 watts average 15.1 watts peak
Configuration	Open construction, base mounted
Torquer	Brushless d-c pulse torqued
Torque level-- roll	6 oz-in.
Torque level-- pitch	3.5 oz-in.
Encoder	Optical, 13-bit serial readout
<u>Boom</u>	
Natural bending frequency	0.06 rad/sec
Diameter	0.57 inch
Thickness	0.002 inch
Material	BeCu
Configuration	Perforated edge lock
Length	160 feet
Weight	2.6 pounds
<u>Tip deployer</u>	
Weight	18.4 pounds
Dimensions	7.5 x 6.7 x 14 inches
Power	Self-contained thermal battery

The COGGS is truss-mounted to avoid contact with the solar paddles and to allow a full ± 60 -degree boom angle of freedom in the deployed condition (see Figure 11.2-1). The mechanical mounting allows for temperature stabilization of the COGGS via heat flow to the vehicle truss. The mechanical mounting also aligns the boom axis to the spacecraft reference axes for proper coupling of the control. The deployer is mounted topside, via the COGGS base ring, to keep the launching forces (acting on the deployer and boom mass) from overstressing the precision gimbal bearings. It is an adaptation of the configuration already used successfully on NASA project RAE/Explorer 38. The mechanical layout of the system is shown in Figure 11.2-2 and the electrical schematic is shown in Figure 11.2-3.

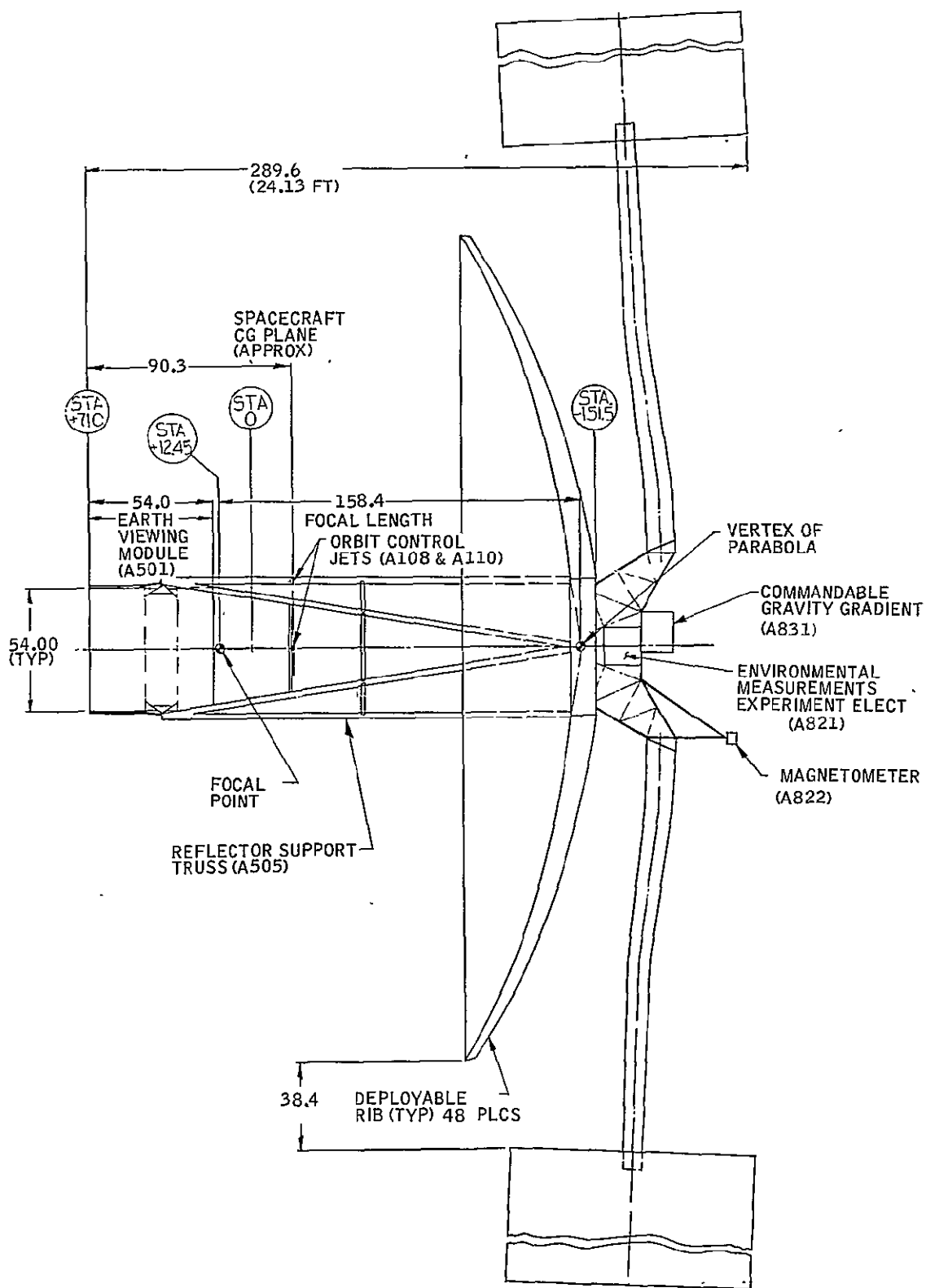


Figure 11.2-1. ATS F&G Deployed Configuration

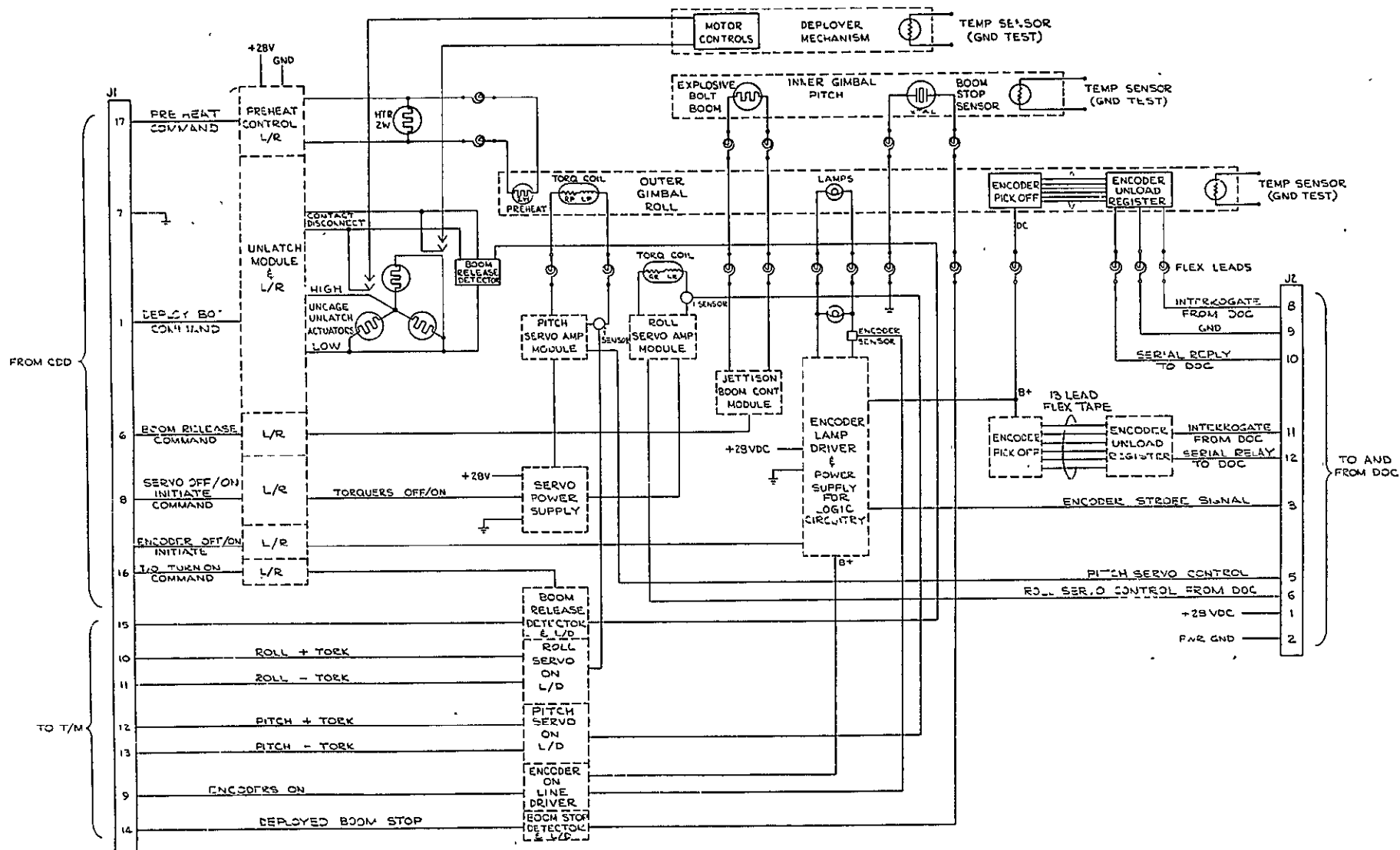


Figure 11.2-3. COGGS System Schematic

The COGGS is characterized by an open-frame construction, with enclosures (called intergimbal assemblies) to protect the encoders and bearings on the gimbal, and an enclosed deployer. The enclosures provide at least partial protection from dust, stray sunlight into the encoders, and a greater thermal stability. The COGGS base ring will be bolted to the truss and will be positioned snug up against an alignment surface. The base ring will have a smooth flat bottom surface for good heat conduction.

COGGS electrical outputs and command inputs will be carried in two cables through two connectors to the COGGS electrical interface. Standardized circuits will be used, and parts with an established acceptance will be selected from the preferred parts lists.

11.2.2 DESIGN CHARACTERISTICS

11.2.2.1 Gimbal/Torquer Assembly

The components on the gimbal/torquer assembly include two bearing systems, an encoder, spiral flex leads, a torquer, and gimbal stop limits. To obtain good gimbal balance and symmetry, these components are clustered into module sets for each end of the gimbal. Since the torque motor and encoders are the heaviest, they logically belong at opposite ends of the gimbal.

An attractive packaging method is to create gimbal component clusters, or intergimbal assemblies. These units are progressively assembled into the inside of a short, thin-shelled, cylindrical barrel which has a high bending moment-to-weight ratio. The construction shown in Figure 11.2-2 features complete module assemblies, progressively rotor-mounted onto a gimbal stub shaft and stator-mounted within a cylindrical external gimbal shell housing. The shell is about 4 inches in diameter and has considerable stiffness (high bending moment) with a wall thickness of as little as 0.060 inch. The shell protects the encoder from stray incident light from the sun in orbit and from room lighting during prelaunch testing. The shell and end cap protect the bearings and encoder optics from dust and contamination. Thermal coatings can be applied to the shell with ease.

11.2.2.1.1 Gimbal Bearings

The selection of bearings for the COGGS gimbal was based on four major requirements:

- 1) The friction level in orbit must be very low (i. e., 10^{-5} lb-ft).
- 2) The gimbal must survive launch acceleration forces and buffetting.
- 3) The gimbal must be prelaunch tested to gain assurance of orbital performance.
- 4) The gimbal prelaunch testing must use the orbital bearings only.

The last requirement is the result of assuming that a potential problem would exist if the orbital bearings were substandard, and their friction problems hidden by other larger forces, torques, or testing factors.

These requirements involve a wide range of somewhat conflicting bearing requirements. With a small increase in friction level, via a larger weight savings, Honeywell bearing specialists have selected a miniature precision ball bearing capable of meeting the application requirements. The optimum bearing has a small pitch diameter, a large number of balls, and the balls have a large diameter. To withstand heavy loading, a large contact area is desirable. This bearing will withstand 200 pounds of radial load during launch, which is 10 pounds per bearing times a 20-g acceleration. This bearing will also meet the 10^{-5} lb-ft friction requirement when the radial force is reduced to less than 0.5 pound per bearing.

The ball-bearing configuration involves the use of the conventional "three-bearing-per-gimbal" configuration. The encoder end is axially restrained, while the torquer end has some freedom to slide with temperature, vibration, and acceleration environments. At gimbal resonance points, this sliding action produces a damping loss and limits structural amplification of the input vibration.

The three-bearing-per-gimbal approach has a second unique feature. The encoder end, with duplex bearings, has half the load per bearing when properly aligned. The friction torque is a function of the radial load to the $4/3$ power. Therefore, halving the load reduces the friction per bearing to less than half of the torque, since $0.5^{4/3} = 0.40$, and the total friction can be 80 percent of that for a single bearing. If proper alignment is achieved, bearing life can also be lengthened.

11.2.2.1.2 Encoder Mounting and Circuitry

The full word, 13-bit encoder is preferred over a serial or incremental encoder. It is anticipated that the encoder will be a complete module which is mounted onto the gimbal as a preassembled and precalibrated unit. The radial bellows-coupled encoder appears to be the best method of coupling. The optical-type encoder has a definite advantage over contact-type encoders in that it does not contribute to the friction. Encoder lamps are available in filament types and gallium arsenide red-light emitters. The gallium arsenide lamp is recommended because it can be strobed and requires less power than the incandescent filament type.

Most of the encoder electronics can be mounted on the heat-sink gimbal base. This includes the programmable power source(s) for the lamps and logic. The one exception is the encoder serializer for the inner gimbal, which must be mounted on the outer gimbal to use a single spiral flex lead instead of 13 spring flex leads on the gimbal axis. The serializer reads out into a temporary memory, such as flip-flops, and then unloads the 13 digits into a single line, using DOC-synchronized (clock) gating. Since the gimbal rotates very slowly and the readout is very fast, the probability of transition error is very small.

11.2.2.1.3 Gimbal Torquer

An attractive approach for the gimbal torquer is the brushless, moving magnet d-c torquer. This torquer is designed for a minimum of bias, hysteresis, and "centering torques". It weighs 1.1 pounds and requires 1.4 watts to obtain 6.0 oz-in of torque. The unit is already designed and tooled, and represents an improvement in weight, power and volume. It must be very accurately centered, however, to avoid applying side forces to the gimbal bearings via magnet pull from unequal leakage fluxes.

11.2.2.1.4 Flex Lead Modules

Flex leads were chosen over such alternate methods as slip rings and ball-and-socket flex joints. Honeywell has confidence in this approach since it so strongly parallels the floated gyro flex lead use. The design uses copper leads, 6 inches long, which are formed into three-turn spiral wraps and mounted in spiral scroll (or "keepers") to limit excursion in vibration.

Ideally, the flex leads exert no force or torque with the gimbal assembled and the encoder at null. This can be accomplished by mechanically manipulating the spiral flex lead "springs" or by special annealing techniques. An adjustable ring is located between the fixed outer case and the fixed rotor-mounted inner terminals. Forces can be nulled by translating the adjustable ring. Torques can be nulled by rotating the adjustable ring.

11.2.2.1.5 Caging and Latching Function

The gimbal must be restrained from rotating off null and/or banging the stops during launch; therefore it is caged. The boom and deployer must be restrained from breaking free of the spacecraft during launch buffeting; therefore, it is latched. It is desirable to keep the deployer weight and g reaction forces off the gimbal bearings; therefore, the deployer is supported from the COGGS base ring rather than through the gimbals.

The gimbal is kept from rotating by holding the boom in a fixed position. This means that gimbal torques are restrained by the boom stiffness, and that stiffness has been increased by using a boom liner stub inside the boom. This stub, at launch, extends inside of the deployer for approximately an inch.

Boom latching is accomplished by fastening the deployer case to the vehicle at launch.

11.2.2.1.6 Electronics

The gimbal electronics are mounted on the gimbal base, providing weight savings in minimizing connector and harness weight. The inner gimbal encoder module(s) must be mounted on the outer gimbal to minimize the number of flex spiral leads and the problems that they create.

The design of the electronics for COGGS is dependent on the packaging configuration. The COGGS will use circuitry that consumes minimum power, and will use the gimbal housing for heat sinking of internal components wherever possible. Because the electronics are packaged in the channel surrounding the gimbal assembly and mounted on the spacecraft hub, it was felt that modular assembly of components using cordwood construction would be more appropriate than printed circuitry. These modules would contain piece parts with natural functions grouped together. The space available for electronics is of odd-shaped volumes, and modules can be designed to fit these volumes. Since the electronics contain LIC circuitry, and high-level torquing currents are developed, the use of discrete components is mandatory.

Components on the inner gimbal are electrically connected to the outer gimbal components via flex leads; thus, the circuitry must be mechanized in such a manner that the fewest flex leads are required. Mechanizing the encoder information from a parallel output readout device to a series readout accomplishes a major reduction in the flex lead count.

11.2.2.1.7 Torquer Drive Amps

The electronics necessary to activate the torque winding of each gimbal are designed to provide the power interface to the DOC pitch and roll error commands, each of which consists of logic levels (1 or 0).

The electronics consist of line receivers coupled to balancing and amplification stages in order to provide sufficient drive to the current bridge network consisting of power transistors and silicon-controlled rectifiers for fast switching.

11.2.2.1.8 Encoder Logic

Integrated logic networks consisting of nand/nor gates and flip-flop circuitry sample the 13-line bits from each of the optical encoders. The strobe signal from the DOC converts the parallel readout to a series readout for use by the DOC signal processing circuits.

11.2.2.1.9 Interface Circuitry

Standard line drivers/line receiver networks are incorporated for interface circuits to the DOCs, CCDs and DACUs.

The load interface circuit will be connected directly to the spacecraft power supply.

11.2.2.1.10 Mode Logic

Circuitry is provided to perform certain functions on receipt of commands from the ground via the CDD. These include the commands listed in Section 11.1.3.2.

The gravity gradient boom for the ATS will be a two-axis gimbaled boom vertically deployed from a canister mounted on the spacecraft hub. Because the deployer must function flawlessly in space, an adaptation of the radio astronomy Explorer-A satellite deployment mechanism has been used. The RAE mechanisms have demonstrated their reliability by successfully deploying four 750-foot-long booms in space.

In operation, the boom tip is attached to the COGGS inner gimbal and the mechanism deployed away from the spacecraft on a one-shot-type deployment. The mechanism in the deployed configuration then becomes the boom tip mass. The deployment mechanism contains all components necessary for mechanically extending itself from the spacecraft.

The COGGS boom/deployer assembly is shown in Figure 11.2-4.

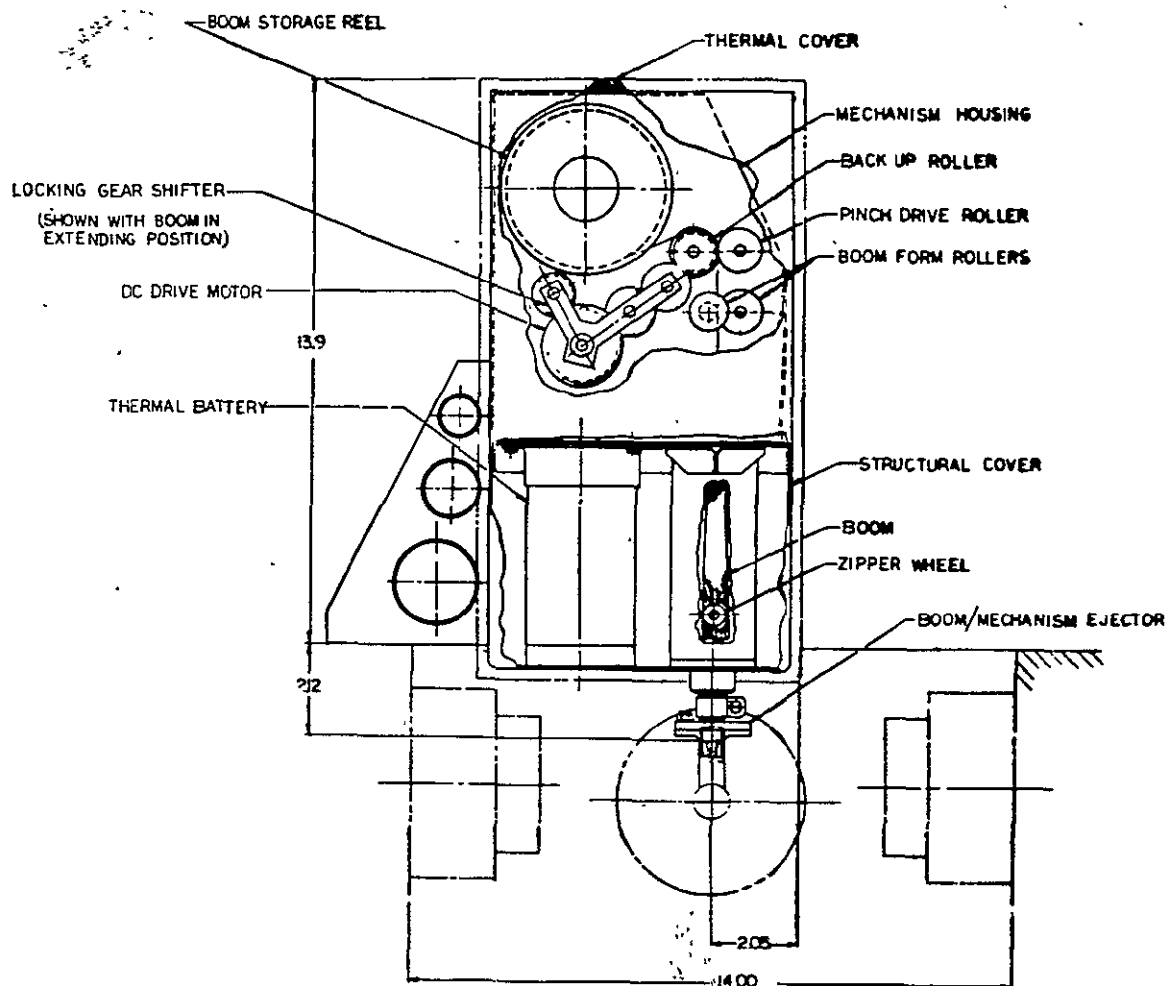


Figure 11.2-4. COGGS Boom/Deployer Assembly

In the deployer design maximum use is made of non-magnetic material and other materials which are especially noted for their ability to perform in the space environment without degradation or malfunction.

The entire deployer mechanism is encased in a thermal insulation blanket. This blanket is made as a sandwich of very thin aluminized Mylar, in sections zippered together by Velcro tape. This sectioning technique allows for last-minute prelaunch access to the deployer mechanism.

11.2.2.2.1 Boom Design

The ATS COGGS boom will be of the RAE edgelock type, approximately 165 feet long. The basic material used in the element is precision rolled beryllium copper. The element is made from 0.002 x 2.00-inch continuous strip which is heat treated to form a 0.570-inch-diameter tube. This technique allows the tube to be wound flat on a storage reel for maximum compactness.

Figure 11.2.5 illustrates the RAE edgelock antenna element prior to heat treating. The locked seam is made possible by punching a series of tabs on the edges of the flat element. Every other tab has its apex flattened, i.e., the distance from the centerline of the element to the edge of the tabs alternates between long and short. When the flattened element springs into a tube, a short tab on one edge of the tube is presented to a long tab on the opposite edge of the tube. The seam is "zippered" by tucking short tabs under long tabs.

The "zipper" (Figure 11.2.6) consists of a narrow crowned roller mounted inside the half-formed tube on the centerline of the seam at a point just before the tabs to be mated touch each other. Opposite tabs move across the roller together as the element is dispensed but are restrained momentarily by the roller from curling into a tubular shape. The shorter tab springs clear of the roller first and curls into its normal position on the centerline of the seam. The longer tab clears the roller last so that when it springs down, it overlaps the shorter tab already on the centerline. The process is reversed on the next pair of tabs. The end result is a row of alternating interlocked tabs.

This edge locking in the deployed configuration provides for greater torsional rigidity and minimizes torsional bending flexibility. Each tab is one inch long in the axial direction. To avoid play between adjacent tabs, an interference fit was achieved through the incorporation of an 11-degree wedge angle.

Ball-bearing-mounted rotating guide rollers are used exclusively to assist the antenna element in making the transition from flat to round. These rotating guides are contoured so as to achieve a smooth transition of the antenna element with no deformations after many cycles. The contoured rollers also provide element bending root fixity. The output end of the guide system contains an adjustable pylon which has the edge locking roller as its tip.

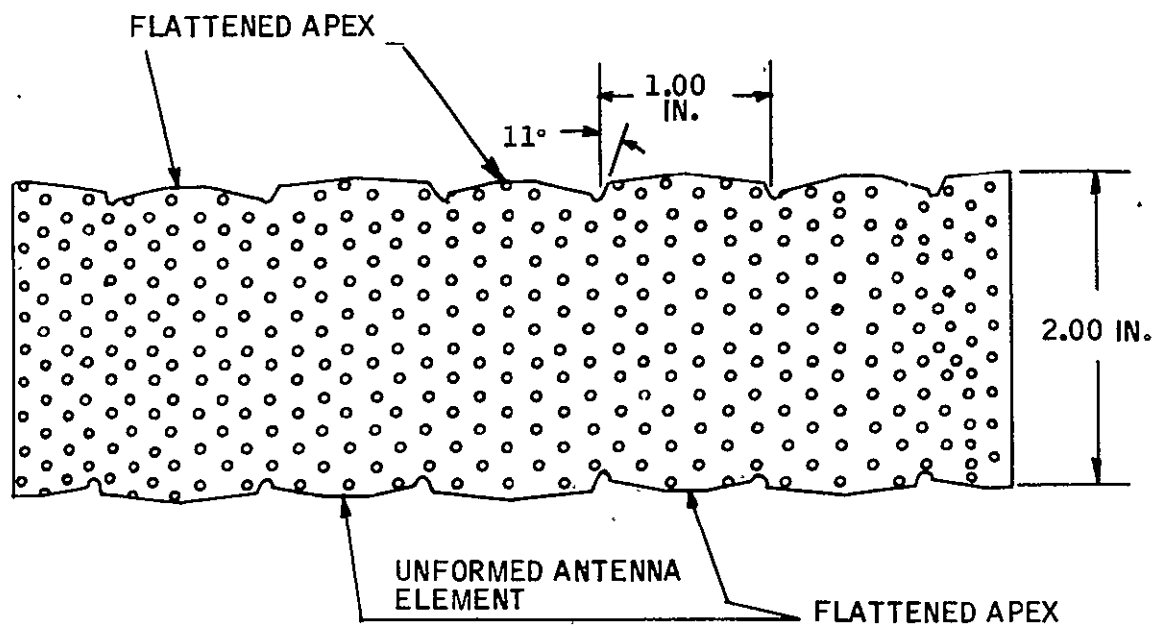


Figure 11.2-5. RAE Edgeloek Antenna Element Prior to Heat Treating

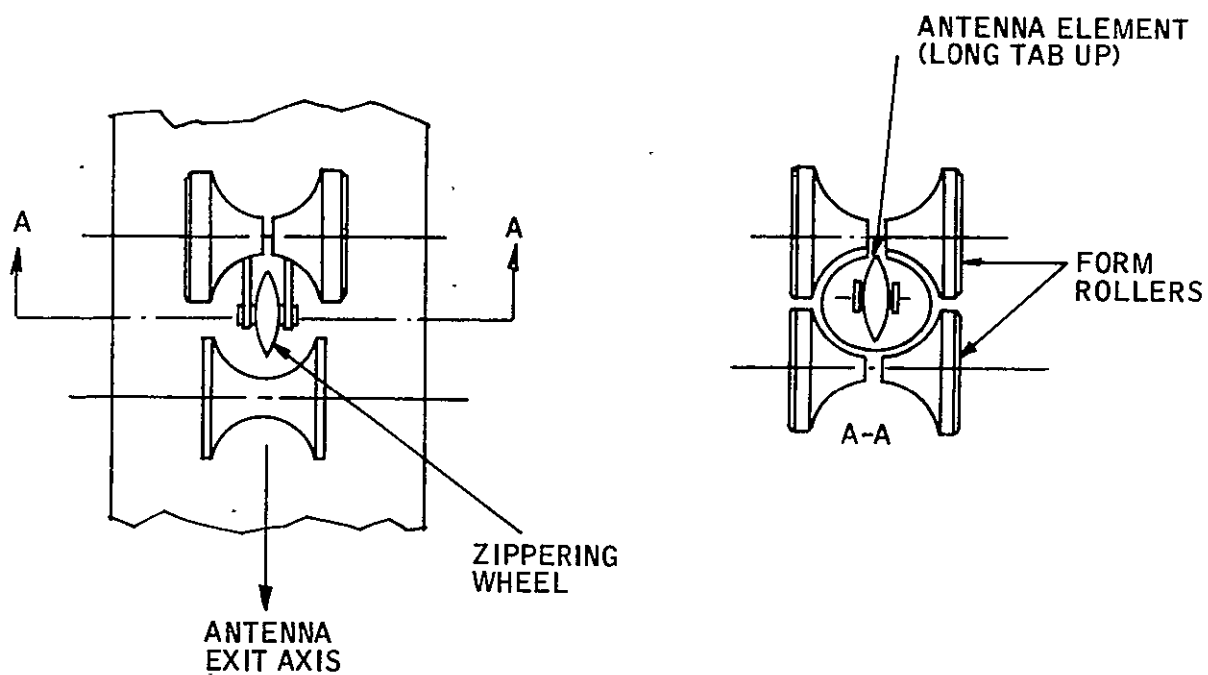


Figure 11.2-6. Antenna Element Edgeloek Zipper

11.2.2.2.2 Power Supply

Power for the d-c drive motor is supplied by an internal thermal battery that is deployed with the boom mechanism. The battery is an electrochemical power supply device based on the use of electrolytes of inorganic salts that are solid and nonconducting at normal missile and space vehicle operating temperatures. Upon application of sufficient heat, the electrolyte melts and becomes electrically conductive. Electrical energy may then be withdrawn from the system, with the battery functioning as a standard reduction oxidation process.

Within the battery itself, the heat source is carefully controlled and the correct amount of heat is applied to bring it to its operational temperature range. The heat source may be ignited by flame or hot particles which can be provided either by an electric match or a mechanically initiated primer. The wires to the battery activating device will be cut by a pyrotechnic cutter upon initiation of the deploy signal. This battery can produce 10 watts of power for 10 minutes.

In use, battery life is limited to the length of time the heat is retained in the cell sufficient to keep the electrolyte in the molten condition. As a result of this, most thermal battery applications are of a relatively short useful life. Since total life is dependent on heat retention, the battery for the boom mechanism will have a much greater insulation, which will allow for the increased duration mentioned above. The battery will be 2.5 inches diameter by 5 inches long.

11.2.2.2.3 Drive Motor

The drive motor will be a 14-vdc nominal rated motor as previously space-qualified in the RAE mechanisms. This motor is hermetically sealed and is shielded against outside magnetic influences by a "mu" metal can inside the motor housing. The motor is geared to a pinch drive roller through a miniature precision gear train.

11.2.2.2.4 Pinch Drive Roller

In the extend mode of operation, the element is pulled from the storage reel by a pinch drive roller while a small retarding force is applied to the storage reel through a friction clutch. This feature precludes buckling of the element because it is in tension instead of compression.

The pinch roller consists of a silicone rubber elastomer-coated roller which is compressed against a hard backup roller. This particular material has high resistance to wear and provides against outgassing in a vacuum environment. The boom element is squeezed or pinched between the friction drive roller and the hard backup roller. This precompressed friction drive roller squirts the material out of the mechanism at the rate of six inches per second for the total 165-foot boom length. This then requires a total deployment time of 330 seconds.

The friction drive roller is powered through a gear train which is attached to the d-c motor.

For acceptance functional testing, a retraction capability is built into the mechanism in the form of a mechanical gear shifter attached to the shaft on the d-c motor. When the motor is actuated, the torque imparted by the motor shaft will swing the gear shifter into mesh with either the extend drive gear or the retraction gear, depending on which way the motor shaft is rotating. The basic difference between this mode and the extend mode is that the storage reel is driven instead of the pinch drive roller. During the retract mode a slight retarding force is applied by a slip clutch at the pinch drive roller, which maintains element tension between it and the storage reel. This feature ensures a repeatably tight element storage wrap. During launch, the gear shifter will be locked into mesh in the extend boom position; thus, the mechanism will not have an in-flight retraction capability.

The mechanism design incorporates a reel caging device so that the spring-like antenna material will remain tightly wrapped on the storage spool during the launch vibration environment. Prior to deployment of the element, the reel caging device is unlocked, allowing the reel to rotate freely.

Full boom extension is sensed by a subminiature switch which has a spring-loaded contactor wheel that rides against the surface of the element. This contact takes place at the backup roller. Switch actuation occurs when a slotted hole punched in the end of the boom element passes under the switch roller, allowing it to drop into a cutout in the backup roller, thereby opening the drive motor circuit and removing power from the drive motor.

At the end of full boom extension, an impact load occurs when the mechanism stops its movement. This impact imparts to the boom a slight tension load which will be sensed at the spacecraft gimbal by a strain gage or comparable system. This monitoring will provide an indication of the end of boom deployment.

Prior to and during the launch phase, the boom deployer mechanism is solidly attached to the spacecraft structure interface in three places. The attachments consist of a pyrotechnic gas sealing piston-type separation nut, attachment bolt, and bolt catcher (see Figure 11.2-2). These attachments serve as the mechanism caging device during the launch phase.

The separation nut is a pyrotechnic device designed to contain the gasses generated by combustion of the power cartridge. The separation nut is a pneumatic mechanical separation system which relies on gas pressure to release and eject the bolt to which it is attached. The nut maintains complete gas containment, thereby preventing contamination of the adjacent structure and atmosphere.

The bolt is a typical, high-strength MS-type stainless steel. Upon release by the explosive nut, the bolt is blown back into a bolt catcher sleeve. This bolt catcher is designed to retain the bolt after separation and to prevent potential structural and component damage due to fragmentation of the fastener. The bolt is prevented from rebounding by shock absorbers in the cap of the bolt catcher which trap the bolt by a retaining spring.

The boom can be jettisoned from its root fixity at the inner gimbal by means of a separation pyrotechnic. This pyrotechnic is a simple explosive device which when activated will blow free the entire deployment mechanism and boom from its attachment at the spacecraft inner gimbal. The boom is attached to the separation pyrotechnic by simple clamping.

11.2.3 THERMAL CHARACTERISTICS

Thermal control of the commandable gravity gradient subsystem can be accomplished by passive means employing superinsulation and state-of-the-art thermal coatings. Preliminary analysis confirms the feasibility of passive control within tolerable temperature limits for both the in-orbit inert storage period and the operative period. The final choice of coatings and the sizing of radiating areas will depend on the required magnitudes of temperatures as well as the rate of changes in temperature. This will be a function of the bearing design and other subsystem components which have thermal stress requirements.

The COGGS is characterized by the availability of large exterior surfaces, and a relatively low wattage input. Since there is no air present, there is no heat transfer by convection. The gimbals are characterized by a minimum amount of common surface and connections, and consequently have poor metal-to-metal conduction through the point-contact bearings and the long, thin flex leads. The thermal capacity of each gimbal can be manipulated somewhat by selection of materials, and allocation of mass. The thermal energy received from the sun, for a synchronous satellite, varies as an approximate sine function over a 24-hour day.

11.2.4 POWER CHARACTERISTICS

The COGGS interfaces with the spacecraft power supply through a standard load interface circuit. Maximum power consumption is 15.16 watts, as shown in Table 11.2-2.

Average power can be considered to be this value minus the torquer and torquer drivers. Strobing of the encoder will further reduce the peak power by approximately 6 watts.

Allowing a 10 percent dissipation of power in the load interface circuit, the average power is 5.48 watts, as shown in Table 11.2-3.

11.2.5 MASS PROPERTIES

The mass properties are tabulated for the two boom configurations considered. Each of the boom configurations was configured to give a first-mode bending frequency of greater than 0.06 rad/sec. The boom with a 0.765-inch diameter presents a minimum weight configuration (tip mass plus boom). The 0.57-inch-diameter boom is from an existing design. Tables 11.2-4 and 11.2-5 summarize the overall weight of the COGGS using 0.765- and 0.57-inch-diameter booms respectively.

Table 11.2-6 presents a detailed weight breakdown of the gimbal assembly, and Table 11.2-7 presents a detailed breakdown for the boom and deployer mechanism.

Table 11.2-2. Maximum Power Consumption

Item	Power (watts)
Encoder logic	0.32
Torquer	2.80
Torquer drivers	0.28
Load interface circuit	1.1
Power supply	1.5
Encoders	6.0
Mode logic	3.16
Total	15.16

Table 11.2-3. Average Power Consumption

Item	Power (watts)
Encoder logic	0.32
Power supply	1.5
Mode logic	3.16
Load interface circuit	0.5
Average	5.48

Table 11.2-4. Weight Using a 0.765-Inch-Diameter Boom

Component	Weight (lbs)
Gimbal assembly	14.7
Gimbal electronics	3.3
Boom	5.4
Deployer (tip mass)	14.2
Total	37.6

Table 11.2-5. Weight Using a 0.57-Inch-Diameter Boom

Component	Weight
Gimbal assembly	14.7
Gimbal electronics	3.3
Boom	21.0
Deployer (tip mass)	
Total	39.0

Table 11.2-6. Weight Summary - Gimbal Assembly

Component	Weight (lbs)
Intergimbal assemblies	
DC torquer	1.0
13-bit encoder (less modules)	1.8
Flex lead assembly	0.3 per set
Gimbal bearing set	0.1
Encoder bellows adapt	0.25
Bearing plates (2)	0.1
Gimbal stop assembly	0.15
Total per IGA	3.70 x 2 (usage) = 7.40
Base mounting ring	3.50
Inner gimbal structure	0.4
Outer gimbal structure	1.8
PC tapes and OG wiring	0.1
Boom jettison, clamp, stop sensor	0.6
Connectors (2)	0.2
Base wiring and flex tapes	0.3
Miscellaneous clamps, washers, screws	0.4
Subtotal for hardware set less IGA = 7.3 lbs	
Grand total for gimbal assembly (less electronics)	14.7

Table 11.2-7. COGGS Boom/Mechanism Weight Breakdown

Component	Weight (lbs)
Thermal battery (including insulation and connector)	4.00
Battery mounting base	0.21
D-C motor	0.85
Boom (0.57-inch-diameter)	2.60
External structure (including 3 brackets)	3.84
Internal structure	3.57
Bolt catchers	0.69
Reel caging device	0.60
Element form rollers	0.27
Support tube assembly rollers	0.14
Backup and pinch rollers	0.19
Shafting	0.07
Reel	0.60
Gear shifter linkage	0.19
Gear shifter locking mechanism	0.50
Miscellaneous gears	0.10
Motor gear	0.04
Miscellaneous bearings	0.15
Miscellaneous small plates and covers	0.50
Subtotal	19.11
Miscellaneous hardware (10 percent of 19.11)	1.91
Total	21.02

The design reliability goal established for COGGS is stated as 85 percent probability that the COGGS will successfully complete its intended mission and function. This goal pertains to a one-year operating life in orbital environment, preceded by a nonoperating orbital life of up to 18 months.

The reliability of the COGGS (R_S) is computed as

$$R_S = R_1 \times R_2$$

where

R_1 = Reliability of boom deployment

R_2 = Reliability of COGGS for a one-year mission, following successful boom deployment.

In accordance with the ATS-F&G proposal instructions, Honeywell performed reliability calculations based on failure rates as specified in MIL Handbook 217A, as well as failure rates for Hi-Rel space applications. As stated in MIL Handbook 217A, "these failures do not include the effects of preconditioning, screening, vendor selection, special handling, or other quality assurance controls which are in addition to normal controls"; these failure rates are therefore not compatible with the ATS Program part requirements. The actual COGGS reliability is therefore based on the failure rates for Hi-Rel space application. Parallel computations based on MIL Handbook 217A are presented for comparison purposes only.

Based on the calculations made for R_1 and R_2 , the COGGS reliability is:

$$R_S(\text{MIL Hdbk 217A}) = 0.9956 \times 0.4561 = 0.4541$$

$$R_S(\text{ATS Hi-Rel}) = 0.9956 \times 0.9476 = 0.9434$$

AUXILIARY PROPULSION SUBSYSTEM

The Phase C effort has allowed a close coordination of the ACS with the auxiliary propulsion subsystem (APS) interface in requirement setting and in adjusting ACS requirements with APS performance to provide an optimally designed system. The key requirements that were addressed in defining the APS and integrating it with the ACS for the ATS-F&G spacecraft were:

- Simplicity and redundancy of design for high reliability
- Long life (2 years minimum)
- Propellant compatibility with vehicle experiments, antenna, thermal radiation surfaces, and earth, star and sun optical sensors
- Capable of thrust levels from 0.002 lbf to 0.10 lbf covering orbit control (O/C) and attitude control (A/C) requirements
- Capable of functioning at a wide range of flow rates and temperature extremes
- Operating with limited electrical power
- Highest possible specific impulse to minimize system weight
- Propellant must be storable for 2 years minimum mission life
- Impulse bits from 0.0025 lb-sec to several lb-sec with pulse repeatability within ± 10 percent
- System suitability for a total impulse of approximately 10,000 lb-sec

Several possible approaches to the APS were studied. Thrust levels, system total impulse, duty cycle and power requirements were established for these candidate propulsion systems. As studies progressed, the field was narrowed to hydrazine and ammonia, with funded breadboard programs established for further evaluation and determination of the preferred approach. APS performance capabilities were compared with spacecraft requirements and compatibility with other vehicle control systems and components in the process of making a selection. As a result of these studies and tests, the ammonia

system was selected as the preferred approach. This choice was made primarily on the basis of:

- Advanced development and qualification status of ammonia feed systems and thrusters backed by flight experience
- Better minimum attitude impulse bit repeatability
- Feed system does not require zero-g propellant control
- Lower temperatures can be tolerated in the propellant feed lines (i.e., reduced thermal control)
- Compatability of ammonia with sensors, experiments and antenna mesh as well as APS system components

The studies and analysis leading to the selection of the ammonia system benefited further by use of existing ammonia propulsion system capability developed for earlier ATS vehicles, the LES-6 and NRL satellites.

12.1 FUNCTIONAL DESCRIPTION

This section describes the functional requirements, performance, and interfaces established for the APS and the several components of the system. The functional performance of the APS is designed to meet or exceed the functional requirements. Table 12.1-1 summarizes APS requirements versus performance. Table 12.1-2 summarizes component requirements versus performance. The source of the requirements is also referenced in the tables.

12.1.1 FUNCTIONAL PERFORMANCE REQUIREMENTS

The functional performance requirements for the APS and the A/C and O/C thrusters, with consideration given to duty cycle, reliability, system leakage, and telemetry of various pressures, temperatures, and control functions, are discussed below.

12.1.1.1 Overall APS

Overall system requirements as determined by mission analysis for system total impulse, thrust levels, thrust response and repeatability, duty cycles, life, leak rate, telemetry, and system redundancy are discussed in the subsections that follow.

12.1.1.2 Impulse

Impulse requirements imposed on the selected ammonia APS are listed in Table 12.1-3 for the nominal design spacecraft with a deployed weight of 2000 pounds and for the maximum weight spacecraft (for tank sizing only) with a deployed weight of 2450 pounds.

Table 12.1-1. APS Requirement/Performance Summary

APS Parameter	Requirement	Source of Requirement	Performance
Impulse, lb-sec Propellant weight, lb	10,600 flight load 66 flight load	Spacecraft systems analysis	12,700 design capacity 78 design capacity
Thrust: Pitch and roll, lb Yaw, lb	0.050 } Attitude 0.025 } control	See Attitude Control Subsystem Report in Section 12 of this report	0.050 0.025
Thrust: Orbit Control, lb	.002 - .100		0.003
Reliability	Operate for 2 years minimum	Section 5 of APS report	R = 0.95512 for 2 years
Power, when thrusting (O/C thruster), watts	Minimum for 200 sec. I_g	Power allocation analysis and Table 3.1.2; of APS report see also ACS report	30
System leak rate, sec He/sec	10^{-4}	Section 6.6.1.1 of APS report	Less than 10^{-5}
Telemetry	As required to determine system condition	Sections 2.3.4, 3.2.3 and 6.9.4 of APS report	Tank temperature; feed system pressure; O/C thruster heater current
Duty cycle	See Duty Cycle Table 12.1-5 of this report for A/C and O/C jets	Attitude Control Subsystem report	A/C and O/C thruster duty cycle varies for 8 different operational modes; see Table 12.1-5 of this report.

Table 12.1-2. Component Requirement/Performance Summary

Parameter	Requirement	Source of Requirement	Performance
1) Latching valve response Response time, opening, sec Response time, closing, sec Leak rate, maximum, sec He/sec	0.020 maximum 0.020 maximum 5×10^{-6}	ACS Analysis Report and Section 6, 6, 1 of APS report	0.012 nominal 0.006 nominal 1×10^{-8}
Flow rate, lb/sec, attitude control jets	5×10^{-4} for 0.050-inch orifice	Based on 0.050 lbf at $I_{sp} = 100$	5×10^{-4} plus, for 0.057-inch orifice
Flow rate, lb/sec, orbit control jets	15×10^{-4} for 0.015-inch orifice	Based on 0.001 lbf at $I_{sp} = 200$	15×10^{-4} plus, for 0.017-inch orifice
Power, watts, attitude control Power, watts, orbit control	1 2	Minimum Power Schedule Table 3, 1-2 of APS report, see also ACS report	2 1
Actuation cycles, attitude control Regulator cycles Actuation cycles, orbit control	200,000 minimum 500,000 minimum 1000 minimum	APS Evincive Data Report	2,000,000 plus 2,000,000 plus 2,000,000 plus
2) Latching valve response Response time, opening, sec Response time, closing, sec Actuation cycles Leak rate, maximum, sec He/sec Flow rate, lb/sec, maximum	0.40 maximum 0.40 maximum 10,000 minimum 10^{-3} 1×10^{-3}	APS Evincive Data Report	0.0045 0.005 20,000 minimum 10^{-8} or better 1.87×10^{-3} with 0.077-inch orifice
3) Command signal	On-off power for RPFA(2). Open-close latching valves (6) Prime and/or backup jets (6, ± each axis) On-off O/C jets-heaters and valves (8) On-off A/C jet valves and override (6, ± each axis)	ACS Report	Provided
4) Propellant tank capacity and ground level measurement	Size for 78 pounds of propellant	ACS Analysis and Evincive Data	Designed for 78 pounds of propellant and measured ultrasonically
5) Control pressure transducer Control range psig Cycle life Proof pressure, psi Burst pressure (capsule) psi Temperature range, operating °F Electrical excitation, vdc Electrical output, vdc	15 to 30 5×10^5 450 675 +13 to +127 26 0 to 5	Sections 2, 3 and 6, 3 of APS report Required as pressure sensor for regulation system	Mechanical range 0 to 300 psig 3×10^6 500 (based on 250 psi rated range) 750 -65 to +260 28 ± 4 0 to 5
6) Fuel pressure transducer Pressure range, psia Overpressure, psia Burst pressure, psia Excitation, vdc Accuracy at ambient, percent	0 to 300 350 675 26 ±2 of full scale	Required output for telemetry Engineering judgement	0 to 300 450 1500 24 ± 4 ±1 of full scale
7) Thermistors Temperature range, °F Sensor accuracy, percent	+13 to +127 3	Fairchild-Hiller specification and engineering judgement	-50 to +350 2
8) Valve heater (regulator valves only) Heat dissipation, watt	0.5	Experience on ATS-D&E determined that gas exposure to valve seat is preferred	0.5
9) O/C line-heater tapes	2 watts distributed	Fairchild-Hiller thermal analysis	2 watts distributed

Table 12.1-3. APS Impulse Requirements

Parameter	Characteristic Velocity (ft/sec)	Impulse (lb/sec)	
		Nom. Wt. (2000 lb)	Max. Wt. (2450 lb)
a) Orbit control			
Initial orbit adjust	30	1875	2285
East-west stationkeeping (2 years)	14	870	1070
East-west station repositioning	60	3700	4500
North-south stationkeeping (1 month)	14	870	1070
Additional orbit maneuvers	10	635	805
Orbit control subtotal	128	8000	9800
Orbit propellant weight (lb) ($I_{sp} = 200$ sec) (heater power limited to 30 watts)	---	40	49
b) Attitude control			
Impulse (2 years) (lb/sec)	---	2600	2900
Propellant weight (lb) ($I_{sp} = 100$ sec)	---	26	29
c) Total impulse (lb/sec) O/C + A/C	---	10,600	12,700
d) Total propellant weight (lb) O/C + A/C	---	66	78

The spacecraft jet response time from signal-on to 90 percent rated thrust and signal-off to 10 percent rated thrust is as follows for the A/C and O/C thrusters:

- Attitude control thruster response and repeatability requirements for all control modes except extended jet-only limit cycle and SAPPSAC modes are as follows: (1) minimum equivalent square-wave pulse duration 0.100 second; (2) maximum thrust rise response 0.035 second and decay response 0.100 second; (3) impulse repeatability within ± 20 percent.
- Jet-only and SAPPSAC requirements for response are: (1) 0.035-second rise response and 0.100-second decay, with repeatability within 0.010-second rise and decay; (2) pulse duration 0.050 second minimum; (3) impulse repeatability within ± 10 percent.
- Orbit control thruster response and repeatability requirements for orbit control are: (1) maximum thrust rise and decay response 5.0 seconds; (2) pulse repeatability not critical and demands only that rated thrust be obtained within ± 10 percent in 5 seconds after "on" command.

12.1.1.3 Thrust

Thrust levels derived from torque and lever arm requirements are listed in Table 12.1-4.

Table 12.1-4. Thrust Levels and Lever Arms

Thruster	Thrust (lb)	Lever Arm (ft)	Torque (ft-lb)
Pitch and roll	0.050	5.6	0.280
Yaw	0.025 (each of two)	2.0	0.050
Orbit control	0.003	N/A	N/A

12.1.1.4 Duty Cycle

Table 12.1-5 lists estimated duty cycle requirements of the APS in various control modes which were used as design guides.

Table 12.1-5. Thruster Duty Cycle in Control System Operational Modes

Operational Mode	Attitude Control Thrusters	Orbit Control Thrusters (Ammonia)
1) Damp residual rates	Three axes all continuous ON at same time. Pitch: 4.25 minutes; roll: 4.05 minutes; yaw: 1.64 minutes.	N/A
2) Initial acquisition	Continuous ON time in following sequence: Pitch and yaw both: 1.31 minutes; roll: 1:31 minutes; yaw: 0.3 minute; followed by 8 below.	N/A
3) Initial orbit correction	Limit cycling; Pitch 0.3 percent ON time, yaw 0.6 percent ON time. Pulse width of 0.200 second ON. Roll as in 8 below.	7.2 days total ON time (one or several burns) (1875 lb-sec at 0.003-lb thrust)
4) Wheel unloading	Pulse 1 second ON and 60 seconds OFF. Pitch: train 5 pulses once per day. Roll: train 2 pulses twice per day. Yaw: train 2 pulses once per day.	N/A
5) Stationkeeping E-W (2 years)	Same as 3 above.	3.5 hours each month (870 lb-sec)
6) Stationkeeping N-S (1 month)	Same as 3 above.	1.35 hours (each mode every 12 hours) (870 lb-sec)
7) Additional orbit maneuvers	Same as 3 above.	635 lb-sec
8) Repositioning E-W	Same as 3 above.	7.2 days (each end)
9) System backup	Limit cycling. Pitch, roll and yaw individually ON or in combinations up to all three axes ON at same time. 0.100 second ON and 1000 seconds OFF to 0.300 second ON and 100 seconds OFF.	N/A

The APS is to be designed for a mission lifetime of at least two years, with five years as a design goal. Tests have been conducted on an ammonia system similar to the ATS-D&E hardware for 1.5 years at Avco to determine compatability and operation of the system. The system is exercised for thrusting and gas generation once a week and to date continues to perform satisfactorily. TRW Systems has operated an ammonia system, designated the ACSKS, for two years with a propellant capacity of 22.75 pounds and a thrust level of 0.020 pound. This system has operated satisfactorily during this time and is presently being modified to meet the ATS-F&G requirements. These life tests serve as the backup for choosing identical or similar parts for the two-year mission of ATS-F&G.

The system will contain parallel redundant tankage, pressure regulation assemblies, A/C thrusters, and O/C thrusters as shown in Figure 12.1-1. A detailed analysis of the reliability of the complete APS is given in Section 5 of the APS Phase D Proposal, Vol. I-D.

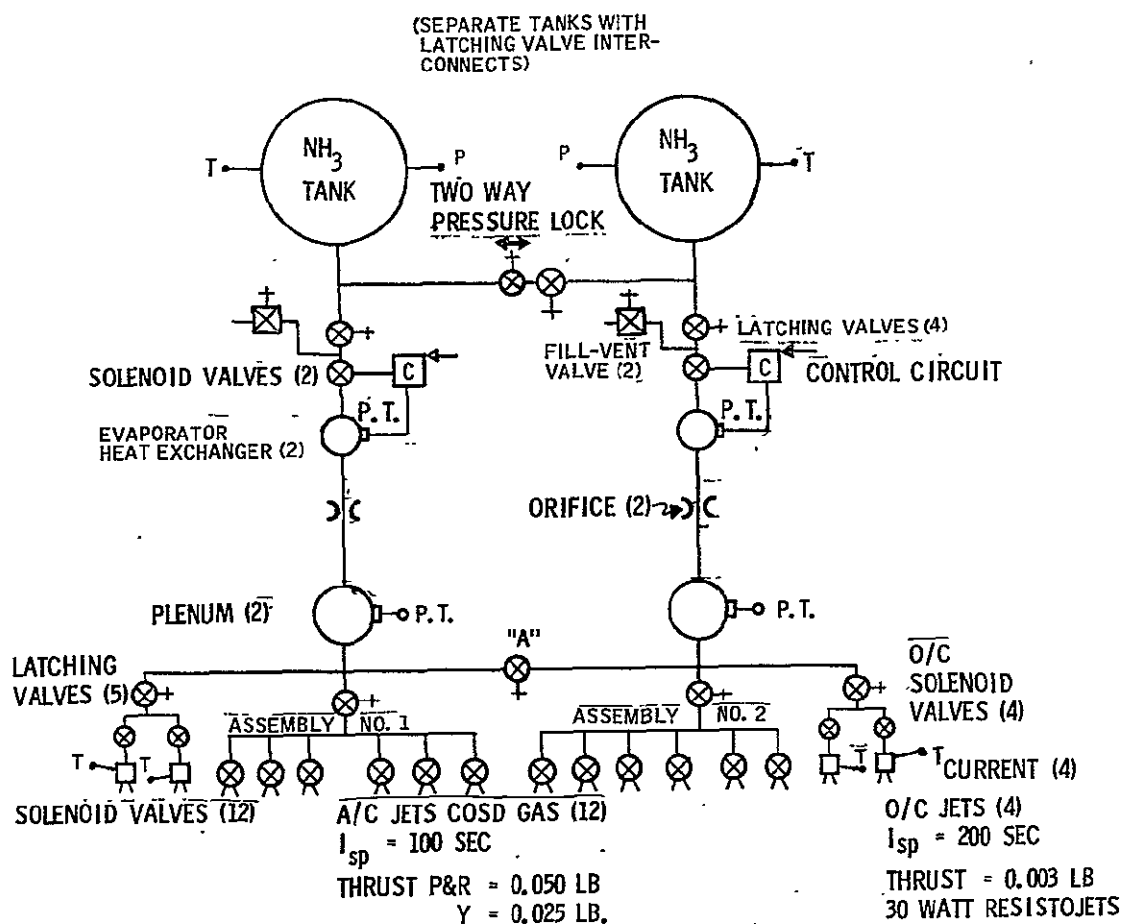


Figure 12.1-1. Ammonia APS Mechanization Diagram

12.1.1.5.1

Attitude Control Thrusters

The 12 A/C thrusters will be redundant so that at least one means of thrust backup for each axis will be provided based on ground command. All solenoid valves will be located on the central APS assembly. Thrust nozzles will be located on the spacecraft as shown in Figure 12.1-2. Jet control action for the two groups of thruster assemblies, each of which is isolated by a latching valve, is as follows:

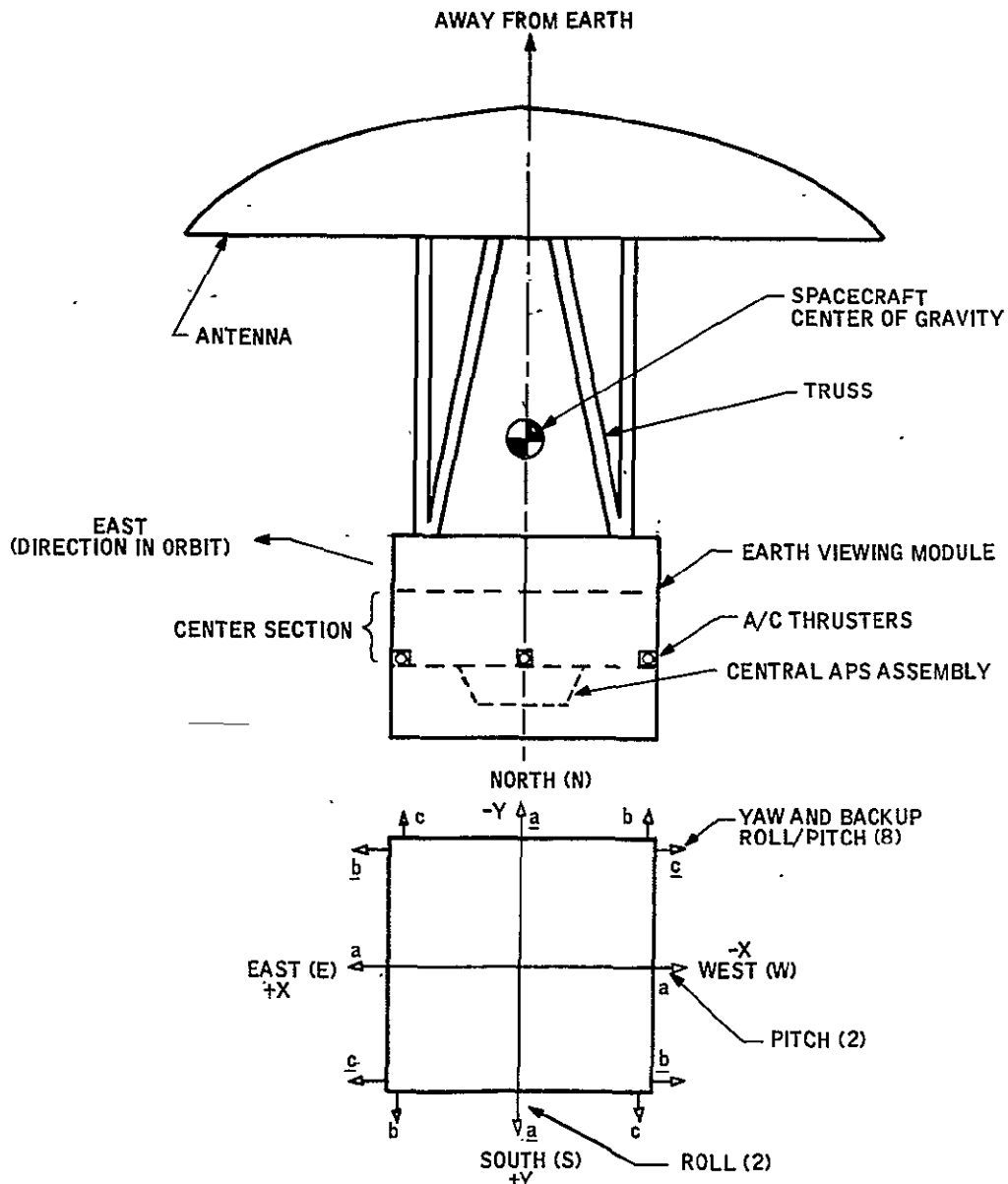


Figure 12.1-2. Location and Redundancy of Attitude Control Thrust Nozzles

<u>Control Action</u>	<u>Thruster Assembly No. 1</u>	<u>Thruster Assembly No. 2</u>
- Pitch	Ea (prime)	Eb and Ec
+ Pitch	Wa (prime)	Wb and Wc
- Roll	Nb and Nc	Na (prime)
+ Roll	Sb and Sc	Sa (prime)
- Yaw	Nb and Sb	Eb and Wb (prime)
+ Yaw	Nc and Sc	Ec and Wc (prime)

Ground commands can be used to provide jet only attitude control.

12. 1. 1. 5. 2 Orbit Control Thrusters

The four O/C thrusters will expel gaseous ammonia plus free nitrogen and hydrogen. They will be heated by an electrically-powered resistance element located immediately upstream of the nozzle throat. Figure 12. 1. 3 shows that the nozzles for these thrusters will be located on the truss in the plane of the spacecraft center of gravity, so that the specified line of thrust of each nozzle will pass through the spacecraft center of gravity. The solenoid valves for the O/C jets will be located in the central assembly of the APS within the center section of the EVM.

The control system will be mechanized so that thruster redundancy is provided by rotating the spacecraft about the yaw axis. All control of the O/C jet heaters and valves is by ground command.

12. 1. 1. 6 Power

The electrical heaters in the O/C thrusters are sized to consume no more than 30 watts each. The power for the jet control valves will be 1 watt each for orbit control and 2 watts each for attitude control.

12. 1. 1. 7 System Leak Rate

The leak rate of the APS will not exceed 10^{-4} scc/sec of helium. This leak rate is based on nominal valve leak rate data of 5×10^{-6} scc/sec of helium per valve, and that 10^{-4} scc/sec of helium represents 0.65 lb of ammonia in a 2-year period.

12. 1. 1. 8 Command Telemetry

All necessary commands for powering the APS, for selecting redundant elements for control of the O/C jets, and for ground override control of the A/C jets are provided (Table 12. 1-2). The APS will contain sensors which provide signals that represent critical temperatures and pressures to indicate propellant consumption. The signals will be conditioned as necessary to provide the specified signals for telemetry.

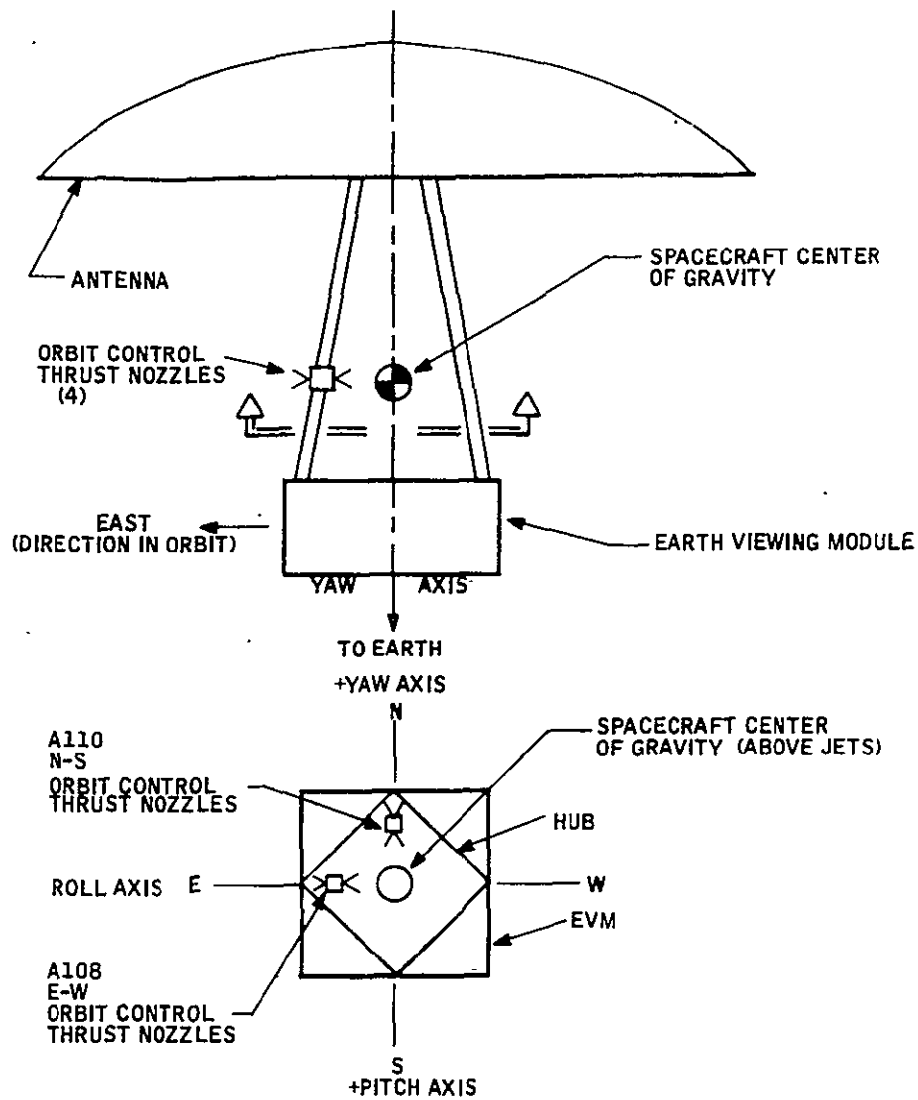


Figure 12.1-3. Location of Orbit Control Thrust Nozzles

12.1.2 COMPONENT FUNCTIONAL REQUIREMENTS

The functional requirements of the several components and subassemblies of the regulated pressure feed assembly, A/C and O/C thrusters, and command/telemetry signals are discussed in the paragraphs that follow.

The design and procurement specifications for operating hardware will contain functional requirements more stringent than the system requirements to assure ample margins in meeting extremes of system operation.

12.1.2.1 Regulated Pressure Feed Assembly

The required system propellant supply of 66 pounds of ammonia will be equally divided between two cylindrical tanks that provide a working pressure varying with temperature from 73 psia to 212 psia.

The feed system pressure control level will be set in the range of 20 to 24 psia with ammonia gas supplied to the A/C and O/C solenoid control valves in the temperature range of +40° to +100°F. For these temperatures the flow rate range will be 15×10^{-6} lb/sec on the low side to 10×10^{-4} lb/sec on the high side per half-system. The maximum flow requirement will be sustained for 98 seconds. An intermediate flow requirement of 50×10^{-5} lb/sec will be sustained for 243 seconds, and the minimum flow shall be maintained for 7.2 days. In the temperature range of 13° to 127°F, the system will be required to provide flow for pitch and roll of a 0.3-percent duty cycle, for yaw of a 0.6-percent duty cycle, and one O/C jet on 100-percent duty cycle for 10 minutes.

Solenoid valve requirements for the regulated pressure feed assemblies, A/C thruster, and O/C thrusters are listed in Table 12.1-6.

Table 12.1-6. Ammonia Solenoid Valve
Functional Requirements

Item	Requirement
Medium Orifice ⁽¹⁾	Ammonia gas 0.050 inch, minimum ⁽²⁾
Operating pressure	20 psia, normal to 300 psi maximum
Power (at 24 vdc) per coil	3 watts, maximum ⁽³⁾
Response time, opening	0.020 second, maximum
Response time, closing	0.020 second, maximum
Actuation cycles	200,000 cycles, minimum
Orbit life (duration of operation)	2 years, minimum
Leak rate, maximum	5×10^{-6} sec Hc/sec
Temperature range	25 to 127°F
Weight	Minimum
Screen, inlet filter, rating	25 microns
Flow rate of ammonia gas, maximum	5×10^{-4} lb/sec

- (1) Equivalent sharp-edge orifice diameter.
- (2) 0.050-inch orifice applies to solenoid valve for regulated pressure feed assembly and A/C thrusters. Minimum orifice size for O/C thrusters will be 0.015 inch.
- (3) 3 watts applies to the solenoid valve with 0.057-inch orifice. The 0.015-inch orifice valve will require 2 watts maximum.

Latching valve requirements for selecting either of 2 bands of 6 A/C thrusters, one or both regulated pressure feed assemblies, a pair of O/C thrusters, and separation of the complete system into equivalent half-sections are listed in Table 12.1-7.

12.1.2.2 Attitude Control Thrusters

Vacuum thrust levels will be as specified in Table 12.1-8 within the specified operating temperature range.

Table 12.1-7. Ammonia Latching Valve
Functional Requirements

Item	Requirement
Medium	Ammonia gas
Operating pressure	130 psia normal to 300 psi maximum
Voltage	24 $\pm$ 2 volts dc
Current pulse	2 amperes
Response time, opening	0.040 second, maximum
Response time, closing	0.040 second, maximum
Actuation cycles	10,000 cycles, minimum
Orbit life (duration of operation)	2 years
Leak rate, maximum	10 ⁻⁵ scc He/sec
Weight	Minimum
Flow rate	0.0012 lb/sec
Screen, inlet filter, rating	25 microns
Pressure drop	3 psia

Table 12.1-8. Thrust Level and Response Time for A/C and O/C Thrusters (Including Solenoid Valves)

Thruster Module	Thrust Level		Thrust Response Time	
	Pitch and Roll Thrust (lb)	Yaw Thrust (lb)	Valve Signal to 90% Thrust Rise (sec)	Valve Signal to 10% Thrust Decay (sec)
Attitude control	0.050 $\pm$ 0.002	0.025 $\pm$ 0.001	0.035 max	0.100 max
Orbit control	0.003 $\pm$ 0.00015		5.0 max	5.0 max

Thrust response time with nominal feed pressure and excitation voltage, and with thruster temperature within the specified operating range, will be as specified in Table 12.1-8. Note also the A/C thruster solenoid valve and functional requirements in Table 12.1-6.

The thruster response listed in Table 12.1-8 is based on A/C thruster solenoid valve functional requirements as specified in Table 12.1-6 plus the addition of a 3/16-inch O.D. tube, two feet long, between valve and jet nozzle for A/C jets.

The specific impulse of the thruster will be 100 seconds minimum in a vacuum.

The repeatability of the delivered total impulse bit with the minimum electrical input pulse width of 0.030 second will be $\pm$ 10 percent from the average impulse bit throughout the specified operating temperature.

12.1.2.3

Orbit Control Thruster

The vacuum thrust of the nozzle with electrically-heated gas will be 0.003 ± 0.00015 lbf.

Thrust response time with nominal feed pressure and excitation voltage, and with thruster temperature within the specified operating temperature range, will be as specified in Table 12.1-8. This response is based on a 1/8-inch line, 5 feet long, between valve and nozzle.

The vacuum specific impulse of the thrust nozzle at 0.003 lbf and with the specified electrical power input will be 200 seconds minimum.

Each nozzle will be adjustable ± 5 degrees from the c.g. plane perpendicular to the X-axis. The nozzle will also be adjustable in the X-axis within ± 0.5 inch. Thrust misalignment will be controlled within ± 0.5 degree.

12.1.2.4

Telemetry

Telemetered data will be conditioned to provide a signal of zero to 6.4 vdc over the full range of the variable and of specified impedance.

12.2

FUNCTIONAL OPERATION AND PERFORMANCE

The discussion that follows covers the general operation of the APS in the process of generating and regulating ammonia gas, and controlling the flow through valves and jet nozzles for thrust control. Several modes of operation are detailed using certain latching valves, jets, and redundant regulation as called out in Table 12.2-1 to satisfy a given condition or operational mode.

Table 12.2-1. Normal Operational Modes

Mode	Regulated Pressure Control Circuit	Latch Valve	A/C Valve	O/C Valve
1) Initial fill pump vacuum	Both ON	Interconnect and plenum tank valves closed, All others open ↓	All closed ↓	All closed ↓
2) Topping off charge	Both OFF			
3) Ascent	Both OFF			
4) Damp residual rates	Both ON	Tank and "A" only closed ↓	Use as needed ↓	
5) Initial acquisition	Both ON	↓		↓
6) Initial orbit correction	One ON	Tank only closed ↓		Use as needed ↓
7) Wheel unloading	↓	↓		All closed ↓
8) E-W stationkeep	↓	↓		Use as needed ↓
9) Add orbit maneuvers + N-S stationkeep	↓	↓		↓
10) E-W reposition	↓	↓	↓	↓
11) System backup	↓	↓	↓	All closed

System performance and capability in meeting the vehicle requirements, including operation and performance of individual components and subassemblies, are discussed below.

12. 2. 1 OVERALL SYSTEM

12. 2. 1. 1 Operation of the APS

The auxiliary propulsion subsystem operates with either liquid or gaseous ammonia supplied from two cylindrical tanks to provide ammonia gas to the plenums and associated plumbing at a given set pressure. The valve-nozzle-thrusters are commanded to operate by the actuator control electronics, thus drawing ammonia gas from plumbing and plenum. A mechanization diagram of the ammonia APS is shown in Figure 12.1-1. As ammonia gas leaves the plenum, a pressure drop occurs in the system which is sensed by the regulator loop pressure transducer. When the control pressure is reached, the APS electronic control unit activates the solenoid valve, allowing more ammonia into the preplenum or capillary tubes from the storage tank. As ammonia is admitted through the control valve, to the evaporator-heat exchanger, the control pressure rises to the upper limit of the hysteresis band, turning off the valve and terminating flow.

For zero-g applications, the system must accommodate either liquid or gaseous ammonia flow. As liquid ammonia is approximately two orders of magnitude more dense than gaseous ammonia, techniques have been devised to accommodate the two-phase flow. The Avco approach to this problem is to use a throttling valve or orifice, preplenum, and control orifice to flash the liquid to vapor and then superheat the vapor. TRW uses a multiplicity of small diameter tubes for the required heat transfer area to assure ammonia gas to the jets. Without these elements, inflow of liquid ammonia directly into the plenum chamber would result in prohibitively large pressure excursions per valve operation, in addition to flooding the system with ammonia. In addition to the preplenum or capillary tubes serving as a heat exchanger during liquid and gaseous operation, the enclosed volume of the heat exchanger provides lead time for control valve operation. During liquid metering, the heat of vaporization and necessary superheat must be supplied to the fluid downstream of the regulation valve in the preplenum or capillary tubes. The source of heat for this operation is the sensible heat in the remaining fuel. To extract this energy from the fuel, the preplenum or capillary tubes are in direct contact with the propellant or the storage tank. Thus, the modes of conduction, condensation/conduction are available for this heat transfer process.

The APS is controlled in part by the APS electronic control unit and the ACS. The operation of all latching valves and the A/C solenoid valves will be controlled by the ACS or ground commands to separate controllers. The APS electronic control unit will control the pressure regulation systems, O/C thrusters (hot and cold), process all housekeeping telemetry required to assess the satisfactory operation of the APS, and condition power for the thruster operation.

Normally, only one-half of the APS operates at a time, with storage tank crossover latching valves closed and all other latching valves open. The other latching valves are used only in the event of a leak failure to seal off the leaking component. During the high flow rates associated with initial acquisition, latching valve "A" is closed and both channels of the regulated pressure feed assembly (RPFA) are used.

The "normal" operational modes of the APS are listed in Table 12.2-1. The table lists the condition or position, (i. e., open, closed) of the various valves for each of the control system modes of operation. Thus, the system provides attitude control thrust for pitch, roll, and yaw (two thrusters) and orbit control thrust (one nozzle at a time) singly or in any combination of these three attitude control axes plus orbit control.

The operation of the system with failed components is illustrated by the data listed in Table 12.2-2. The component which is assumed to have failed is listed in the first column, while the other columns indicate the action necessary to return the system to operation. Note that the redundancy used allows system operation with as many as three failed components in some cases.

12.2.1.2 System Total Impulse

The APS provides a total impulse capability of 12,700 lb-sec (78 pounds of ammonia) in the propellant tanks. The tanks are sized so that the ullage volume is zero at 150°F. This provides a design margin of approximately 23°F, since the system will not be exposed to temperatures exceeding 127°F. If this design margin were not used, there would be the danger of rupturing the high-pressure side of the system due to the thermal expansion of the liquid ammonia. The use of a pressure relief valve to accommodate this expansion is considered undesirable because of the possibility of leakage through the valve. A relief valve will be used with the filling equipment and retained as part of the system until an accurate weight or measure of liquid in the tank is made.

The actual planned propellant flight load for the 2000 -pound ATS-F&G vehicle is 66 pounds of ammonia. The actual ullage volume of propellant tanks based on the design for 78 pounds with loading to 66 pounds is 18.5 percent.

12.2.1.3 Duty Cycle

The APS thrusters are capable of providing a continuous range of duty cycles over a wider range than required (Table 12.1-5). The valves (and thus impulse bit) are capable of pulse widths shorter than required, and the breadboard tests indicate the system is capable of performance within specified requirements with continuous high-flow rates for the periods required.

12.2.1.4 System Leak Rate

The APS system leak rate is expected to be very much less than that allowed. The allowed leak rate of 10^{-4} scc/sec of helium (based on

Table 12.2-2. Operating Modes (With Failed Elements)

Failed Element	Regulated Pressure Control Electronics	Latch Valves	Attitude Control Valves	Orbit Control Valves	Remarks
A) One failed element					
1) Either regulated pressure branch	Failed branch OFF. Normal branch ON.	Failed branch closed. Normal branch open.	N/A	N/A	Control functions fully operational
2) A/C thruster in section	N/A	Failed branch closed. Normal branch open.	- Pitch Ea + Pitch Wa - Roll Nb & Nc + Roll Sb & Sc + Yaw Nc & Sc - Yaw Nb & Sb	N/A	See above, but use A/C thrusters listed. Inhibit yaw correction until pitch is complete.
3) A/C thruster in section	N/A	Failed branch closed. Normal branch open.	- Pitch Eb & Ec + Pitch Wb & Wc - Roll Na + Roll Sa + Yaw Ec & Wc - Yaw Eb & Wb	N/A	See above, but use A/C thrusters listed. Inhibit yaw correction until roll is complete
4) O/C thruster in either branch	N/A	Failed branch closed. Normal branch open.	N/A	Use thruster in operational branch	Rotate spacecraft 90° about yaw for orbit control
B) Two failed elements					
5) 1 and 2 above	*	*	*	*	*
6) 1 and 3 above	*	*	*	*	*
7) 1 and 4 above	*	*	*	*	*
8) 2 and 4 above	*	*	*	*	*
9) 3 and 4 above	*	*	*	*	*
C) Three failed elements					
10) 1, 2, and 4 above	*	*	*	*	*
11) 1, 3, and 4 above	*	*	*	*	*

*See referenced item numbers above.

an equivalent ammonia leak rate of 0.72 scc/hour) would result in the loss of 0.65 lb of ammonia during the two-year mission. This leak rate is equivalent to 16 valves leaking at the maximum allowed rate of 5×10^{-6} scc/sec of helium. Previous test experience indicates that the chosen valves are most likely to leak in the range of 10^{-8} to 10^{-6} scc/sec of helium.

12.2.1.5 Life

The APS equipment is designed (as a goal) to operate in space for 5 years in contrast with the mission requirement of 2 years. The design goal is based on life testing of components and system. The life testing to 2 million cycles without degradation on the control valves is in the order of 10 times the required cycle life for a 2-year mission. The accumulated operating time on a representative system under simulated conditions has exceeded 1 year, with tests continuing and with no performance degradation noted to date.

12.2.1.6 Command

Command signals are as follows:

- On-off power for RPFA(2)
- Open-close latching valves (8)
- Prime and/or backup jets (6; $\pm$ each axis)
- On-off O/C jets -- heaters and valves (8)
- On-off A/C jet valves and override (6; $\pm$ each axis)

12.2.1.7 Telemetry

The APS telemetry meets all the requirements specified. The data telemetered is as follows:

- 1) Temperature of No. 1 tank
- 2) Temperature of No. 2 tank
- 3) Regulated pressure 1
- 4) Regulated pressure 2
- 5) Tank pressure 1*
- 6) Tank pressure 2*

*Consideration is being given to telemetering the commanded state of the latching valves, which A/C jets have been selected, O/C tube temperature, and the two tank pressures; these are not at present included in the telemetry schedule list.

- 7) Current of heated thruster (north)
- 8) Current of heated thruster (south)
- 9) Current of heated thruster (east)
- 10) Current of heated thruster (west)
- 11) Orbit correction thruster temperature (see footnote on previous page)

12.2.2 COMPONENTS

12.2.2.1 Regulated Pressure Feed Assembly

The major functional elements of the RPFA are shown schematically in Figure 12.2-1. Examination of the schematic shows that the ammonia propellant flows from a tank through a latching valve, control solenoid valve, preplenum and flow control orifice, or capillary tube heat exchan-

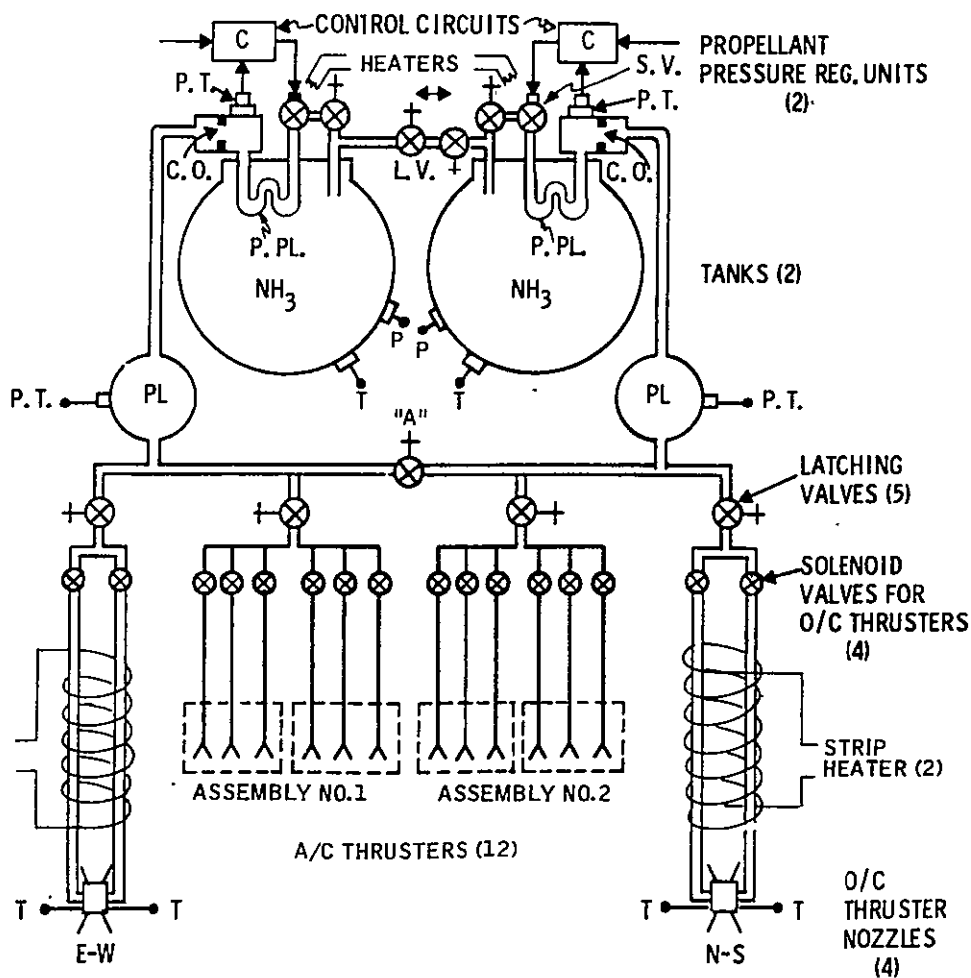


Figure 12.2-1. Regulated Pressure Feed Assembly

ger, and through the main plenum to a three-way branch, through latching valves, and to the thrust control solenoid valves. The RPFA thus provides: (1) propellant storage in 2 cylindrical tanks; (2) the means to convert propellant from liquid to gas at a set pressure; (3) the range of propellant flow rates required by the thrusters; and (4) the means for controlling flow to groups of thrusters.

12.2.2.1.1 RPFA Operation

Normally, all latching valves remain open, except the crossover latching valves between the storage tanks and valve "A" are closed. The crossover latching valves between the tanks and valve "A" are to protect against loss of complete system propellant in the event of a tank leak. Valve "A" can be closed to protect against a leak downstream of the regulators. Valve "A" is normally open to permit the primary jets of thruster assemblies 1 and 2 to operate from one RPFA. Valve "A" will be closed under high-flow requirements during the acquisition mode, so that each regulation system can operate independently to share the operating load and during failure diagnosis.

Since the spacecraft operates in a zero-gravity field, either liquid or gaseous ammonia may be taken from the tank and then passed through the solenoid valve and into the heat exchanger. The system is designed to be fully operational with either liquid or gaseous ammonia flowing from the tank. The purpose of the preplenum or capillary tubes is to provide sufficient heat transfer area to ensure gaseous ammonia leaving the heat exchanger and entering the main plenum. The preplenum or capillary tubes may be described as an "evaporator-heat exchanger". By proper sizing of the evaporator-heat exchanger, the system is designed so that over the specified range of operating temperatures there will always be gaseous ammonia entering the plenum. The purpose of the plenum is to store the gaseous ammonia at the set pressure for use by the thrusters, with sufficient capacity to limit the cyclic operation of the regulating control valve.

Basically, the operation of the regulated pressure unit is cyclical. Normally, the control solenoid valve is closed, and when a thruster is operated, the gas flowing out of the plenums and through the thrusters lowers the pressure in the plenums. This drop in pressure is sensed by the pressure transducer (on the outlet of the capillary tubes or preplenum). The flow control circuit then opens the control solenoid valve, allowing more propellant to flow into the plenums until the pressure rises sufficiently for the pressure transducer to command the control circuit to close the control valve. Thus, we see that the flow rate into the plenums exceeds the flow rate out of the plenums. The equipment is designed to ensure that the input flow to the plenums will always exceed the maximum output flow through the thrusters under any anticipated combination of temperatures and liquid or gas flowing into the preplenum or capillaries. Thus, the control solenoid valve opens and closes at intervals to recharge the plenum with gaseous ammonia.

12.2.2.1.2 RPFA Performance

The feed system will as a minimum be operable over a temperature range from 40° to 100°F, with controlled supply to the A/C and

O/C jets at either 20 or 24 psia. The gas flow rate capability with the two regulators operating in parallel is from zero lb/sec to 200×10^{-5} lb/sec, with either gas or liquid withdrawal from the tank and for any mixture ratios inbetween. This flow rate can be sustained for 1.6 minutes using the redundant circuits in parallel. This flow capability has been confirmed by breadboard tests on a single regulator feed assembly which maintained flow for approximately 5 times the duration required by mission analysis. The minimum flow can be maintained for the required time period of any maneuver limited only by the quantity of propellant allocated for that maneuver.

12.2.2.2 A/C Thruster

The A/C thruster assemblies generate thrust of the required magnitude, duration, and response for inertia wheel unloading, three-axis attitude control in yaw, pitch and roll, with redundant pairs of yaw thrusters acting as backup to both the pitch and roll functions.

12.2.2.2.1 A/C Thruster Operation

The A/C thrusters consist of valve, feed line, and nozzle assemblies. The thrust control valves are mounted on the central structure of the APS assembly, with the nozzles mounted outboard near the sides and corners of the EVM. The tubing from the valve outlet to the nozzle constitutes a chamber volume.

When the solenoid valve is opened, gaseous ammonia passes through the valve into the volume between the valve seat and the nozzle throat, thus raising the chamber pressure. As the gas passes the nozzle throat into the exit cone, the gas expands, converting temperature and pressure energy into kinetic energy. Thrust results from the expulsion of gas from the nozzle at high velocity. Nozzle thrust is proportional, during rise and decay, to chamber pressure.

12.2.2.2.2 A/C Thruster Valve Performance

The performance capability of the A/C thrusters exceeds the requirements. The expected maximum number of cycles on any one A/C thruster valve during the two-year mission is 200,000 cycles; this compares with the valve chosen for use on the thruster which has demonstrated by test (qualification) over 2 million cycles.

The valve used in this thruster is capable of pulse widths down to approximately 0.015 second, whereas 0.030 is the specified requirement. The specified requirement of 0.030 second is approximately two-thirds of the planned minimum pulse width of 0.050 second.

The nozzle exit area ratios are planned at 100-to-1, with high nozzle efficiency so that the expected specific impulse will only be 4 or 5 percent below the theoretical. This will result in an average specific impulse of over 100 seconds, depending on the temperature of the supplied ammonia.

12.2.2.3

O/C Thruster

The O/C thruster assemblies generate thrust of 0.003 lbf at 200 seconds specific impulse with intermittent or continuous duty. The direction of the thrust vector passes through the spacecraft center-of-gravity, to impart a change in the orbital position and velocity of the spacecraft. Each assembly provides means for increasing the specific impulse of the thrust nozzles and preventing propellant condensation in the feed tubes.

12.2.2.3.1

O/C Thruster Operation

Each of the two required thruster modules consists of two sets of valves, tubes, heating elements, and thrust nozzles. The thrust control valves are mounted on the central structure of the APS assembly, with the nozzles mounted in pairs back-to-back on the truss between the EVM and the antenna opposite the spacecraft center-of-gravity. The tubing between the valve outlet and the nozzle will be thermally controlled.

The operation of the O/C thruster is identical to that of the A/C thruster except that the O/C thruster contains an electrical heater to raise the temperature and thus the specific impulse of the expelled gaseous ammonia. Also, the tubing between the valve and nozzle is warmed with a low-power heater and insulated to prevent condensation of ammonia inside the tube.

The thrust nozzle throat is approximately 0.012 inch in diameter for a chamber pressure of 20 psia. The thruster valve equivalent orifice size is 0.017 inch in diameter.

12.2.2.3.2

O/C Thruster Performance

The O/C thruster in the APS meets or exceeds all performance requirements. The minimum required specific impulse is 200 seconds. For 0.003-pound thrust and 30 watts power input, the actual specific impulse for the Avco thruster is 200 seconds at 24 psia. The TRW thruster is 220 seconds at 20 psia nozzle box pressure.

The expected minimum number of cycles on any one O/C thruster valve during the two-year mission is 10,000 cycles, whereas the valve chosen for use in the thruster has demonstrated by test qualification over 2 million cycles. The Naval Research Laboratory has run this same-valve design over 14 million cycles, with the leak rate still within requirements.

12.2.2.4

Solenoid Valve for Regulator and A/C Thruster

The solenoid valve selected for attitude control and pressure regulation subsystems is a Wright Components, Inc. model 15457. This valve design is the same configuration, with slight modification, as model 15398, which was qualified and flown on the ATS-D&E program. The valve has an ethylene propylene rubber (EPR) seat with a metal stop. The EPR material has proven to be the best elastomer for ammonia compatibility through extensive testing at various companies in industry.

12.2.2.4.1 Operation

When energized with 26 vdc from the actuator control electronics, the solenoid valve develops an electromagnetic field which attracts the armature-poppet assembly to close the magnetic gap (stroke) of the valve. As the armature-poppet assembly moves, it opposes a closing spring. The valve then opens, permitting ammonia gas to flow through the valve in a coaxial manner. When the valve is deenergized, the electromagnetic field decreases to zero, allowing the compressed spring to return the armature-poppet to the opposite end of its stroke, which closes the valve. A slight preload or compression of the EPR takes place at valve closing before the armature-poppet contacts a metal-to-metal stop. This compression of the EPR is designed to be of such an amount to assure minimum leakage but not so much as to cause fatigue of the elastomer.

12.2.2.4.2 Performance

At Honeywell's request, Avco tested the model 15457 valve for response, leakage, cycle life, and electrical characteristics. The valve performance data is given in Appendix VI of the APS report. The valve performed well within specifications, indicating low leakage in the order of 2.5×10^{-9} scc He/sec at the conclusion of 2 million cycles.

The results of these tests indicate that the valve is an excellent choice for both pressure regulation and A/C thruster flow control.

Experimental performance data for the model 15457 valve is summarized in Table 12.2-3.

Table 12.2-3. Solenoid Performance Data at 74°F (Model 15457)

Parameter	Upstream Pressure	
	30 psig	300 psig
Liftoff voltage	13.0 vdc	21.1 vdc
Dropout voltage	7.8 vdc	12.1 vdc
Open response (24 vdc)	12.5 msec	50.0 msec
Close response (24 vdc)	5.0 msec	3.0 msec
Power at 24 vdc	1.63 watts	1.63 watts
Leakage	2.5×10^{-9} scc He/sec	2×10^{-9} scc He/sec
Cyclic capability	Over 2×10^6 cycle	---

12.2.2.5 Solenoid Valve for O/C Thruster

The O/C solenoid valve model 15398 is the same configuration as model 15457 discussed above. The basic difference is that the model 15398 valve has a smaller diameter valve seat and therefore requires a lower driving power to operate. This valve has been qualified and flown on the ATS D&E program.

12.2.2.5.1 Operation

The operation of this valve is identical to that of model 15457 discussed above.

12.2.2.5.2 Performance

The performance of model 15398 is very similar to model 15457, except that it will handle about 1/12 the flow for the same pressure drop. This valve is ideally suited for the flow control of the 0.003-pound thruster in orbit control.

12.2.2.6 Latching Valve

The use of latching valves in the APS is planned as a means of protecting against the primary valve failure mode of leakage past the valve seats. Including latching valves in the flow lines allows for series seats in every flow path, thus providing protection against "open" leakage-type failures.

Fuel tanks will be plumbed together through a line containing two latching valves "back-to-back." Tests have shown that pressure differentials greater than 50 psi (higher pressure on the outlet side) cause the valve to unlatch from the closed position. Back-to-back valves prevent flow in the crossover line at high pressure differentials.

A normally-closed squib valve has also been considered for use in place of the two latching valves. This approach may, after further analysis, be adopted; however, this limits the system to one regulator failure, with no means of limiting propellant loss once the valve is opened if a leak in the opposite storage and flow regulation system develops. If the mode of failure just described occurs, and latching valves are used instead of the normally-closed squib valve, the latching valves can be pulsed open and then closed as required to provide sufficient propellant for control and the leak without being pressurized continuously. This will conserve propellant and allow leakage only when jet thruster operation is required.

12.2.2.6.1 Latching Valve Operation

The latching valve proposed for this application is a Wright Components, Inc. model 15478. Avco tested this valve for Honeywell with satisfactory results. The valve was cycled for 20,000 cycles, with intermediate and final leak checks indicating leakage in the order of 1.5×10^{-8} scc He/sec. The operation is similar to the Wright solenoid valve, except that two solenoid coils are required, one to open the valve and one to close it. The valve is held open by a spring-loaded ball and detent and held shut by the ball and detent, spring, and hydraulic force.

12.2.2.6.2 Latching Valve Performance

The Wright Components, Inc. model 15478 latching valve was selected on the basis of data obtained in recent tests. The valve performance meets all the requirements of flow, leakage, orifice size, cyclic life,

and compatibility with ammonia. Of particular interest is the cyclic life. In general, the latching valve will not be required to operate more than 5000 cycles during the mission life. During operational tests, the valve was cycled 20,000 times with no deterioration in operation, leakage, or detent wear.

12.3

FUNCTIONAL INTERFACE DESCRIPTION

The functional interfaces of the APS include:

- Electrical interfaces with the ACE
- Mechanical interfaces with the spacecraft structure
- Thermal interfaces with the spacecraft

The installation and mounting portion of the mechanical interface is described in detail in Section 12.4.1.2.

The input interface signals to the APS consist of commands to various parts of the system, electrical power, and heat. The output interface signals from the APS include thrusts from the A/C and O/C thrusters, and telemetry data signals.

Table 12.3-1 lists these interfaces, inputs, and outputs.

Table 12.3-1. Interface Summary

Interface Category	APS Interfaces With	Inputs	Outputs
Commands	ACE	See Table 2.3-2	Thrusts
Electrical power	ACE	See Table 2.3-2	N/A
Telemetry	ACE	See electrical power	See Table 2.3-3
Thermal	Spacecraft	See Section 2.3.6	N/A
Mechanical ⁽²⁾	Spacecraft	N/A	N/A

(1) Thrust acts mechanically upon spacecraft.

(2) See Section 3.1.3 for installation and mounting requirements.

12.3.1

SPACECRAFT INTERFACES

The APS command and electrical power input/output functions interface with other elements of the spacecraft as shown in Figure 12.3-1.

12.3.2

INPUT INTERFACE SIGNALS

The input command and electrical power input interface signals from the actuator control electronics to the APS are listed in Table 12.3-2.

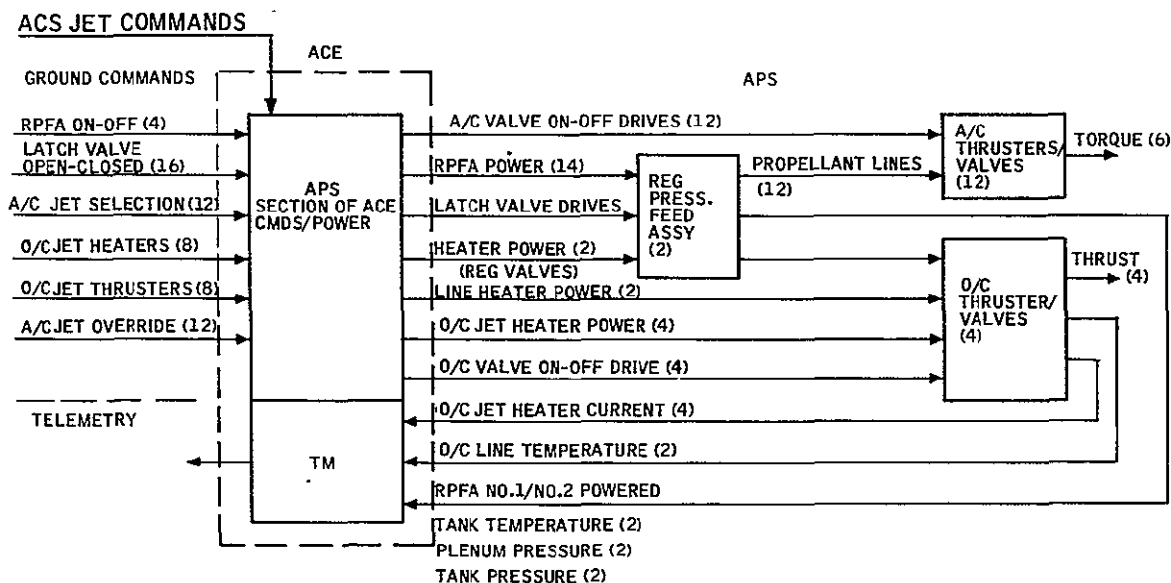


Figure 12.3-1. APS Interface Signals

Table 12.3-2. Input Interface Signals (From Actuator Control Electronics)

Item	Signal Description	Axis	Level
1	Solenoid Valve ON-OFF	+ Pitch	
2		- Pitch	
3		+ Roll	
4		- Roll	
5		+ Yaw	
6		+ Yaw	
7		- Yaw	
8		- Yaw	
9		+ Yaw	
10		+ Yaw	
11		- Yaw	
12		- Yaw	
13	No. 1 pressure control circuit		+ 26 vdc
14	No. 1 pressure control circuit		
15	No. 2 pressure control circuit		
16	No. 2 pressure control circuit		
17	Latch valve ON pressure No. 1		
18	Latch valve OFF pressure No. 1		
19	Latch valve ON pressure No. 2		
20	Latch valve OFF pressure No. 2		
21	Latch valve ON O/C E and W		
22	Latch valve OFF O/C E and W		
23	Latch valve ON O/C N and S		
24	Latch valve OFF O/C N and S		
25	Latch valve ON A/C section No. 1		
26	Latch valve OFF A/C section No. 1		
27	Latch valve ON A/C section No. 2		
28	Latch valve OFF A/C section No. 2		
29	Latch valve ON crossover tank		
30	Latch valve OFF crossover tank		
30A	Latch valve ON crossover plenum		
30B	Latch valve OFF crossover plenum		
31	Solenoid valve O/C thruster N		
32	Solenoid valve O/C thruster S		
33	Solenoid valve O/C thruster E		
34	Solenoid valve O/C thruster W		
35	Heaters for pressure regulator solenoid valves (2)		
36	Heaters for O/C feed tubes		
37	Heater O/C jet N		
38	Heater O/C jet S		
39	Heater O/C jet E		
40	Heater O/C jet W		

12.3.3

OUTPUT INTERFACE SIGNALS

The output interface signals from the APS to the actuator control electronics are listed in Table 12.3-3. The APS also has thrust outputs that act directly on the spacecraft. These thrust outputs are described in Section 12.1.1.3.

12.3.4

TELEMETRY

Propellant tank temperature, tank pressure regulated pressure and O/C thruster temperatures are the data items selected for telemetering to the ground. Table 12.3-3 summarizes the pertinent data.

Table 12.3-3. APS Output Interface Signals to ACE

Item	Function Monitored	Parameter Value	Signal Type ⁽¹⁾	Sampling Rate ⁽²⁾
1	Temperature, No. 1 tank	-11 to +41 $\pm 3^{\circ}\text{C}$	Analog	One per 48 sec
2	Temperature, No. 2 tank	-11 to +41 $\pm 3^{\circ}\text{C}$	Analog	One per 48 sec
3	Tank pressure No. 1 tank	40 to 300 psi $\pm 5\%$	---	One per 48 sec
4	Tank pressure No. 2 tank	40 to 300 psi $\pm 5\%$	---	One per 48 sec
4A	Regulated pressure feed assembly No. 1, powered	On-Off	Digital	One per 48 sec
4B	Regulated pressure feed assembly No. 2, powered	On-Off	Digital	One per 48 sec
5	Regulated pressure 1	20 to 40 $\pm$ psia	Analog	One per 3 sec
6	Regulated pressure 2	20 to 40 $\pm$ psia	Analog	One per 3 sec
7	Heater current, E jet	30 watts	Analog	One per 48 sec
8	Heater current, W jet	30 watts	Analog	Ent.
9	Heater current, N jet	30 watts	Analog	One per 3 sec
10	Heater current, S jet	30 watts	Analog	When firing
11	O/C line temperature	1v to 4v	Analog	---

⁽¹⁾Corresponds to 0 to 6.4 vdc to telemetry from actuator control.

⁽²⁾Can be sampled once every 3/8 sec in dwell mode.

12.3.5

POWER INTERFACE

All electrical power required by the APS is supplied by the propulsion electronics section of the actuator control electronics. The input interface list (Table 12.3-2) includes all power and signal interfaces.

12.3.6

THERMAL INTERFACE

The thermal interface for conducting heat into the APS is defined by the four mounting surfaces of the support structure casting. This is more clearly shown in a layout drawing (see Figure 12.4-1). Radiation from the spacecraft is the other means of heat transfer to the APS.

The average heat rate needed for a 2-year mission to vaporize 66 pounds of ammonia is 0.564 watt.

When an O/C thruster is on for periods of up to 2.2 days continuous, the heat rate needed is 12 watts.

In the event that no heat is supplied by conduction or radiation during the highest flow requirement of pitch, roll and yaw operating simultaneously for 4.25 minutes, the propellant equilibrium temperature will drop a total of 4°F.

The heat input to the APS by radiation, assuming a 5°F difference between the APS equipment and parts of the spacecraft, will be 17 watts. This assumes the support structure to have an area of 14 square feet and an emissivity of 0.85.

For the case of conducted heat, assuming the conductivity of wrought magnesium alloy is taken as 60 Btu/hr/ft²/°F/ft, the length of the support as 2 inches, the area of contact at each of four support points as 0.5 inch square, and if the temperature differences across these supports is 2°F, then the heat transfer rate is 35 watts. This is a significant rate of heat transfer compared with the required rate of 12 watts during extended operation of an O/C thruster.

12.4 PHYSICAL DESCRIPTION

The APS is fully redundant and packaged as an integral subsystem, with all components except the A/C and O/C thrusters mounted to a single support platform at the center of the EVM. System requirements for thruster response are sufficiently long so that all thruster control valves can be separated from the thruster nozzles by appropriate tube lengths and mounted with other components on the APS support platform. The connecting tubes between valves and thrust nozzles act as thrust chambers that increase the L^* in varying amounts. The single APS package concept lends itself to assembly by the supplier rather than by the subsystem manager and component integrator, permitting the best assemblage of parts, system testing, and cleanliness control. A layout drawing of the proposed system is shown in Figure 12.4-1. Figure 12.4-1 shows strap-mounted tanks; however, further studies indicate that a polar or flange-mounted tank would allow more freedom for dimensional changes caused by pressure variations which become significant when the tank reaches this size. The plenum tanks may be mounted in either manner, and the final selection will be based primarily on the ease of packaging and modifications that are inevitable before the overall EVM package design is frozen.

The redundant regulator pressure feed assembly will consist of two parallel cylindrical ammonia storage vessels, two evaporator heat exchangers, pressure regulation and control loops, plenum chambers, distribution tubing, latching and jet control valves. Either half-system is capable of maintaining spacecraft control and will provide one-half of the required mission total impulse.

The present plan is that Honeywell will supply the APS platform to the subsystem supplier and directly handle all interface problems between FHC structures, EVM packaging, and accommodation with APS package design.

The propellant tanks shown in the layout differ dimensionally from those presently planned for the system. This change to a

larger diameter of 12.6 inches and approximately 25 inches in length was made to permit the use of qualified titanium forgings at a reduced cost.

The fill and drain valve connections will be located between the pressure regulation assembly (PRA) latching and control valves. This arrangement requires propellant to be loaded and withdrawn through the PRA latching valves, but provides a significant advantage in that the fill and drain valves can be isolated if leakage occurs. Two valves are used to allow independent filling of each tank (with the two isolation latching valves closed). The APS half of the fill and drain valves will be located at some position on the spacecraft skin near the attitude control nozzles.

The evaporator heat exchanger will be made up of either a dual helical coil mounted and supported from a cover plate that attaches to a four-inch tank opening or capillary tubes bonded to the outside surface of the propellant tanks.

12.4.1 OVERALL APS

Figure 12.4-2 shows the APS installed in the EVM of the spacecraft. Figure 12.4-3 shows schematically the elements of the system.

The A/C thruster nozzles are located on the outside near the corners of the spacecraft. The control valves are located inside the EVM and packaged in the regulator pressure feed assembly.

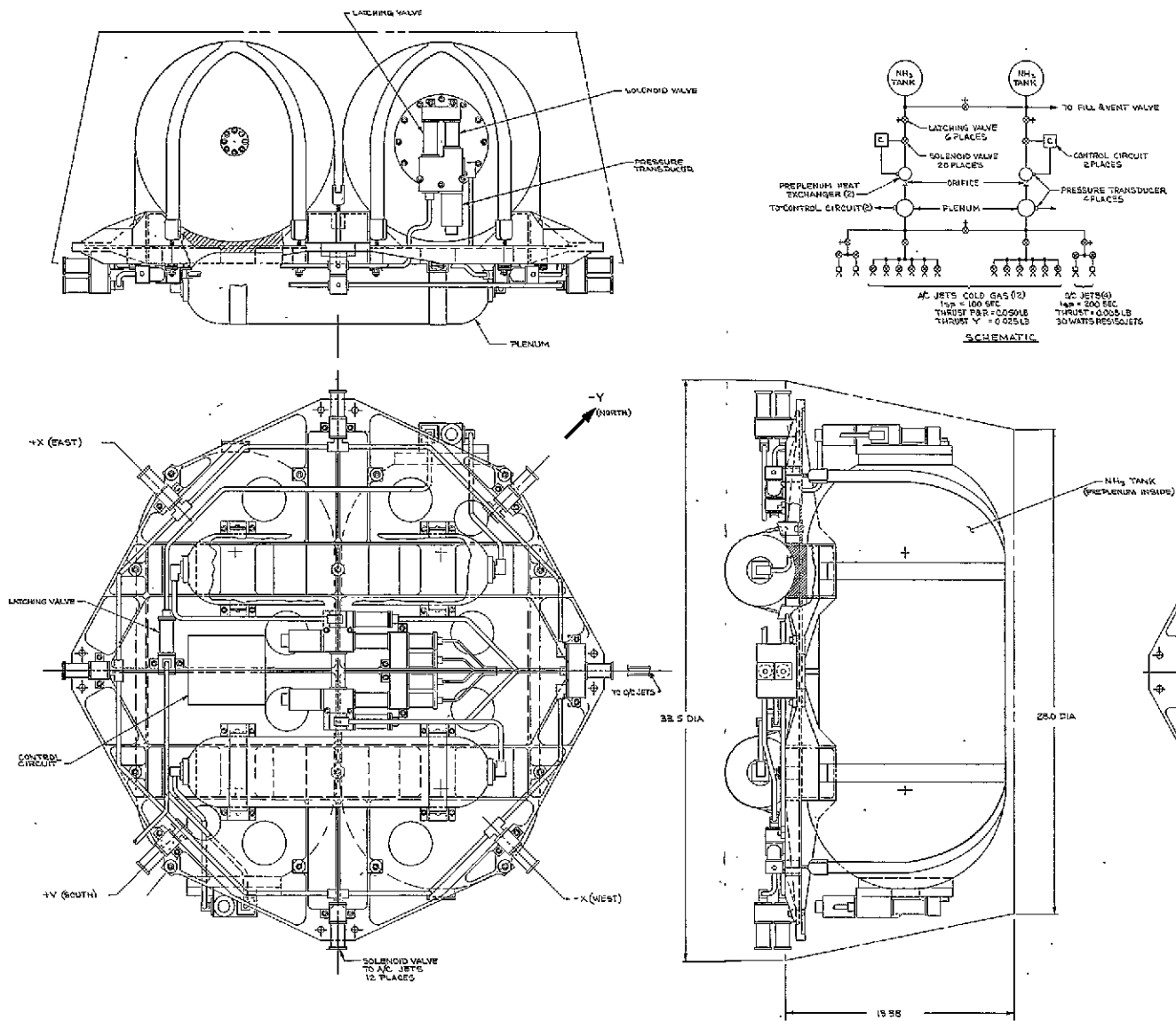
The O/C thruster modules are mounted approximately 3 feet above the top of the EVM on the trusses connecting the antenna and the EVM. These thrusters are aligned to act through the center-of-gravity for orbital translation.

12.4.1.1 Weight

Table 12.4-1 lists the estimated weights of major subassemblies of the APS and gives the total estimated weight with and without propellant.

Table 12.4-1. APS Weight Breakdown

Major Subassembly	Weight (lb)
Regulated pressure feed assembly (2)	50.83
A/C thruster nozzles (12)	1.77
O/C thrusters (2)	3.60
Dry weight subtotal	56.20
Propellant weight	66.00
Total	122.2



FOLDOUT FRAME 1

Figure 12.4-1. Regulated Pressure Feed Assembly -- Cylindrical Tanks

12-29
FOLDOUT FRAME 2

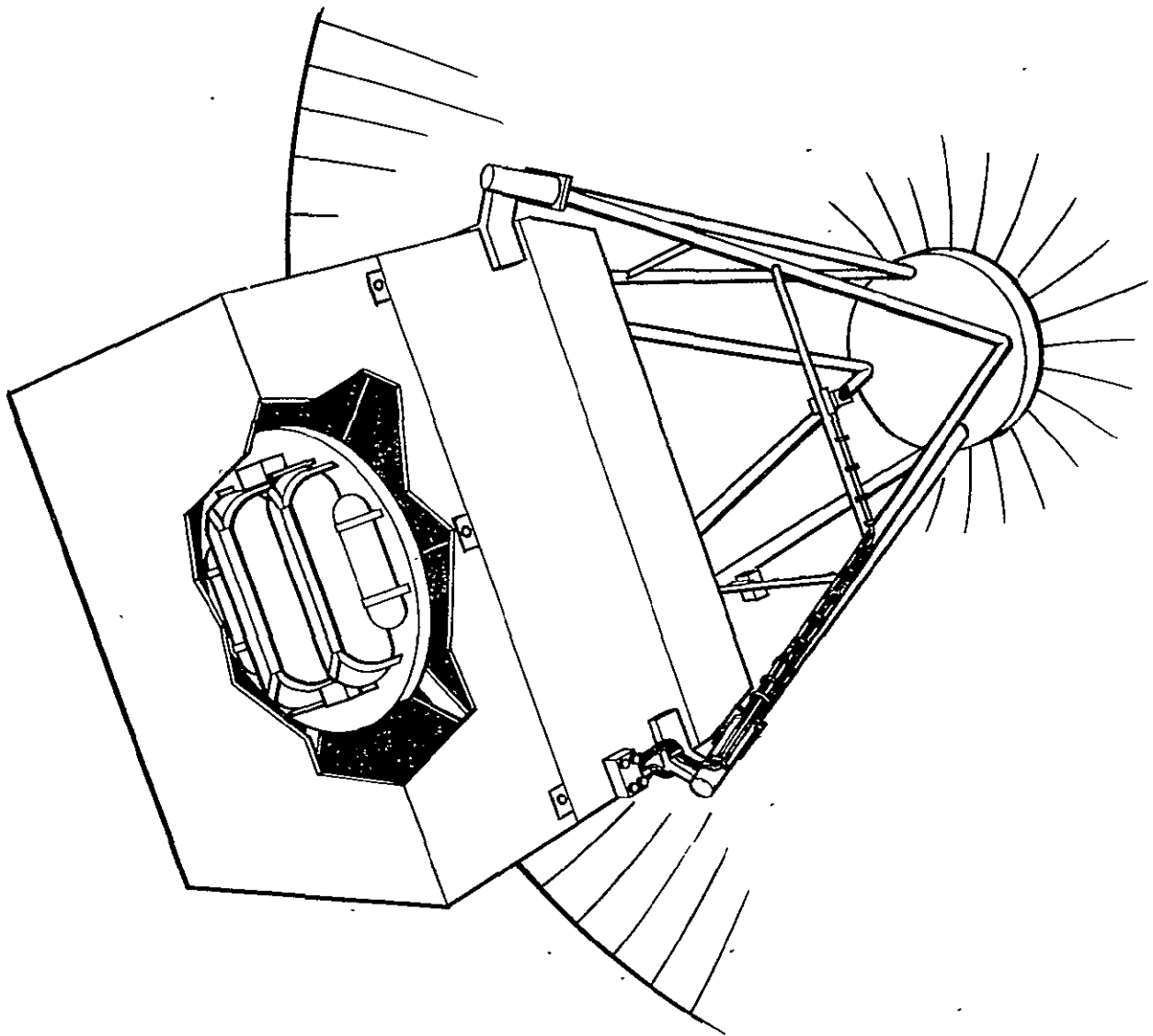


Figure 12.4-2. ATS Spacecraft Showing Location of Auxiliary Propulsion Subsystem in Earth Viewing Module

12.4.1.2 Installation and Mounting

The regulated pressure feed assembly, the four A/C thruster valves, all latching valves, and the 12 A/C thruster assemblies will be manufactured and all tubing connections sealed and assembled while in a contamination-controlled clean room. The O/C thruster and its associated feed tubing can be shipped disassembled from the pressure feed assembly, since the valves for the O/C thrusters are in the feed assembly which is sealed at the clean room facility. If the more detailed interface analysis of the EVM and APS indicates that the APS can be installed with the O/C thrusters and lines completely assembled, this will be done.

The APS will be pressurized with ammonia when shipped to Honeywell. The assembly will be installed in a shipping frame that simulates the spacecraft. This APS assembly will then be carefully removed from the

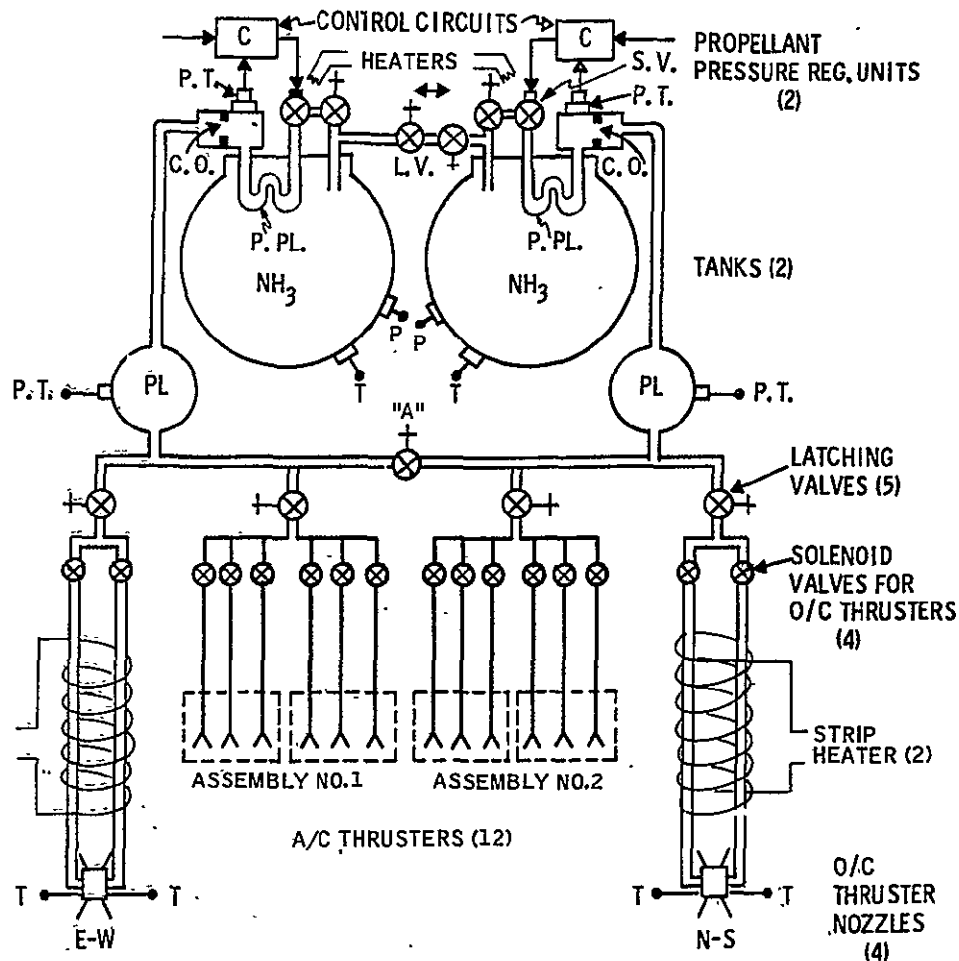


Figure 12.4-3. Major Subassemblies of the APS

shipping frame and transferred to the spacecraft center section and bolted into place. This is the same assembly procedure that was used to install the gas system in the Mariner Mars spacecraft.

12.4.2 COMPONENT PHYSICAL DESCRIPTIONS

The two major component assemblies in the APS are:

- Feed and attitude control thruster assembly
- Orbit control thruster and tube assembly

12.4.2.1 Feed and Attitude Control Thruster Assembly

The major components of each feed and attitude control thruster assembly include:

- Mounting support structure
- Propellant tank
- Plenum tank

- Evaporator heat exchanger
- Pressure control circuit electronics
- Solenoid valves
- Latching valves
- Control pressure transducer
- Fill and drain valve
- Attitude control thruster assembly

The A/C thruster assembly is described under a separate heading, although it is assembled and shipped as a part of the feed and attitude control thruster assembly.

The proposed approach to the feed and attitude control thruster assembly design is shown in Figure 12.4-1. As indicated earlier, this layout drawing was made before more detailed designs were determined for components of the APS. Therefore, the descriptive material which follows is to be considered more complete and up to date than the layout shown in Figure 12.4-1.

The components that make up the regulated pressure feed assembly (RPFA) are all mounted on the APS platform.

12.4.2.1.1 Mounting Support Structure

The preliminary design of the selected approach for the mounting support is shown in Figure 12.4-1. The detailed design remains to be established; however, the estimated weight of the magnesium alloy support is 6.0 pounds.

To minimize the cost of the few pieces needed, the support will be machined from wrought stock rather than making use of other fabrication techniques such as casting.

12.4.2.1.2 Propellant Tank

The propellant tank proposed differs dimensionally from that suggested (see Figure 12.4-1). The diameter was increased to 12.6 inches and the length was decreased to 21.6 inches. This change was made to allow the use of qualified titanium forgings at a reduced procurement cost.

In addition, polar mounting will be used for these tanks. With this mounting technique, one end of the tank is held securely while the mounting bar of the other end is allowed to move in the axial direction within a teflon or equivalent bushing. The tanks are thus unrestrained and no further compensation needs to be provided for dimensional changes due to pressure and temperature variations.

The estimated finished weight of the titanium propellant tank is 6.2 pounds, and the estimated internal volume is 2005 cubic inches.

12.4.2.1.3 Plenum Tank

A cylindrical or a spherical-type plenum tank is suitable. Final packaging details will determine the choice. The cylindrical titanium plenum tank has a volume of 200 cubic inches, is 4.1 inches in diameter by 17.75 inches in length, and has an estimated weight of 0.76 pound. The spherical plenum tank is 7.3 inches in diameter.

The plenum tank mounting is similar to that used to mount the propellant tank.

12.4.2.1.4 Evaporator Heat Exchanger (Preplenum)

The flow requirement for the ATS-F&G APS is approximately 10 times the flow of the ATS-D APS; the former therefore required more heat exchanger surface area for heat transfer. The Avco approach uses 1/4-inch stainless-steel tubing wound on a three-point supporting structure.

Other designs, such as the TRW capillary heat exchanger, use long capillary tubes for the preplenum.

12.4.2.1.5 Control Circuit Electronics

The control circuit electronics for both half-systems are contained in a package of approximately 4 x 4 x 7 inches. The weight of this package and the APS cabling harness is 1.9 pounds.

12.4.2.1.6 Solenoid Valves

Both the large (2-watt) Wright Components, Inc. model 15457, and the small (1-watt) model 15398 solenoid valves have the same external dimensions, and the same estimated weight (0.25 lb). The model 15398 was qualified and flown on the ATS-D&E program. These valves have an EPR seat with proven ammonia compatibility.

12.4.2.1.7 Latching Valve

As a result of the preliminary latching valve evaluation funded by Honeywell, the Wright Components, Inc. model 15478 latching valve is recommended.

12.4.2.1.8 Control Pressure Transducer

The pressure transducer used for monitoring the evaporator heat exchanger is a derivative of the model 3255 manufactured by Servonic Instruments, Inc.

The possibility of changing just the pressure port fitting to provide a low leak seal with better control of contamination (to avoid threads inside the O-ring seal) is being investigated.

12.4.2.1.9 Fittings

All flow connections will be welded or flanged. AN and MS fittings will be avoided in the interest of contamination control. Simple flange seals with O-rings have proven very reliable on the ATS-D&E program. All O-rings were physically examined; samples were chemically analyzed and carefully cleaned. This procedure has resulted in exceptionally low leakage with the versatility of manufacture and assembly.

12.4.2.2 A/C Thruster Assembly

The A/C thruster assembly consists of a model 15457 solenoid valve, feed tube, and nozzle. The solenoid valve is mounted in the control APS package as part of the feed and attitude control thruster assembly, while the nozzle is mounted near the external surface of the EVM. The feed tube connects the valve outlet to the nozzle inlet.

The stainless-steel nozzle will be mounted on the EVM with a three-point method of attachment so that thermal control washers and shims can be used if necessary. The nozzle throat diameters are approximately 0.045 and 0.031 inch, with an exit area-to-throat area ratio of 100 and an exit cone divergence half-angle of 15 degrees. The ammonia temperature, pressure, and Reynolds numbers in the nozzle throat permit the use of standard nozzle design. The tube connection to the nozzle inlet will be either welded or brazed.

Functional performance is acceptable with standard 1/8-inch stainless-steel tubing for the orbit control feed between the valves and nozzles.

The roll/pitch lines will consist of 11 inches of 3/16-inch tubing, and the yaw lines will consist of 23 inches of 3/16-inch tubing. These line sizes should result in thrust rise times of approximately 16 and 32 milliseconds and fall times of 30 and 60 milliseconds respectively. If the minimum equivalent square impulse width is assumed to be 100 milliseconds, the repeatability of the minimum impulse bits should be no problem with either yaw or pitch thrusters. The lines will be tied down where possible. Sufficient bends will be provided to allow for platform distortion with excessive pull at the tube ends.

The tube connection (to the valve) will be either welded or brazed to a stainless-steel flange, with provision for an O-ring seal to the valve outlet.

12.4.2.3 O/C Thruster and Feed Tube Assembly

A preliminary layout of the O/C thruster and feed tube assembly is shown in Figure 12.4-4. Vector position is adjusted vertically with spacers. The thrust vector may also be adjusted ± 5 degrees in any direction from the nominal normal by turning the ball mount and vector adjusting screws.

The design of the O/C thruster assembly is shown in Figure 12.4-5. The assembly incorporates two functionally independent thrusters with a common mounting bracket. Thruster alignment is provided by a ball

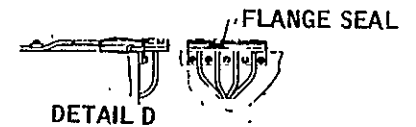
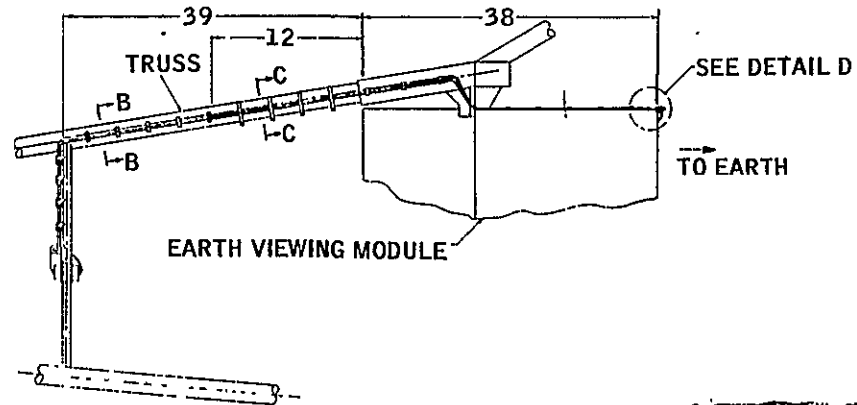
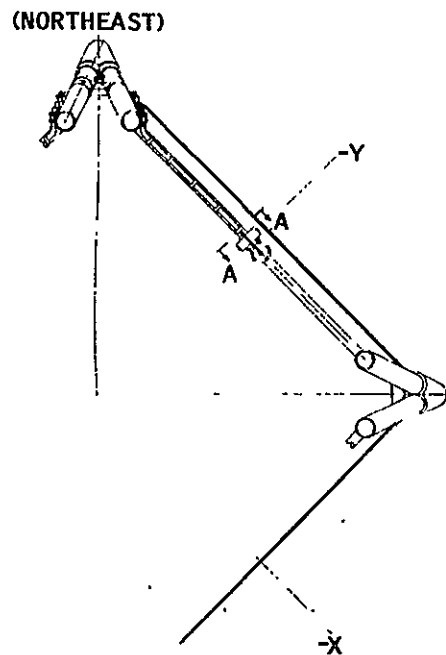
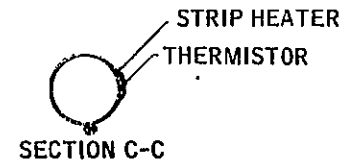
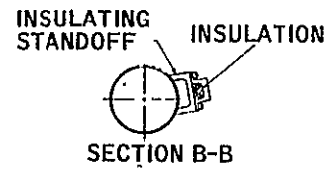
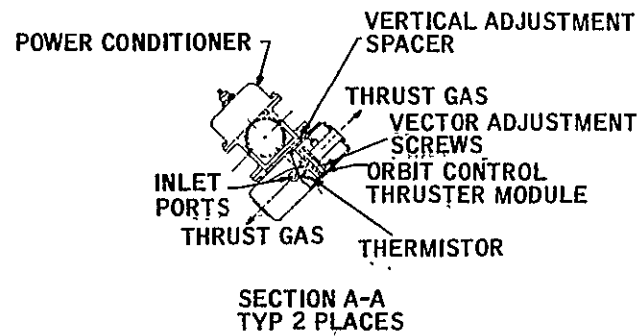


Figure 12.4-4. Location and Mounting of Orbit Control Thrusters and Feed Tubing

and socket-type mounting. This feature allows angular adjustment to any position within 5 degrees of the nominal thrust axis. Alignment on the spacecraft will be accomplished by optical means. The thruster design will provide an alignment reference so that a mirror can be attached to the nozzle which will be perpendicular to the thrust axis to within ± 0.5 degree. The thruster bracket includes slots to allow a total thruster adjustment of 1 inch along the spacecraft axis of symmetry to account for uncertainties in the designed spacecraft center-of-gravity. Propellant lines are designed for sufficient flexibility to allow this adjustment.

Each O/C thruster assembly will have a single electrical connector with a three-pin configuration. The return lead from the heaters will be common to both thrusters to minimize current limiting and temperature sensing electronic components. Since the external portion of each thruster is electrically conducting, the thrusters are electrically isolated from the spacecraft structure. Individual pneumatic connections for each thruster will be provided in the form of mechanical fittings. Pneumatic lines are electrically isolated from the thrusters as shown in Figure 12.4-5.

The heater element (see Figure 12.4-6) will consist of a double helix of resistance wire similar to that found in ordinary household light bulbs. The heater surface area will be optimized for efficient heat transfer to the propellant. The wire diameter will be selected for rigidity, electrical resistance, and tolerance to sublimation.

The resistance value will be compatible with the spacecraft supply voltage. Fortunately, the most promising materials for use with ammonia propellant in the temperature range of interest have sufficiently high dielectrical resistivities to result in practical heater element designs. Resistivities of the materials under consideration are shown in Figure 12.4-7. The use of a noble metal alloy (Rh or Rh-1r) may allow functional checkout of the heater element in the atmosphere.

The development thrusters have used alumina electrical insulators which also direct the propellant flow near the heater element. Unfired alumina, which is readily machinable, is commercially available and is fired following machining...

All other thruster components will be fabricated from conventional oxidation-resistant materials, such as stainless-steel, nickel, or the superalloys. All are readily machinable and easily joined by brazing or electron beam welding.

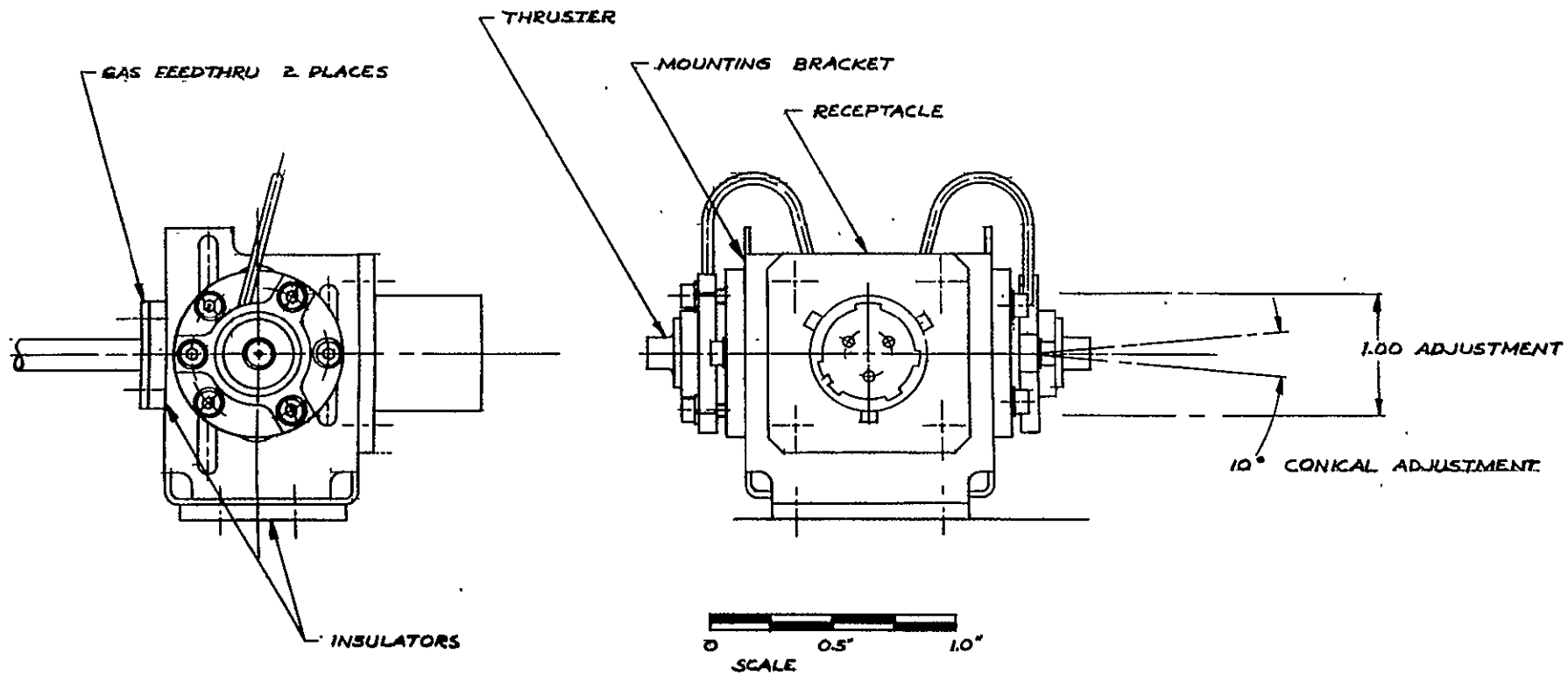


Figure 12.4-5. Orbit Control Thruster Design (TRW)

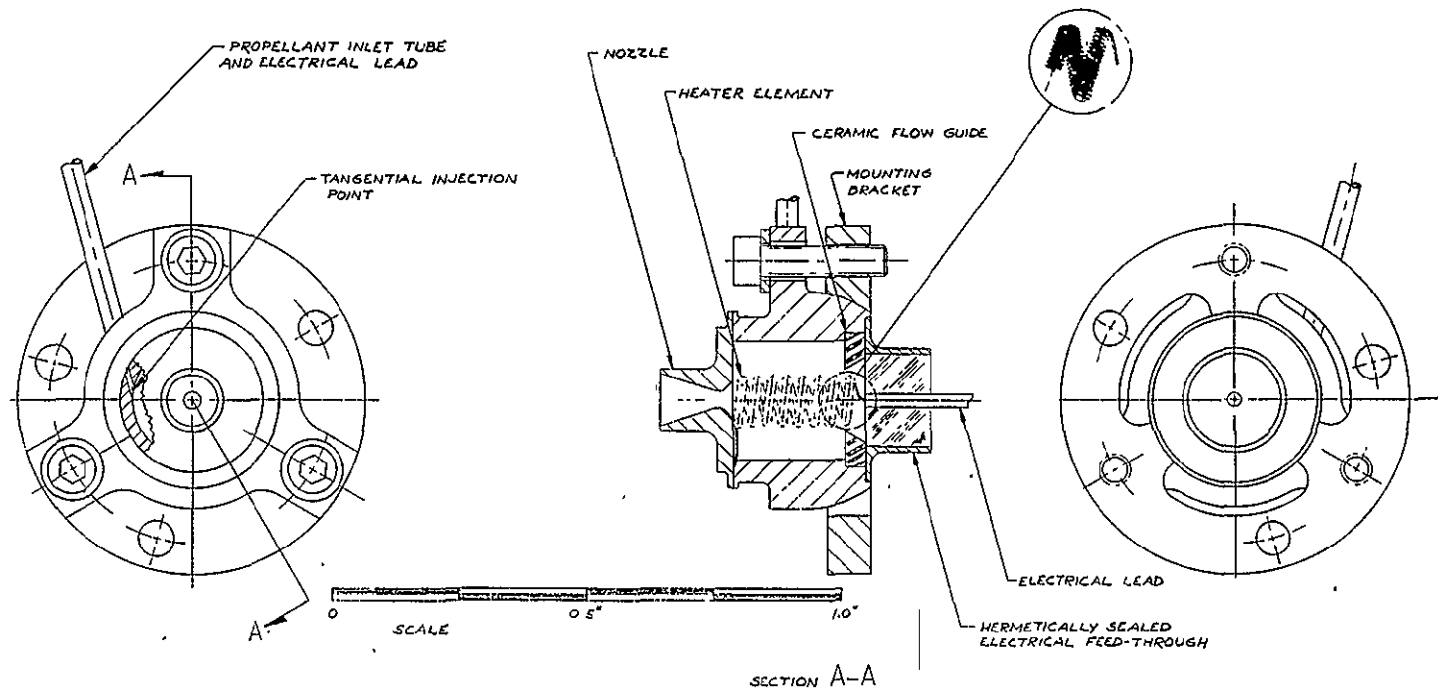


Figure 12.4-6. Orbit Control Thruster Design (TRW)
Showing Location of Heater Element

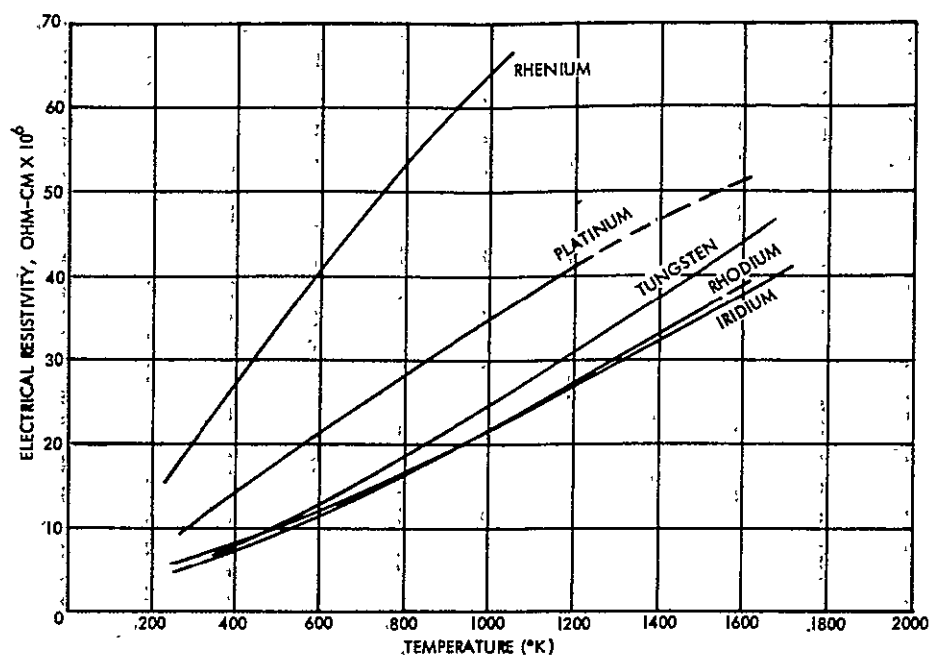


Figure 12.4-7. Electrical Resistivities of Heater Materials

SECTION XIII

TELEMETRY AND COMMAND SUBSYSTEM

13.1 INTRODUCTION

The Telemetry and Command (T&C) Subsystem is the lifeline of the ATS F&G spacecraft and is the connecting link between the operations controller and spacecraft functional subsystems. To perform its function properly over the life of the mission, the T&C Subsystem must be evolved with an excellence of design and attention to detail which enables positive control of spacecraft functions under the most adverse conditions.

A generalized diagram of the spacecraft/ground station orientation is shown in Figure 13.1-1. (A more detailed summary of the ground station configuration is shown in Appendix E of the Technical Proposal.) In the normal mode of operation, commands generated within ATS Operations Control Center (ATSOCC) at GSFC are teletyped (with phoned verification) to the transmitting ground station. The commands are then manually set into octal thumbwheels on the ATS Command Console. When the transmit button is pressed, the console automatically inserts the sync bits and the dialed-in-spacecraft and command addresses into the proper bit positions in the 64-bit command word format as it is stepped out serially to the transmitter modulator. A timed-tone execute capability requirement which is used for an optimum-attitude control experiment has been identified for the ATS F&G spacecraft. This capability will require a real-time link between the computers at GSFC and the transmitter of at least one ground station (Rosman).

The telemetry data is presently received from the spacecraft by the ground stations and recorded on an analog tape recorder. The ground stations have some quick-look capability. The analog tapes are reduced weekly at GSFC. Because of the increased complexity of the ATS F&G spacecraft and its experiments, it will be necessary to establish a real-time type status board at ATSOCC-GSFC, along with either real-time relay of the telemetry data from the receiving sites or a receiving facility directly at GSFC. It is also recommended that any recording of the telemetry data be performed on a digital tape recorder.

A requirement has also been established for the ATS F&G T&C Subsystem to have the capability of acting as a transponder for a voice-bandwidth analog mode. To allow a low-powered, transportable ground transmitter to exercise this mode, the high gain parabolic reflector aboard the spacecraft has been used advantageously to receive this signal. Normal command functions are received only by spacecraft receivers associated with omnidirectional antennas.

The T&C Subsystem design described in this report was configured to achieve the highest possible reliability and to maintain simplicity and flexibility. No single failure will prevent commanding or telemetering any of the defined functions aboard the spacecraft. Multiple-failure survivability design is used at critical circuit points. The design is consistent with the Aerospace System Standards and is also responsive to special mission functions, such as the SAMOC/SAPPSAC (timed tone) and special data (voice-bandwidth analog) modes. The subsystem has been designed to remain operationally similar to ATS A-E spacecraft and to offer a data-word command

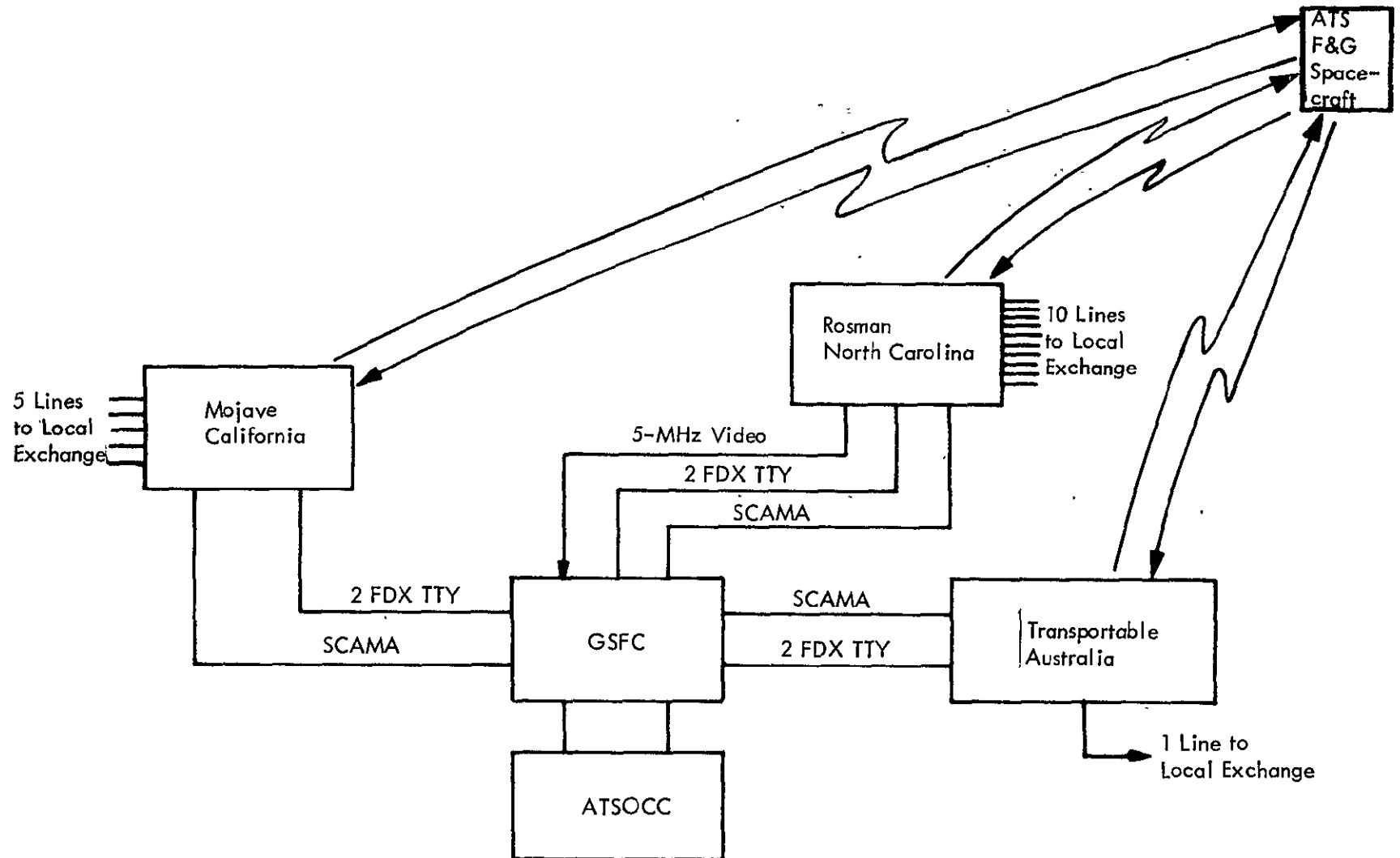


Figure 13.1-1 ATS Communication Networks

capability, multiple timed-tone or voice bandwidth analog modulation of the command carrier, and simultaneous transmission of normal telemetry, EME data, and voice-bandwidth analog data on the telemetry carrier. The subsystem is also designed to employ standardized outputs from the command distribution unit (CDU) and inputs to the Data Acquisition and Control Unit (DACU), while continuing to satisfy specialized subsystem requirements such as unique data word transfer to the Digital Operational Controller (DOC), telemetry synchronization pulses to the ACS and interferometer, timer function for the experiments, and time code generation.

The resultant T&C design represents a highly reliable T&C Subsystem, which complies with Aerospace Systems Standards, satisfies all general and specialized ATS F&G T&C requirements, offers readily implemented flexibility and expansion capability, has minimum departure operationally from ATS A-E (minimizing personnel retraining requirements), and maximizes use of presently available ATS A-E ground hardware (transmitters, receivers, DHE) for minimum cost impact on the ground station. A block diagram of the T&C Subsystem is shown in Figure 13.1-2.

Table 13.1-1 highlights the functional parameters of the command subsystem. The command subsystem receives, decodes, and distributes discrete and data word commands to the spacecraft subsystems. It has a capacity of 512 redundant discrete command addresses and 63 data word addresses. Each data word transmission carries nine bits of data. It employs one of two assigned VHF carriers which is AM modulated by the FSK 0 and 1 command tones. The 0 and 1 tones are themselves 50-percent AM modulated by a 128-Hz sine wave clock, which is demodulated in the Command Decoder/Distribution (CDD) unit and used for all timing within the CDD and for data word transfer to the subsystems. Before command execution, ground personnel verify that the proper command was received by the CDD via telemetering of the 6-bit data word address, 9-bit command word, and a single bit for "true" or "false" 8-bit spacecraft address. Two redundant CDDs are employed, each with its own spacecraft address, so that only one is employed at any one time. All execution of discrete and data word commands occurs only upon receipt of an execute tone from the ground station.

Seven additional execute tones can, in a special mode, amplitude-modulate the command carrier and be accepted and decoded by the CDD for timed-tone control capability of the Attitude Control System (ACS) jets and wheels. A discrete command is used to enable the timed-tone detectors. Six of the tones are used for real-time control of the addition and subtraction jets on the three attitude axes. The seventh is used as a holding tone and must be present during the entire time-toned mode of operation. Absence of this holding tone disables the timed-tone detectors and logic.

A third signal, amplitude-modulated by a voice-bandwidth analog signal, can be present on the second VHF carrier frequency. This signal is demodulated by the receiver connected to the prime focus feed and switched via the Data Switching Unit (DSU) to a telemetry transmitter for retransmission to the ground via the prime focus feed antenna. This mode of operation provides the simplex push-to-talk voice coordination between ground stations, as required in support of some planned experiments.

The Telemetry Subsystem functional parameters are highlighted in Table 13.1-2. The DACU is a fixed-format encoder with dwell capability. It accepts digital and analog data from the spacecraft subsystems and experiments, digitizes each analog sample in a 9-bit ADC, and sequences each telemetry point into its pre-assigned position in the telemetry format. The DACU uses output from the spacecraft

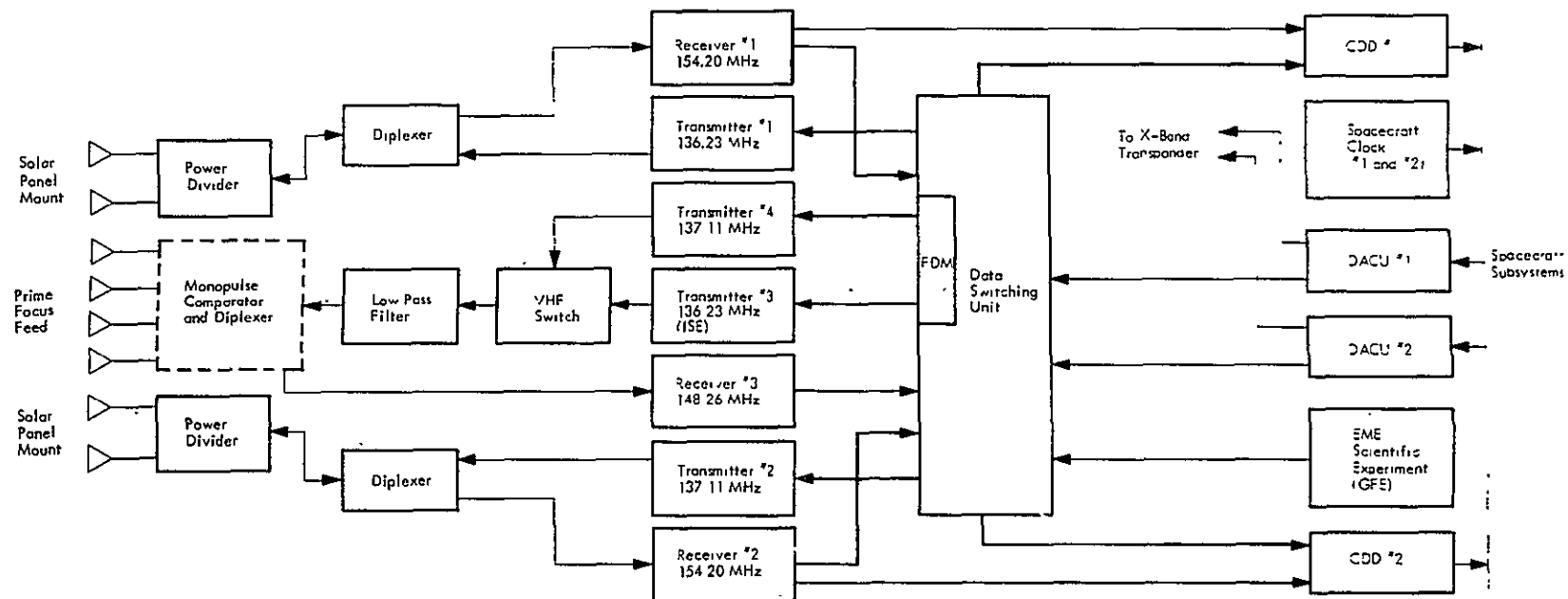


Figure 13.1-2. T&C Subsystem Functional Block Diagram

Table 13.1-1. Command Subsystem Parameters

Characteristics	ATS F&G
Modulation	FSK/AM/AM
Frequencies	148.26 MHz, 154.2 MHz
Spacecraft Antennas	Omnidirectional mounted on solar panels. Directional via prime focus feed and parabolic reflector.
Clock	128 Hz AM modulation of 0 and 1 tones
Command Word Length	64 bits
Bit Rate	128 bps
Command Word Format	
Introductory 0's	27 bits (1-27)
Sync bit(s)	2 bits (28, 29)
Spacecraft Address	8 bits (30-37)
Data Word Address	6 bits (38-43)
Command Address	9 bits (44-52)
Trailing 0's	12 bits (53-64)
Command Discrete Addresses	512 max
Data Word Addresses	63 max
Verification	Telemetry 16 bits
Command Execution	Execute Tone
Digital Word Capabilities	9- bit word
Digital Word Execution	Data word is stepped to the selected subsystem as it is received by the command decoder. It is acted upon by the subsystem only upon receipt of an execute tone from the ground which is transferred to the same subsystem by the CDD.
Timed-tone Control	7 additional execute tones for SAMOC/SAPPSAC jets and wheels control.
Technology	Optional qualified IC & MSI technology where weight, power & volume savings and reliability increase is required.
ATSOCC/Ground Station Data Link	Command Link for real-time control of Timed-tone Execute tones.
Command Console	Similar to present OGO/ATS command console with addition of two octal thumbwheels for data word address insertion and modification for SAMOC/SAPPSAC timed tone command capability.
Special Data Mode	AM modulation of the command carrier by voice bandwidth analog signals.

Table 13.1-2. Telemetry Subsystem Parameters

Characteristics	ATS F&G
Frequencies	136.23 MHz, 137.11 MHz
Spacecraft Antennas	Near omnidirection mounted on solar panels. Directional via prime focus feed and parabolic reflector.
Modulation	PM on omni-associated transmitters, FM/PM or PM on prime focus feed-associated transmitters.
Formatting	Fixed format with dwell capability
Bit Rate	390 bps nominal
Word Length	9-bit
Minor Frame Length	128 words
Minor Frame Period	Approximately 3 seconds
Subcommutation	16 words, 16 deep
Total Format Capacity	368 nine-bit words. Each word carries one digitized analog telemetry point or nine digital telemetry points.
Analog Channels	304
Digital Channels	513
Sync	First 3 words
Subframe Identification	Word 4
Calibration	Words 5-7
Command Verification	Words 8-11
Time Code Tag	Four words, once per minor frame
High Rate Telemetry, Special Data Mode	2000-bps EME data and voice bandwidth analog signal in the prime mode of operation are frequency division multiplexed along with the normal telemetry information onto the carrier of one of the two telemetry transmitters associated with the prime focus feed antenna (FM/PM). In a backup mode, either the EME or voice bandwidth analog data can modulate the prime focus feed-associated transmitter directly (PM), and either the EME or normal telemetry data can modulate the omni-associated transmitters directly (PM).
Technology	Optional qualified IC and MSI where weight, volume, power, and reliability indicate a need.
Ground Station/ATSOCC Data Link	Real-time link and/or VHF antenna, receivers, and DHE at ATSOCC desirable.

clock as an input to frequency dividers and logic functions which generate the switching functions required to establish the telemetry format. The switches driven by this format generator are MOSFET gates. In the dwell time, the first 16 words are repeated in each minor frame, irrespective of which word is being dwelled on. Thus, sync, status, calibration, command verification, and other telemetry points of prime importance are present for ground station use in each minor frame, independent of DACU mode of operation.

Four telemetry transmitters are employed: two at each of the two assigned frequencies. Two transmitters are associated with the VHF antenna at the prime focus feed and two associated with the two omni-antennas on the solar panels. The transmitters are individually commanded ON or OFF from the ground.

The prime mode of operation is FM/PM, with normal telemetry, EME, and voice bandwidth analog data frequency division multiplexed onto the carrier of one of the two transmitters associated with the prime focus feed. This mode takes advantage of the parabolic reflectors high gain to give both the EME and voice bandwidth analog channels adequate margin for high-quality link performance. Normal telemetry or EME data can modulate either of the omni-associated transmitters (PM) in backup modes, and EME or voice bandwidth analog data can modulate either of the prime-focus feed associated transmitters (PM). Normal telemetry data can also use the X-band transponder as a backup downlink.

The voice bandwidth analog channel can be used as a real-time monitor of the command channel. This feature could be used to assess the noise in the command channel during periods of no command transmission, or it could be used to monitor receipt of either the normal command signal or the SAMOC/SAPPSAC execute tones. Thus, the SAMOC/SAPPSAC experiment could be terminated if interference signals were present which could inadvertently actuate the enable ACS jets or wheels.

13.2 T&C SUBSYSTEM DESCRIPTION

13.2.1 FUNCTIONAL DESCRIPTION

This subsection describes the functional operation of the T&C Subsystem when operating on discrete commands, data word commands, and in the timed-tone mode, normal and dwell telemetry modes, high-rate EME telemetry mode, and special data mode (voice-bandwidth analog channel).

The T&C Subsystem shown in Figure 13.1-1 will employ two VHF command frequencies (148.26 MHz and 154.20 MHz) and two VHF telemetry frequencies (136.23 MHz and 137.11 MHz). The primary command frequency, 154.20 MHz, will be used for all command functions and will be received by command receivers 1 and 2, which are connected to omnidirectional, monopole pair antennas located on the solar panels. The secondary command frequency will be used for command backup special data mode operation (voice bandwidth analog signal) and will be received by command receiver 3 which is associated with the prime focus feed antenna.

The primary telemetry link will be a frequency division multiplexed (FDM) link associated with transmitter 3 or 4 and the prime focus feed antenna. The FDM unit is designed to multiplex normal data (output of the DACU), EME data, and the Special Data functions onto one downlink carrier (136.23 MHz if transmitter 3 is used and 137.11 MHz if transmitter 4 is used).

This scheme allows simultaneous transmission of the EME and Special Data functions. Adding the normal telemetry function to the multiplexed channel also augments the link margin for this function. A detailed description of the FDM link's modulation indices, modulating frequencies, and link performance is given later in this section. If the spacecraft is maneuvering so that the directional beam is no longer pointing toward the earth, the normal telemetry will be switched by ground command to transmitter 1 or 2 for transmission to earth via the omnidirectional antennas. The DSU can also be commanded to switch the EME to either transmitter 1 or 2 (whichever is not being used for the normal telemetry function) for transmission to earth in a degraded mode (higher bit error rate). The Special Data function cannot be switched to the omni-associated transmitters because the link has inadequate margin to support a recoverable voice-bandwidth analog signal.

The FDM unit is housed in the DSU. In case of a failure in the FDM unit, either the EME function or the Special Data function can be switched directly as the input to transmitter 3 or 4. The normal telemetry function would be commanded to one of the omni-associated transmitters.

13.2.1.1 Command Function

A functional block diagram of the Command Subsystem is shown in Figure 13.2-1. The Command Subsystem will receive, decode, and distribute discrete and data word commands to the spacecraft subsystems. It has a capacity of 512 discrete command addresses and 63 data word addresses. Each data word transmission carries nine bits of data. It employs a VHF carrier which is AM-modulated by the FSK 0 and 1 command tones. The 0 and 1 tones are AM-modulated by a 128-Hz sine wave clock, which is demodulated in the CDD and used for all timing within the CDD and for data word transfer to the subsystems. Before command execution, ground personnel verify that the proper command was received by the CDD via telemetering

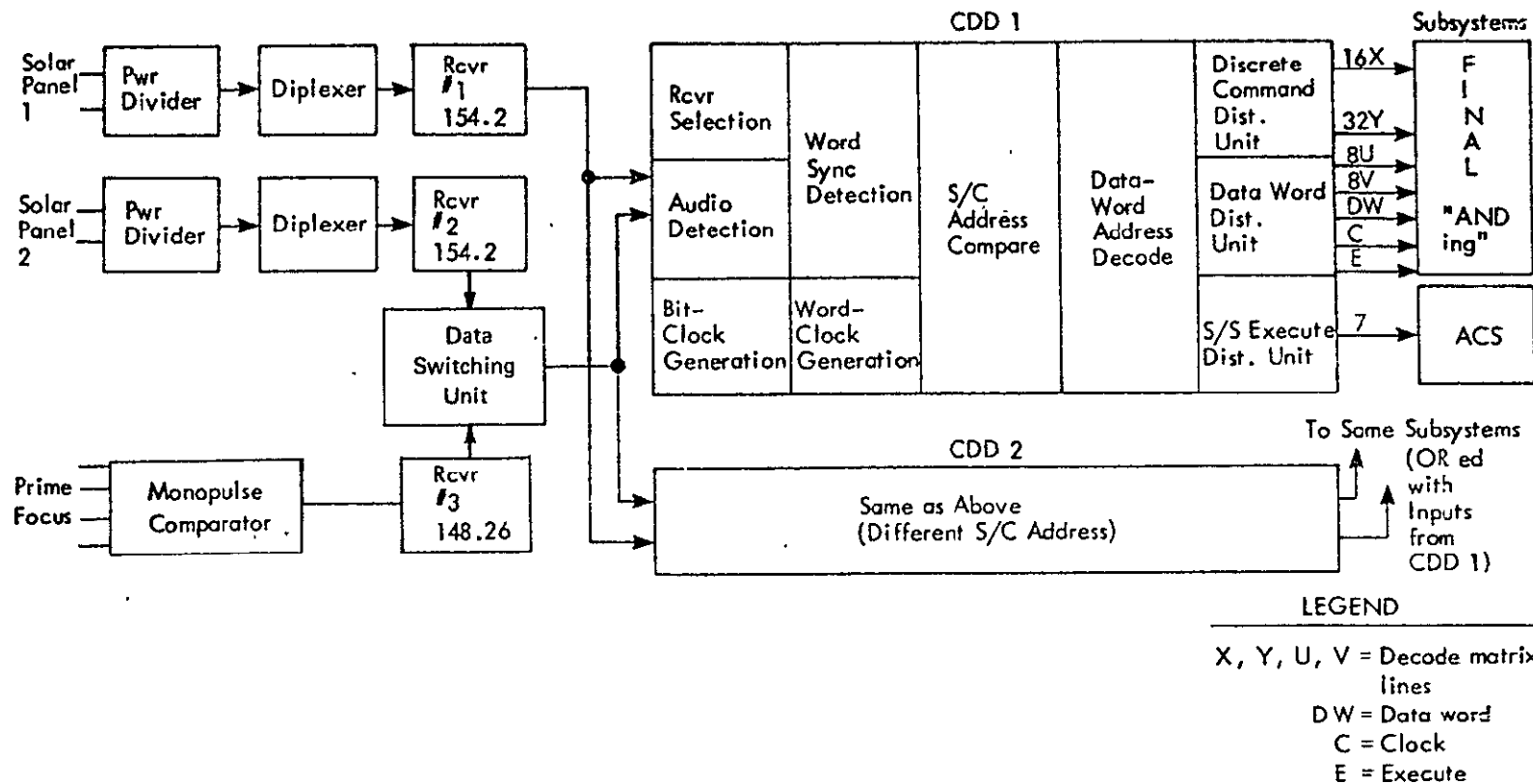


Figure 2.1-1. Command Subsystem Functional Block Diagram

of the data word address, 9-bit command word, and a single bit for true or false 8-bit spacecraft address. Two redundant CDDs are employed, each with its own spacecraft address, so that only one is employed at any one time. All execution of discrete and data word commands occurs only upon receipt of an execute tone from the ground. In a special mode, seven additional execute tones can AM-modulate the command carrier and are accepted and decoded by the CDD for timed-tone control capability of the ACS jets and wheels. A discrete command is used to enable the time-tone detectors. Six of the tones are used for real-time control of the addition and subtraction jets on the three attitude axes. The seventh is used as a holding tone and must be present during the entire time-toned mode of operations. Absence of this holding tone disables the timed-tone detectors and logic. The secondary frequency command carrier is amplitude-modulated by a voice-bandwidth analog signal. This signal is demodulated by the command receiver and is switched via the DSU to the FDM unit and telemetry transmitters 3 or 4 for retransmission to the ground. This signal cannot activate the CDDs.

The Command Word format is shown in Figure 13.2-2. This figure indicates the ranges of the octal thumbwheels required on the ground command console to set in the spacecraft address, discrete/data word address, and command address/data word.

Table 13.2-1 lists the 10 tone frequencies assigned to the 0, 1, execute, and SAMOC/SAPPSAC tones. An EMI computer analysis of these tones showed this assignment to be free of second-order intermodulation product interference. The study is described in Appendix B of the Technical Proposal.

Table 13.2-1. Command Tone Frequency Assignment

<u>Function</u>	<u>Frequency (Hz)</u>
Clock	128
0	7400
1	8600
Execute	5790
Holding (SAMOC/SAPPSAC)	1447
+Pitch (SAMOC/SAPPSAC)	3000
-Pitch (SAMOC/SAPPSAC)	4550
+Roll (SAMOC/SAPPSAC)	5155
-Roll (SAMOC/SAPPSAC)	3621
+Yaw (SAMOC/SAPPSAC)	1025
-Yaw (SAMOC/SAPPSAC)	1860

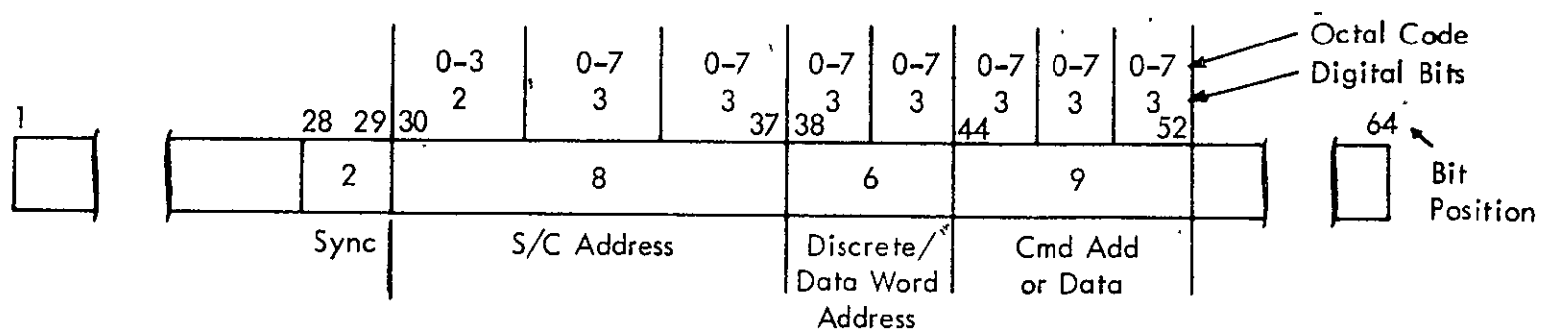


Figure 13.2-2. Command Word Format

13.2.1.2

Telemetry Function

A functional block diagram of the Telemetry Subsystem is shown in Figure 13.2-3. In the normal telemetry mode, one of the two DACUs is commanded ON, along with one of the redundant spacecraft clocks. The DACU samples the input lines from the digital and analog telemetry points in a predetermined sequence, digitizes the analog levels sampled in a 9-bit ADC, and serializes the digital bits into a nominal 400-bps biphase bit stream, which is switched by the DSU to the FDM unit or one of the two transmitters associated with the omnidirectional antennas under control of ground command.

A DACU enters the dwell mode under control of ground command, which uses a data word command to select the DACU dwell word. In this mode, the first 16 words of the 128-word minor frame are still telemetered as in normal mode (to retain sync, status, calibration, in a command verification) but the remaining 112 words are repetitions of the word dwelled upon. When this is a minor frame word, the selected word is repeated 112 times each minor frame (3 seconds). In the case of dwell on a subcom, the 16-word subcom is repeated 7 times in the 3-second minor frame period. The DACU returns from the dwell mode to normal mode upon ground command. Both DACUs can be activated simultaneously, and if desired, one can be in normal mode while the other is in dwell, both can be in normal mode, or both can be in a dwell mode on the same or different words and transmitted via different telemetry transmitter frequencies.

High-rate (1800-bps) telemetry data from the EME and voice-bandwidth analog (Special Data mode) data also modulate the downlink carriers. The prime mode of operation employs a frequency division multiplexed combination of the three modulation sources. The FDM signal phase modulates one of the two transmitters associated with the prime focus feed antenna. In backup modes the normal telemetry and EME data can modulate the omni-associated transmitters, and the Special Data signal and EME data can directly modulate the prime focus feed-associated transmitter.

13.2.1.3

Command Requirements

The following tabulation summarizes the discrete and data word address requirements of each subsystem. A complete listing of each command function is shown in Appendix A of the Technical Proposal.

	<u>Discrete</u>	<u>Commands</u>
		<u>Data Word Addresses</u>
Attitude Control	156	10
Telemetry and Command	43	3
Communications	8	1
Electrical Power	56	0
Structures	5	0
Experiments	<u>89</u>	<u>6</u>
Totals	357	20

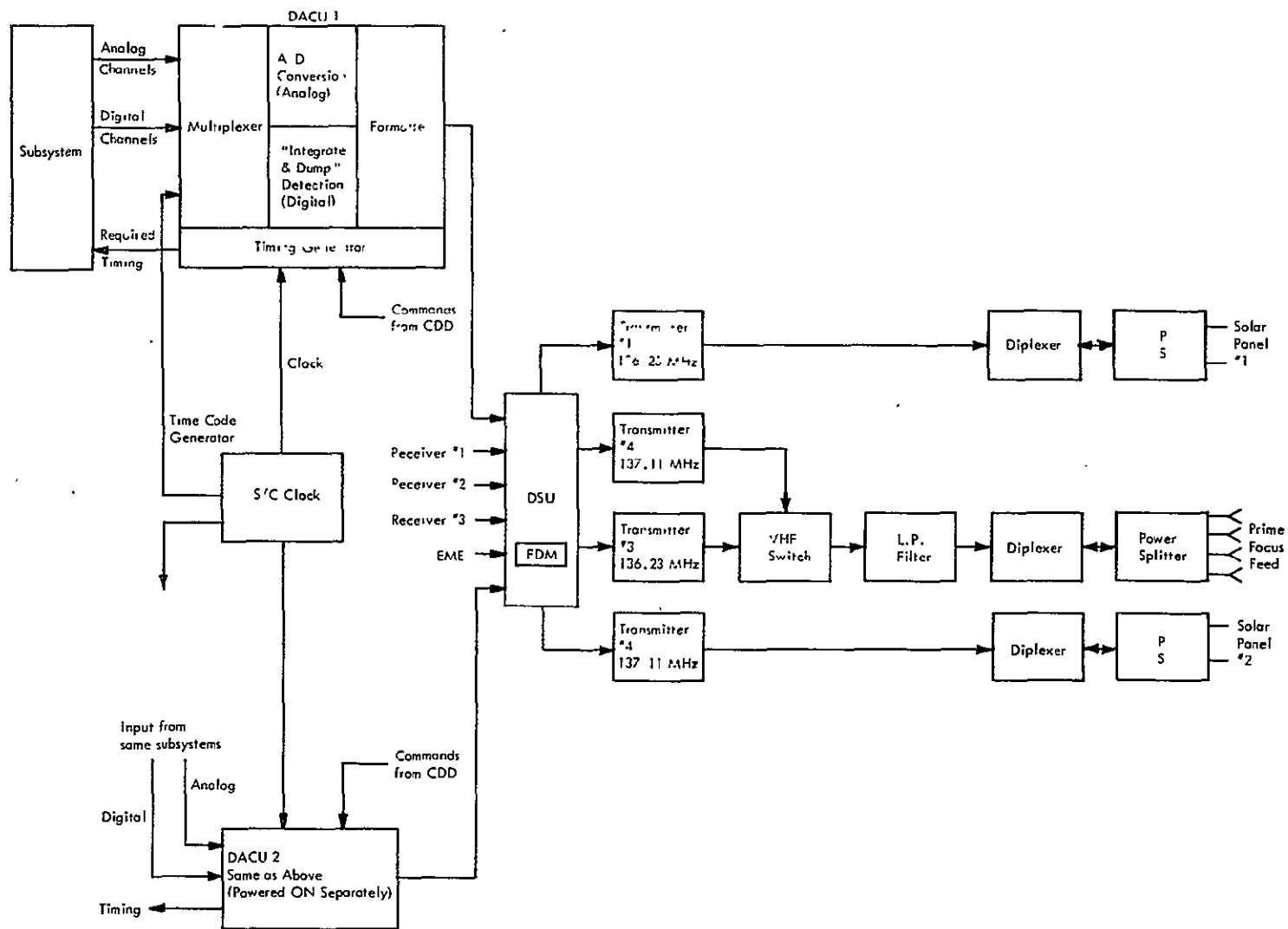


Figure 13.2-3. Telemetry Subsystem Functional Block Diagram

13.2.1.4

Command Address Assignments

Figure 13.2-4 shows the assignment of octal command addresses by A number for discrete commands. The assignments are made on the 16 x 32 matrix, which represents the operation of the 16-X and 32-Y lines which are ANDed in the subsystems as the final stage of address decoding. Assigning the addresses as shown in square or near-square blocks results in the minimum parts count and highest reliability figures for the array command distribution system chosen for ATS (on the basis of the tradeoff study shown in subsection 6.2.2 of the Technical Proposal).

The address assignments are seen to be broken into two segments on the 16 x 32 array. The left segment represents assignments to commands associated with functions other than ON-OFF commands to the Line Interface Circuits (LICs), and the right segment is commands to these LICs. They must be separated, since the LIC commands have ground isolation requirements which demand additional circuitry in the CDD that is not required by non-LIC commands. The address for each command is found in Appendix A of the Technical Proposal.

For example, A153 command functions are assigned octal command addresses 252-254, 272-274, and 312-314. When the spacecraft receives command address 010111011, (273), it decodes the first 5 bits as Y₁₁ and the last four bits as X₁₁. The Y₁₁ and X₁₁ lines are enabled during presence of the execute tone, and the function at address 273 in A153 is activated.

The size of the command group allocated to each A number (which is assigned more than two commands) includes several spare assignments. This spare allocation can be used in the future by the simple addition of an AND gate in the subsystem (e.g., one-fourth of an IC which measures about 1/4 x 1/2 x 1/8 in.). No change would be required in the CDD electronics or the subsystem terminator circuitry. This versatility (ease of expansion or change) is a characteristic of the array distribution scheme. Figure 13.2-5 shows the data word address assignments. Here the 6-bit data word address is broken down into two 3-bit segments which are decoded to enable 1 of 8 U lines and 1 of 8 V lines. Octal address 55 has been designated the "discrete" tag, which tells the CDD the following nine bits are a discrete command address rather than a data word.

13.2.1.5

Telemetry Requirements

The following tabulation summarizes subsystem analog and digital telemetry requirements of the ATS F&G spacecraft. A complete tabulation of each telemetry function is shown in Appendix A of the Technical Proposal.

	<u>Analog</u>				<u>Digital</u>	
	1/3 SPS		1/4 SPS		1/3 SPS	1/48 SPS
Attitude Control	4	10*	43	20*	104	72
T&C	3	10	0		72	36
Communications	0		5	4	13	104
Electrical Power	0		24	8	0	36
Structures	0		8	6	0	9
Experiments	21	30	69	36	22	72
Subtotal	28	50	149	74	211	329
Total			301			540

*Dedicated spares

Discrete															Discrete Low Level Isolated																			
	Y ₀	Y ₁	Y ₂	Y ₃	Y ₄	Y ₅	Y ₆	Y ₇	Y ₈	Y ₉	Y ₁₀	Y ₁₁	Y ₁₂	Y ₁₃	Y ₁₄	Y ₁₅	Y ₁₆	Y ₁₇	Y ₁₈	Y ₁₉	Y ₂₀	Y ₂₁	Y ₂₂	Y ₂₃	Y ₂₄	Y ₂₅	Y ₂₆	Y ₂₇	Y ₂₈	Y ₂₉	Y ₃₀	Y ₃₁		
X ₀	000	020	040	060	100	120	140	160	200	220	240	260	300	320	340	360	400	420	440	460	500	520	540	560	600	620	640	660	700	720	740	760		
X ₁	001	021	← Command Address (Octal)														401						500	520									761	
X ₂	002	022			A*			731				188										500	520		241								762	
X ₃	003	023			241																	500	520						190				763	
X ₄	004	024												272								500	520		401								764	
X ₅	005	025					416					831										500	520										765	
X ₆	006	026																				500	520										766	
X ₇	007	027		067	107	127								741								507			153		154			276		310	767	
X ₈	010	030		070	110	130			821			761										510			169		188						777	
X ₉	011	031		071	111	131								791								511							178			272	771	
X ₁₀	012	032		072	112	132																512			141		142						772	
X ₁₁	013	033		073	113	133			781			153										513			185		186		4XX		201		202	73
X ₁₂	014	034				500																514							205		4XX		406	4
X ₁₃	015	035						185		186				801								515			402		425		411		413		415	5
X ₁₄	015	036				550		201		205		154										516												776
X ₁₅	015	037	057	077	117	137	157	202	77	217	237	257	277	317	337	402	57	377	417	437	457	477	517	537	557	577	617	637	657	677	717	737	757	777

Figure 13.2-4. Command Address Assignments

Octal Data Word Address

	U ₀	U ₁	U ₂	U ₃	U ₄	U ₅	U ₆	U ₇
V ₀	00	10	20	30	40	50	60	70
V ₁	01		185		761	310		71
V ₂	02			201	202	205		72
V ₃	03			781	821			73
V ₄	04					54		74
V ₅	05				45	DIS	65	75
V ₆	06					56		76
V ₇	07	17	27	37	47	57	67	77

Figure 13.2-5. Data Word Address Assignment

13.2.1.6

Telemetry Address Assignment

Figures 13.2-6 and 13.2-7 show, respectively, the allocation of the minor frame and subcom telemetry words to the spacecraft A numbers. Each word is designated to contain 1 analog point (A) or 9 digital points (D). The 105 minor frame words available for assignment are numbered 7 through 111. Words 0 through 6 are specified for sync, status, and calibration use. Words 0 through 15 are telemetered once each minor frame, regardless of whether the DACU is in the normal or dwell mode.

Words 112 through 127 are the 16 subcom channels. Telemetry points assigned to a word on one of these channels are sampled once every sixteenth minor frame, or once every 48 seconds. If the DACU were commanded to dwell on one of these channels, each point on that channel would be sampled 7 times in each 3-second minor frame period.

A complete list of the specific telemetry address assignments by function is shown in Appendix A of the Technical Proposal.

13.2.2

LINK CALCULATIONS

The link margin calculations for the telemetry and command links are shown in tables 13.2-2 and 13.2-3, respectively. The modes listed in the telemetry link are further defined as follows. The prime mode is a frequency division multiplexed signal transmitted via the 30-foot parabolic dish. This signal contains EME, Special Data, and normal telemetry frequency division multiplexed onto the downlink carrier

	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
	D Sync	D Sync	D Sync	D Status	A Cal 201	A Cal 201	A Cal 201	D Ad Ver 236	D Ad Ver 236	D Ad Ver 237	D Ad Ver 237	A S	A S	A S	D Doc 1 185	D Doc 2 186
Word #	16	17	17	19	20	21	22	23	24	25	26	27	28	29	30	31
A# assigned this word	A Laser 761	A MMW 771	A HRC 731	A RFI 741	A S	A Laser 761	A S	A S	A S	A S	A Pol. Sens 154	D Earth Sen 178	D Earth Sen 178	D Earth Sen 178	A S	A S
	32	33	34	35	36	37	38	39	40	41	42	43	44	45	46	47
	A Laser 761	A MMW 771	A Rad 801	A S	A T&FD 781	A S	A S	A S	D Interf. 272	D Interf. 272	D Interf. 272	D Interf. 272	D Interf. 272	D Interf. 272	D Interf. 272	D Interf. 272
	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63
	A Laser 761	A S	A HRC 731	A RFI 741	A T&FD 781	A S	A S	A S	A S	A S	A 276	A 310	A 310	A 276	A S	A S
	64	65	66	67	68	69	70	71	72	73	74	75	76	77	78	79
	A Laser 761	A S	A Rad 801	A RFI 741	A T&FD 781	A S	A S	A S	A S	A S	A Pres Plen 113	A Pres Plen 114	A S	A S	A S	A S
	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95
	D Laser 761	A MMW 771	A HRC 731	A S	D T&FD 781	A S	A S	A S	A S	A S	A 154	A S	A S	A S	A S	A S
	96	97	98	99	100	101	102	103	104	105	106	107	108	109	110	111
	D Laser 761	A MMW 771	A Rad 801	D DFI 741	A S	D Laser 761	A S	A S	D SCC 205	D SCC 205	D SCC 205	D SCC 205	A S	A S	A S	A S

Notes: D = Digital
A = Analog
S = Spare

Figure 13.2-6. Minor Frame Telemetry Word Assignments

Subcom Channel

Word# within the channel

A# assigned this word

	112	113	114	115	116	117	118	119	120	121	122	123	124	125	126	127
1	A 153	D 190	D 272	A S	A 188	A 320	A 401	A 402	A 500	A 320	A 761	A 761	A 731	A 791	A 801	A 741
2	A 154	A 108	A S	A S	A 188	D 320	A 401	A 402	A 500	A 320	A 761	A 761	A 731	A 791	A 801	A 741
3	A 169	A 109	A S	A S	A 188	A S	A 416	A 416	A 500	A S	A 761	A 761	A 731	A 791	A 801	A 741
4	A 169	A 110	A 276	A S	A S	A S	A 416	A 416	A 500	D 320	A 761	A 761	A 731	A 791	A 801	A 741
5	A 169	A 111	A 276	A S	D 108	A 320	A 416	A 416	A 500	A S	A 761	A 761	A 731	A 791	A 801	A 741
6	A 153	A 113	A 276	A S	A 153	D 320	A 416	A 416	A 500	D 320	A 761	A 761	A 731	A 791	A 801	A 741
7	A 154	A 114	A 276	A S	A 154	A S	A 416	A 416	A 500	A S	A 761	A 761	A 731	A 791	A 801	D 741
8	D 178	A 115	A 276	A S	A 169	A S	A 416	A 416	A 500	D 320	A 761	A 761	A 731	A S	A 801	D S
9	D 178	A 116	A 276	A S	A 169	A 320	A 416	A 416	D 500	D 320	A 761	A 761	A 731	A S	A 801	A S
10	D 178	A 161	A 276	A S	A 169	D 320	A 416	A 416	D 2XX	D 321	A 761	A 761	A 731	A S	A 801	A 781
11	A 141	A 163	A 276	A S	A 153	A S	D 416	D 416	D 2XX	D 331	A 761	A 761	A 731	D 771	A 801	A 781
12	A 141	A 162	A 276	A S	A 153	D 310	D 416	D 416	D 2XX	D 331	A 761	A 761	A 731	A S	A 801	A 781
13	A 142	D 16X	A S	A S	A 154	D 320	D 7XX	D 7XX	D 2XX	D 350	A S	A 761	D 731	A S	D 801	A 781
14	A 142	A S	A S	A S	A 154	A S	A 415	A 417	A S	A S	A S	A 761	A S	A S	A S	A 781
15	D 141	A S	A S	A S	D 154	A S	A 416	A 416	A S	A S	A S	A 761	A S	A S	A S	A 781
16	D 178	A S	A S	A S	A S	A S	A S	A S	A S	A S	A S	D 761	A S	A S	A S	D 781

Note: D = Digital
A = Analog
S = Spare

Figure 13.2-7. Subcommutated Telemetry Word Assignments

Table 13.2-2. Telemetry Links

S/C Transmitter			2 W		
S/C Losses			1.5 dB (prime focus feed)		
			2.5 dB (solar panel ant.)		
Ground Station Ant. Gain			21 dB		
Ground Station Syst. Temp.			1290°K, nominal		
			SNR		
	<u>Mode</u>	<u>Rc'd Power</u>	<u>TM</u>	<u>EME</u>	<u>SDL</u>
Normal Mission	FDM to Reflector (+16 dBI)	-126.3 dBW @ 136.23 MHz	+ 25.9 dB	+ 24.3 dB	+ 28.1 dB
Ascent	TM to Solar Panel Ant. (-10 dBI)	-156.3 dBW @ 136.23 MHz	+ 12.4 dB		
Alternate	TM & EME direct to Solar Panel Ant. (-3 dBI)	-149.3 dBW	+ 19.4 dB @ 136.23/137.11 MHz	+ 12.4 dB @ 137.11/136.23 MHz	
Contingency	TM to Solar Panel (-3 dBI) EME or SDL to Reflector (+16 dBI)	-149.3 dBW @ 136.23 MHz	+ 19.4 dB		
		-130.3 dBW @ 137.11 MHz		+ 31.4 dB	30.4 dB

Table 13.2-3. Normal Command Links

Command Transmitter:				2.5 kW		
Command ERP:				+ 48.5		
S/C RF Losses:				2.5 dB		
S/C Threshold:				-120.0 dBm		
<u>Phase</u>	<u>Frequency</u>	<u>Function</u>	<u>S/C Antenna</u>	<u>Received Signal</u>	<u>Predetection CNR</u>	<u>SNR</u>
Ascent/ Acquisition	154.20 MHz	PCM Command	Solar Panel Ant. (-10 dBI)	-134.8*	+15.5	+31.8
Mission	154.20 MHz	PCM Command	Solar Panel Ant. (-3 dBI)	-125.8 dBW	+24.5	+40.8
SAMOC/ SAPPSAC	154.20 MHz	PDM Tone	Solar Panel Ant. (-3 dBI)	-125.8	+24.5	+32.5**
SDL	148.26	AM Analog	VHF Prime Focus (+16 dBI)	-108.3	+42.0	+49.8
Mission	154.20 MHz	Contingency SAMOC/ SAPPSAC	Solar Panel (-10 dBI)	-134.8	+15.5	+23.5
Mission	148.26	PCM Command	Prime Focus (+16 dBI)	-108.3	+42.0	+44.3

*Assumes worst case command ERP = 47.5 dBW

**27.5 dB if PCM Command Function exercised simultaneously;
45 dB if prime focus is used.

within the DSU. In the "contingency" mode, either the EME or the Special Data signals, but not both, can be transmitted via the 30-foot parabolic dish. Normal telemetry will be transmitted via the VHF antennas on the solar panels. The "Alternate" mode assumes EME and normal telemetry are transmitted via the omnidirectional VHF antennas located on the solar panels. The Ascent mode applies during liftoff, parking orbit, and transfer orbit. During the early liftoff period, the signal is transmitted via the VHF antennas through an RF window in the shroud.

The command link parameters are listed in Table 13.2-3. All normal commanding is done via the omnidirectional VHF antennas with the prime focus being used only as a backup mode. The Special Data Link (SDL) is a non-mission critical voice-bandwidth-analog link. The link calculations are treated in extensive detail in subsection 2.2 of the Technical Proposal.

13.2.3 FDM-VHF TELEMETRY LINK

A significant criterion for arriving at the FDM configuration was the desire to limit the RF bandwidth to that allocated for the ATS F&G program at VHF, i.e., 30 kHz; therefore, narrowband FM modulation and premodulation filtering was used. The FDM baseband spectrum extended in this case to 11.8 kHz. For narrowband phase modulation of the transmitter, the RF bandwidth requirement for the FDM modulated link is 23.6 kHz. This allows 6.4 kHz (+3.2 kHz) for any remaining allocations. Since the Doppler shift will be about zero, the 6.4 kHz has been allocated to transmitter stability (about 0.0025 percent). This change to the presently specified transmitter specification requirement of 0.003 percent is negligible.

Link performance and the resultant signal quality were evaluated for all three signals that were present simultaneously. Normally this is not required, i.e., there will be no SDL* information transmission for a large percentage of the time. When the SDL link is not in use, the SDL VCO input will be squelched, and power will be transferred in part to the other two data channels. Approximately 1.5-dB improvement can be obtained for the EME and TM channels.

Alternative combinations of power distribution can be made in the event of changing requirements. The present distribution of the 2-watt transmitter power is fairly consistent with accepted standards.**

The performance of the 2-watt FDM/PM, VHF, spacecraft-to-ground link using the high gain antenna is presented in Table 13.2-4. The FDM frequency allocation is illustrated in Figure 13.2-8. To conserve spectrum and stay within the 30-kHz RF bandwidth allocation, premodulation filtering has been provided for each of the data sources. Frequencies given in Figure 13.2-8 are filter cutoff frequencies. Linear phase filters (Bessel) are required for both EME and TM data, with rolloff of at least 18 dB per octave. The SDL data filter also has an 18-dB per octave rolloff, but linear passband phase is not required for this analog signal. EME data is at 1800 bps in split phase form. Filter cutoff frequencies are 130 Hz to 2.7 kHz. EME data will modulate the PM carrier directly.

* SDL - Special Data Link (Voice-bandwidth analog signal).

** GSFC Aerospace Telemetry Standards, Part III, Section 1, par. 4.2.2, p. 9.

Table 13.2-4. FDM-VHF (136 MHz) Telemetry Link Calculations

	Worst Case	Nom.	Best Case
Transmit Power (dBW)	3	3	3
S/C Diplexer & Filter, Connectors & Cable, & Mismatch Losses (dB)	-2.0	-1.5	-1.0
S/C Antenna Gain (dB)	17	18	19
Free Space Attenuation (@ 136 MHz) (22,300 s. m.) (dB)	-167.8	-167.8	-167.8
Ground Antenna Gain (dB)	21	21	21
Ground Line and Connector Losses (dB)	1	1	1
Total Received Power (dBW)	-127.8	-126.3	-124.8
Receiver Noise Density (dBW/Hz)	-192.1	-197	-200
Receiver Noise in 30-kHz RF Bandwidth (dB)	-147.3	-152.2	-155.2
Carrier-to-Noise Ratio at Receiver Input Terminals (dB)	19.5	25.9	30.4
Carrier Tracking Channel			
Carrier-to-Noise Ratio at Mod Loss of -2.2 dB (dB)	17.3	23.7	28.2
Carrier-to-Noise Ratio in a 20-Hz Loop Bandwidth (dB)	19.1	25.5	30
Required Carrier-to-Noise Ratio (dB)	11	11	11
Margin (dB)	38.1	14.5	49
FME Channel			
Carrier-to-Noise Ratio at Mod Loss of -8.0 dB (dB)	11.5	17.9	22.4
Signal Energy-to-Noise Power Density in 2.7-kHz Bandwidth (dB)	19.1	25.9	30.3
Required E/N (dB)	14	14	14
Margin (dB)	5.1	11.9	16.3
TM Channel			
Carrier-to-Noise Ratio at Mod Loss of -13 dB (dB)	6.5	12.9	17.4
Subcarrier-to-Noise Ratio for Predetection Bandwidth of 1.2 kHz	17.5	23.9	28.4
Required Subcarrier Signal-to-Noise Ratio	12	12	12
Subcarrier Margin over Threshold (dB)	5.5	11.9	16.4
TM Channel			
E/N ₀ in Post-detection Bandwidth of 600 Hz (dB)	20.5	26.9	31.4
Required E/N ₀ (dB)	11	14	14
Margin	6.5	12.9	17.4
SDI Channel			
Carrier-to-Noise Ratio at Mod Loss -3.4 dB (dB)	16.1	22.5	27.0
Subcarrier-to-Noise Ratio in Predetection Bandwidth of 4.8 kHz (dB)	21.1	27.5	32.0
Required Subcarrier Signal-to-Noise Ratio (dB)	12	12	12
Subcarrier Margin (dB)	9.1	15.5	20
Post-detection S/N in 2.4-kHz Bandwidth (dB)	24.1	30.6	35
Required S/N	13	13	13
Margin	11.1	17.6	23

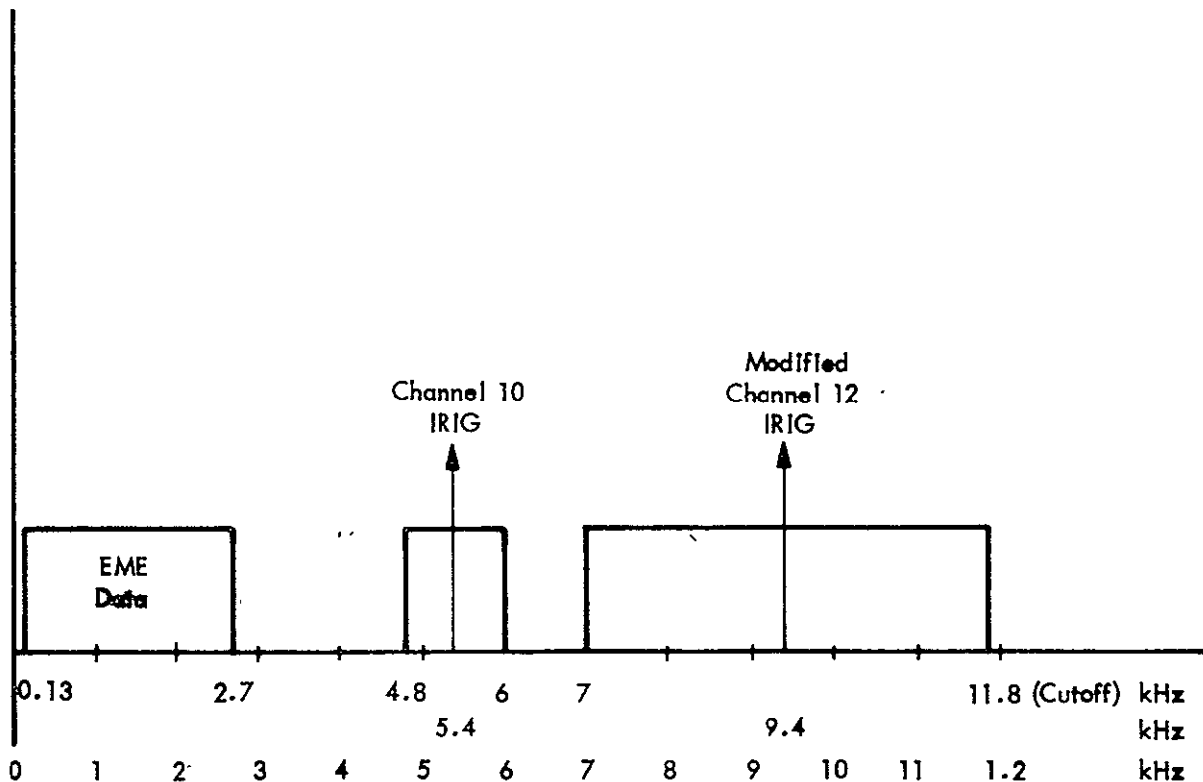


Figure 13.2-8. FDM Frequency Allocations

Filtering for the split-phase 400-bps TM data is from 25 Hz to 600 Hz. This data will FM-modulate the IRIG channel 10 SCO with a modulation index of about 0.6. The modulation spectrum of this SCO will be bandpass-filtered to include 4.8 kHz to 6 kHz.

The special data (analog signal) is restricted to 2.4 kHz and will FM-modulate a 9.4-kHz SCO. The modulation index will be 0.6 and the modulated spectrum will occupy the 7-kHz to 11.8-kHz region.

13.2.3.1 Link Parameters and Performance

The 2-watt spacecraft transmitter feed loss through the switch and filter is taken as 4 dB. The 30-foot dish provides circular polarization, and the antenna gain is not expected to exceed 19 dB. The worst case indicates a 2-dB antenna gain decrease to account for operation during large angle offset of the spacecraft, e.g., during a scanning mode.

Path loss is not expected to change significantly. Ground receiver antenna gain has been supplied by NASA* as 21 dB (which is about 1 dB below the

*ATS-F Transmission Links Data, Rev 4, Goddard Space Flight Center, NASA (March 15, 1969).

maximum gain points of the SATAN).^{*} Dual-diversity ground reception is assumed, with no polarization loss. An antenna-to-preamplifier feed loss is held to approximately 1 dB. The resultant worst-case received signal power is expected to be about -127.8 dBW and 3 dB higher in the best case.

Noise power considerations for the receiver are based on the equivalent antenna temperature which is assumed to vary between 300°K and 4000°K, i.e., minimum galactic background and hot-spot temperatures,^{**} respectively. A ground receiver noise figure of 3 dB is taken as nominal^{***} with 4 dB as maximum. Table 13.2-4 shows a worst-case noise power spectral density of -192.1 dBW/Hz. An improvement of about 8 dB is obtained in the best-case calculation. Worst-case received power is -127.8 dBW and is thereby being presented to the receiver terminals in a 30-kHz bandwidth at a carrier-to-noise ratio of 19.5 dB (worst case).

13.2.3.2 Modulation Parameters

The modulation index of the main carrier due to each channel is selected to provide nearly equal signal-to-noise ratios at the output of the subcarrier filtering networks. The deviation on the main carrier is chosen as one radian (rms), a value which provides nearly equal distribution of carrier power and channel power and ensures that better than 98 percent of the power will be in a band of 30 kHz around the VHF carrier. To provide near equal signal-to-noise ratios at the output of the channel bandpass filters, the peak deviation of each channel must be proportional to the square root of the bandwidth of each channel. Using this approach, the following results are obtained and used in the link calculations.

It may be seen from Table 13.2-5 that 71 percent of the total power is in the residual carrier and sidebands, while 29 percent is wasted on higher order products.

Table 13.2-5. Power Distribution

	Mod Index (Peak)	Percent of Total Power	Power Retained (watts)	Mod Loss (dB)	Bandwidth (kHz)
Carrier		-61	1.2	-2.2	
EME	0.5	-8	0.16	-8.0	2.7
TM	0.27	-2.3	0.05	-13.0	1.2
SDL	0.8	-2.3	0.46	-3.4	4.8
TOTALS		94	1.9		

^{*} Satellite Tracking and Data Acquisition Network Facilities Report, NASA-GSFC (June 1964), p. 2-68.

^{**} George E. Mueller, "A Pragmatic Approach to Space Communication," Proc. IRE, Vol. 48 (April 1960), pp. 557-566.

^{***} Stadan Facilities Report, NASA-GSFC (June 1964), p. 2-30.

13.2.3.3 Carrier Tracking Channel

For the coherent detection of the phase-modulated carrier, it is necessary to transmit a residual carrier of sufficient power to allow the phase detector to lock on the signal. The relative transmitted reference carrier power level may be found by obtaining the square of the product of the zero order Bessel functions (first kind) for each signal's modulation index. This is shown (see Table 13.2-5) to provide a reference carrier power of 1.2 watt. The carrier-to-noise ratio in a 20-Hz loop bandwidth is calculated from the carrier-to-noise ratio at the receiver input, diminished by the modulation loss of -2.2 dB and enhanced by the bandwidth improvement of 31.8 dB. This results in a worst-case ratio of 49.1 dB. Assuming detection of the phase-modulated transmission by the use of a phase-lock ground receiver (e.g., a combination of the standard ground station General Dynamics Receiver and Electronics 215 phase demodulator), approximately a 9-dB phase lock loop signal-to-noise ratio is required to limit rms tracking error to the acceptable value of 20 degrees.* Applying a 2-dB margin to this, the resultant requirement of 11 dB is more than adequately met by the 38.1 dB worst case condition indicated in Table 13.2-4.

13.2.3.4 EME Channel

The power in the EME sidebands was calculated to be about 8 percent of the total power or 0.16 watt. Of interest in this channel is the signal energy-to-noise power density, E/N_0 . This value was obtained from the carrier-to-noise ratio, diminished by the modulation loss of -8.0 dB and enhanced by the bandwidth improvement (i.e., 7.9 dB).

Assuming the electronics unit is used in demodulation, coherent detection can be obtained for EME data. This is about 3.5 dB more efficient than the noncoherent mode** for the telemetry data, i.e., requiring about 10.5 dB for 1 in 10^6 bit error rate. However, if a penalty of 3 dB for nontheoretical performance of the ground PCM console*** is included, a 14-dB threshold can be established for this signal. A worst-case margin of 5.4 dB is indicated.

13.2.3.5 TM Channel

As seen in Table 13.2-4, the worst-case value of 17.5 dB provides 5.5-dB margin over FM threshold. To determine bit error rate for the telemetry digital signal, a calculation of E/N_0 had to be made. This was calculated by enhancing the predetection signal-to-noise ratio by the 3-dB post-detection bandwidth improvement. A worst-case value of 20.5 dB is obtained for E/N_0 . It can be shown** that for noncoherent detection, a signal quality of 14 dB will provide bit error rate of better than one in 10^6 . Using this as a threshold, the worst-case telemetry performance has a 6.5 dB margin.

* STADAN Facilities Report NASA-GSFC (June 1964), p. 2-15.

** Comparison of the communication efficiency of phase-modulated and frequency modulated PCM telemetry systems, Contract No. NAS5-1474, Final Report, SGC 74 R-6, 29 Nov. 1961 (p. 6).

*** PCM data handling equipment Magnavox Company, Contract NAS5-3199, p. 1-3.

The worst-case signal-to-noise ratio of about 21.1 dB provides 9.1-dB margin over the approximate FM threshold of 12 dB. After FM detection and filtering (FM improvement is 0 dB), the worst-case SDL signal-to-noise ratio ranges from 24.1 dB (worst case) to 35.0 dB (best case). It has been shown that a threshold for received signal quality* is obtained at about 13 dB. Threshold here is defined loosely as the signal quality in which further improvement results in a proportionally smaller improvement in intelligibility. In the reference paper the threshold of 13 dB represents an intelligibility of about 80 percent to 90 percent. Therefore, worst-case condition provides a margin over this threshold of about 11.1 dB.

The T&C Subsystem units are housed in 19 A-numbered cases. The physical characteristics (weight, size, and power) of each unit (component) and of the total subsystem are summarized in Table 13.2-6. Each table entry is justified in Section 3 of the Technical Proposal, where the physical implementation of each A-numbered component of the T&C system is discussed.

The mechanical design of each T&C Subsystem component is directed toward a minimum weight and volume configuration which maintains the structural integrity necessary for the ATS F&G space environment. Package design and heat transfer and environmental effects analysis techniques used successfully on other IBM space programs such as OAO (PPDS & SSCU), Gemini, Titan III, and Saturn were employed in the mechanical design for the T&C Subsystem.

In addition, consideration is given to mounting provisions, spacecraft/component form factor for spacecraft installation, in determining the packaging configuration of the T&C Subsystem components.

All components of the T&C Subsystem, with the exception of the VHF antenna units, will be planar-mounted to the support mounting plates or shelves within the Earth Viewing Module (EVM) of the ATS spacecraft. The antenna unit will be located on each solar panel at the far end of the 30-foot reflector. The exact optimum location of the antenna has been determined experimentally by modeling. The exact location of other T&C Subsystem components within the EVM is shown in the Structures volume of this report.

Figure 13.2-9 shows an overview of the interface requirements of the T&C Subsystem. Hardwire connections between T&C unit boxes, shown as single lines, are labeled as to the number of wires which the line represents. The origin of inputs to the DACU and destinations of outputs from the CDD (subsystems A-numbers) are not shown in this diagram. That level of detail is shown in Section 3 of the Technical Proposal. A complete table of pin assignments for each T&C number is shown in Appendix D of the Technical Proposal.

* J. Nicherson, "A Study of the Effect of Frequency Translation Error on Intelligibility of Speech in the Presence of Noise," Rome Air Development Center, USAF, Contract AF 30 (602) 1818, November 5, 1968.

Table 2.11-1. Mass Properties Estimates

<u>A</u>	<u>Component</u>	<u>Weight (lb)</u>	<u>Size (in.) (L x W x H)</u>	<u>Power Requirements (watts)</u>	<u>Power Dissipation (watts)</u>
201	T/M Data Acquisition and Control Unit 1	11.4	10.7 x 7.7 x 4.1	7.3	7.3
202	T/M Data Acquisition and Control Unit 2	11.4	11.5 x 9.6 x 4.0	7.3	7.3
205	Spacecraft Clock	2.8	9.1 x 4 x 2.4	2.3	2.3
211	VHF Transmitter 1	0.4	4.75 x 2.1 x .78	5.0	3.0
212	VHF Transmitter 2	0.4	4.75 x 2.1 x .78	5.0	3.0
213	VHF Transmitter 3	0.4	4.75 x 2.1 x .78	5.0	3.0
214	VHF Transmitter 4	0.4	4.75 x 2.10 x .78	5.0	3.0
216	VHF Diplexer 1	1.0	3.75 x 4 x 3.25	N/A	0.4
217	VHF Diplexer 2	1.0	3.75 x 4 x 3.25	N/A	0.4
221	VHF Low-Pass Filter	0.2	1/2 dia. x 5 long	N/A	0.4
222	VHF Relay	0.1	1.34 x 1.31 x .5	N/A	N/A
225	VHF Antenna Unit 1	0.7	3 x 3 x 1**	N/A	0.3
227	VHF Antenna Unit 2	0.7	3 x 3 x 1**	N/A	0.3
231	VHF Receiver 1	1.3	6 x 4 x 2.5	0.8	0.8
232	VHF Receiver 2	1.3	6 x 4 x 2.5	0.8	0.8
233	VHF Receiver 3	1.3	6 x 4 x 2.5	0.8	0.8
236	Command Decoder/Distributor 1	12.1	12 x 8.7 x 3.5	11.5 (peak)	10.3 (peak)
237	Command Decoder/Distributor 2	12.1	12 x 8.7 x 3.5	11.5 (peak)	10.3 (peak)
241	Data Switching Unit	4.6	8 x 4.7 x 3.4	2.5	2.5
		63.6		37.4*	

* Includes: 1 DACU, spacecraft clock, 2 transmitters, 2 receivers (1 in standby @ 0.2W), 1 CDD (1 in standby @ 2.0W), and the DSU.

The 11.5-W peak requirement of the CDD has a duty cycle of less than 1 percent.

** Plus two 22-inch whips, each.

Figure 13.2-9. Overview of Interface Requirements of T&C Subsystem

- 13.3 T&C SUBSYSTEM COMPONENTS
- 13.3.1 COMMAND DECODER/DISTRIBUTION (CDD) UNIT
- 13.3.1.1 Functional Description
- 13.3.1.1.1 Requirements

The Command Decoder/Distribution Unit (CDD) is the heart of the Spacecraft Command Subsystem. It demodulates FSK, clock, execute, and timed-tone mode tones received from the command receivers, decodes the command addresses carried by the FSK tones, distributes the data word commands, and executes tone to the proper locations within the subsystems. To put the CDD in perspective a description of the Command Subsystem, emphasizing the CDD's requirements and functions, follows.

One of the most important operational functions of the command subsystem is the transmission of discrete commands. Two command carrier frequencies have been allocated: 154.20 MHz (primary) and 148.26 MHz (secondary). The primary frequency is employed for command purposes, and is received by the omni-associated receivers on the spacecraft. (The secondary frequency is the prime uplink for the Special Data function, and is received by the prime focus feed-associated receiver; the Special Data information can be transmitted at the prime frequency as a backup mode.) At its inputs each CDD has an astable oscillator which alternately samples the outputs of the Command Receivers at a rate of approximately 15 Hz. Upon detection of a signal on one of these two input lines (a 0 tone modulated by the 128-Hz clock signal), the CDD locks onto the appropriate command receiver and enters a hold state in anticipation of the sync bits of the command word. The proper spacecraft address and command address are set at the command console, and the data word address thumbwheels are set to 5-5, signifying a discrete command. The transmit button is pressed, and the command console automatically transmits the thumbwheel settings into the command format sequence. Both redundant CDDs establish word sync upon receipt of the sync bits and both examine the 8 bits of S/C Address. Only the CDD which was addressed will process the remainder of the command word. The other CDD disables itself upon recognizing it has not been addressed. The addressed CDD decodes the 6 bits of data word address and sets a discrete latch upon decoding the 5-5 address. The 9-bit command address is decoded and enables one of the 16 X-line drivers and one of 32 Y-line drivers. The contents of the data word address and command address registers are telemetered to the ground for verification. Upon positive verification, ground personnel modulate the command carrier with the execute tone, which the CDD detects and applies power to the X & Y line drivers. One single subsystem location where the enabled X & Y lines enter the same AND gate receives the execute pulse for as long as the execute tone is present. Multiple execution of the same command is possible by simply repeatedly pulsing the execute tone the desired number of times. The same X and Y lines will remain enabled as long as the 0 tone, modulated by the 128 Hz clock tone, is present and the next command word transmission has not been initiated. Upon dropping of the clock modulation or upon detection of the sync bits of the next command transmission, all registers are cleared. A maximum of two commands per second could be processed by the CDD. This would preclude ground verification of the command and would necessitate machine input of the commands to the command console instead of the thumbwheel insertion described. Normal command rate with verification is approximately one command every ten seconds.

Data word commands are handled by the command console in the same way as discrete commands, except that the data word address is an octal number other than 5-5 (0-1 through 7-7) and the data word itself is set by the three command address thumbwheels. Upon detecting a data word address other than 5-5, the CDD enables one of 8 U-lines and one of 8 V-lines in the data word distribution unit.

The 9 bits of data are stepped into the addressed subsystem as they are received in parallel with their loading the 9-bit command address decoder. Before enabling the execute tone, ground station personnel would normally verify the contents of the two registers in the CDD via telemetry. The CDD detects the execute tone and transfers an execute pulse to the selected subsystem which received the data word, at which time the subsystem accepts the contents of its register.

To initiate the timed-tone mode of operation, a discrete command (enable jets or enable wheels) is transmitted, which enables the 7 timed-tone detectors and associated electronics in the CDD. One of the 7 tones is a holding tone and must be brought up simultaneously with (or before) the execution of the enabling discrete command. The remaining tone channels in the CDD then convert tones as received to bilevel execute pulses on a real-time basis, for direct control of the 6 ($\pm P$, $\pm R$, $\pm Y$) attitude control jets or wheels, as commanded.

13.3.1.1.2 CDD Functional Implementation

13.3.1.1.2.1 Flow for Command Word Processing. The flow diagram of the CDD is shown in Figure 13.3-1. The receiver select logic connects the CDD to the receiver that has signal output (0 tone modulated by the 128-Hz sinewave clock signal). The 128-Hz AM on the 0 and 1 tones is demodulated and shaped into a square wave bit-clock. The word sync logic detects the first two consecutive 1's following the introductory 0's and labels them bits 28-29, establishing the word clock reference. Bits 30-37 are sensed by the spacecraft address compare logic during bit period 37 to determine if the command is addressed to this decoder. If the compare function is not true, processing of the remainder of the command word is inhibited. The 6-bit shift register is sensed during bit period 43 by the data word address decoder logic. If the particular 6-bit combination assigned to signify a discrete command is sensed, the DIS latch is set. In this situation, the contents of the 9-bit shift register will be sensed during bit period 52 by the X-Y line decoder logic. The 9-bit command address will be decoded and 1 of the 16 X lines and 1 of the 32 Y lines will be enabled. The receipt of an execute tone enables power to the X and Y drivers, transferring the command pulse to the particular location in the spacecraft, where the enabled X and Y lines input the same AND gate.

All other assigned 6-bit combinations of the data word address cause one of the eight U lines and one of the eight V lines to be enabled. In this situation the 9-bit data word (bits 44-52) is gated to the addressed subsystem shift register in parallel with the 9-bit shift register in the CDD. Upon receipt of the execute tone from the ground, the CDD gates the execute pulse to the same subsystem. For a DOC data word, the contents of the 6-bit shift register is shifted serially to the DOC during bits periods 53-58 (following transfer of the 9-bit word).

The S/S (SAMOC/SAPPSAC) filters, threshold detectors, gating, logic, and drivers are enabled upon receipt of a discrete command (enable jets) and are held enabled only as long as the holding tone (one of the seven S/S tones) is present.

13.3.1.1.2.2 Timing for Command Word Processing. The processing of a command word is performed essentially in a real-time manner in that the spacecraft address is compared during receipt of its eighth bit (bit 37), the data word address is decoded during receipt of its sixth bit (bit 43), the (discrete) command address is decoded during receipt of its ninth bit (bit 52), and the data word is transferred to the addressed subsystem shift register as received, when received by the CDD (in parallel with insertion into the 9-bit shift register in the CDD). The only delayed function is the transfer of the 6-bit data word address to the DOC (only if a DOC data word has been transmitted) during bit periods 53-58, after the transfer of the 9-bit data word.

Upon receipt of an execute tone from the ground, addressed discrete commands are executed in real time. Upon receipt of an execute tone by the CDD from the ground, a subsystem that has received a data word is also given an execute pulse in a real-time manner, except that the CDD will not permit transfer of the execute pulse until after the entire data word has been received by the subsystem (bit period 53 in all cases except a DOC data word, bit period 59 in the latter case). S/S tones are converted in real-time to bilevel signals that control the firing of the attitude control jets.

13.3.1.1.2.3 Internal Power Control. The CDD design approach is to turn power on sequentially to the CDD circuit elements, enabling power to each element only during that period of the command sequence when its operation is required. Table 13.3-1 summarizes this sequential power enabling, referring to the functional blocks shown in Figure 13.3-1.

13.3.1.1.2.4 Interrupt. All latches are reset, all enabled lines are disabled, all logic is returned to a logical 0 state, and the CDD is returned to a standby (minimum power) condition upon loss of the command carrier or modulation of the carrier by the clock modulated 0 or 1 tones, or the SAMOC/SAPPSAC holding tone for a period of at least 15.6 milliseconds (two bit periods). A new command can be initiated without having to execute a previous (unwanted or erroneous) command. It is not necessary to employ the interrupt feature to clear the registers; the new command will replace the undesired one in the CDD registers. The S/S circuitry within the CDD is reset upon receipt of a disable jet's discrete command or absence of the S/S holding tone.

13.3.1.1.2.5 Standby Mode. In the standby mode, power is applied only to the receiver select logic, filter driver, (clock) detector, and threshold circuits. The receiver select logic alternately samples the two receiver input lines at a 15-Hz rate. Upon receipt of a modulated command carrier, the CDD leaves the standby mode.

13.3.1.1.2.6 Command Distribution. The command distribution system used in the spacecraft is an "Array" approach whereby the final stage of command decoding is performed in the subsystems. Within the CDD, the first 5 bits of the 9-bit command address is decoded and 1 of 32 Y lines is enabled. The last 4 bits are decoded and 1 of 16 X lines is enabled. Each X and Y is connected to one or more Low Level Command Drivers (LLCD), which are capable of driving up to 8 Low Level Command Terminators (LLCT) distributed throughout the various subsystems (see figures 13.3-2 and 13.3-3). There are 64 LLCDs associated with discrete commands, 19 with data word commands, 7 with SAMOC/SAPPSAC, and 10 spares, for a total of 100.

OR gates at the input to the LLCTs provide the capability of commanding with either of the two CDDs. Each LLCT provides one of the two inputs of 2-input AND gates which perform the final level of decoding.

Table 13.3-1. Power Enabling of CDD Elements

Latch P - enabled during command signal presence Latch Q - enabled upon positive S/C address compare; disabled by Bit 29 Latch E - enabled during execute tone presence. ** All latches disabled by dropping carrier or modulation **	
Functional Block	Power Enabling Requirements
Receiver select logic, Filter Driver, O & I Tone Filters, AM Detector, Clock Integrator	Powered continually.
Decision Logic, Word Sync Logic, 9-bit Shift Register, S/C Address Compare Logic	Latch P
E Tone Filter, 6-bit S. R., Data Word Address Decoder	Latch Q
X-Y Line Decoder	DIS (1 output of the Data Word Address Decoder)
X-Y Drivers	$DIS \cdot Latch\ E \cdot \overline{t_{29-52}}$
Data Word Distribution Unit & Drivers	$DIS \cdot \left(\overline{t_{44-52} + (t_{53-48} \cdot DOC)} + E \cdot \overline{(t_{29-52} + t_{53-58} \cdot DOC)} \right)$
S/S Filters, Detectors, Gating Logic & Drivers	S/S Holding Tone, after being enabled by discrete Enable Jets Command.

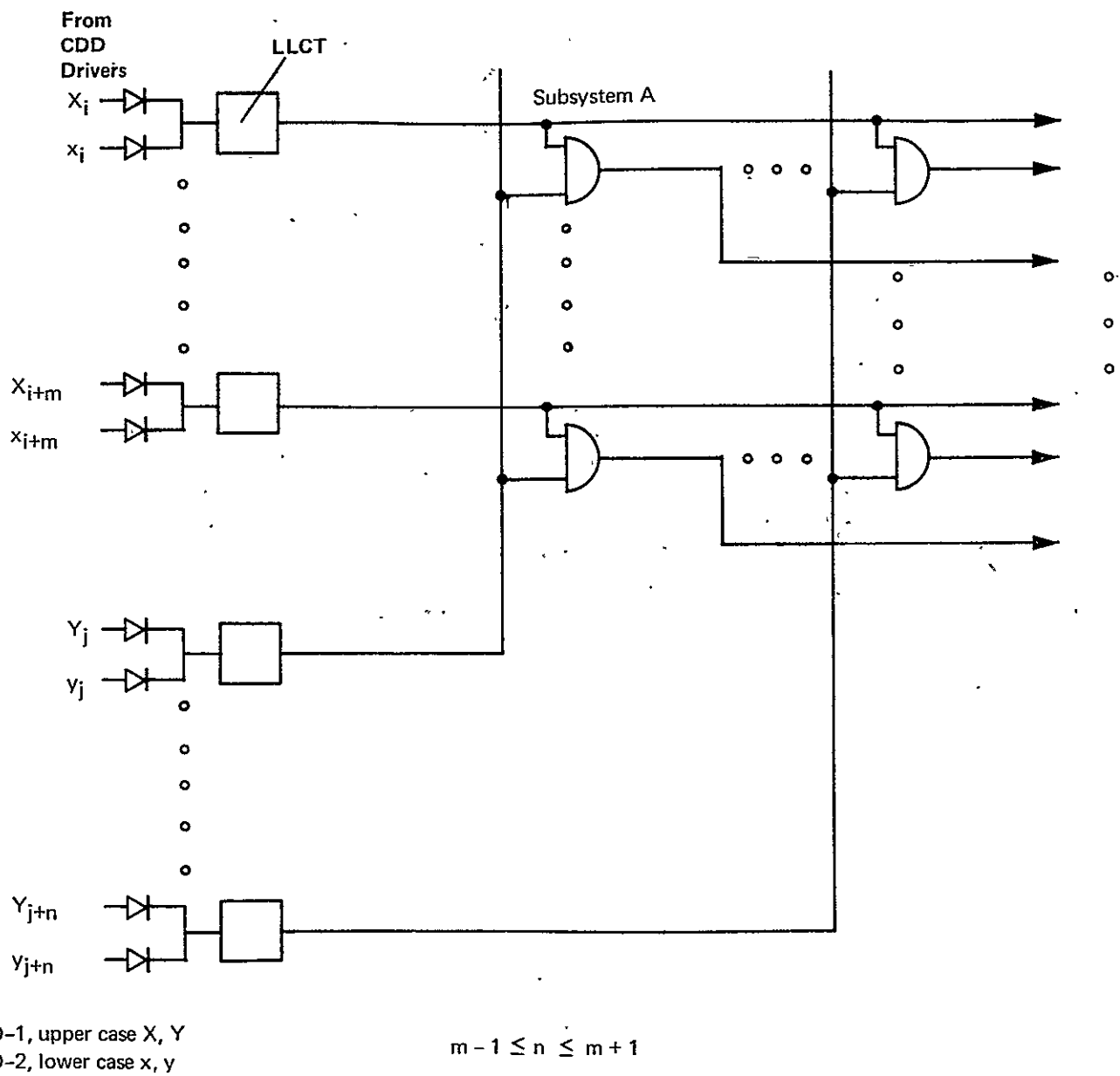
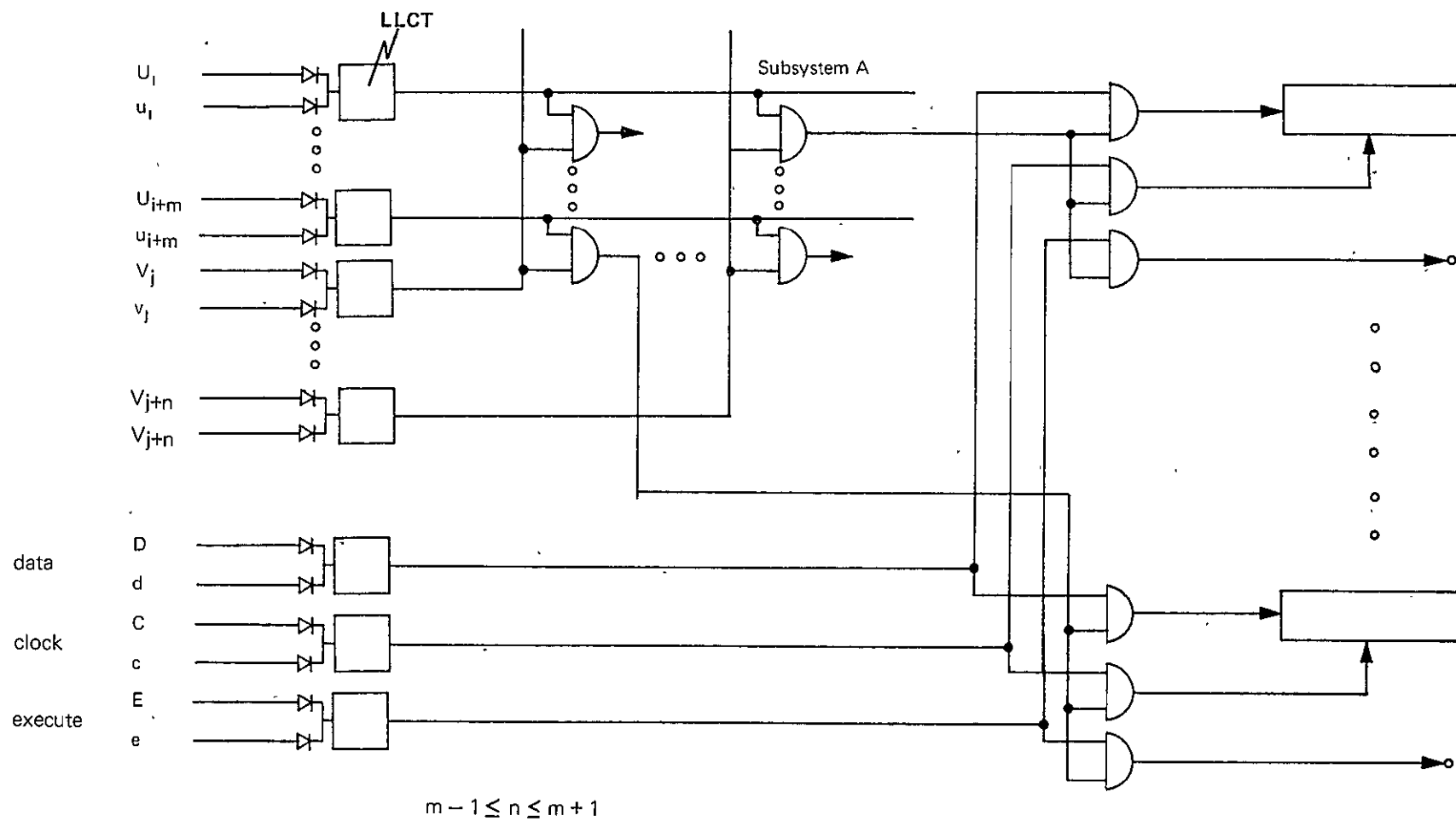


Figure 13.3-2. Termination Configuration for Discrete Commands



CDD-1, uppercase letters
CDD-2, lower case letters

Figure 13.3-3. Termination Configuration for Data Word Commands

During presence of the Execute tone, power is enabled to the LLCs, and the output of the LLCs associated with the decoded X and Y lines assume a logical 1 state. The output of the unique AND gate in a subsystem associated with the intersection of the X and Y lines shifts to a logical 1 state.

Data word command distribution also employs this array approach. The approach was selected after extensive tradeoff analyses showed it to require fewer piece parts and to offer higher reliability, noise immunity, and flexibility than other command distribution schemes considered. This analysis is reported in section 6.2.2 of the Technical Proposal.

The LLCs and LLCTs have been breadboarded and are discussed in detail in sections 3.1.1, 3.1.2 and 3.1.3 of the Technical Proposal.

13.3.1.2 Design Description

A block diagram of the CDD is shown in Figure 13.3-4. Detailed diagrams of each block are shown and described in sections 3.1.1 and 3.1.2 of the Technical Proposal.

The discrete command distribution unit is shown divided into isolated and nonisolated sections. The nonisolated LLCs develop their signal voltages with reference to a common signal return, and are used for all discrete functions except power on-off commands. The power on-off commands, which go to the Line Interface Circuits, must have the signal levels referenced to power-common. To maintain isolation between the signal and power commons, isolation circuits are inserted in the CDD between these LLCs and the logic that drives them. This group of drivers is then termed the isolated discrete distribution unit.

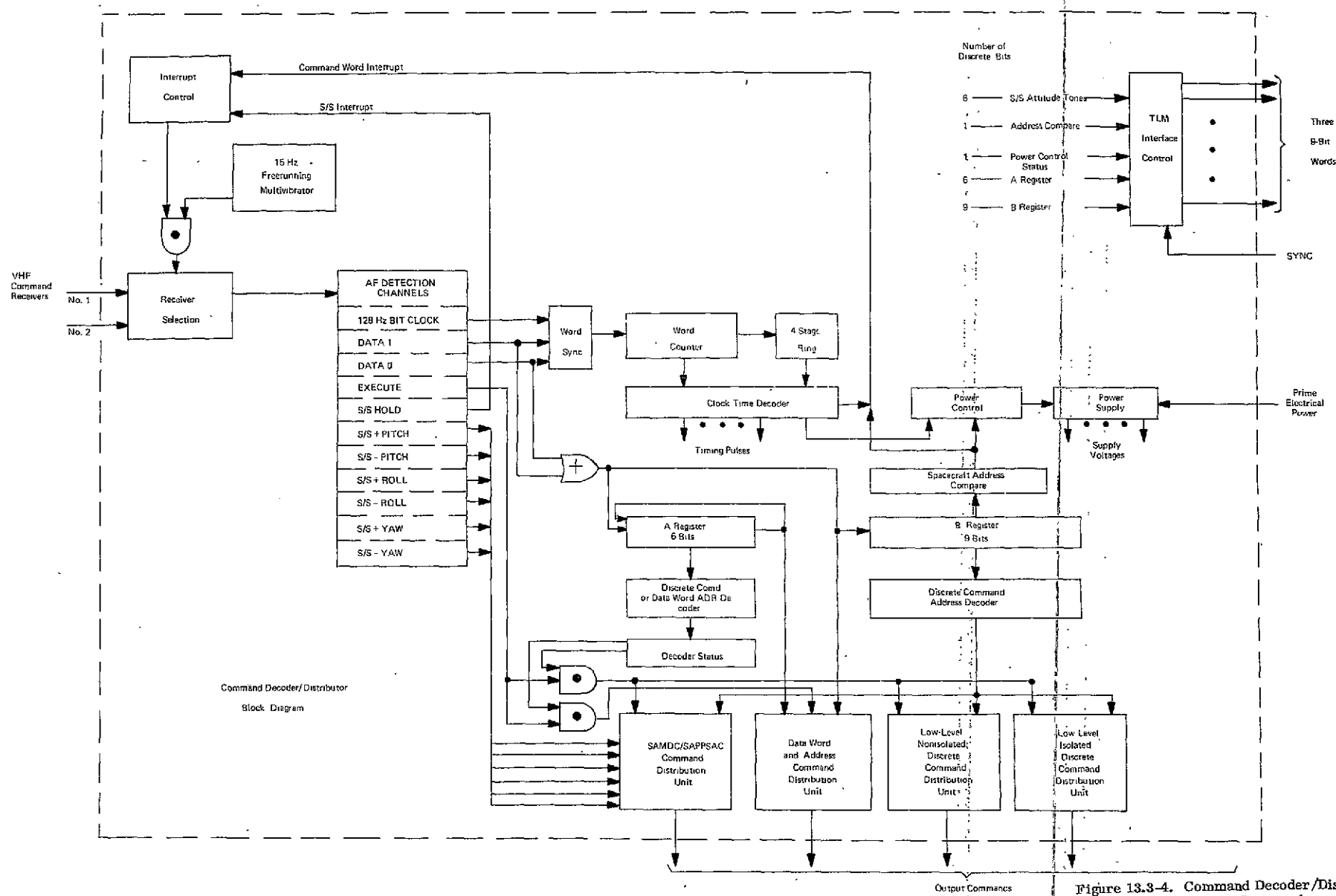


Figure 13.3-4. Command Decoder/Distributor Timing Commands

FOLDOUT FRAME 1

13-37/13-38

FOLDOUT FRAME 2

13.3.2 DATA ACQUISITION AND CONTROL UNIT

13.3.2.1 Functional Description

The functional diagram of one of the redundant DACUs (Figure 13.3-5) shows the fundamental functional units required in DACU design. Timer and program generation logic sequentially steps the multiplexer through the fixed format shown in Figure 13.3-6. The telemetry functions assigned to these format words are discussed in Appendix A of the Technical Proposal. Dwell mode capability commanded from the CDDs allows inspection of limited data points at increased sampling rates. Analog multiplexing is required for both single-ended and balanced high accuracy sources, while digital inputs are single-endedly applied in serial and parallel fashions. Calibration voltages are required to maintain analog system accuracy. Synchronization and status codes are added to the format via the digital multiplexer serializer. The multiplexed analog inputs are digitized by a 9-bit analog-to-digital converter (ADC). The 9-bit converter is necessary to meet high accuracy channel requirements. The digitized analog and digital data, in the form of 9-bit words, is then telemetered to the ground station by the VHF transmitter via the data switching unit (DSU). Power is supplied by a commandable regulator that isolates the DACU from the spacecraft main power bus. Timing is supplied to the DACU from the redundant spacecraft clock. A backup clock in the DACU ensures continued multiplexing in the event both spacecraft clock inputs fail.

An estimate of digital and analog telemetry points to be sampled by the DACUs is given by rates in Table 13.3-2 and by functions in Table 13.3-3. In addition to sampling various data sources at the minimum rates specified, the DACU must also identify the location of data within the format. Synchronization requirements

Table 13.3-2. Telemetry Points by Sampling Rates

	Analog	Digital
Once Per Minor Frame (1/3 sec)	28	211
Once Per Major Frame (1/48 sec)	149	329

Table 13.3-3. Telemetry Points by Function

	Analog				Digital	
	1/3		1/48		1/3	1/48
Attitude Cont.	4	10*	43	20*	104	72
Telem. & Cond.	3	10	0		72	36
Communications	0		5	4	13	104
Electrical Pwr	0		24	8	0	36
Structures	0		8	6	0	9
Experiments	21	30	69	36	22	72
	28	50	149	74	211	329
	301				540**	

*Dedicated spares

**This number indicates total bits of telemetry; 60 of these are serial data from the Interferometer and require only one switch. Thus $540 - 59 = 491$ switches within the DACUs are required for the digital inputs.

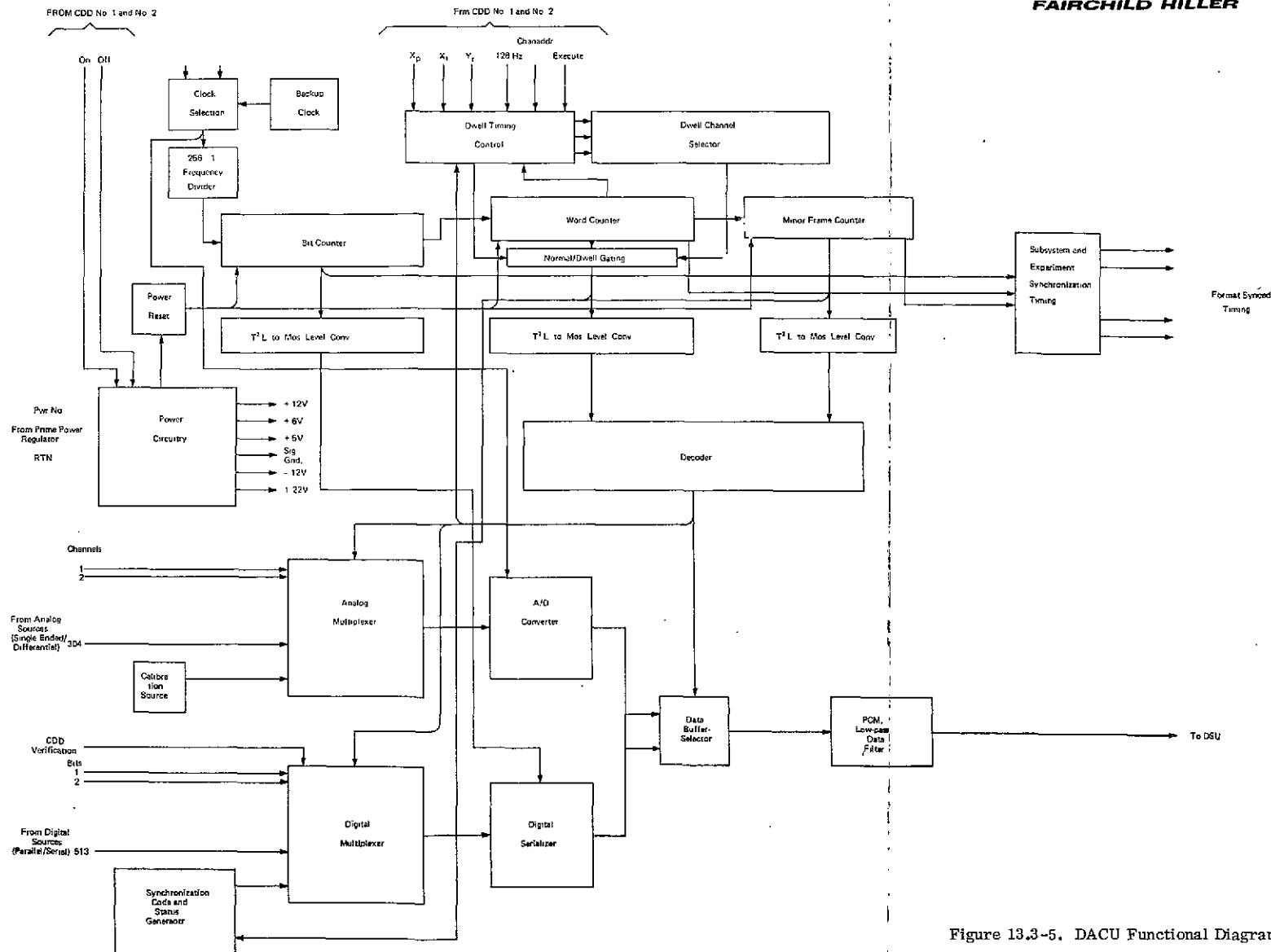
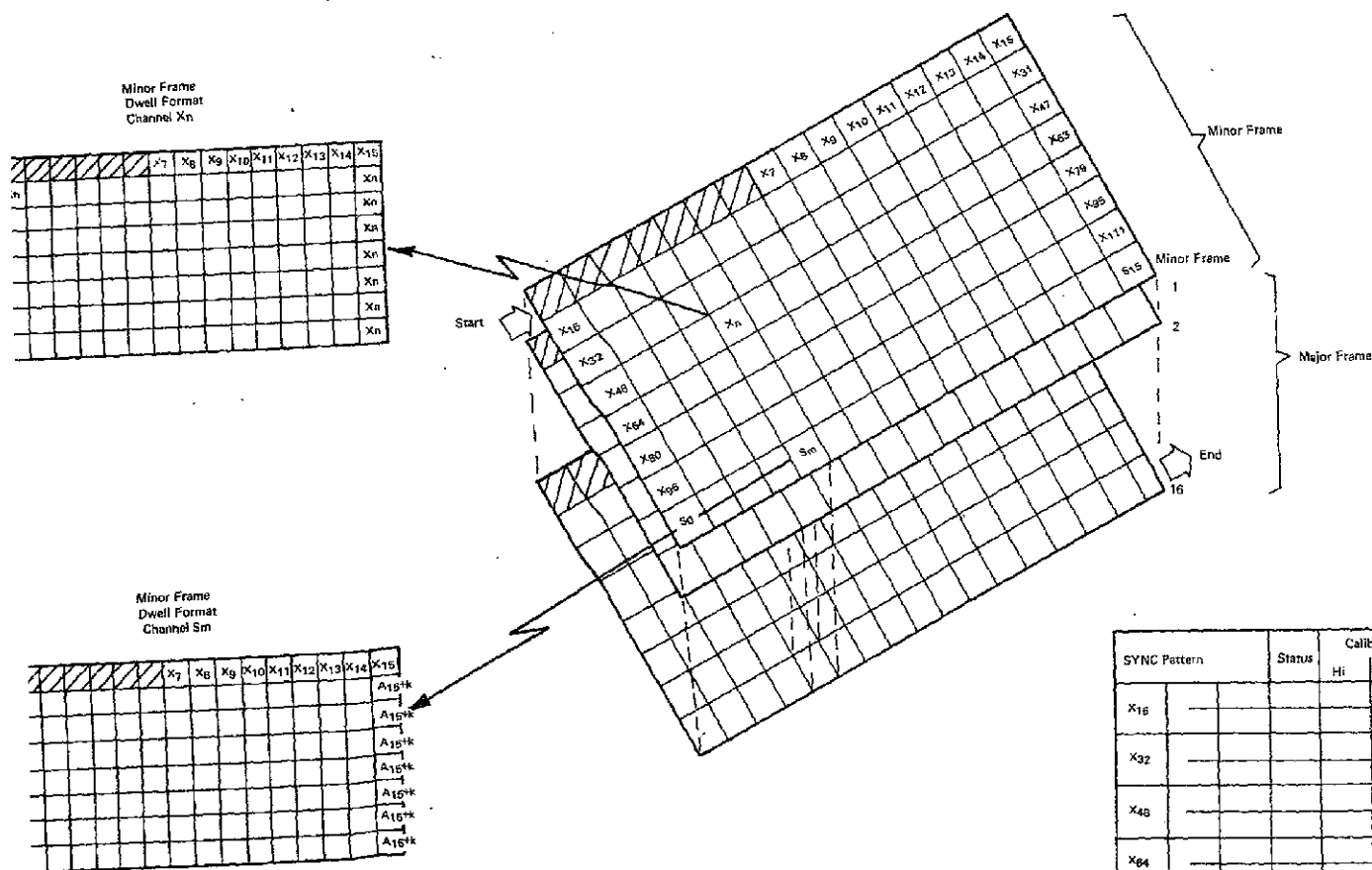


Figure 13.3-5. DACU Functional Diagram



Normal Format (Minor Frame)

SYNC Pattern		Status	Calibration			X7	X8	X9	X10	X11	X12	X13	X14	X15
			Hi	Med	Lo									
X16														X31
X32														X47
X48														X63
X64														X79
X80														X95
X96														X111
S0	S1													S15

Figure 13.3-6. Telemetry Format

FOLDOUT FRAME 1

FOLDOUT FRAME 2 13-43/13-44

are specified in the GSFC Pulse Code Modulation Telemetry Standard, dated January 27, 1966, and the IRIG Telemetry Standards (revised March 1966). For more information on the synchronization pattern used, see the Proposal, Section 3.2.1.2.1. The synchronization pattern must be no more than 33 adjacent bits long, with the suggestion that N-bit frame-sync patterns be selected under the criterion that the probability of displacement of the pattern by ± 1 bit be minimized while at the same time restricting the probability pattern displacement by 2 to $(N - 2)$ bits below a prescribed maximum.

A status word is needed in the format to satisfy the requirements for dwell mode operation and efficient formatting. This word is the fourth 9-bit word in the minor frame. Subcommutation is identified as to format location by status bits. The dwell address is required in the status word while in the dwell mode for data verification at the ground station. Three calibration voltages are required for ADC calibration. These voltages encompass the input range and exercise all ADC digital outputs.

Two 9-bit words per CDD are required in the format to verify commands. Sixteen bits are derived from the digital command word, and one bit verifies the presence of the timed (SAMOC/SAPPSAC) experiment holding tone. These four words are required regardless of dwell/normal mode DACU operation.

Analog channels and rates are shown in Table 13.3-2; approximately 10 channels require differential multiplexing for high accuracy. Digital channels and rates are shown in Table 13.3-2, all inputs are parallel with the exception of a 59-bit serial input from the interferometer. The data output from the DACU is routed via the DSU to one of the VHF transmitters, or as a backup contingency, to the X-band transponder. Premodulation filtering is maximally linear (phase), with attenuation of 36 dB per octave.

In order that certain data may be inspected at increased rates and not dominate the format, a dwell capability is required. The DACU dwells on a specified word upon receipt of a command from the ground.

Format synchronized timing from the DACU is required by the Attitude Control Subsystem to allow internal holding registers to be updated without data changing during the intervals when they are being sampled. A second need for DACU synchronized timing results from the interferometer serial input requirement. The DACU must supply a data clock and an enabling pulse to facilitate this serial transfer. Figure 13.3-7 shows the timing of these two DACU outputs.

The DACU power functional requirements are:

- The power subsystem interface with the DACUs consists of positive regulated buses ($+V_{R1}$, $+V_{R2}$) at $30.5 \text{ Vdc} \pm 2 \text{ percent}$ and return.
- The interface power circuitry must be capable of withstanding power interruption, polarity reversal, overvoltage, under-voltage and transient conditions.
- Regulated buses will have a maximum ripple of 0.5V peak-to-peak measured at the equipment input power connector.
- The DACU will function without component damage for power interruptions.

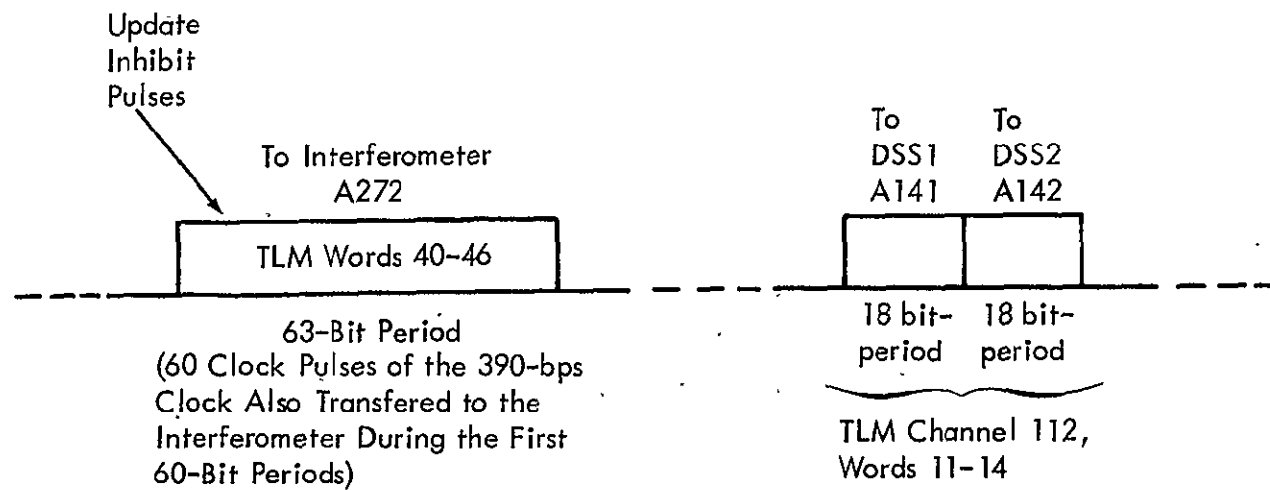


Figure 13.3-7. DACU Output Signal Timing

- The DACU will function as required without circuit component damage for power polarity reversal due to negative transients on the positive power buses and positive transients on the power return bus. Maximum reversed polarity amplitude will be 32V, measured at the equipment input power connector.
- The DACU will function without circuit component damage for (sustained) overvoltage of 35V, measured at the equipment input power connector.
- Undervoltage will in no way damage circuit components.

The program generator is shown in the functional diagram as bit, word, and minor frame counters with reset control, level conversion logic, normal/dwell gating, and decoder. These counters count continuously (even in dwell mode when the dwell channel selector logic superimposes the desired codes into the level converters). The bit counter is of the ring, error correcting type, facilitating the serialization of digital input data with no decoding from the counter. Simple binary down counters are used to generate both word and minor frame timing pulses. Outputs from the word counter to the dwell channel selector enable the superposition of selected dwell channel in each minor frame: normal/dwell gating exclusive ORs, normal counter outputs, and the dwell channel selector outputs. The decoder will operate at FET levels, since it is more efficient to level-convert the fewer counter outputs than the numerous decoder outputs. This results from the desire to use low power TTL in the counter and gating logic to save space and power and the desirability of using FET multiplexers. Decoder functions include the selection of digital and analog channels, power control of analog multiplexers; control of data buffer-selector; and the insertion of sync, status, and calibration bits into the format.

Analog multiplexing is performed by GI MEM 5015s, an MSI device described in Section 3.2.3 of the Technical Proposal. The 19 devices required for full minor frame and subcommutated analog channel capacity allow a total of 304 inputs. Sixteen differential channels are considered for 9-bit accuracy in the minor frame, with the remaining analog capacity of 288 words being single ended. Exact implementation depends on actual requirements and desired number of spares. In addition, a resistor diode network allows +25V fault protection. Multiplexer power strobing is used to reduce power consumption. Quality assurance measures regarding the use of the MEM 5015 are described in Section 5 of the Technical Proposal.

The ADC converter is shown in Figure 13.3-8. It is of the successive approximation type, with a 9-bit capability. A signal buffer is employed at the ADC input to remove common mode errors and to provide ground isolation for differential input from high accuracy channels. An analog signal buffer circuit provides common mode rejection at low frequencies and a 40-dB per decade rolloff above a 3-dB point approximately 180 Hz. A 180-Hz break frequency was determined from timing and settling requirements for the ADC. A timing diagram of ADC operation is given in Figure 13.3-9. Bit rate of 390 Hz is shown with word N and N+1 enabled, under control of the decoder. Settling of the A/D input is allowed in the first 8.5-bit intervals to within 0.1 percent of final value. Then, a start A/D pulse initiates the conversion at a 100-kHz rate. A serial holding register is employed at the ADC output, clocked both at 390 Hz and A/D conversion rates to smooth data flow to the data buffer selector.

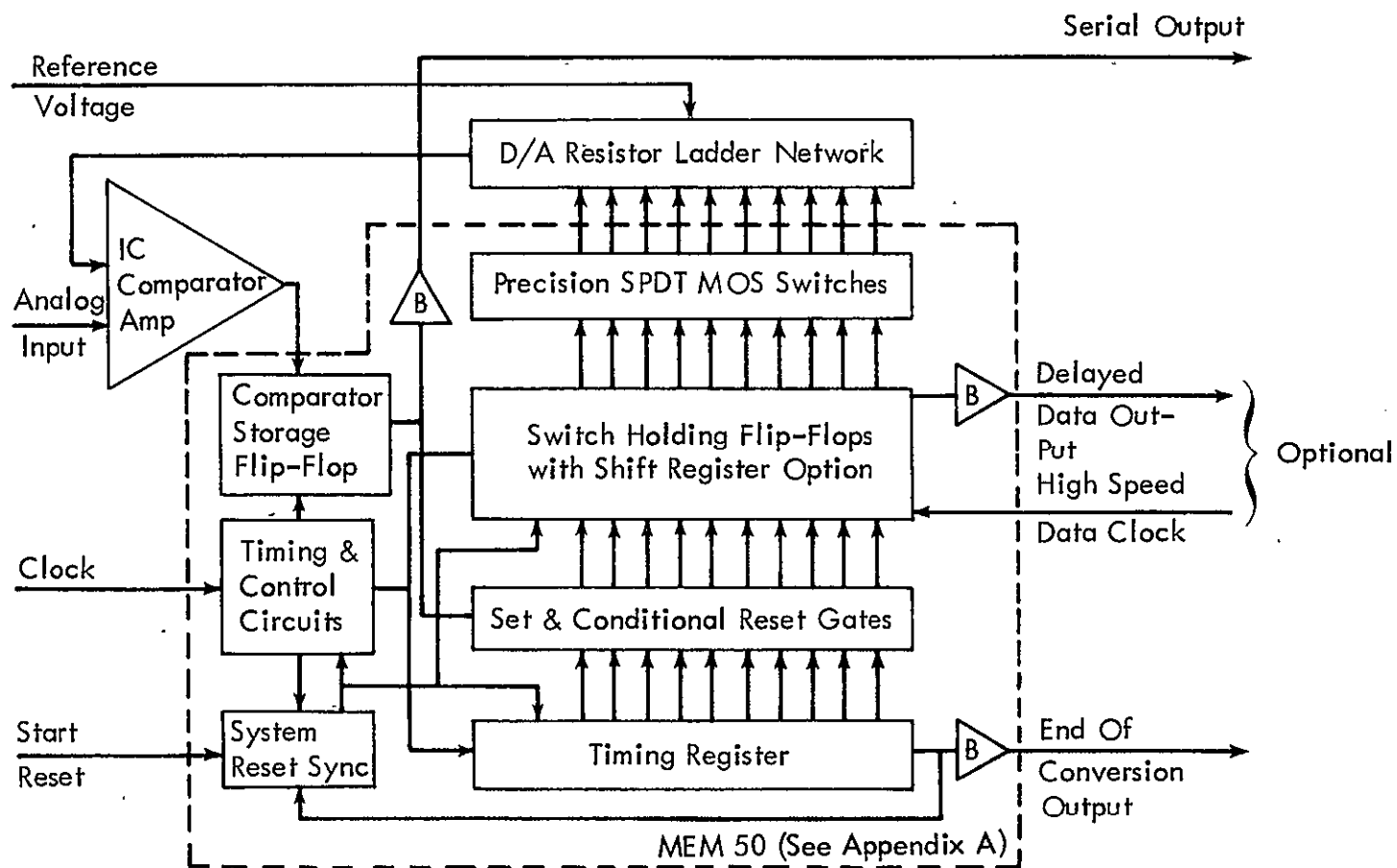


Figure 13.3-8. Analog-to-Digital Converter

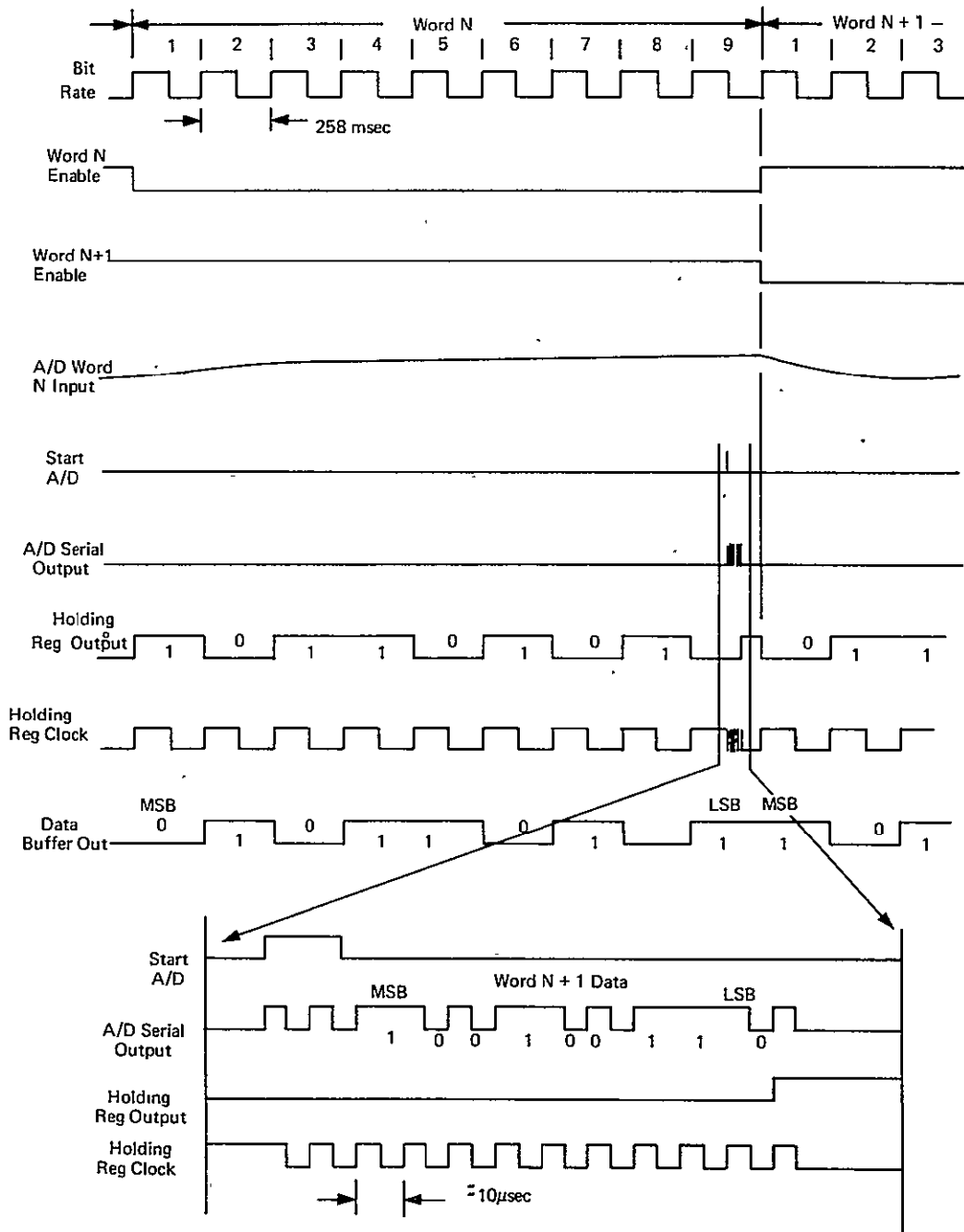


Figure 13.3-9. A/D Converter Timing

The function of the data buffer selector is to multiplex digital and digitized analog data into a coherent, level conditioned, split phase bit stream. A PCM filter conditions the data for modulation of one of the VHF transmitters. The X-band buffer is used to supply an isolated output to the X-band transponder. The calibration source is a voltage reference independent of the ADC. Its outputs are fed into the analog multiplexer as are other analog signals to provide accurate calibration. Synchronization code and status bits are also applied at the first stage of multiplexing. Codes are generated, with no hardware impact, by the manipulation of parallel signal sources at the multiplexer inputs.

Dwell control is accomplished by using a dwell timing control and a dwell channel selector. The CDDs supply a selection signal to indicate which dwell mode is being selected. This is followed by 128-Hz clock, dwell data, and load signals. With this decoder input, the dwell timing control logic determines when to begin actual dwell operation. The dwell channel selector replaces the internal counter sequence with the dwell channel address. The minor frame introductory words are inserted at proper intervals.

The recommended GSFC 27-bit code taken from Appendix A, Table 1, of the "Pulse Code Modulation Telemetry Standard," dated January 27, 1966 is:

111 110 101 101 001 100 110 000 000.

In view of no presently known special code requirements, the GSFC-recommended code is assumed. The flexible DACU design allows pattern changes with minimum hardware impact.

Low, medium, and high calibrations voltages are as follows:

	<u>Voltages</u>	<u>Code</u>
Low	0.265 ± 0.0005	000 011 010
Medium	2.565 ± 0.0005	100 000 000
High	5.095 ± 0.0005	111 111 101

The codes associated with each chosen voltage exercise all levels of the 9-bit A/D output. Zero and full scale voltages are not included since deviations beyond these extremes reveal no information.

Figure 13.3-10 details the characteristics of dwell and normal formats. The normal format sequences through 99 minor frame rate words and 16 subcoms of 16 words each. Identification of data allocation (analog or digital, parallel or serial, single ended or differential) is given. In the dwell mode, the format begins normally for the first 16 words ending with X4. Then, depending on whether dwelling on a minor frame channel, Xn, or subcom, Sm, the formats are as shown.

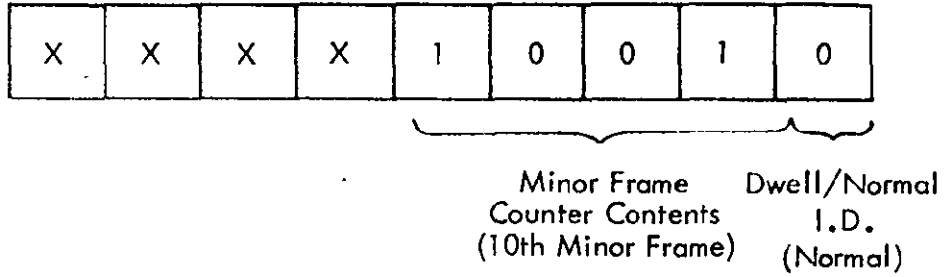
13.3.2.1.1 Input Signals

Analog inputs are 0 to 5.11V with $\pm 25V$ fault voltages allowable without permanently damaging the DACU circuitry. Coding is as follows:

$0.005 \pm 0.005V$ - 000 000 000

$5.115 \pm 0.005V$ - 111 111 111

Normal Mode



Dwell Mode

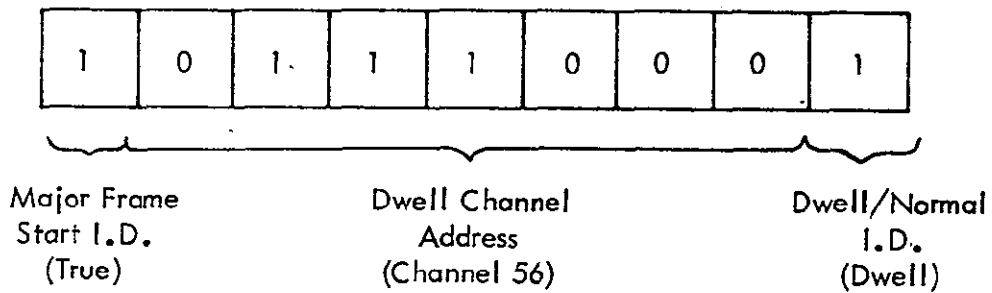


Figure 13.3-10. Status Word Description

Digital inputs are defined as follows:

-2.0 to 0.4V logical 0

2.4 to 8.0V logical 1

The dwell command interface is a low-level serial address transfer from the CDDs. When a DACU is designated the addressee by the 6-bit Data Word Address block in the Command word format, the CDD enables the particular U_i and V_j lines that terminate in an AND gate in the addressed DACU. This enabled AND gate output is used to gate the clock, data word (dwell address), and the execute tone to the DACU from the three respective buses. A commandable regulator is employed at power supply front end to allow low level command interfacing. The power supplies of each DACU may be switched on or off independently. The 100-kHz input is a low level signal interfacing with differential line receivers. Each DACU is capable of receiving clock inputs from either of the redundant spacecraft clock units. The real-time clock outputs are input to the DACU as a digital word for inclusion in the format.

13.3.2.1.2 Output Signals

The multiplexed data outputs to the DSU will be split-phase, 0 and 5V from a source impedance ≤ 1 kilohm. The sync timing outputs to the subsystems will be 0 and 5V from a source impedance ≤ 1 kilohm.

13.3.2.2 Design Description

13.3.2.2.1 Power Circuitry and Power Reset

Each DACU contains circuitry to provide isolation and conversion from the spacecraft to regulated buses ($+V_{R1}$ and $+V_{R2}$). A standard commandable regulator is $\pm 12V \pm 10$ percent, $+5V \pm 10$ percent, and $-22V \pm 5$ percent, while $+6V \pm 7$ percent derived from the other DACU remains on for FET protection. However, should the $+6V$ supply fail, a diode in the substrate line serves to prevent a DACU with a malfunctioning $+6V$ supply from loading the inputs to the redundant DACU. As additional protection, off DACU supply voltages are allowed to drift up to $+6V$ in the manner described below:

- Assumption. If the ground return line of the power supplies (except $+6V$) is isolated from ground and resistively coupled to $+6V$ when the power supply is switched off, then all power supply outputs drift to $+6V$ in the off state.
- Implementation. A PNP transistor is inserted between the switched power supply secondary commoned center-taps and ground, emitter to ground. The base is resistively coupled to the $-22V$ supply and a "pull-up" resistor inserted from base to emitter. The commoned center-taps of the switched power supply are resistively connected to the $+6V$ power output.
- Operation. As the switched power supply is turned off, the $-22V$ output drifts towards ground potential. This allows the pull-up resistor to turn the transistor off, thus isolating the switched supplies from ground. The resistor from the commoned center-taps of the switched supply secondaries is connected onto the $+6V$.

The reset circuit senses the application of power and presets DACU registers to initial states.

13.3.2.2.2 Dwell Logic

The CDD-DACU interface for dwell mode consists of DACU normal or dwell mode selection and dwell address loading interface signals. The dwell address is loaded in the following sequence: (1) a DACU is selected for dwell mode by the CDD distributed interface; (2) dwell load, 128-Hz clock, and dwell digital data are loaded into the channel selector register; (3) an execute command enables the DACU to proceed into the dwell mode at the appropriate time; and (4) a decoder input enables the dwell mode at the start of a minor frame.

Dwell enable, which is generated from the combined inputs of the dwell timing control unit and word counter, is used to insert the dwell address into status word (4), leaving sync, calibration, and words 7 through 15 in the format as under normal operating conditions.

13.3.2.2.3 Clock Interface and Bit Counters

The clock interface encompasses clock selection, 256:1 frequency divider, bit counter, and TTL-to-MOS level converters. The spacecraft clock inputs to the DACU are ORed after interfacing is accomplished. This 99.42-kHz clock is normally routed after conversion to MOS levels to the ADC. However, if neither spacecraft clock is operational, the absence of input is sensed by the discrete circuit shown. This circuit then enables an internal stable multivibrator at $100 \text{ kHz} \pm 5 \text{ per cent}$ to supply the required clock.

The relatively high input frequency is required to allow sufficient settling time for analog filtering at the ADC front end and to allow A/D conversion to take place in 9-bit intervals of the 390-Hz clock. Countdown to 390 Hz is provided by a 256:1 frequency divider consisting of an 8-bit binary down-counter.

The bit counter is an error-correcting ring counter, rippling a "1" by the nine MOS level converters during each word interval. This approach to a bit counter has the added advantage of providing the gate drive to the digital multiplexer serializing digital 9-bit words.

The TTL to MOS level converter chosen has gated inputs. This feature is utilized to inhibit the selection of any bit of a digital word while analog signals are being multiplexed.

13.3.2.2.4 Counter/Decoder

The word counter is a 7-bit binary counter of 128 states. A complete cycle of counts indicates a complete minor frame; therefore, the word counter is used to drive the minor frame counter. This unit counts the 16 minor frames per master frame. Ripple counters are used as opposed to synchronous counters because of their reduced complexity and the low clock rates being used. The decoder is immune to transient intermediate codes propagating through the counter. Counter outputs are employed in status generation and also supply normal decoder inputs.

Normal/dwell gating inhibits or enables the word counter outputs into the TTL to MOS level conversion gating when in dwell or normal modes, respectively. The dual input capability of the TTL to MOS converter is once again used to OR normal or dwell addresses into the decoder.

The decoder itself consists of diode arrays providing 84 outputs from 11 unique inputs. Multiplexer control and other control requirements dependent upon format timing are supplied by the decoder at MOS levels. Special timing and subcom dwell multiplex addressing circuits are also shown.

13.3.2.2.5 Multiplexers

The multiplexer includes analog multiplexing, digital multiplexing and serialization, calibration source, and synchronization code and status generation.

The analog multiplexing is accomplished using MEM 5015 LSI MOS devices, each having a 16-channel capacity. Isolation between 8-channel segments (an added failure effects feature) is afforded by MEM 2009 MOS FET gating at the multiplexer output. Each MEM 5015 is power-strobed to conserve power. Only the negative supply is switched in order that the positive substrate voltage may be retained to prevent input loading. (Removal of the positive voltage would cause the inputs to see a low common mode impedance.) Resistor-diode networks on both analog and digital multiplexers provide $\pm 25V$ fault protection at the inputs.

An analog calibration source employs a reference diode supplied from an operational amplifier. The op-amp power supply variation rejection allows a stable voltage to be maintained across the Zener reference diode. A potentiometer-resistor network permits calibration voltage adjustment. Precision resistors used in the voltage divider do not require fine adjustments to produce "medium" and "low" outputs once "high" adjustment is made.

The digital multiplexer is designed for minimum failure effect. The shorting of a single input MOS FET causes the loss of no more than 9 bits of information. Serialization of a 9-bit digital word is first accomplished with gate inputs from the bit counter. Words are selected under decoder control in the second state of digital multiplexing.

The generation of sync pattern is straightforwardly implemented by "hard wiring" the selected pattern to a group of 27 input MOS FETS. The bit counter and decoder sequence through the pattern to produce the serial 27-bit code.

13.3.3 SPACECRAFT CLOCK

13.3.3.1 Functional Description

The spacecraft clock requirements are shown in Figure 13.3-11. The spacecraft clock is redundant within a single unit and supplies redundant signal outputs. Two commandable power supplies controlled by the CDD are used to make either of the oscillator-frequency dividers operative. The oscillator supplies a stable clock for telemetry, for time period counting, and for real-time code generation. A frequency divider is required to reduce the oscillator frequency to a desired clock interval.

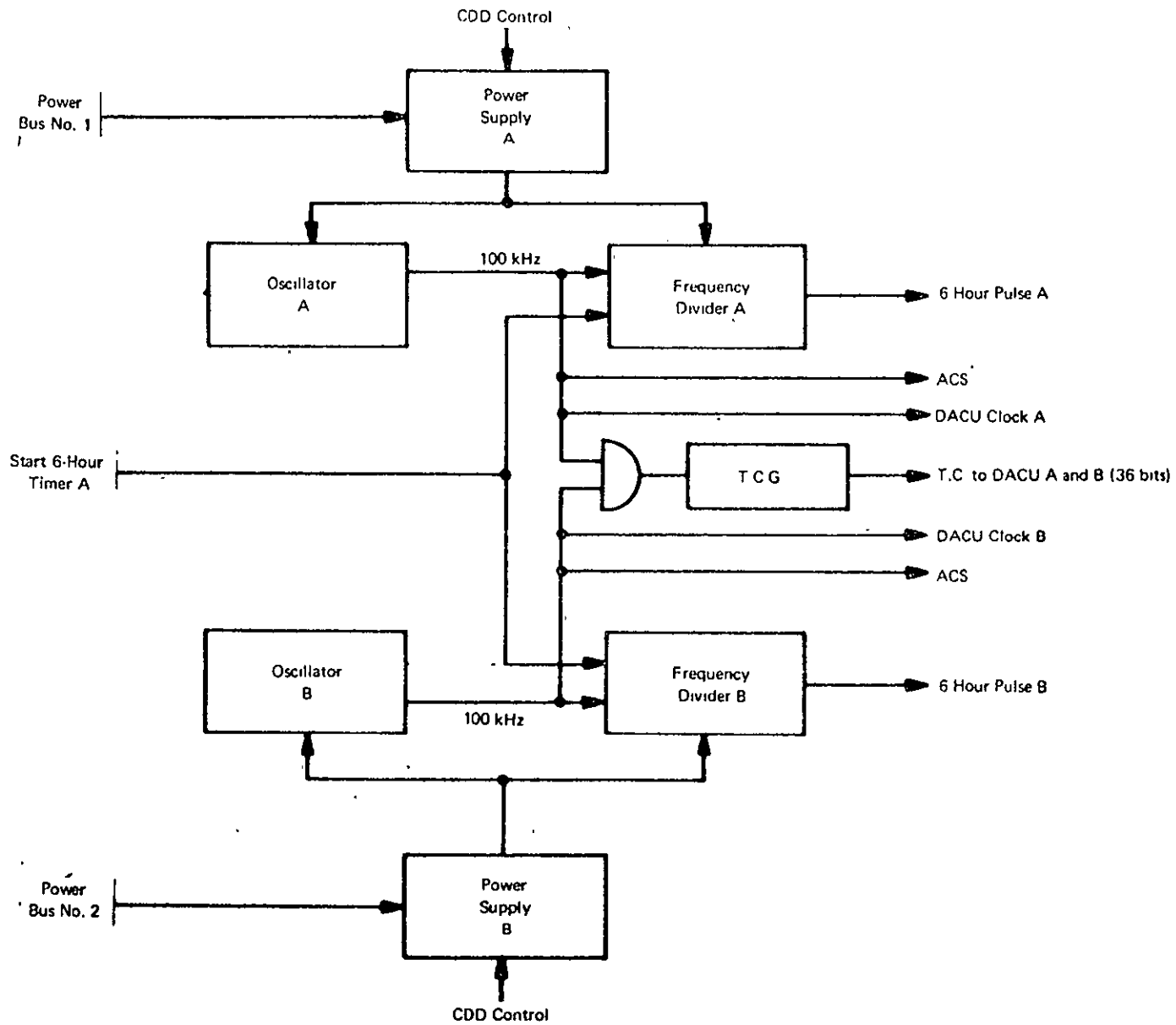


Figure 13.3-11. Spacecraft Clock Block Diagram

Four clock outputs are provided: a 100-kHz signal to the DACUs and the ACS, a six-hour time delayed pulse for automatic radio beacon experiment turnoff, and a 36 bit real-time code for inclusion in the data format. IRIG standards require a long-term bit rate stability of equal to or better than 1 percent. This requirement is exceeded by the 10^{-5} ppm of the 1.6000 MHz master oscillator. Short-term stability is also specified. Time displacement of bit start times in an interval of T relative to the nominal start times established by an average over the preceding interval T shall not exceed 0.1 bit period. The interval T shall be taken as 10 times the maximum period between assured bit transitions.

The time code generator included in the spacecraft clock consists of a 36-bit BCD code which tags each telemetry minor frame to the nearest second. The stability requirements specified above exceed the accuracy required as an input to this time code generator. In the event both oscillators are commanded OFF, the time code generator can be updated by ground command when an oscillator is again activated.

13.3.3.1.1 Detailed Functional Performance

A 100-kHz symmetrical clock is supplied through four line drivers from one of the spacecraft clock sections to the DACUs and the ACS. This frequency allows an accurate time code to be generated by a simple dividing chain. A second output is a 50-msec pulse (6-hour pulse A or B) occurring 6 hours after a 50-msec low level input pulse (start 6-hour timer A or B) has been received from the CDD. The third output is a low level positive true real-time code of 36 bits.

The variation of transmitted bit rate shall be less than 10^{-3} ppm; therefore, the oscillator frequency will be controlled to within 10^{-3} at a nominal frequency of 1.6000 MHz. The time displacement of the 100-kHz signal bit start times in an interval T relative to the nominal start times, established by an average over the preceding interval T, will not exceed 0.05-bit period. The interval T will be taken as 10 times the maximum period between assured bit transitions.

13.3.3.1.2 Interface Requirements

Output clocks and pulses are low level signals with interface characteristics compatible with the DACU, ACS, and radio beacon unit. Input power and start-six-hour-timer inputs shall also possess the same low level interfaces. Table 13.3-4 indicates pin requirements.

Table 13.3-4. Spacecraft Clock Pin Requirements

<u>Signal</u>	<u>No. of Pins</u>	<u>Type Wire</u>
Power	8	
100-kHz	8	twisted pair
TLM	2	
TCG	42	
6-Hour Timing	4	
CDD Interfaces	28	
	92	

13.3.3.1.3 Input Output Signals

The DACU and ACS clock output is a symmetrical square wave at 100-kHz, with signal levels of 0 and 5V. Under normal operating conditions in the DACU, rise and fall times measured at the DACU are less than 125 μ secs.

The signal characteristics of the six-hour pulse are a 50-msec + 10 percent pulse occurring 6 hours after a start-six-hour-timer command. Signal levels are 0 and 5V with 1 msec minimum rise and fall times.

A 50-msec (nominal) start-six-hour-timer command initiates the countdown of the 100-kHz clock to a six-hour interval. Interface signal levels are -5 and 5V.

The real-time code is a 36-bit parallel low level output to the DACU digital multiplexer. The 36 bits are BCD coded into decades representing:

- Seconds
- Seconds/tens
- Minutes
- Minutes/tens
- Hours
- Hours/tens
- Days
- Days/tens
- Days/hundreds.

Update of the time code generator is accomplished by using a serial 9-bit command interface. The assigned bit pattern is as follows:

<u>Bit</u>	<u>Function</u>
1	Spare
2-5	Update data
6-9	Address of decade to be updated.

Each power supply located in the spacecraft clock supplies telemetry information to indicate which clock unit is operative. Telemetry points are

- Power Bus 1 ON/Power Bus 1 OFF
- Power Bus 2 ON/Power Bus 2 OFF

13.3.3.2 Design Description

13.3.3.2.1 Spacecraft Clock Circuit Operation

A μ A710 crystal controlled oscillator design supplies a 1.6000 MHz clock to a 4-stage binary counter which in turn provides the 100-kHz DACU clocks.

The sequence of events relating to the generation of the six-hour pulse is as follows. The 100 kHz clock signal is divided down to 1 pps by a series of five decade counters. This pulse drives the six-hour timer and the TCG. A start-six-hour timer pulse from the CDD enables a chain of $\div 10$ and $\div 6$ counters. These counters are previously initialized to all zeros. Six hours later, the counter returns to all zeros with a transition at the output flip-flop firing a one shot that inhibits further counts. The output of the shot is a 50 msec $\pm$ 10 percent pulse used as the six-hour pulse output to the Radio Beacon. This 6-hour pulse's leading edge is in error by no more than 10^{-3} ppm.

The time code generator generates a 36-bit time code for insertion into the telemetry format. The TCG counts in BCD and, as such, its output is compatible with the IRIG 36-bit time code format. See the Technical Proposal for further details.

13.3.4 DATA SWITCHING UNIT

13.3.4.1 Functional Description

The DSU functional configuration which satisfies the requirements of the T & C Subsystem is shown in Figure 13.3-12. DACU 1, DACU 2, or the EME can each be switched to the FDM unit or either omni-associated transmitter. The special data function uplink transmission can be received by any one of the three VHF receivers; therefore, the DSU can switch any receiver's output to the FDM unit. FDM bypass switches allow the EME or special data functions to directly modulate transmitters 3 or 4 in the event of a failure in the FDM unit.

Requirements include a distributed low level CDD interface, redundant control and switching units, simplex FDM, and four commandable regulated supplies (one for each transmitter).

Conditions and switching modes are summarized below:

- Input subsystems are DACUs 1 and 2, EME(ISE) and Receivers 1, 2, and 3.
- Outputs are applied to VHF transmitters 1, 2, 3, and 4.
- Transmitters 1 (2) and 3 (4) are never on at the same time, since they are operating on the same frequency.
- Each DACU and the EME data must be switchable into any transmitter.

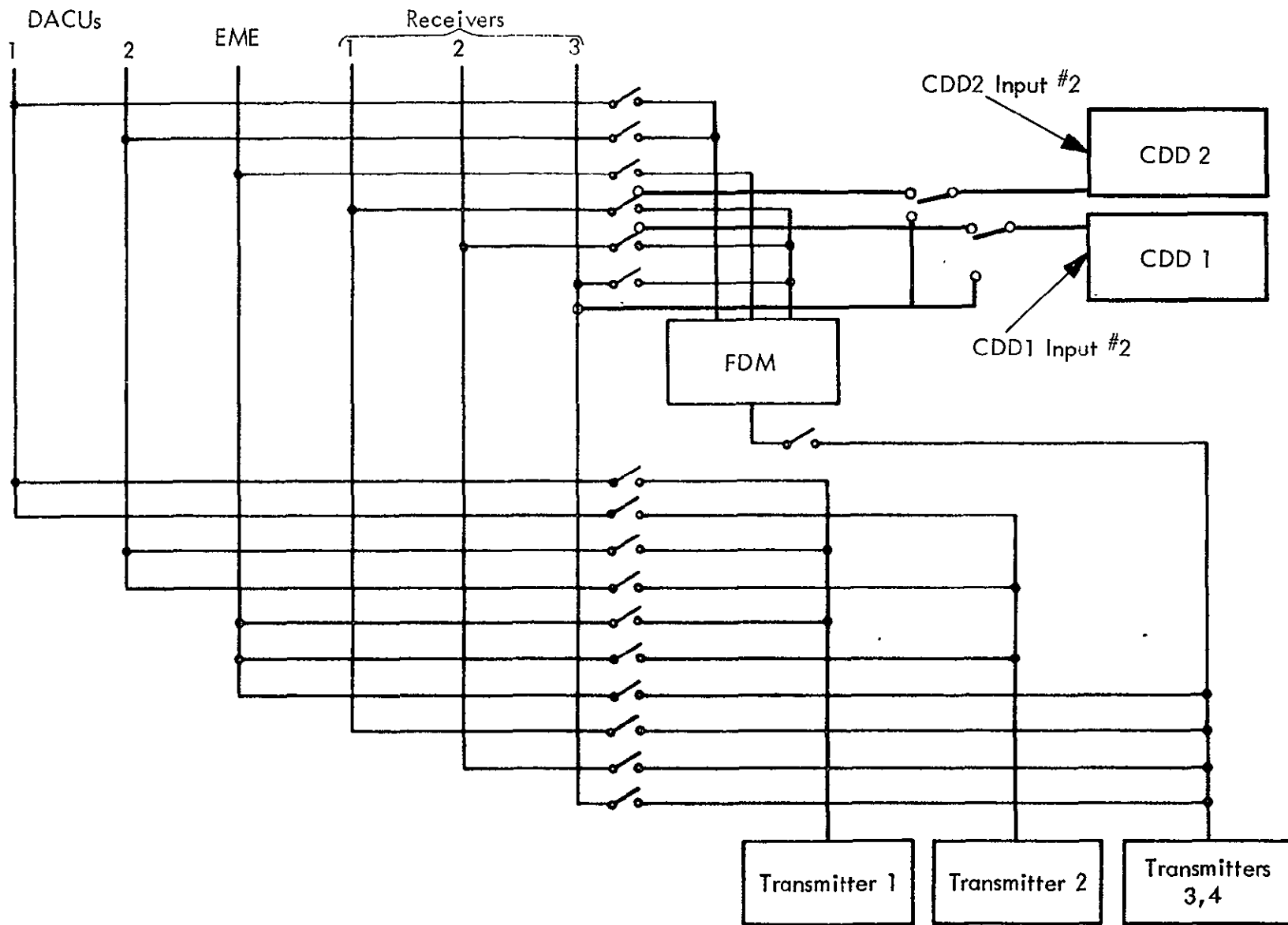


Figure 13.3-12. DSU Switching Functions

- All inputs are switchable to the FDM which is in turn switchable to XMTRS 3/4.
- Normal cross-strapping of Receivers 1 and 2 with CDDs 1 and 2 can be modified to allow Receiver 3 to be an input to either CDD.

The CDD interfaces with the DSU at low level -5 and +5V signals, utilizing transistorized interfaces. Additionally, the interface is a partial decoder for the distributed command system. Using this approach, a coincidence of two CDD command pulses is required for activation of a switching function. All switched signals are low level, with levels and impedance defined in subsection 3.4.1.2 of the Technical Proposal. The power subsystem interface with the DSU consists of positive regulated buses (V_{R1} , + V_{R2}) at 30.5 Vdc $\pm$ 2 percent and return. The power interface circuitry within the DSU is capable of withstanding power interruption, power polarity reversal, overvoltage, undervoltage, and transient conditions.

13.3.4.2 Design Description

The DSU consists of magnetic latching relays, commandable regulators, and a frequency division multiplex system. The FDM is the prime down-link mode. It must be provided sufficient backup for highly reliable function over the two year mission life; therefore, the DSU is implemented full redundant, with FDM bypass switches allowing the EME or Special Data to modulate transmitters 3 or 4 directly.

The DSU switching functions are implemented using Teledyne magnetic latching TD-5 can dpdt relays. A simplex DSU is configured to implement the following basic switching:

<u>Connect</u>	<u>To</u>
Either DACU or EME	XMTR 1
Either DACU or EME	XMTR 2
Either DACU and/or EME and/or RCVR 1, 2, or 3	FDM System input
Receiver 3	CDD 1 or 2

Relays are also provided to bypass the FDM and connect any receiver or EME directly to XMTR 3/4. These switching functions are repeated in the redundant DSU. The FDM is simplex; however, either DSU has the capability to switch the mission essential functions around it.

Only one set of dpdt relay contacts is used to perform the switching functions. The second set of contacts on each relay has its set contact connected to +V and its reset contact connected to ground. The wiper goes to both DACUs for redundant telemetry to the ground station. Thus, DSU status is continuously monitored.

The FDM unit, whose functional design is described in Section 13.2.3, is packaged within the DSU. The implementation of the two voltage controlled oscillators (VCOs), Adder/Isolator, and bandpass filter which comprise the FDM unit is shown in Figure 3.4-9 of the Technical Proposal.

For packaging convenience, the commandable regulators for transmitters 1 through 4 are contained in the DSU housing along with their command terminators. A diode OR circuit is provided on the transmitter 3 and 4 ON command terminators to provide a pulse to the coaxial switch between transmitters 3 and 4 outputs and the prime focus feed antenna so that only the energized transmitter is connected to the antenna.

The command termination array for DSU 1 and 2 uses the relay coil to perform the AND function. In the command matrix several coils are diode ORed to two or three different Y lines. This enables 2 or 3 different commands to energize that particular coil.

13.3.5 T & C VHF EQUIPMENT

13.3.5.1 Command Receiver

The VHF command receiver is a solid-state AM receiver which is used to demodulate command and other audio signals received via the VHF antennas, and to provide audio outputs to the command decoder and special data channels. The received signal is an FSK/AM/AM signal at a carrier frequency of 154.20 MHz or 148.62 MHz. Each receiver has two isolated audio outputs. One of the outputs of receivers 1 and 2 is dedicated to CDD 1 and 2, respectively. The second output of these receivers is normally cross-strapped within the DSU to the opposite CDD, but can be switched to the FDM unit or transmitter 3 or 4 if used in the Special Data mode. Receiver 3 output enters the DSU where it can be switched to the FDM, Transmitter 3 or 4, or either of the two CDD units. Detailed characteristics of the VHF receivers are listed in Section 3.5.1.1.1 of the Technical Proposal. A summary of these characteristics is shown in Table 13.3-5.

Table 13.3-5. Receiver Characteristics

Frequency	154.20 MHz or 148.26 MHz
Noise Figure	8 dB max at standard source temperature of 290°K
Modulation	AM
IF Bandwidth	30 $\pm$ 3 kHz at -3 dB; 110 kHz max at -60 dB
Image Rejection	> 80 dB
Input Impedance	50 Ω
Sensitivity	
Input Level	
Spurious Response	> 60 dB rejection outside the 110 kHz passband
Output Number	2 isolated outputs
Level	6.0 $\pm$ 0.6 volts peak to peak
Frequency	0.3 Hz to 10 kHz
Power Input	180 mW/840 mW at 30.5 VDC (standby/interrogate)
Output Z	10 k Ω

13.3.5.2

Telemetry Transmitters

The VHF Telemetry Transmitter is a solid-state, fixed frequency, phase-modulated unit with a transmitted power of two watts. The T & C subsystem uses four telemetry transmitters, two at 136.23 MHz and two at 137.11 MHz. The modulating signal is either PCM telemetry data, a voice bandwidth analog signal, or a composite FDM signal. Each transmitter is powered by a commandable Load Interface Circuit housed within the DSU. A detailed discussion of transmitter characteristics can be found in Section 3.5.2.1.1 of the Technical Proposal. A summary of these characteristics is shown in Table 13.3-6.

Table 13.3-6. Transmitter Characteristics

Frequency	Fixed Tuned, 136.20 MHz and 137.23 MHz
Phase Noise	6 degrees max at a maximum phase deviation of 2.8 radians peak to peak
Power Output	2 watts minimum
Output VSWR	≤ 2 to 1
RF Bandwidth	≤ 30 kHz
Output Impedance	50 Ω
Modulation	2 radians peak to peak phase modulation
Input Signal	6.0 $\pm$ 0.6 volts peak to peak
Input Impedance	10 k Ω
Power Input	5 watts (when energized)

13.3.5.3

VHF Passive Components

The diplexer in ATS F & G T & C subsystem provides 60 dB of isolation between the transmitter and receiver, with an insertion loss of 1 dB.

The Low Pass Filter provides suppression of spurious and harmonic outputs from the VHF primary feed transmitter to preclude RFI to other S/C systems. The filter provides 60 dB attenuation at 315 MHz and above with an insertion loss of 1 dB at f_0 (136.20 or 137.11 MHz).

13.3.5.4

VHF Antennas

The VHF antenna configuration is a monopole pair located at the extremities of the polar panels. It is designed to transmit and receive circularly polarized RF energy at frequencies from 136 MHz to 154 MHz. The antennas are near-omnidirectional, with patterns which offer 95 percent coverage for the telemetry function and 100 percent coverage for the command function when the polar panels are in the deployed state. The antenna configuration is shown in Figure 13.3-13 and a histogram of the composite antenna patterns for the fully deployed condition is shown in Figure 13.3-14. A typical polar plot antenna pattern for this configuration is shown in Figure 13.3-15. Additional antenna patterns and a complete evaluation of the RF link coverage are shown in Section 3.5 of the Technical Proposal.

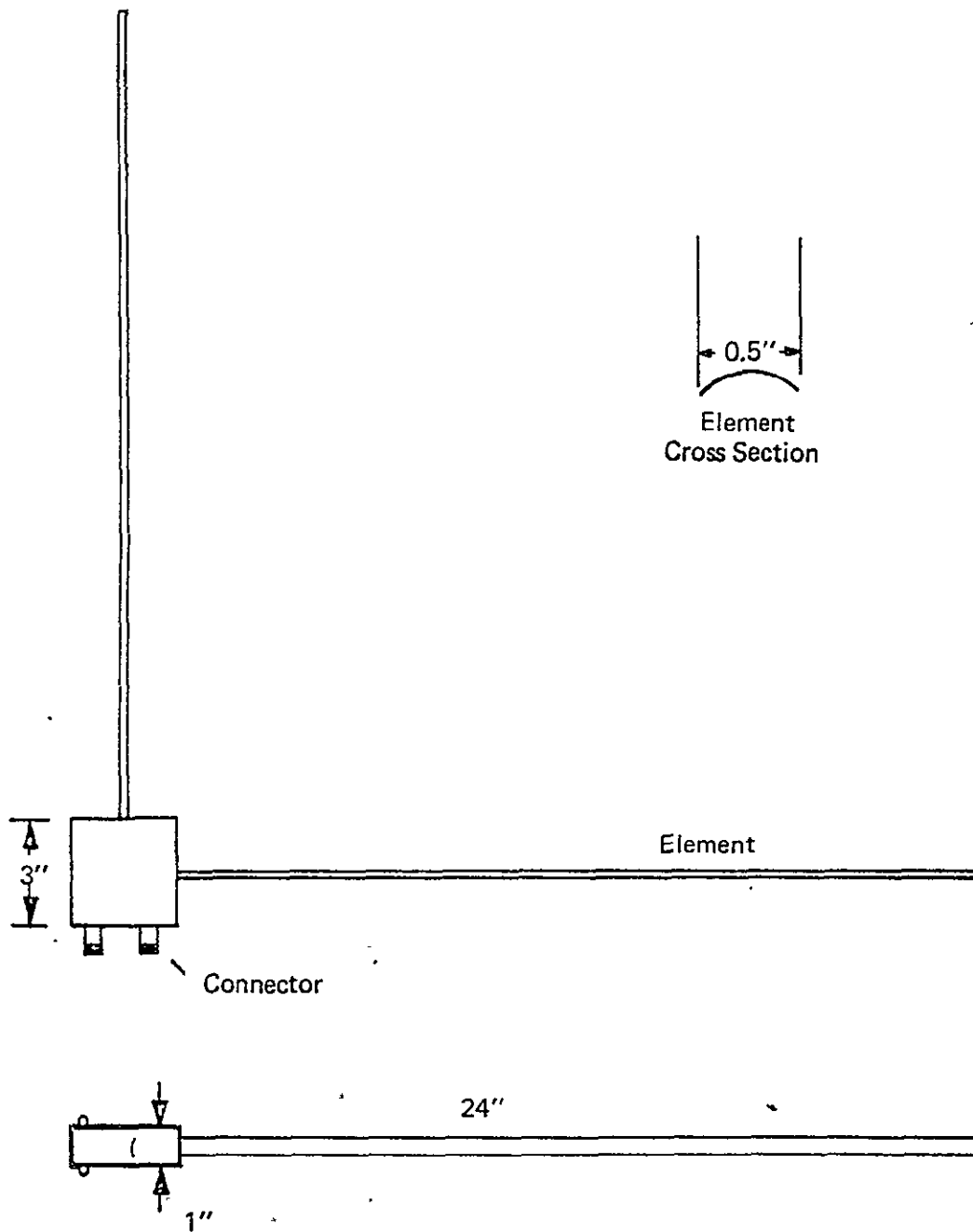


Figure 13.3-13. Antenna Unit

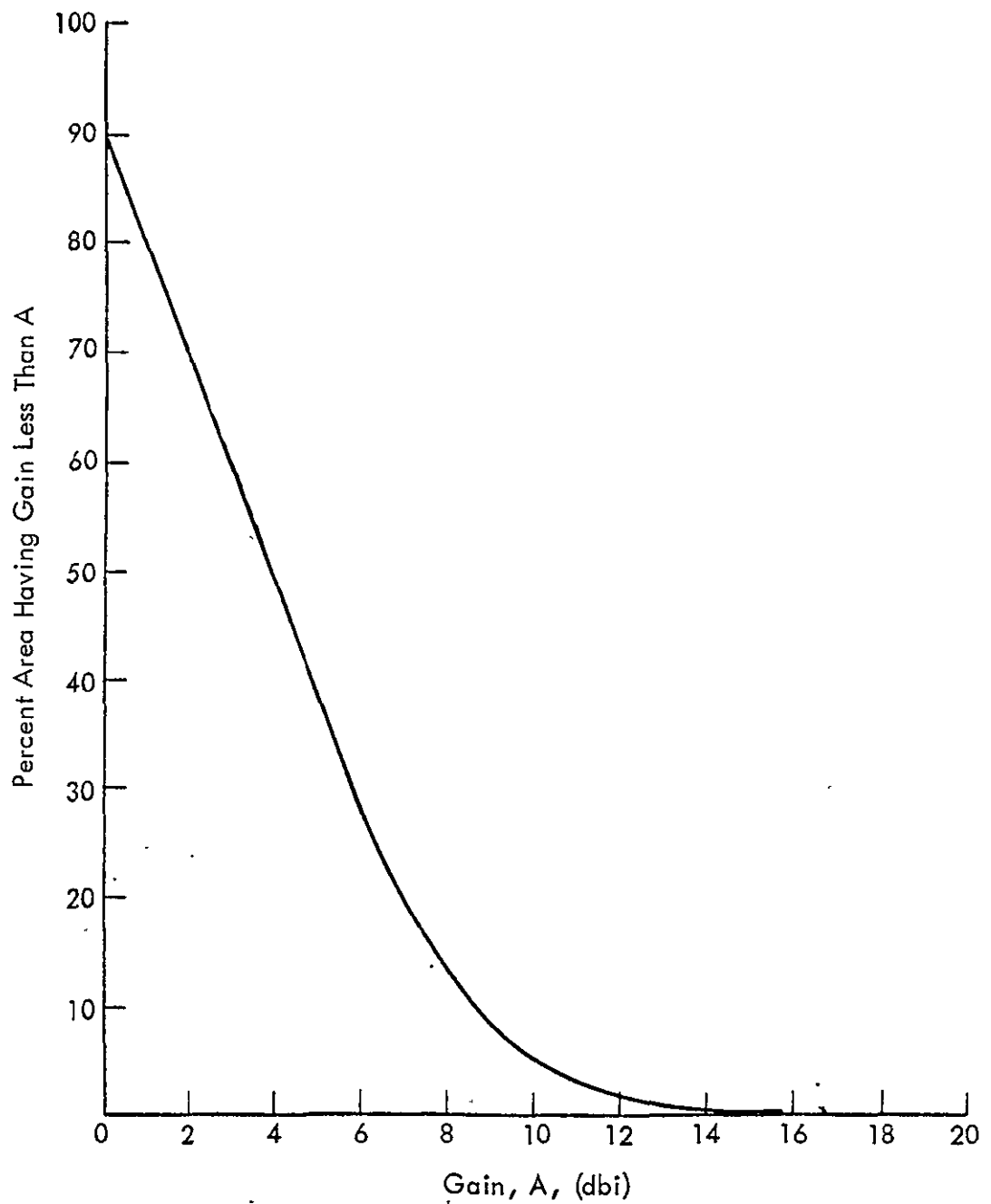


Figure 13.3-14. Composite Antenna Pattern Histogram Deployed Spacecraft
(Both VHF Antennas Operational)

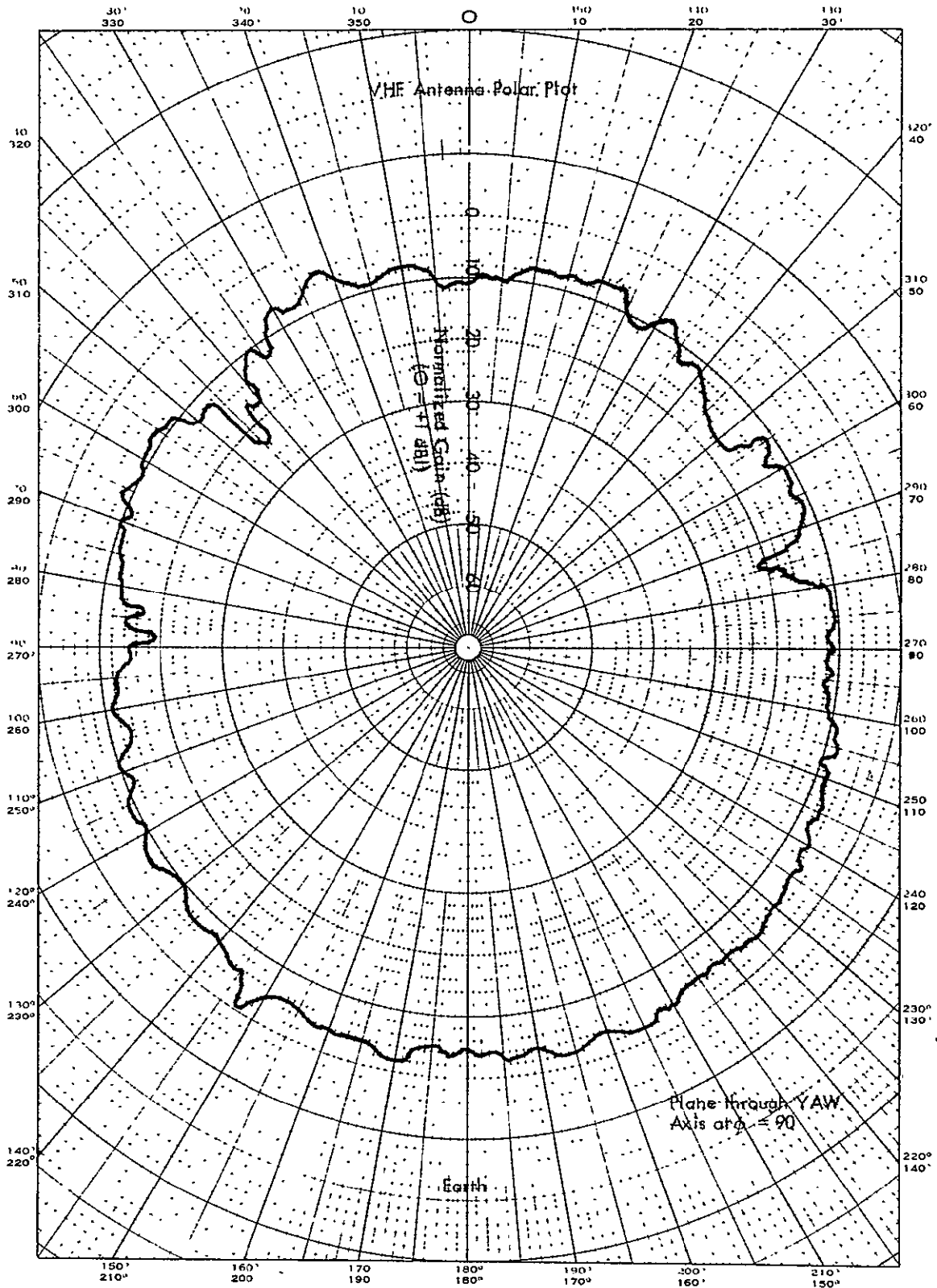


Figure 13.3-15. Typical Polar Plot Antenna Pattern

SECTION XIV

ELECTRICAL POWER SUBSYSTEM

14.1 DESIGN OBJECTIVES

The electrical power subsystem has been designed to fulfill the specific needs of the ATS F&G mission while imposing as few of its own limitations as possible on the mission. Of particular significance in the ATS F&G mission is the wide variation of electrical power levels required by the many different operational modes of the experiments slated for the ATS-F spacecraft. Also, the design is flexible enough to accommodate a possible new set of experiments for the ATS-G spacecraft. The wide variation in power level, the unique physical configuration of the spacecraft, the need for long-life trouble-free performance, the requirement for design flexibility, and many other requirements have determined the final selection for the ATS power subsystem design. These design objectives are defined and discussed in the following paragraphs.

14.1.1 VOLTAGE AND POWER REQUIREMENTS

Certain spacecraft loads have clearly defined operating voltage requirements. Foremost among these are the GFE experiment packages, which are specified¹ to operate from +28 volts ± 2 percent as measured at the input connector of the experiment packages. Of equal importance is the largest single load, the solidstate UHF power amplifier, which requires an operating voltage of +25 volts to avoid over-stressing the power transistors in the final class-C stage. The remaining spacecraft loads are not sensitive to an absolute value for their operating voltage, either because the design can tolerate an operating voltage other than +28 volts, or because additional conditioning is provided at the load.

The highest demands on the power subsystem occur during the prime communications experiments such as PLACE experiment, the spacecraft requires a total normal conditioned power of 432.2 watts; for the ITV experiment, 406.3 watts. These high power loads determine the capacity of the solar array.

The peak conditioned power loads are 506.9 watts for PLACE and 563.8 watts for ITV when the X-band monitor is being used. Although the prime power for these peaks will be derived from the batteries as well as the solar array, the periods are short (in most cases less than one minute), and the battery demands during these peak-load periods do not exceed the battery demands during occult. The occult loads are used to determine the required capacity of the battery.

The Experiment Interface Specification also requires a ripple voltage less than 10 mv rms on the input power line for each GFE experiment. This

specific requirement, as well as the general need for avoiding electromagnetic interference (EMI) dictated the use of active isolation circuits in the power line to each load system.

14.1.2

SOLAR ARRAY PANELS

The configuration of the ATS F&G spacecraft with its 30-foot parabolic reflector imposed many restrictions on the location and configuration of the solar array panels. The solar panels have been located outboard of the reflector in such a manner that the reflector casts a minimal sun shadow on the array at any time throughout a day or a year (considering the maximum angle of the spacecraft to the plane of the ecliptic). Conversely, the panels cast a minimal sun shadow on the reflector to prevent undesirable reflector thermal gradients resulting in distortions and loss of rf gain. Furthermore, the panels are designed to minimize the disturbance torques around the spacecraft center-of-mass caused by solar pressure in each of the three control axes. Finally, the solar array panels are configured to provide the desired power output at all times throughout the day (i. e., at all anticipated solar angles including offset pointing) to avoid imposing power restrictions that would limit the flexibility of experiment operations. The solar array design provides a constant output throughout the day with minimum support from the batteries. The solar array has been designed to yield a minimum number of mechanical deployment joints and no mechanical deployment sections are used on the panels.

The solar array design provides sufficient effective area to develop the power required to support the normal power load requirements of the experiments. As a primary design goal the array is sized to support the ITV experiment in its normal mode (without the X-band monitor) during any sunlit four-hour period of the day. At the same time it will support the Environmental Measurements Experiment (EME) for twenty-four hours per day at any time of the year except during occult, for at least two years in orbit. All other operating modes of the spacecraft require less power, except the PLACE experiment which demands 8 percent additional power.

As a secondary design goal, the solar array supports the PLACE experiment for at least the first year in orbit, and is able to support the PLACE experiment at any time during the second year, either by drawing power from the battery, or by turning off the EME experiment during the period of peak load. With the power available from the array sized in accordance with the above goals, all other combinations of anticipated experiments can be accommodated during the two year mission.

The detail design features of the array provide protection from the anticipated environments (such as radiation, abrasion, thermal gradients), and

facilitate fabrication, testing, handling, and repair. The selection of the solar cells series-and-parallel configuration has considered the required voltage output, the physical limits of the panels, the effects of failure modes, and the effects of solar angle variations (including temperature effects). The solar array panels have been modularized into submodules to reduce manufacturing costs and simplify testing.

14.1.3

SPACECRAFT BATTERY CONFIGURATION

The spacecraft battery configuration will support the "house-keeping" subsystems during ascent, acquisition, and occult; and also during failure

modes that might result in temporary loss of solar array power. As a secondary function, the batteries provide support to the solar array during short-duration peak loads that exceed the capacity of the array.

Nickel-cadmium batteries are used in this configuration, and the batteries are sized so that the maximum depth of discharge for any single battery (two battery system) does not exceed 40 percent during normal operation in order to minimize the weight and still achieve the required discharge cycles. Charging techniques that require any portion of the solar array to be dedicated solely to this purpose have been avoided to permit maximum flexibility in the use of the solar array power to support the spacecraft mission requirements. Charge rates equal or slightly exceed the C/10 rate in order to maintain the batteries safely at full capacity. Charge control is maintained to satisfy the battery-temperature/voltage-charging requirements to prevent over-charging or undercharging the batteries.

14.1.4

POWER SUBSYSTEM PERFORMANCE

The reliability goals set for the power subsystem require a design that will assure a reliability of at least 0.95 for two years of spacecraft operation in orbit. Although it can be assumed that the solar array output will continue to degrade with time because of the radiation environment, the design of the power subsystem precludes any other item that would result in further degradation of the subsystem performance beyond that determined by the solar array output or the battery charge/discharge cycle capabilities. The design goal is to provide continued operation within the above restriction for at least five years.

To achieve maximum confidence in the power subsystem, well-proven, flight-tested design techniques have been selected to the maximum extent possible within the particular constraints imposed by the ATS F&G mission requirements. For example, fixed, rather than rotating, solar panels are used. Also, regulation techniques, such as shunt regulators that require extensive wiring between the solar array panels and the Earth Viewing Module (EVM), have been avoided because of the long wire runs and the flex joints occasioned by the need for deploying and positioning the solar panels outboard of the reflector. Furthermore, the power subsystem design provides sufficient redundancy such that any single wire or component can fail without causing a loss of the mission; the command subsystem will remain powered as long as power is available from any source on the spacecraft. Finally, the power subsystem design is readily testable at the manufacturing facility and at the launch pad, both to minimize cost and to establish confidence in the power subsystem performance.

SUMMARY OF DESIGN OBJECTIVES

In review, the design objectives that are imposed on the power subsystem can be summarized as follows:

- Provide +28V $\pm 2\%$ to GFE experiments (ripple less than 10 mv rms).
- Provide + 25V for the solid state rf power amplifiers.
- Provide line isolation circuits for each load.
- Locate solar panels to avoid mutual shadow interference with the reflector or other spacecraft elements.
- Configure solar panels to minimize solar disturbance torques.
- Configure solar panels to maintain a constant power output at all times throughout the day with minimum battery support.
- Size solar panels to support combined ITV and EME experiment loads at end of 2 years.
- Size battery for support of "housekeeping" loads during occult when no solar power is available.
- Limit battery depth of discharge to less than 40% for a single battery (normal operation).
- Establish battery charge rates that equal or slightly exceed C/10 to maintain full battery capacity.
- Provide system design reliability that exceeds 0.95 for 2 years.
- Provide an operating life that exceeds 5 years, with performance reduced only by solar array degradation and/or battery life.
- Provide redundancy where needed to avoid loss of mission with a single-point failure.
- Provide a back-up power connection to the batteries for the command subsystem during failure modes.
- Allow for ease of testing of the power subsystem at the manufacturing facility and at the launch pad.

14.2.2

LINE CONDITIONING

Many methods of line conditioning were considered for this application. Previous ATS spacecraft limited the line voltage by use of a solar array shunt limiter. However, the application of these techniques becomes cumbersome when applied to the ATS F&G configuration because of the nearly 40-foot separation between the deployable solar array panels and the power conditioning equipment in the EVM. Partial shunt regulation requires the addition of many costly taps on the solar array panels as well as many long lines from these taps to the control devices. The cost of designing and implementing a tightly-controlled partial solar array shunt voltage regulator or limiter, together with the difficulty of performing adequate dynamic testing on a solar array or regulator simulator, led to the early rejection of both the shunt limiter and the shunt regulator as an optimum method for load-bus voltage limiting or regulation.

The recommended configuration shown in Figure 14.2-2 depends upon the use of switching-type, non-dissipative, buck regulators to limit the load bus voltage variations and to provide current amplification within the EVM at the load end of the power transmission lines. In this configuration, the solar array design is not compromised by the need for additional shunt taps, the wiring from the deployable solar array panels is minimal, and the power subsystem design is not critically sensitive to the prediction of the peak power point at end of life. By allowing the solar array bus voltage to "seek its own level", the full power capability of the solar array is available to the spacecraft at all times; thus, more power is available to the mission during the critical early life of the spacecraft than would be available with a shunted solar array. Furthermore, because the solar array voltage is designed to be considerably higher than the final load voltage, the power loss due to the voltage drop in the solar array power lines is reduced, and the size and weight of the lines themselves can be reduced.

Recent developments in the design of switching-type, non-dissipative regulators, together with the availability of new parts, have resulted in the manufacture of spacecraft power regulators designed to operate at input voltages up to 125 volts. As a consequence, the solar array can be designed to provide a voltage sufficiently high to charge a battery whose discharge voltage exceeds that required for the regulated output bus voltage without overstressing the switching regulator inputs. This technique completely eliminates the need for boost-type regulators or chargers and instead, allows the use of the more reliable and less complex buck regulators and chargers.

The power subsystem design makes full use of the recent improvements in regulator designs by selecting a nominal solar array peak power point voltage at approximately 50 volts. This is more than sufficient to charge a 28-cell nickel-cadmium battery with a nominal charge voltage of 43 volts and nominal discharge voltage of 33.0 volts. With the battery output diode-coupled to the solar array bus, the resultant design is the essence of simplicity; consisting of two types of buck, non-dissipative, regulators; one (or more) in-line regulators to control the output bus;

and an off-line regulator configured as a battery charger. Figure 14.2-2 depicts the basic elements of the configuration for the power subsystem.

The unregulated solar array bus is normally above 50 volts. This bus voltage is sufficient to back bias the diode connecting the battery to the solar array bus and to provide the battery charge voltage. The charger is current limited and regulated to a nominal +43 volts, which is temperature compensated to avoid both overcharging and undercharging the battery. The unregulated bus is supported by the battery at a voltage greater than +31.5 volts (allowing for wiring, diode, and connector voltage drops) when sufficient solar array power is not available for the loads. When the solar array output voltage peaks due to low temperatures and light loads on a new array, values as high as +99 volts can be reached for brief periods. Thus the 50-volt nominal unregulated solar array bus can vary between +31.5 volts during occult and +99 volts for a few minutes just after occult.

Two types of in-line, buck-type, non-dissipative regulators are used in the power subsystem configuration, these two types differ principally in their power rating and their regulated output voltage. The +25 volt, 280-watt regulator is used to regulate the bus supporting the rf power amplifiers which require a 25-volt input to avoid overstressing the rf power transistors. The 30.5-volt, 250-watt regulator is used to regulate the bus supporting the remainder of the spacecraft loads, including the GFE experiments. The value of +30.5 volts was selected to allow for a 0.5 volt line drop and a 2.0-volt drop across individual load interface circuits (LIC's) that control and isolate each load, and provide final regulation at +28 volts $\pm$ 2 percent.

14.2.3 LOAD INTERFACE CIRCUITS (LIC)

Except for the rf power amplifiers and the command receivers and decoders, each load center in the spacecraft incorporates a standard load interface circuit for its interface power connection to the 30.5 volt bus. These LIC's perform the following functions:

- Provide a regulated 28 volts at the load
- Provide line isolation
- Provide overload protection
- Provide ON/OFF command control
- Provide overvoltage protection
- Provide undervoltage protection

Except for the GFE experiments, each LIC is physically located within the component it controls or within a component that controls other components. Separate LIC's required for the GFE experiments are located as close to their experiments as practical to minimize line drops and noise pickup. Because of the location

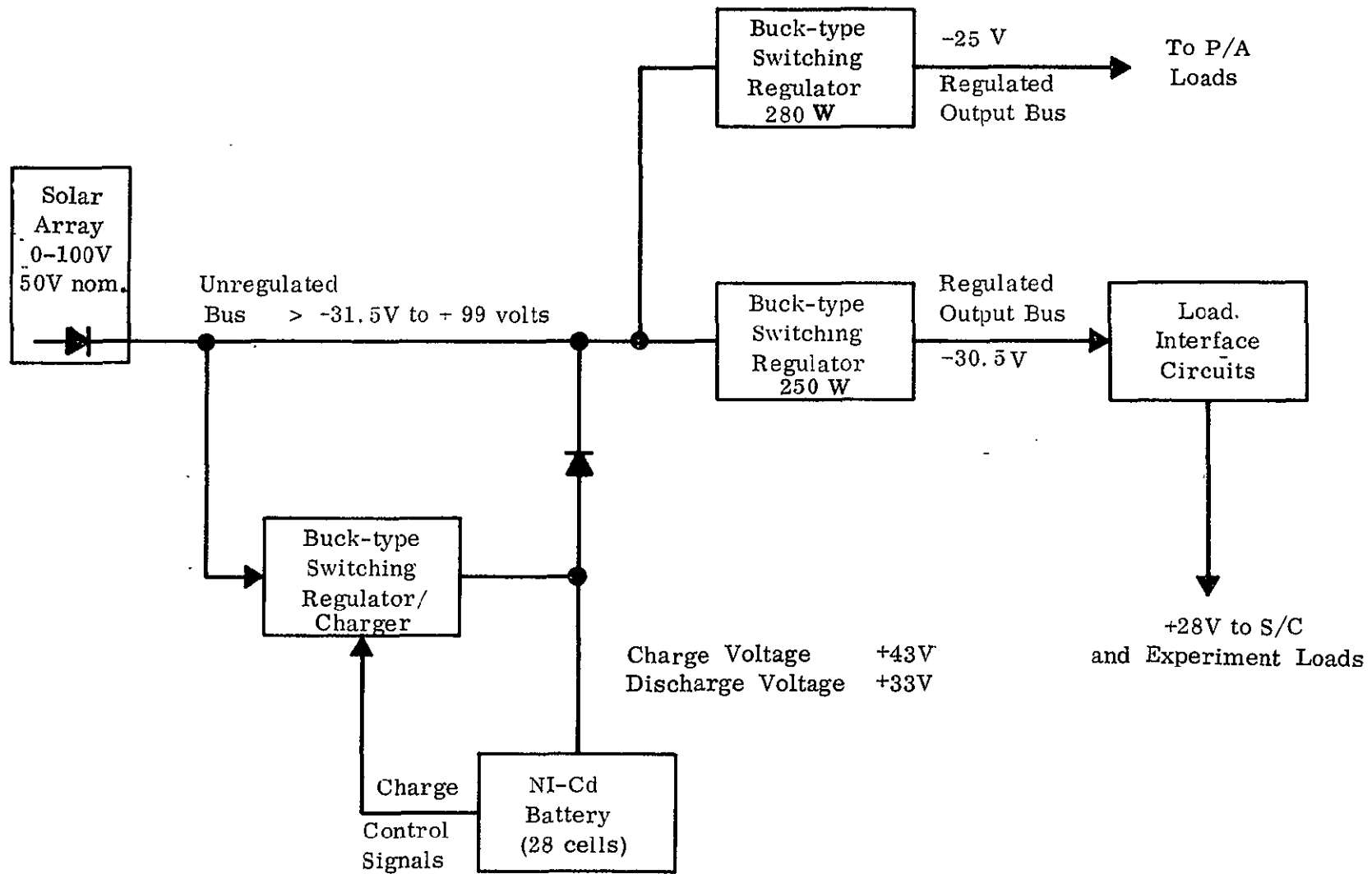


Figure 14.2-2. Simplified Block Diagram of Power Subsystem

of the Environmental Measurements Experiment (EME) on the hub (approximately 17 feet from the EVM, the line regulation is specified at ± 3 percent, rather than ± 2 percent, to allow for the additional line drop.

Two types of LIC's have been designed for the ATS F&G application, differing only in power handling capacity. Both circuits use a simple series pass transistor as the control element. The 15-watt LIC uses a 2N5153 pup transistor as the series pass transistor, while the 60-watt design uses a 2N5005 pup transistor.

14.2.4 SUMMARY OF DESIGN APPROACH

In summary, the power subsystem configuration shown in Figure 14.2-2 provides isolation for all of the high power (rf amplifier) +25 loads from the other spacecraft loads by operating them directly from a dedicated non-dissipative regulator. The remaining loads, including the GFE experiments, are isolated from each other by the load interface circuits. Because final regulation is controlled by LIC's, the 30.5-volt regulator acts essentially as a preregulator to limit the voltage drop (and therefore power dissipation) across the series regulators in the LIC's. Thus, high efficiency is maintained in the power subsystem by using non-dissipative regulators to regulate the few large loads directly, and to limit the voltage to the many distributed series regulators (LIC's) that support the remaining lower power spacecraft and experiment loads.

14.3 SELECTED DESIGN

The selected power subsystem described in the following paragraphs is based on the design objectives in paragraph 14 and the design approach in paragraph 14.2. Refer to Figures 14.3-1 and 14.3-2 for functional block diagrams of the selected design. The power subsystem consists of the following units:

- Two solar array panels
- Two battery chargers
- Two 28-cell nickel-cadmium batteries
- Three 30.5 volt, 250 watt regulators
- Two 25.0 volt, 280 watt regulators
- Eight LIC units (for GFE experiments)
- One Power Control Unit (PCU)

The Power Control Unit (PCU) houses all of the circuitry shown in Figure 14.3-1 except for the discrete components listed above and the battery disconnect plugs which are part of the harness subsystem. The Power Control Unit functions as a line distribution center.

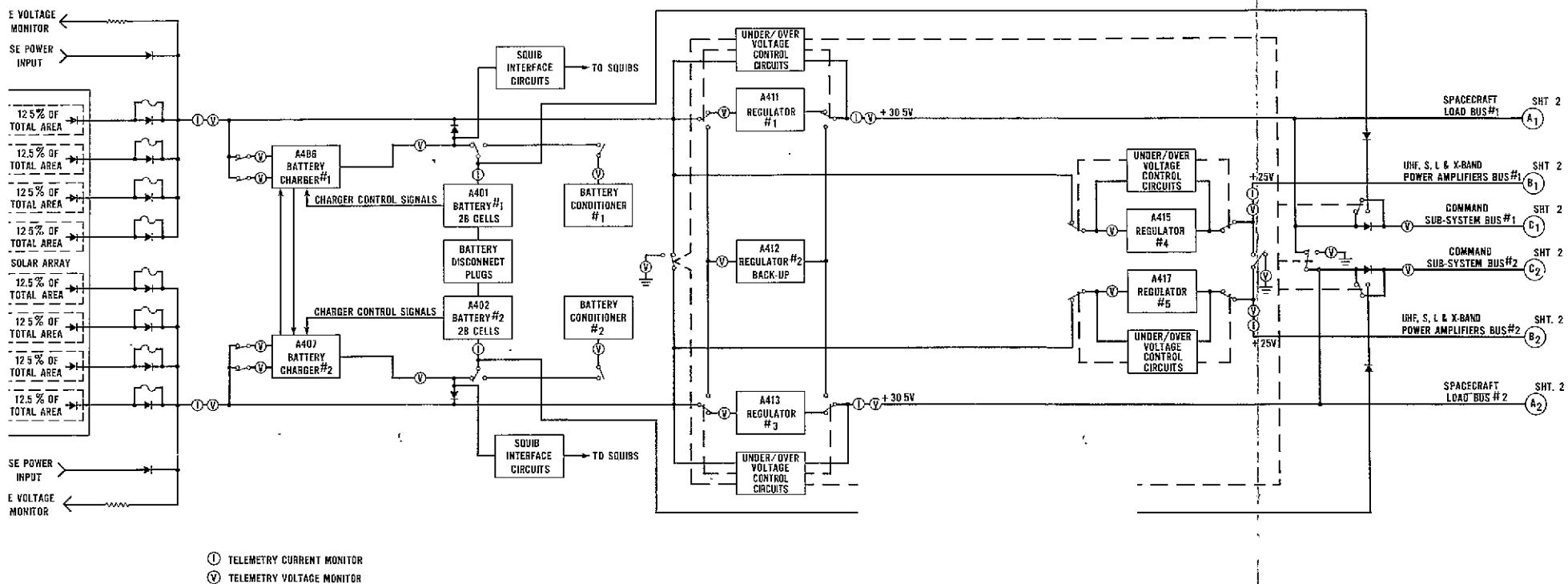


Figure 14.3-1. ATS Power Subsystem Block Diagram

14-13/14-14

FOLDOUT FRAME 1

FOLDOUT FRAME 2

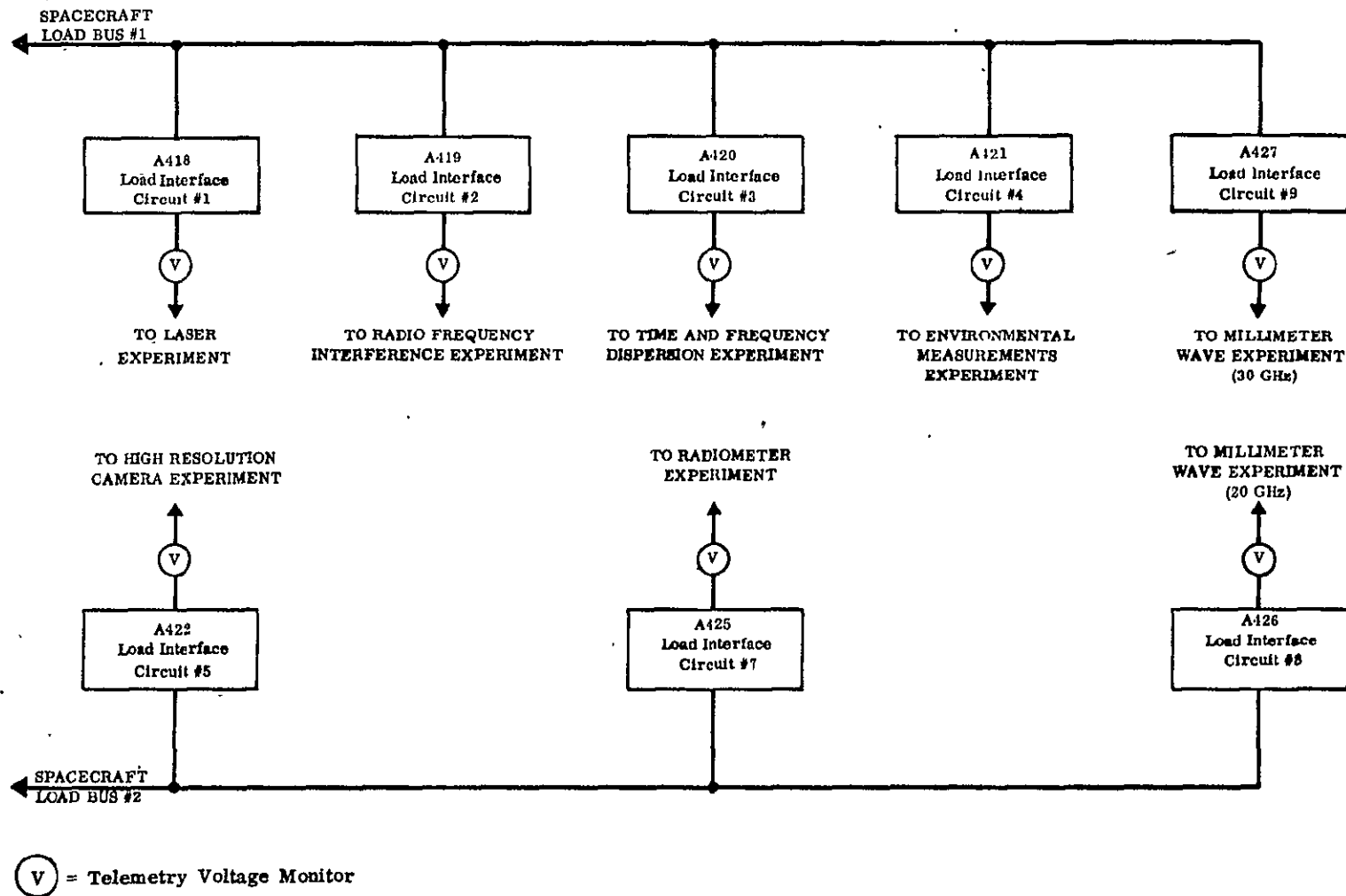


Figure 14.3-2. Power Subsystem Block Diagram, Experiment LIC's

The solar array consisting of two hemicylindrical panels provides a cross-sectional area which is symmetrical about the roll and yaw axes and as close to the center of mass on the pitch axis as possible to minimize the disturbance effects of solar pressure. The effective area and configuration of the two panels provides complete 360-degree coverage so that the power output on a daily basis remains essentially constant. The orientation of the active portions of each panel is such that together they form a cylinder with respect to the sun as shown in Figure 14.3-3. The absolute magnitude of the array output is influenced by one or more of the following:

- Seasonal variations in solar angle incidence.
- Offset pointing of spacecraft Z axis.
- Radiation Degradation
- Temperature.

The total area of the array is 208 sq. ft. with a cell area of 179 sq. ft. resulting in a packaging efficiency of 85.5 percent which is reasonable and obtainable. The total weight excluding the support booms is 175 pounds. The array contains 41,600, 2 x 2 cm, solar cells. The cells are connected in a series-parallel matrix to provide the required voltage and current throughout the environmental variations of illumination, radiation, and temperature. Each panel is made up of 10 replaceable modules as shown in Figure 14.3-4. Each module has eight strings made up of four 2 x 2 cm cells connected in parallel, and 65 cells connected in series. Two of these modules are series connected to obtain 130 cells in series and 8 parallel strings. These strings of cells are interconnected to minimize magnetic flux, while the interconnected parallel strings are staggered around the array by 18° to minimize the power loss at any given time of the day. This arrangement results in a loss of only 1/4 of a panel, or 1/8 of the total array should a feed line-short or open circuit occur.

Each of the parallel strings of solar cells operates at a temperature different than the other electrically-paralleled strings. The temperature of each of these strings varies with the sun orientation, hence time of day. Since the end-of-life voltage is fixed at that required to recharge the batteries (50 volts), the number of series solar cells was optimized such that the composite solar array, not any individual string, operated at its maximum power point at the two-year solstice design goal.

The solar array capability is a function of time in orbit as well as time of year (solstice or equinox). Figure 14.3-5 shows the solar array voltage vs. current (EI) curves (see Figure 14.3-6 for power) for a new array at equinox and a two-year degraded array at solstice with 4-degree offset pointing for ITV operation. The power required at the solar cells for the ITV experiment is 487 watts and is derived as shown in Figure 14.3-7. The array has been designed to exceed this requirement and is capable of supplying 498 watts throughout the illuminated 24-hours orbit with less than 1 percent variation due to daily solar orientation.

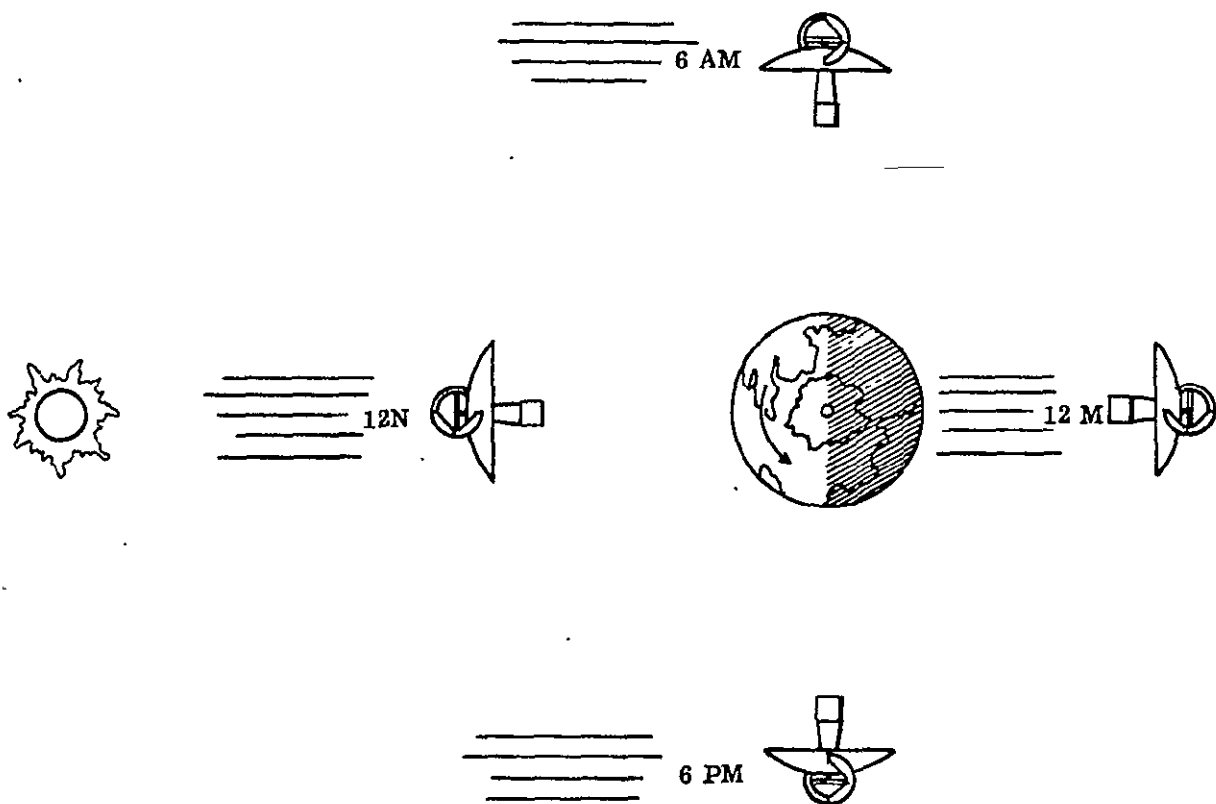


Figure 14.3-3. Solar Array in Respect to Sun

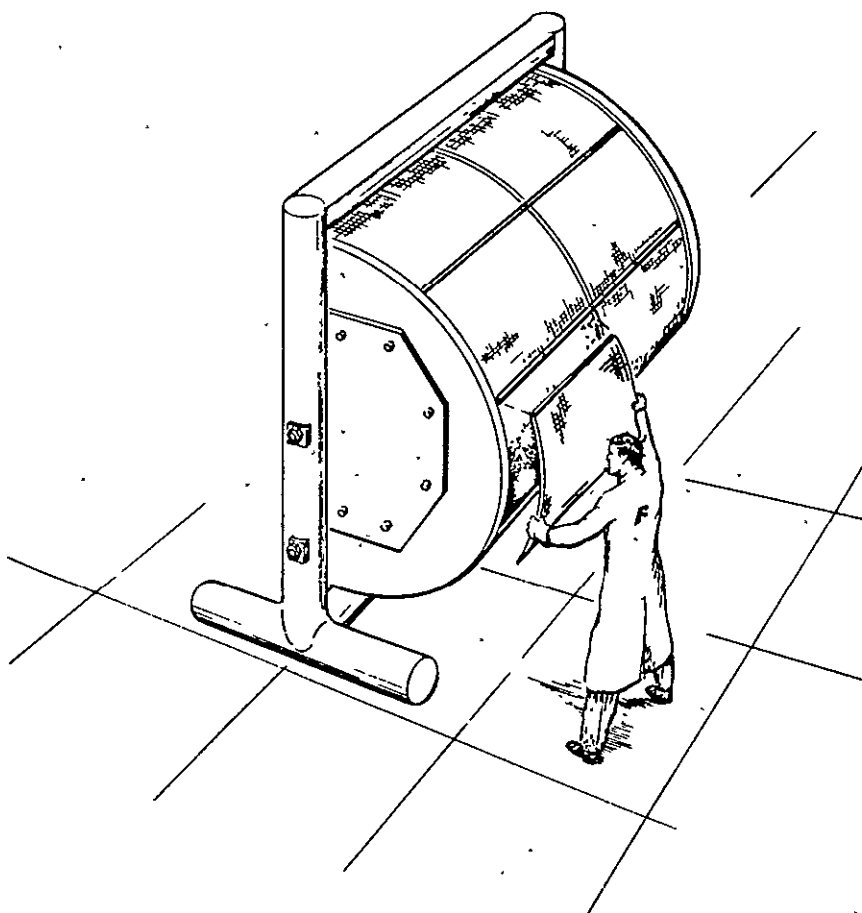


Figure 14.3-4. Solar Array Replaceable Modules

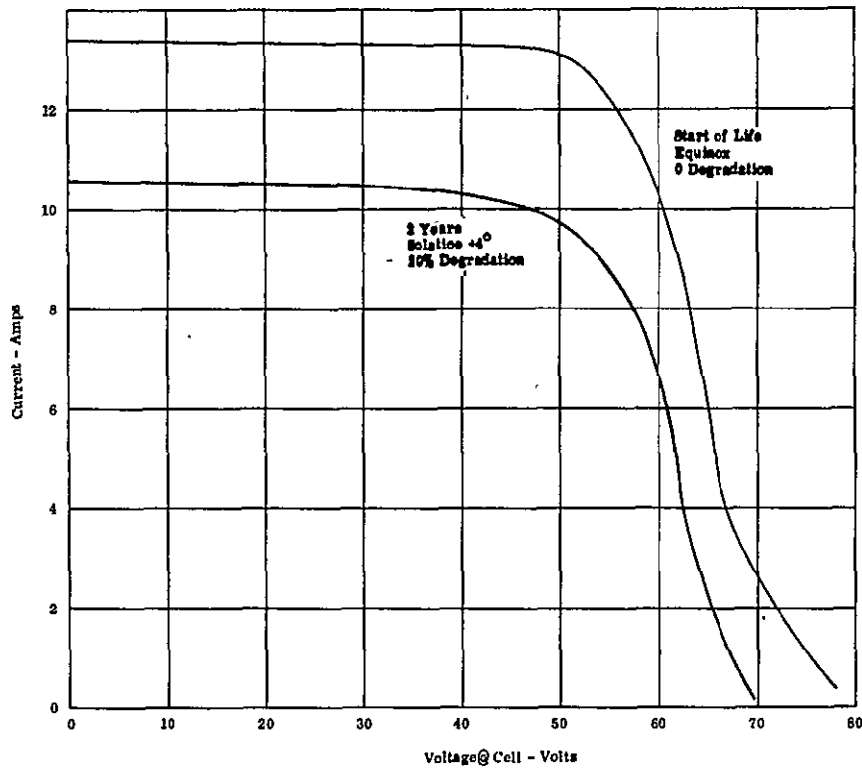


Figure 14.3-5. Solar Array Voltage vs. Current

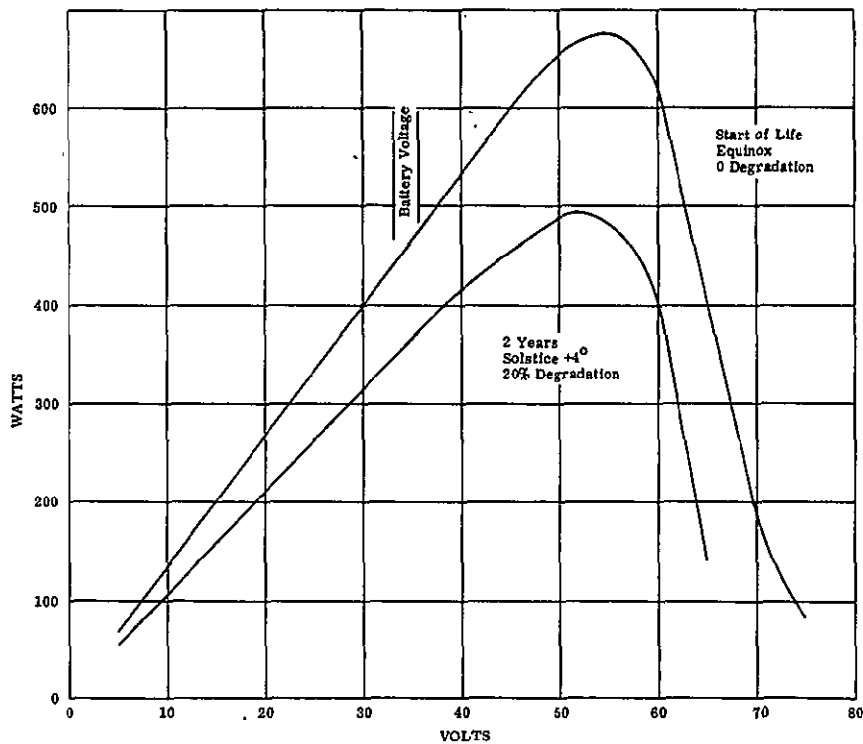


Figure 14.3-6. Solar Array Power Derivation Curves

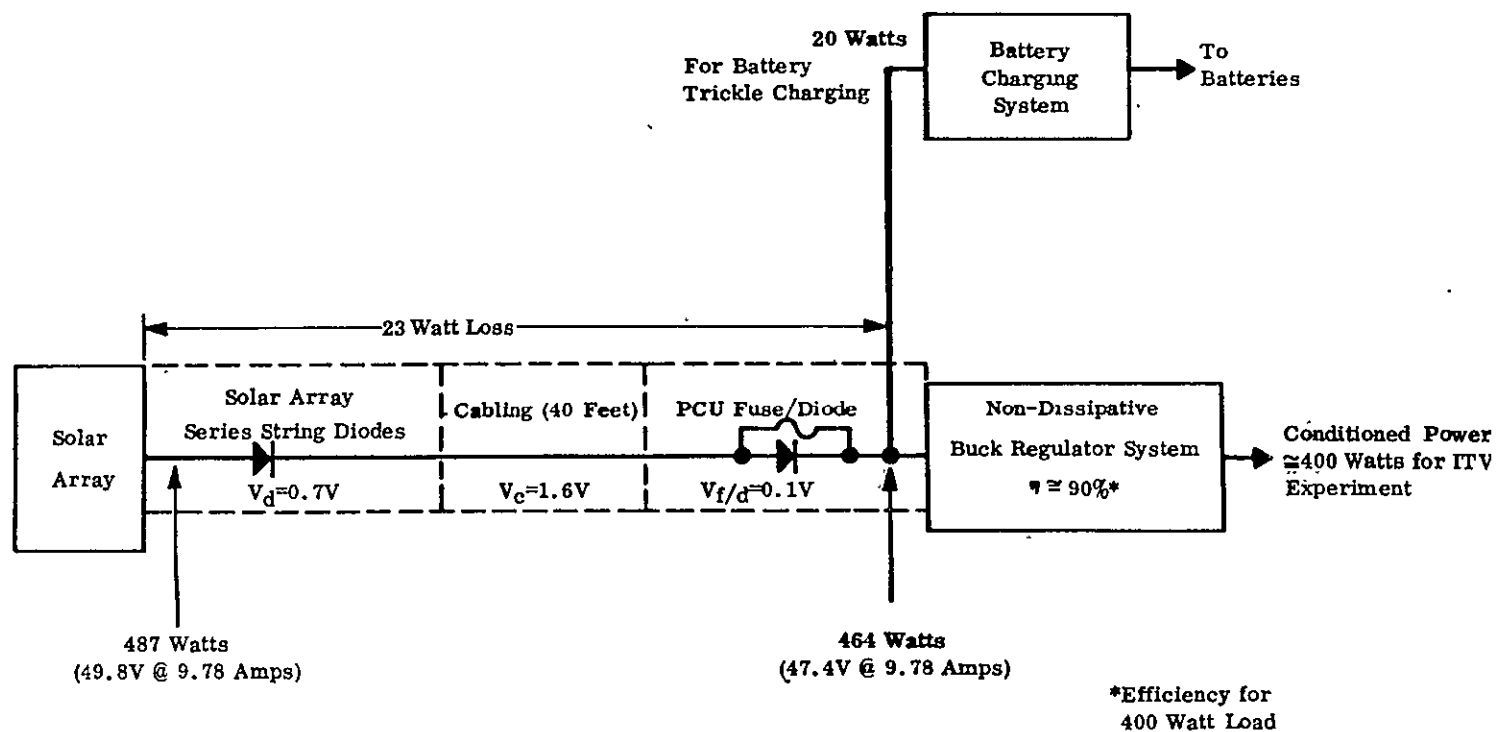


Figure 14.3-7. Solar Array Power Requirement for ITV Experiment

Present data and predictions of solar radiation damage range from the rather conservative 12 percent predicted by Mr. Warren of GSFC to the pessimistic 46 percent degradation of the GSFC document "ATS Spacecraft Expected Environment". For the 2-year ATS F&G, it is recommended that the more moderate value, based upon the measured data presented in Proposal Volume I E, of 20 percent be utilized. This recommended degradation level of 20 percent includes no additional contingency other than the inaccuracy of the solar activity predictions. The 20 percent degradation level provides for expected moderate solar flare activity for two years without undue over-design penalties resulting from pessimistic predictions.

Filter and adhesive losses, soldering fabrication losses, together with cell mismatch and interconnect losses are representative of other contingency-type losses which constitute up to a 7 percent loss in addition to the 20 percent radiation degradation. Thus, the solar cell total losses expected during manufacture and throughout the two-year period in orbit require a degradation factor for design purposes of 27%.

14.3.2 POWER BUSES

The power subsystem is fully redundant so that the loss of any single component will not result in loss of the mission or even (in most cases) in loss of performance. As shown in Figure 14.3-1, the eight portions of the solar array are diode-coupled in the Power Control Unit to form two main solar array busses, each of which is supported by a battery. The diodes are by-passed with heavy fuses to avoid taking an unnecessary voltage drop (power loss) across the diodes during normal operation. However, should one of the long lines from the array short to ground, current from the batteries would be sufficient to blow the fuse, thereby allowing the diode to isolate the shorted line. In this manner, the array losses would be limited to a 12.5 percent reduction in area for a shorted solar array power line. Should the short then disappear, the array would regain full power output except for the slight voltage drop (0.7 volt) across the diode whose bypass fuse had been blown by the temporary short.

Under normal operation the two unregulated solar array busses are coupled together through an automatically controlled bus-tie relay, with ground-command override. However, should a fault occur on either solar array bus, the two busses will be separated to restrict the fault. In like manner, the two regulated spacecraft load busses normally are operated in parallel, but will be separated (automatically or by command) should a fault occur on either one. The two regulated rf power amplifier busses can also be operated in parallel by command control.

A third pair of output busses is provided for the command receivers and decoders to allow them direct access to the battery output terminals in the event a fault occurs on either or both spacecraft load busses. The command subsystem components are designed to operate over a wide range of input voltages (24 to 45 volts). Upon correction of the fault, the command subsystem can be reconnected to the regulated bus through ground-command control.

14.3.3

BATTERY DISCONNECT PLUGS

Because no commands are provided to remove the command subsystem from the power lines, prelaunch power removal is accomplished by means of battery disconnect plugs that break the ground (negative) connection to each battery. External power for prelaunch testing can be supplied to either one of the unregulated power busses through the umbilical power terminals, (labeled "GSE power input" in Figure 14.3-1). Although provision is made for an umbilical line voltage monitor, one advantage of a buck regulator system over a solar array shunt regulator system is that no regulation is required on this line so long as it remains above the nominal line voltage.

14.3.4

LOAD REGULATORS

Two of the 30.5-volt spacecraft load regulators (A411 and A413) are normally ON and connected to their respective busses, thus providing a combined regulation capacity of 500 watts at +30.5 volts. The third regulator (A412), which is normally in standby, functions as a back-up for either of the two main +30.5 volt regulators. Switch-over occurs when a fault in either regulator is "sensed" by the appropriate over/under voltage control circuits. These circuits incorporate the logic circuits required to compare the output voltage with the input voltage and evaluate the performance of the regulator. Ground command override is also provided as a means for switch-over to the back-up regulator.

The rf power-amplifier regulators (A415 and A417) provide an output voltage of +25 volts at a maximum capacity of 280 watts each, which is sufficient to support the four rf amplifiers (X, S, L, UHF) in any of the planned operating modes. Since each regulator can support the full load, they are redundant. Normal operation occurs with only one ON and the other in standby, although it is possible to command them both ON with each regulator supporting a separate bus. The rf power amplifiers connected to the +25 volt busses also have ON/OFF control circuits located within each power amplifier.

The spacecraft system loads are divided between the two main 30.5 volt busses in such a manner as to take advantage of the redundancy inherent in the various spacecraft subsystems (see Figure 14.3-8). As a consequence, should one bus fail, all critical spacecraft systems can be maintained operative by the components connected to the second bus.

The power subsystem provides power to 40 load centers on the two main 30.5 volt busses. Each of the load centers is controlled and regulated by a load interface circuit (LIC). Except for the GFE experiments, each LIC is incorporated into the component that it controls or within a component that controls other components. The command decoders use a special LIC configuration without command turn-off control and with automatic turn-on when the power bus is above 22 volts. The command receiver input power circuit is a simple resistor/zener diode network.

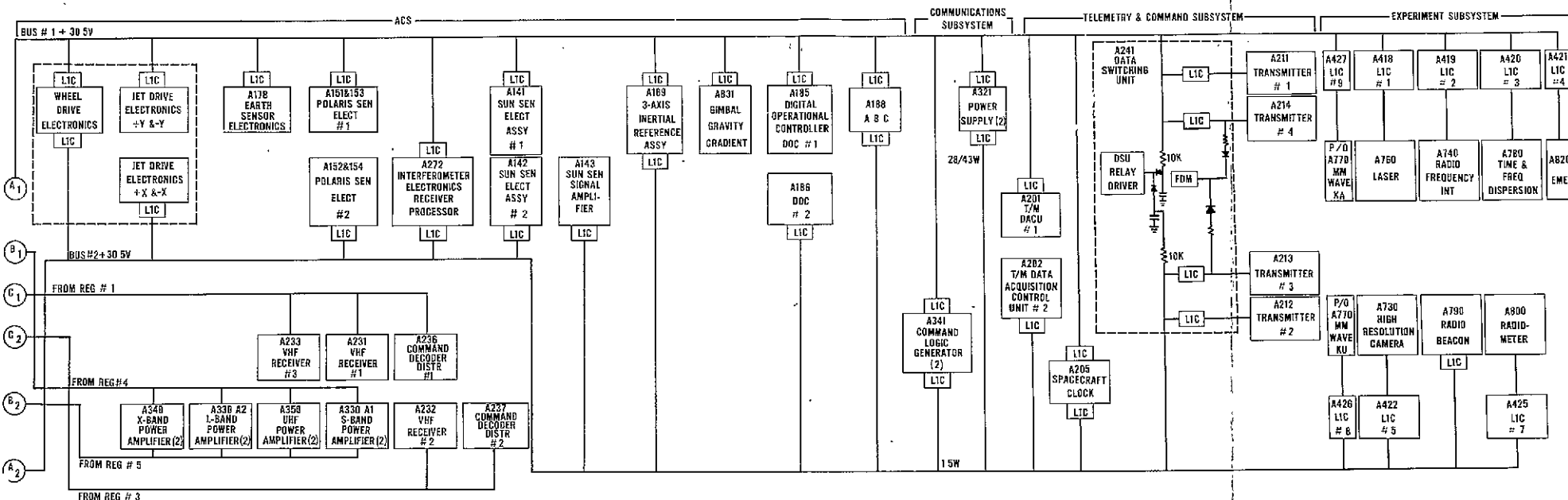


Figure 14.3-8. ATS Power Interface Block Diagram

14-23/14-24

FOLDOUT FRAME

FOLDOUT FRAME 2

14.3.5**BATTERY CHARGERS**

Each battery charger (A406 and A407) consists of one trickle charger and one main charger. Each trickle charger operates on a 50 percent duty cycle and both are not on simultaneously. This technique permits a current of C/30 (.4 amperes) to be delivered to each battery without loading the array with a C/15 (.8 amperes) load normally encountered when simultaneously trickle-charging two batteries. That is, the .4 ampere current is delivered first to one battery then to the other. In this manner the battery sees a higher, more efficient trickle charge rate while the solar array sees the lower current rate. In the event that a main charger should fail, the trickle charger can be employed as a backup charger.

The chargers function as non-dissipative, tapered current, temperature compensated, charge controllers. The initial constant current portion of the charge is accomplished at a C/10 (1.2 amperes) rate. The charge rate is then modified as a function of battery terminal voltage and battery temperature. The trickle chargers provide the 50% duty cycle, C/30 charge rate throughout the sunlit portion of the orbit. Following peak loading discharge or occult discharges of the batteries, the main chargers initially provide an additional C/10 charge rate unless disabled by ground command. Under this charging condition the batteries are receiving C/10 plus 50% of a C/30 charge or essentially an energy level equal to C/8.5 (1.4 amperes). The main charger terminates its contribution at a predetermined voltage and current depending upon battery cell temperature thus the charging reverts back to only the trickle charger.

Charger control techniques that require the use of third and fourth electrode cells to minimize overcharging were considered during ATS F&G Phase B/C, but have not been implemented in the proposed ATS F&G power subsystem design. The ATS F&G power subsystem has reduced the dependence on battery power for mission success by the use of the hemicylindrical solar array and does not require battery support on a daily basis as did the cruciform solar array configuration. Therefore, the number of battery cycles that would have been required to support a minimum size cruciform array has been reduced by 2920 cycles (365 days times 4 cycles for two years) with the hemicylindrical array. This reduction in battery cycles, the limited flight history (OAO) of third electrode cells and also the fourth electrode cells (ATM - not flown), and the short development time for the ATS-F power subsystem has lead to the selection of the flight proven current-limited, temperature compensated charger.

14.3.6**BATTERY**

The battery is provided principally to support the spacecraft loads during absence of solar array power during the occult and for peak loads that exceed the solar array capacity. By careful selection of the system voltage levels (i.e., the solar array, the regulated output, and the battery voltages), the selected design has eliminated the need for any boost circuits, either in the charger or in the battery line. Furthermore, the battery is coupled directly to the line through a diode, rather than through a more complex battery discharge control circuit that would be required with a boost charger.

Accounting for a 1.0-volt minimum drop across the main regulators, the unregulated bus must be supported by the battery at greater than 31.5 volts to maintain 30.5 volts at the output. Assuming a 0.7-volt drop in the battery coupling diode and a 0.5-volt distributed line loss, the battery output voltage must remain at least 32.7 volts during normal discharge. The nickel-cadmium batteries (A401 and A402) incorporate 28 cells of 12 A-H capacity in order to provide the required line support voltage, thereby avoiding the complexity of a battery booster circuit. The battery will have a nominal discharge voltage equal to 33 volts. The occult loading for the batteries is shown in Figure 14.3-9.

14.3.7 SUMMARY OF SELECTED POWER SUBSYSTEM

The selected power subsystem design can be characterized by the following features:

- High Efficiency at Peak Loads - The solar array transmission line operates at nearly double the final regulated voltage, thereby keeping line losses down. The high power loads are regulated directly by high efficiency, non-dissipative, switching regulators. No series diodes are required in the main solar array power busses.
- High Reliability - This simple configuration requires the fewest components. Full redundancy throughout the system provides complete single-failure back-up. Command sub-system has direct access to the battery terminal in the event of multiple failures.
- Low Cost - All regulators and chargers are of the same basic design and similar to those developed for other spacecraft applications. The nickel-cadmium battery cell is a conventional two terminal design. Solar array costs are minimized through modular construction and no shunt control taps are required on solar array. No shunt-tapped solar array shunt regulator simulator is required for subsystem dynamic testing at the manufacturing facility or at the launch pad. There is no need for remote sensing control for umbilical voltage regulation during pre-launch check-out.
- Full Performance - The power subsystem fulfills all the objectives identified previously in paragraph 14.1 and exceeds the regulation and ripple requirements of the GFE experiments. Although it is tailored to provide maximum efficiency for the present ATS F&G configuration, it is sufficiently flexible to allow the operation of any combination of loads up to the limit of the solar array output.

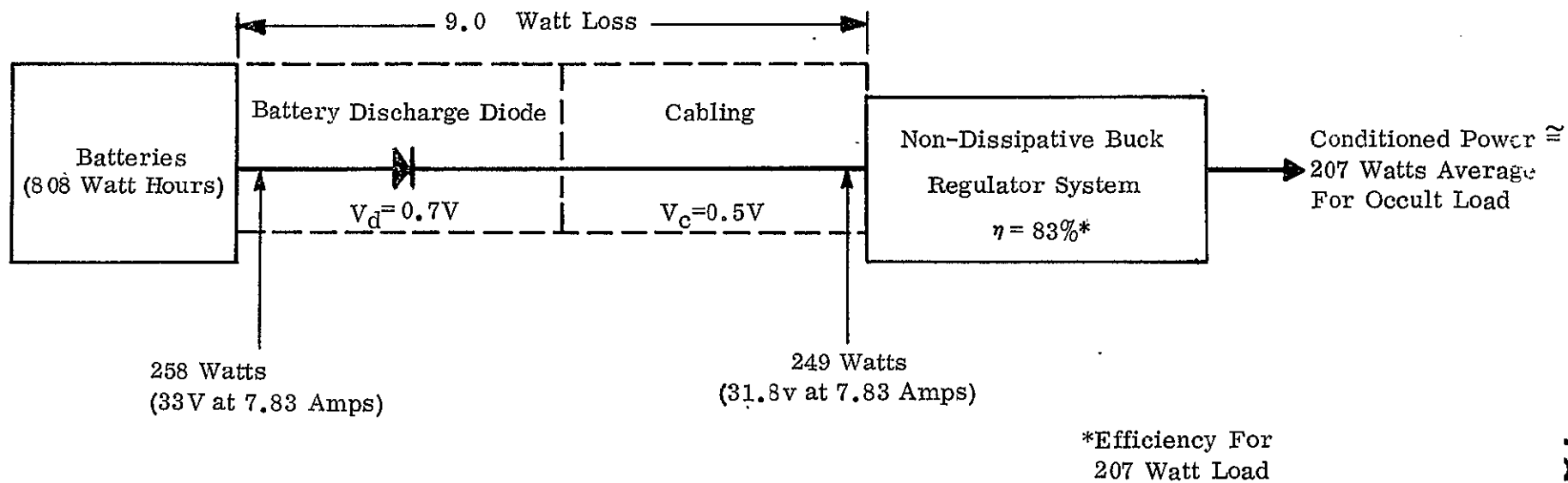


Figure 14.3-9. Battery Loading for Occult Load

Table 14.3-1 lists the vital statistics of the power subsystem and the location of the power subsystem components within the EVM is shown in Figure 14.3-10.

14.4 CONDITIONED POWER PROFILE

The conditioned power profile for the ATS F&G spacecraft is based on the division of the spacecraft into the three major subsystems; experimental packages (GFE), the Interferometer, and the Power Control Unit. These major subsystems loads are defined as follows:

- Telemetry and Command Subsystem
- Attitude Control Subsystem
- Communications Subsystem

A summary of the conditioned dc power required for each of the subsystems and the individual components within the subsystems is given in Figure 14.4-1. The conditioned power requirements are defined as the power required at the output from the regulator system shown in Figure 14.3-7.

14.4.1 TELEMETRY AND COMMAND SUBSYSTEM

The conditioned dc power for the Telemetry and Command Subsystem is shown for three different operating levels which correspond to the peak, normal, and low power states of the subsystem. The individual components for the Telemetry and Command Subsystem are as shown in Figure 14.4-1 together with their conditioned dc power requirements that contribute to each of the three operating levels.

14.4.2 ATTITUDE CONTROL SUBSYSTEM

The Attitude Control Subsystem is divided into five major operating modes. These modes are:

- Ascent Mode
- Acquisition Mode
- Operational Mode
- Nominal Back-up Mode
- Occult Mode

Each of the five major operating modes for the Attitude Control Subsystem has three different conditioned dc power requirements. These power requirements are the peak power, the normal (or expected) power, and the low (or standby) power, required during the mode. Figure 14.4-1 shows the individual components of the Attitude Control Subsystem (ACS) and their conditioned dc power requirements that contribute to the total subsystem operating loads.

Table 14.3-1. Power Subsystem Component Characteristics

	SOLAR ARRAY	BATTERIES	CHARGERS	REGULATORS	LOAD INTERFACE CIRCUITS	POWER CONTROL UNIT
	Solar Cells • N/P Silicon • 10.5% Min. Avg. Eff. • 10 OHM-CM • 12 MILS Thick • 2 x 2 CM • Sintered AG-Ti Contacts Number of Cells • 41,600 Cover Glass • 6 MILS Thick • Fused Silica • 400# Cut Off Filter • Anti-Reflective Coating Interconnect • Expanded Silver Mesh Adhesives • Silgard 182 For Cover Glass • Silastic 140 For Solar Cells Losses • 20% For Orbital Environment • 7% For Fabrication	Nickel-Cadmium • 12 A-H Cells • 28 Cells Per Battery • Hermetically Sealed Cells • Cycle Life ~ $\geq 1000 \pm 20,000$ Cycles Depth Of Discharge • $\leq 40\%$	Trickle Charging • C/30 (0.4 AMP) Charge Rate • Duty Cycled-50% Main Charging • C/10 (1.2 AMPS) Charge Rate • Temperature Compensated	Two Power Ratings • 250 Watts At +30.5 Volts • 230 Watts At +25 Volts Characteristics • $\pm 1.3\%$ Regulation • 89% to 92% Efficiency • 300 Millivolts Peak to Peak Output Ripple • Overload and Short Circuit Protected • Overvoltage Clamp At Output • Load Sharing Circuitry For Parallel Operation • EMI Protected	Two Power Rating • 0 to 15 Watts • >15 to 60 Watts Characteristics • +28 Volts Output • $\pm 2\%$ Regulation • Turn-On And Turn-Off By Command • Automatic Turn-Off At 38 Volts Input • Automatic Turn-Off At 24 Volts Output • Automatic Turn-Off At 1 AMP Max. Overload For 15w LIC • Automatic Turn-Off At 4.5 AMPS Max. Overload For 60w LIC • Ripple Rejection Of 1000: 1 At Low Frequency • Ripple Rejection Of 100: 1 At High Frequency	Functions • 9 Automatic Switching Functions For Fault Removal • 34 Commandable Switching Functions For Control Of Power System Components • 25 Analog and 16 Discrete Telemetry Monitor Points To Show Complete Status Of Power Subsystem • 16 Diodes For Isolation Of Circuits • Battery Reconditioning • GSE Power Input • Squib Power Interface
Size	208 Sq. Ft.	16.5 x 7.5 x 5.5 In.	6 x 5.5 x 3.5 In.	7 x 6 x 3.5 In.	4 x 3.5 x 3 In.	12 x 12 x 8 In.
Weight Per Unit	175 lbs.	44 lbs.	4 lbs.	4.8 lbs.	1 lb.	25 lbs.
Number Required	1	2	2	3 250 Watt Units 2 280 Watt Units	8 60 Watt Units	1
Total Weight	175 lbs.	88 lbs.	8 lbs.	24 lbs.	8 lbs.	25 lbs.
						Per System 328 lbs.

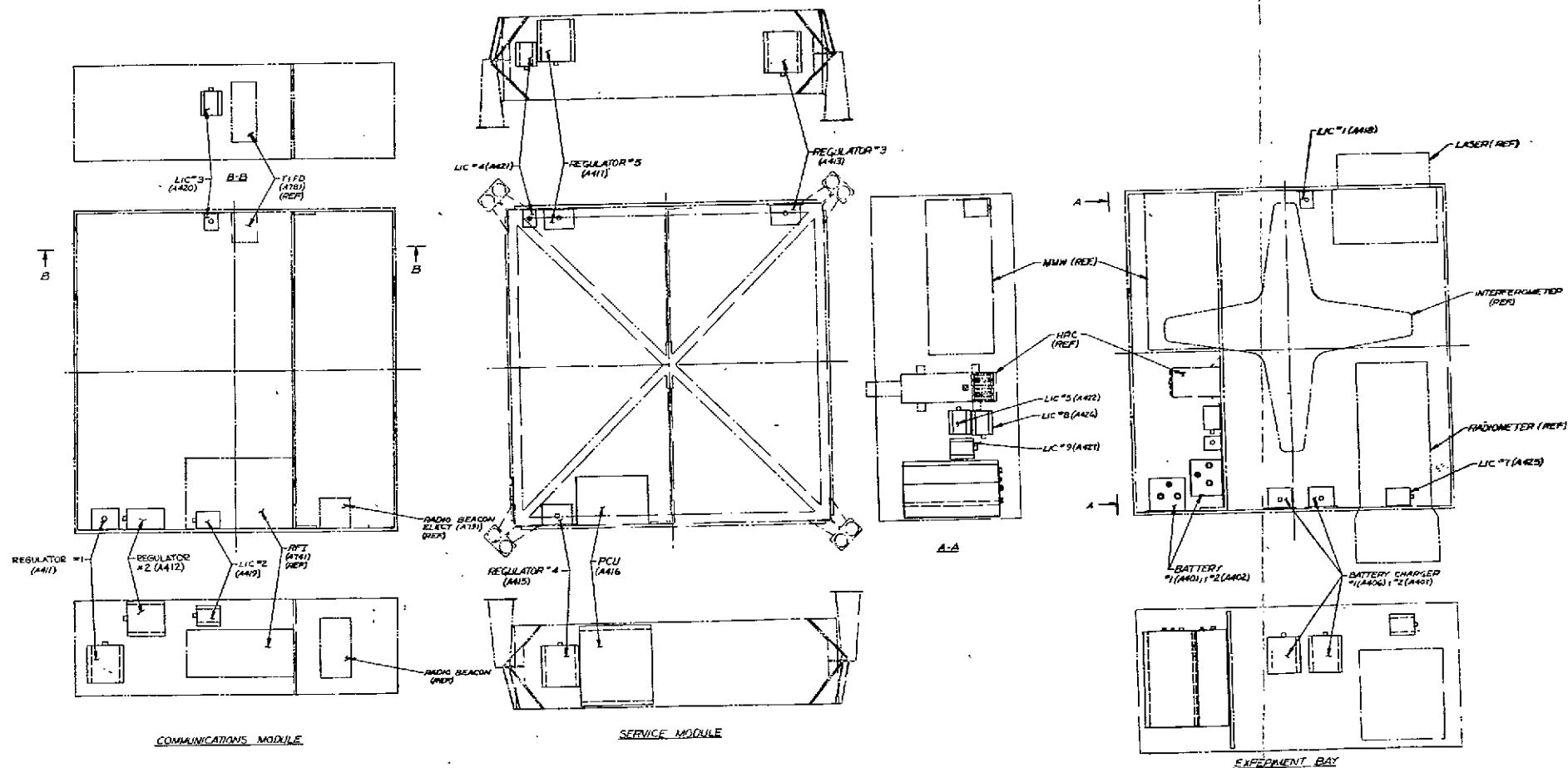


Figure 14.3-10. ATS F&G Power Subsystem Installation Design Layout

14-31/14-32

FOLDOUT FRAME 1

FOLDOUT FRAME 2

14.4.2.1 Ascent Mode

During the ascent mode the peak, normal, and low conditioned dc power requirements of the ACS are 97.9 watts, 82.5 watts, and 55 watts, respectively. This power is required to ensure that the Inertial Wheels, Polaris Sensor, 3-axis IRA, and the Earth Sensor (which contain motor driven parts) will survive the ascent phase vibrations and acceleration by operating these components during the ascent phase.

14.4.2.2 Acquisition and Operational Modes

The Acquisition and Operational modes for the ACS are further divided into submodes of operation:

- Acquisition Mode
 - a. Rate Nulling/Deployment/Coarse Sun Acquisition
 - b. Fine Sun/Earth/Polaris Acquisition
- Operational Mode
 - a. Nominal (Wheels/Jets)
 - b. Jets only
 - c. Precise 3-Axis Rate Stabilization (Low Jitter Mode)
 - d. Orbit Control

Each of these submodes determine the peak, normal, and low power requirements for the ACS components as shown in the following paragraphs.

14.4.2.2.1 Acquisition Mode

The Rate Nulling/Deployment/Coarse Sun Acquisition submode brings the +X axis of the spacecraft to within 30 degrees of the sun line; and the Fine Sun/Earth/Polaris Acquisition submode completes the acquisition procedure by first rotating the +X axis of the spacecraft to coincide with the sun line, then rolling about the +X to find the earth, and finally yawing about the Z axis to find Polaris. Rate Nulling/Deployment/Coarse Sun Acquisition must be accomplished by the onboard batteries since the solar panels will not yet be aligned with the sun, while the Fine Sun/Earth/Polaris Acquisition submode of the acquisition sequence will receive power from the solar panels. The total acquisition procedure requires a maximum of 98 minutes of which the first 42 minutes must be accomplished with battery power at a normal level of 112.7 watts.

14.4.2.2.2 Operational Mode

The four Operational submodes of the Attitude Control Subsystem are used during the normal mission operations of the spacecraft.

The Nominal (Wheels/Jets) submode is the primary mode of operation for the Attitude Control Subsystem. As seen from the Component Summary Chart, the normal conditioned dc power for this mode is 88.6 watts. The Jets Only submode is a back-up mode to the Nominal (Wheels/Jets) submode and the normal operating power for the Jets Only submode is 67.2 watts.

The Precise 3-Axis Rate Stabilization is used when a very stable, low jitter, spacecraft platform is required. This submode is used during the operation of the VHR Camera and the Radiometer. The normal conditioned dc power for this submode is 88.6 watts which is identical to the conditioned Nominal (Wheels/Jets) submode.

The Orbit Control submode is used during the stationkeeping maneuvers. This submode requires 97.3 watts for normal operation which is slightly higher in conditioned power than the Nominal (Wheels/Jets) and Jets Only submode.

14.4.2.3 Nominal Back-Up Mode

The Nominal Back-Up mode for the ACS requires a peak conditioned dc power of 61.6 watts which is nearly twice the low power required for the mode. Factors contributing to the difference in operating power are easily seen in the component summary chart.

14.4.2.4 Occult Mode

The Occult mode requires 88.6 watts of conditioned dc power for normal operation. The Component Summary Chart shows that the Occult mode of operation is identical to the Nominal (Wheels/Jets) operating submode. During the occult, the satellite passes behind the earth and into the earth's shadow; therefore, power is not available from the solar panels. The 88.6 watts must be maintained by battery power and may last as long as 1.15 hours thus becoming a dominant factor in the sizing of the on-board batteries.

14.4.3 COMMUNICATIONS SUBSYSTEM

The Communications Subsystem modes of operation and the conditioned dc power of the individual components of the subsystem are shown in Figure 14.4-1. The operational modes of the Communication Subsystem correspond to the communication experiments that are performed with the communication components and also the components required to support the Experiment Packages.

14.4.4 EXPERIMENT PACKAGES

The Experiment Packages summary shows two power requirements for each package. The power "without load interface circuits" is the power required for the actual experiment. In order to provide voltage regulation to the individual experiments, load interface circuits (series dissipative regulators) are added to the power subsystem. The power loss in the regulator and the power required by the experiment are included in the conditioned dc power for the experiment package.

14.4.5 TOTAL CONDITIONED POWER PROFILES

Figure 14.4-2 is the profile for the total conditioned normal power for the spacecraft during ascent, acquisition, orbit control, occult, and some of the many possible operational modes. The ordinate on the left hand side of the diagram is the conditioned dc power required for each of the modes. The total conditioned dc power

required for the normal operation of each subsystem has been profiled from the data in Figure 14.4-1. Figure 14.4-3 is the profile for the total conditioned peak power for the spacecraft.

The major power loads for the power subsystem are the PLACE and ITV experiments. The PLACE experiment will be performed primarily during the first year of the mission and the ITV experiment will be performed up to the end of the two year mission. Figure 14.4-2 shows the normal conditioned power required for the PLACE experiment and the ITV experiment. The solar array must provide the 432.2 watts for PLACE and the 406.3 watts for ITV. The capability of the power subsystem to perform these experiments over a five year period has been profiled in Figure 14.4-4 for a solstice launch and in Figure 14.4-5 for an equinox launch. The curves of Figures 14.4-4 and 14.4-5 assume a linear 20 percent degradation for the solar array during the first two years and an additional 10 percent during the last three years. The figures show the change in power output from equinox to solstice sun angles of 23.5-degrees plus the 4-degree offset pointing required for the ITV experiment.

TELEMETRY & COMMAND SUBSYSTEM

COMPONENT	DC POWER WATTS		
	PEAK	NORMAL	LOW
Data Acquisition & Control Unit #1	7.3	1.3	7.4
Data Acquisition & Control Unit #2	0	0	0
VHF Transmitter #1	5.0	0	5.0
VHF Transmitter #2	5.0	0	0
VHF Transmitter #3	0	0.0	0
VHF Transmitter #4	0	0	0
VHF Receiver (3)	2.4	2.4	2.4
Command Decoder/Distributor #1	11.5/2.9	2.9	2.0
Command Decoder/Distributor #2	2.9/11.5	2.9	2.9
Data Switching Unit	2.5	2.5	2.5
Spacecraft Clock	2.3	2.3	2.3
TOTAL	38.0	25.3	23.6

EXPERIMENT PACKAGES	DC POWER WITHOUT LOAD INTERFACE CIRCUIT	DC POWER WITH LOAD INTERFACE CIRCUIT
VHF Camera	25W	25W
Time & Freq. Dispersion	26W	31W
Radio Freq. Interference	50W	66W
Laser	30W	33W
Environmental Measurements Experiment	46W/50W	50.8W/55W
Radiometer	20W	22W
MM Wave		
20 GHz Mode	44W	48W
30 GHz Mode	57W	63

Radio Beacon	10.6W
Power Control Unit	13.5W
Interferometer	10.3W
Gimballed Gravity Gradient	10.5W

COMMUNICATIONS SUBSYSTEM

COMPONENT	DC POWER WATTS
1. X Band Preamps (each of 2)	0.26
2. S Band Preamps (each of 2)	0.25
3. L Band Preamp	0.25
4. VHF Preamps (each of 2)	0.15
5. IF Amplifiers (each of 3)	2.5
6. X Band Power Amplifier	80
7A S Band Power Amplifier	70
7B L Band Power Amplifier	125
8. UHF Power Amplifier	200
9. Monopulse Detector and Driver	0.5
10. Power Converter (note 1)	-
11. Synthesizer	-
A. Low Power Mode	11
B. High Power Mode	25
12. Beacon (X Band)	0.2
13. Wideband Data Unit (VCOs)	0.5
14. Command Distributor	1.5
15. X Band Monopulse Modulator	1.0
16. S Band Monopulse Modulator	1.0
17. VHF Monopulse Modulator	0.5
18. S Band Switching Array	4.5

OPERATING MODES	BANDS USED	COMPONENTS USED	TOTAL POWER WATTS
DRS	X, S	1, 2, 5 (2), 6, 1A, 10, 11B, 12, 13, 14, 15	193.9
PLACE	X, L	1, 2, 5 (2), 6, 7B, 10, 11B, 12, 14	243.7
X Band Monopulse	X	1 (2), 5 (1), 9, 10, 11A, 14, 15	19.6
S Band Monopulse	S	2 (2), 5 (1), 9, 10, 11B, 14, 15	36.8
VHF Monopulse	VHF	4 (2), 5 (1), 9, 10, 14, 17	6.1
ITV (with X Band Video Monitor)	X, UHF, VHF	1, 5 (2), 6, 8, 10, 11A, 12, 14	300.6
VHF Camera And Radiometer	X	6, 10, 11A, 12, 13, 14	85.2
MM Wave	X	9, 10, 11A, 12, 13, 14	50.2
Laser	X	1, 5 (1), 6, 10, 11A, 12, 13, 14	36.3
Time & Freq. Disp	X	6, 5(1), 10, 11A, 14	27.3
Radio Freq. Inter.	X	6, 14	61.5
Environmental Measurements Exp.	VHF	T & C	- - -
ITV	UHF, VHF	1, 5 (1), 6, 10, 11A, 12, 14	217.6
R & RR	X	1, 5 (2), 6, 10, 11A, 14	97.5

Note 1: Power loss in converter is equal to 15% of total transmitter power excluding power amplifiers (items 6, 7, 8)

* Alternate 3-Axis Attitude Reference Subsystems (ARS) not shown:
Earth Sensor/Yaw Gyro
Monopulse/Polaris Sensor or Yaw Gyro
Single Station Interferometer/Polaris Sensor or Yaw Gyro
Dual Station Interferometer (Ground command control)
Laser/Polaris Sensor or Yaw Gyro
** 50 Watts for 100 milliseconds to Activate Latching Valves in Failure Mode is not included in peak values.

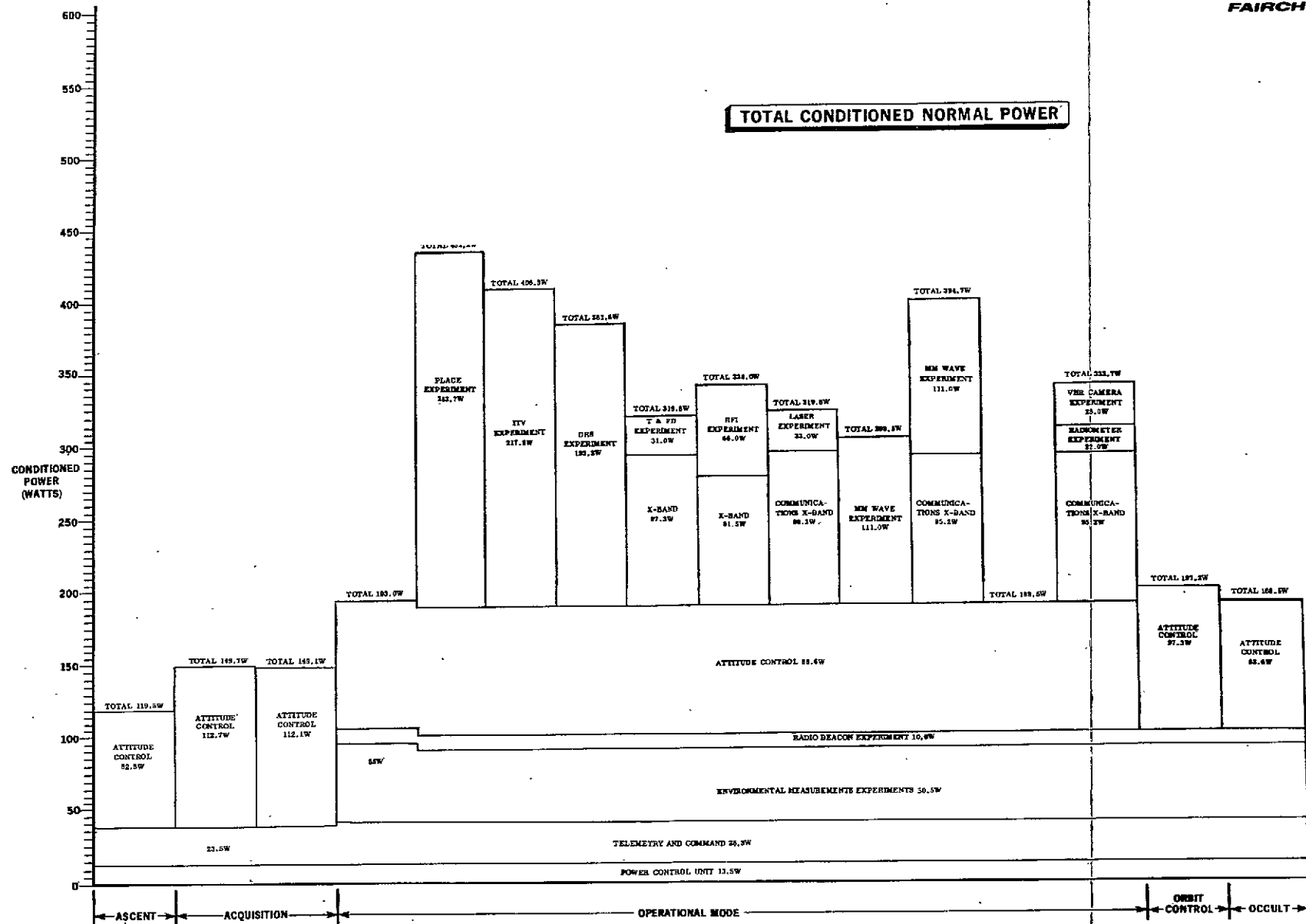
ATTITUDE CONTROL SUBSYSTEM

ITEM		ACQUISITION MODE																								OPERATIONAL MODE												Nominal Backup Mode (Analog Backup Controller) (5% Jet Duty Cycle)			Occult Mode (1.15 Hours)		
		Ascent Mode (5.6 Hours)			Rate Nulling/Deployment/ Coarse Sun Acquisition (26% AC Jet Duty Cycle) 42 min						Fine Sun/Earth/Polaris (10% AC Jet Duty Cycle) 55 min.						Nominal (Wheels/Jets) *			Jets Only* (5% AC Jet Duty Cycle)			Precise 3-Axis Rate Stabilization			Orbit Control (10% AC Jet Duty Cycle)																	
		PEAK	NORMAL	LOW	PEAK	NORMAL	LOW	PEAK	NORMAL	LOW	PEAK	NORMAL	LOW	PEAK	NORMAL	LOW	PEAK	NORMAL	LOW	PEAK	NORMAL	LOW	PEAK	NORMAL	LOW	PEAK	NORMAL	LOW															
Attitude Control Jets	0	0	0	8	1.6	0	8.0	0.8	0	8.0	0	0	8.0	0.4	0	8	0	0	8	0.8	0	8	0.8	0	8.0	0.4	0	8	0.8	0	8.0	0.4	0	8	0	0							
Orbit Control Jets	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0									
Propellant Feed System **	0	0	0	3.0	5.0	5.0	6.0	4.0	4.0	6.0	4.0	4.0	6.0	4.0	4.0	6.0	4.0	4.0	6.0	4.0	4.0	4.0	7.0	5.0	5.0	6.0	4.0	4.0	6.0	4.0	4.0	4.0	4.0	4.0									
Inertia Wheels	16.6	16.6	0	0	0	0	0	0	0	45.0	16.5	13.4	0	0	0	45.0	16.5	13.4	0	0	0	45.0	16.5	13.4	0	0	0	0	0	0	0	0	0	0									
Actuator Control Electronics	11.0	11.0	0	12.7	11.4	11.4	12.7	11.4	11.4	20.3	16.7	16.7	12.0	11.4	11.4	20.3	16.7	16.7	12.0	11.4	11.4	16.7	15.5	14.4	14.4	6.0	5.7	5.7	25.3	16.7	16.7	16.7	16.7										
Sun Sensor Electronics	0	0	0	3.6	3.6	3.6	3.8	3.6	3.6	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0									
Polaris Sensor	18.8	0.4	0.4	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	25.0	9.6	9.6	9.6									
Earth Sensor	10.6	10.6	10.6	14.9	10.6	10.6	14.9	11.8	10.6	14.9	14.9	14.9	10.6	14.9	14.9	10.6	14.9	14.9	10.6	14.9	14.9	10.6	14.9	10.6	10.6	10.6	10.6	10.6	10.6	10.6	10.6	10.6	10.6	10.6									
3-Axis IRA	44.0	44.0	44.0	44.0	44.0	44.0	44.0	44.0	44.0	44.0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0									
DOC	0	0	0	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9	26.9									
ABC	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0									
TOTAL	87.9	82.5	55	136.10	112.76	111.1	141.1	112.1	110.1	146.1	88.6	81.2	92.8	67.2	62.5	148.1	88.6	61.2	123.0	97.3	88.5	61.6	35.8	35.4	146.1	88.6	91.2																

Figure 14.4-1. Power Profile Component Summa

FOLDOUT FRAME 1

FOLDOUT FRAME 2



FOLDOUT FRAME 1

Figure 14.4-2. Total Conditioned Normal Power

FOLDOUT FRAME 2

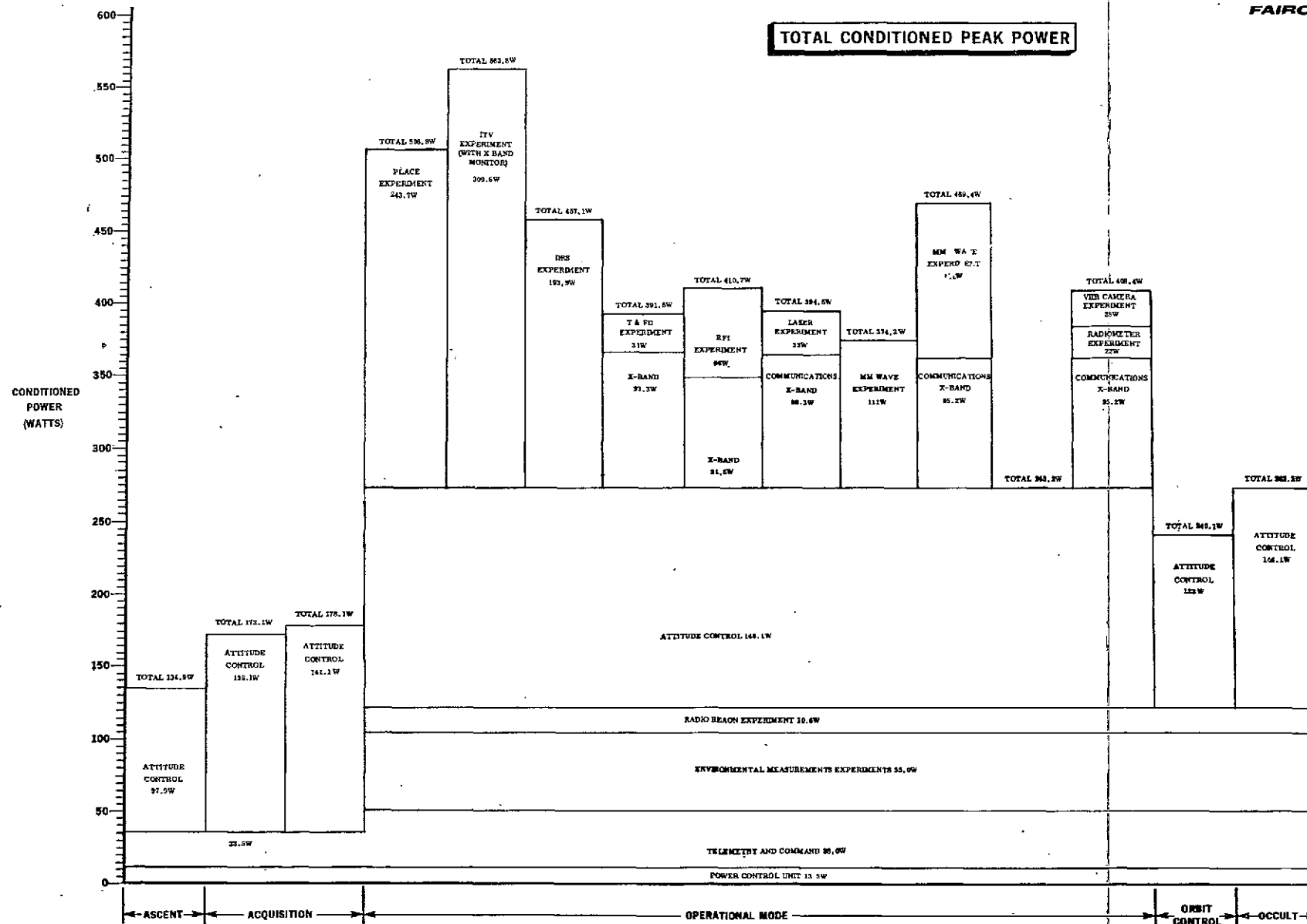


Figure 14.4-3. Total Conditioned Peak Power

FOLDOUT FRAME 1

FOLDOUT FRAME 2

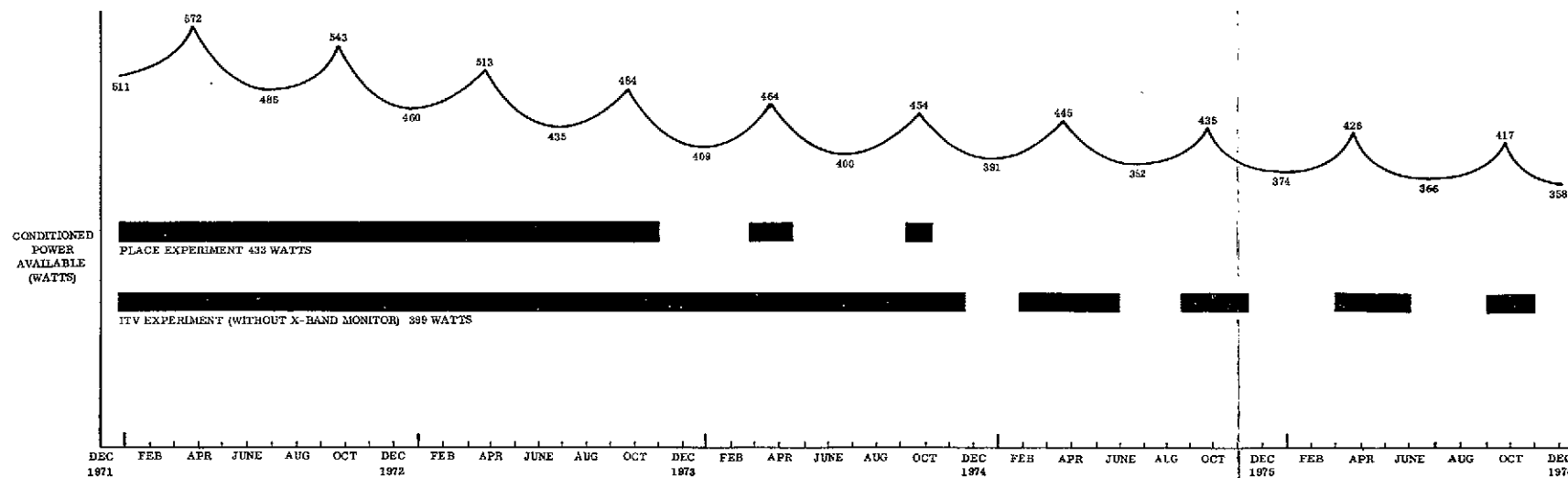


Figure 14.4-4. PLACE and ITV Power
Profile for Solstice Launch

14-43/14-44

FOLDOUT FRAME 1

FOLDOUT FRAME 2

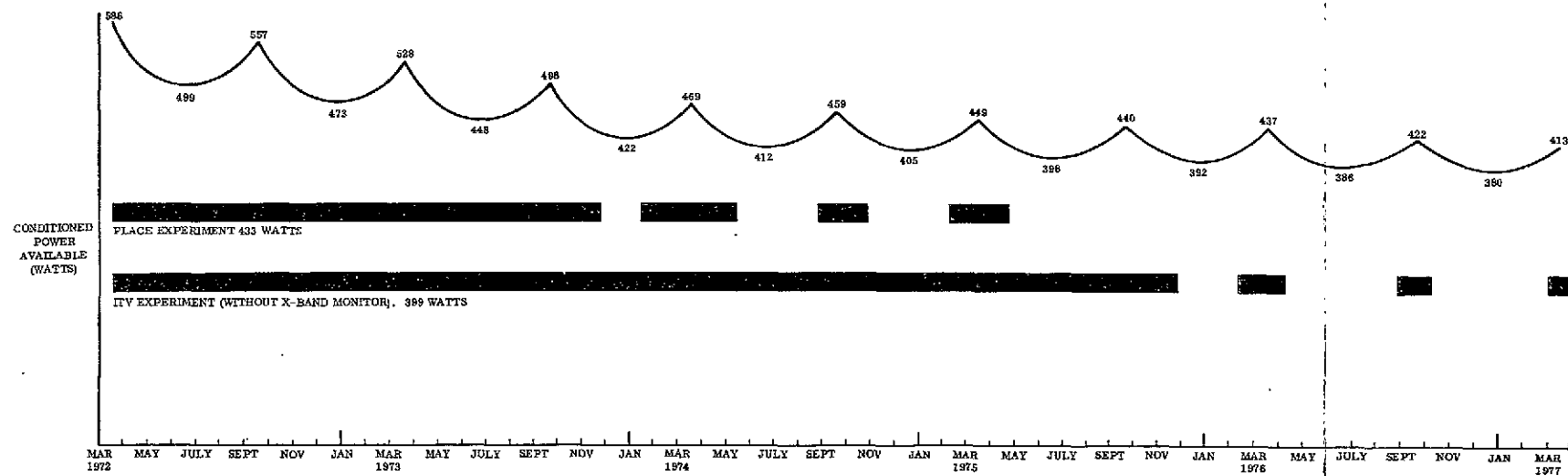


Figure 14.4-5. PLACE and ITV Power Profile for Equinox Launch

FOLDOUT FRAME /

FOLDOUT FRAME 2

14.4.6 DEPTH OF DISCHARGE

Calculations have been made on the depth of discharge presently expected for the specified batteries. The two power modes of concern for battery discharge are the Occult and Ascent modes. The depth of discharge for these modes is shown below:

- Occult Mode - Using the 188.5 watt (1.0 hour) and 263.2 watt (0.15 hour) occult loads from the power profile, (Figures 14.4-2 and 14.4-3) for 28 cells of 12 ampere hours each, and assuming a power conversion efficiency of 80 per cent, the energy discharge will be

$$\begin{aligned} E \text{ discharge} &= \frac{(188.5 \text{ w} \times 1.0 \text{ hr.}) + (263.2 \text{ w} \times 0.15 \text{ hr.})}{0.8} \\ &= 285 \text{ watt-hours} \end{aligned}$$

With the assumed battery capacity of 33.6 volts x 12 ampere hours (equal to 404 watt-hours), the depth of discharge (DOD) will be

$$\begin{aligned} \text{DOD} &= \frac{285 \text{ watt-hours}}{404 \text{ watt-hours}} = 100 \text{ per cent} \\ &= 70.6 \text{ per cent} \end{aligned}$$

for one battery the Depth of Discharge is 70.6 per cent; for a two battery system, 35.3 per cent.

- Ascent Mode - The depth of discharge calculation for the ascent mode is based upon the power profile (Figures 14.4-2 and 14.4-3) and the following assumptions:
 - a. Time intervals are as shown in Figure 14.4-6.
 - b. The normal load from Figure 14.4-2 of 119.5 watts is supported by the batteries for 70 per cent of Phase I and II.
 - c. Sufficient solar array power is available during the initial period of Phase III to supply load power and to recharge the batteries.
 - d. During the last 0.15 hour of Phase III, the batteries will support the 119.5 watt load and during the first 42 minutes (0.7 hour) of the acquisition phase, the batteries will support the peak load of 173.1 watts for 10 per cent of the time period and the remaining time will be at a load of 149.7 watts.

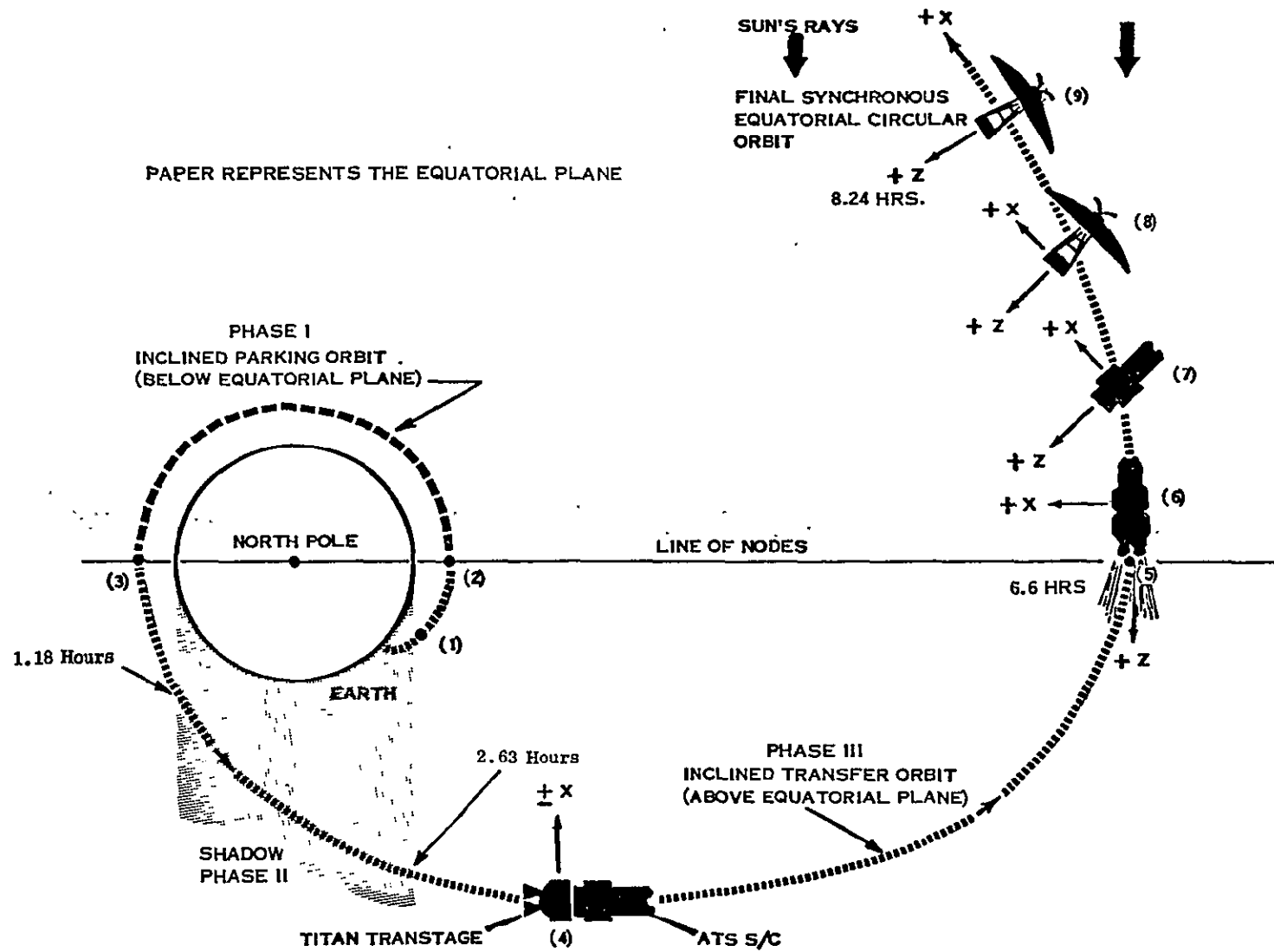


Figure 3.2-3. Ascent Sequence

- e. Solar panels are within 30 degrees of final solar orientation 42 minutes after start of acquisition (sufficient array power available to prevent further battery discharge).

The depth of discharge during Phase I and Phase II is then given by

$$\begin{aligned} E \text{ discharge} &= \frac{(119.5 \text{ w} \times 2.63 \text{ hrs.}) \times 0.7}{0.8} \\ &= 274 \text{ watt-hours} \end{aligned}$$

The depth of discharge for each battery is

$$\begin{aligned} \text{DOD} &= \frac{274 \text{ watt-hours}}{808 \text{ watt-hours}} \times 100 \text{ per cent} \\ &= 33.9 \text{ per cent} \end{aligned}$$

With a recharge time equal to

$$\begin{aligned} \text{Recharge Time} &= \frac{274 \text{ watt-hours}}{\frac{1.18 \text{ volts/cell} \times 28 \text{ cells}}{2 \text{ batteries} \times (1.2 = 0.4/2) \text{ amperes}}} \times 1.3 \\ &= 3.85 \text{ hours} \end{aligned}$$

The depth of discharge during Phase III and the acquisition phase is given by

$$\begin{aligned} E \text{ discharge} &= \frac{(119.5 \text{ w} \times 0.15 \text{ hr.}) + (173.1 \text{ w} \times 0.07 \text{ hr.}) + (149.7 \text{ w} \times 0.63 \text{ hr.})}{0.8} \\ &= 1.55 \text{ watt-hours} \end{aligned}$$

The depth of discharge for each battery at time of array deployment is

$$\begin{aligned} \text{DOD} &= \frac{155 \text{ watt-hours}}{808 \text{ watt-hours}} \times 100 \text{ per cent} \\ &= 19.2 \text{ per cent} \end{aligned}$$

As shown by the previous calculations, the 28-cell battery with 12 ampere hour rating has sufficient capacity for the ascent and occult modes with a depth of discharge above the design objective of 40 per cent for one battery.

14.4.7 BATTERY CHARGING RATES

The energy removed from the 28-cell, 12A-H batteries (using an 80 per cent efficient discharge path and a 0.15 hour occult) is as follows:

$$E \text{ discharge} = 285 \text{ watt-hours}$$

or

$$\frac{285 \text{ watt-hours}}{1.18 \text{ volts/cell} \times 28 \text{ cells}} = 8.64 \text{ Ampere-Hours}$$

The amount of time available for recharging is greater than 18 hours (24 hours less 4 hours for ITV less 1.15 hour occult). For a battery temperature of 20 degrees $\pm$ 10 degrees C, an energy replacement of 130 per cent would be required to accomplish this. Over an 18 hour period, this would set a lower charge rate limit of

$$\frac{8.64 \text{ ampere hours} \times 1.30}{2 \text{ batteries} \times 18 \text{ hours}} = 0.312 \text{ amperes/battery}$$

This value of 312 milliamperes is a rate below the recommended recharge rates to recover full battery capacity. A minimum realistic charging rate to obtain 100 per cent recharge is between C/10 and C/5 (1.2 to 2.4 amperes). Therefore, the recharge time using the ATS charger is

$$\frac{8.64 \text{ ampere hours} \times 1.30}{2 \text{ batteries} \times (1.2 + 0.400/2) \text{ amperes}} = 4 \text{ hours at } 1.4 \text{ amperes/battery}$$

During the remaining 20 hours, the charge rate will be reduced to a maintenance rate equivalent to 400 milliamps (C/30) at a 50 per cent duty cycle rate.

14.5 COMMANDS

The commands for the power subsystem listed in Table 14.5-1 will provide the command control required for normal operation of the power subsystem, and at the same time, will provide sufficient command control for back-up control during possible malfunctions. Of the 50 commands shown in Table 14.5-1, 16 are implemented to provide ON/OFF control for the GFE experiments.

All the loads that derive power directly from the 30.5-volt spacecraft load bus 1 and bus 2 are required to have a load interface circuit (LIC) between the load and the 30.5-volt busses. These load interface circuits are the same type as those developed for the GFE experiment loads.

Table 14.5-1. Power Subsystem Commands

No.	Command
1	Battery #1 Main Charger ON
2	Battery #1 Main Charger OFF
3	Battery #2 Main Charger ON
4	Battery #2 Main Charger OFF
5	Battery #1 Trickle Charger ON
6	Battery #1 Trickle Charger OFF
7	Battery #2 Trickle Charger ON
8	Battery #2 Trickle Charger OFF
9	Battery #1 Connect
10	Battery #1 Disconnect
11	Battery #2 Connect
12	Battery #2 Disconnect
13	Regulated +30 .5 v Bus #1/#2 Connect
14	Regulated +30 .5 v Bus #1/#2 Disconnect
15	Regulated +25 v Bus #1/#2 Connect
16	Regulated +25 v Bus #1/#2 Disconnect
17	Battery #1 Conditioner ON
18	Battery #1 Conditioner OFF
19	Battery #2 Conditioner ON
20	Battery #2 Conditioner OFF
21	Solar Array Bus Connect
22	Solar Array Bus Disconnect
23	Regulator #1 ON
24	Regulator #1 OFF
25	Regulator #3 ON
26	Regulator #3 OFF
27	Regulator #4 ON
28	Regulator #4 OFF
29	Regulator #5 ON
30	Regulator #5 OFF
31	Connect Command Subsystem #1 to Regulated Bus #1
32	Connect Command Subsystem #1 to Battery #1
33	Connect Command Subsystem #2 to Regulated Bus #2
34	Connect Command Subsystem #2 to Battery #2

Table 14.5-1. Power Subsystem Commands (Cont'd)

No.	Command
35	MM Wave (20 GHz) Experiment ON
36	MM Wave (20 GHz) Experiment OFF
37	Laser Experiment ON
38	Laser Experiment OFF
39	Radio Frequency Interference Experiment ON
40	Radio Frequency Interference Experiment OFF
41	Time and Frequency Dispersion Experiment ON
42	Time and Frequency Dispersion Experiment OFF
43	Environmental Measurements Experiment ON
44	Environmental Measurements Experiment OFF
45	High Resolution Camera Experiment ON
46	High Resolution Camera Experiment OFF
47	Radiometer Experiment ON
48	Radiometer Experiment OFF
49	MM Wave (30 GHz) Experiment ON
50	MM Wave (30 GHz) Experiment OFF

For the GFE experiments, the LIC's are separate components of the power subsystem; for the remaining spacecraft loads, the LIC's are packaged with the load or within a component that controls other loads. The LIC's provide the ON/OFF command control for all 30.5-volt bus loads, except for the command receivers and decoders. The command receivers use a simple resistor/zenor diode for their 0.8-watt power regulator, and the command decoders use a special form of the standard LIC that provides overload protection and automatic turn-on when the input bus is above 22 volts.

The high-power rf power amplifier loads connected to the +25 volt busses will have the ON/OFF command control circuits for the individual load located within the load package. These loads can also be turned ON and OFF by command control of the regulators (A415 and A417) that power these +25 volt busses.

The relays that are controlled by command for the power subsystem are as shown in Figure 14.3-1 and the load interface circuits are shown in Figure 14.3-2. Sixteen commands are required by the power subsystem for the LIC's, and these commands interface directly with the units shown in Figure 14.3-2 for the specific experiment load. The remaining 34 commands control the relays shown in Figure 14.3-1; all the relays shown in Figure 14.3-1 are located in the Power Control Unit.

14.6 TELEMETRY MONITORS

The telemetry monitors for the power subsystem listed in Table 14.6-1 will provide the operational data required for the power subsystem, and at the same time, will provide sufficient diagnostic data in the event of malfunctions.

14.6.1 OVERALL FUNCTIONS

The telemetry monitors are needed to perform the following:

- Provide status indications to verify that command functions have been executed.
- Provide measurements of parameters that might directly affect the power subsystem performance.
- Provide data needed for normal, no-failure, mission operations.
- Provide data required during failure modes in order to perform the necessary corrective action.

Of the 48 monitors listed in Table 14.6-1, 26 are analog and 22 are discrete. The placement of these monitors is shown in Figures 14.3-1 and 14.3-2. Figure 14.3-2 shows the discrete monitors (total of eight) at the output of each load interface circuit for the experiment. Figure 14.3-1 shows the placement of the remainder of the monitors, except for the solar array temperature monitors which are located on the solar panels (two per panel) and the battery temperature monitors (one per battery). Typical circuits for the telemetry monitors are shown in Figure 14.6-1 (voltage monitor), 14.6-2 (discrete monitor), and 14.6-3 (temperature monitor). The eight current monitors listed in Table 14.6-1 will be magnetic amplifier type current sensors such as those manufactured by Magnetic Controls Company for the Pegasus, Sert II, and OAO spacecrafts.

14.6.2 SPECIFIC FUNCTIONS

The specific functions performed by the telemetry monitors for the power subsystem components are as follows:

- Solar Array
 - a. Measure temperature at two locations on each solar panel
 - b. Measure solar array current for solar array bus 1 and bus 2
 - c. Measure the solar array voltage on solar array bus 1 and bus 2
 - d. Indicate the status for the bus tie relay between the solar array bus 1 and bus 2
- 30.5-Volt Regulators
 - a. Indicate ON/OFF status to show that these regulators are connected to the solar/battery power sources
 - b. Measure the output current on spacecraft load bus 1 and bus 2
 - c. Measure the regulated output voltage for each regulator at the spacecraft load bus 1 and bus 2
 - d. Indicate the status of the bus-tie relay between the spacecraft load bus 1 and bus 2
 - e. Measure the voltage level to the command subsystem to indicate either a 30.5 volt or battery power source connection

- 25-Volt Regulators
 - a. Indicate ON/OFF status to show that these regulators are connected to the solar array/battery power sources
 - b. Measure the output current on power amplifiers bus 1 and bus 2
 - c. Measure the regulated output voltage for each regulator at the rf power amplifiers bus 1 and bus 2
 - d. Indicate the status of the bus-tie relay between the rf power amplifiers bus 1 and bus 2
- Battery
 - a. Measure the temperature at the hot spot on each battery
 - b. Measure the charge/discharge current for each battery
 - c. Measure the battery voltage for each battery
 - d. Measure the battery reconditioning voltage for each battery
 - e. Indicate the status of the battery connect/disconnect relay for each battery
- Charger
 - a. Indicate the ON/OFF status of the main charge mode for each charger
 - b. Indicate the ON/OFF status of the trickle charge mode for each charger
- Load Interface Circuits
 - a. Indicate the ON/OFF status for the GFE experiments connected to the eight LIC's

The selection of the telemetry monitors has been kept to a minimum level consistent with the parameters that must be monitored for sufficient coverage of the power subsystem functions.

Table 14.6-1. Power Subsystem Telemetry Monitors

No.	Monitor	Type
1	Solar Array Bus #1 Current	Analog
2	Solar Array Bus #2 Current	Analog
3	Power Amplifiers Bus #1 Current	Analog
4	Power Amplifiers Bus #2 Current	Analog
5	Battery #1 Charge/Discharge Current	Analog
6	Battery #2 Charge/Discharge Current	Analog
7	Spacecraft Load Bus #1 Current	Analog
8	Spacecraft Load Bus #2 Current	Analog
9	Solar Panel Temp #1	Analog
10	Solar Panel Temp #2	Analog
11	Solar Panel Temp #3	Analog
12	Solar Panel Temp #4	Analog
13	Battery #1 Temp	Analog
14	Battery #2 Temp	Analog
15	Battery #1 Voltage	Analog
16	Battery #2 Voltage	Analog
17	Solar Array Bus #1 Voltage	Analog
18	Solar Array Bus #2 Voltage	Analog
19	Regulated Bus #1 Voltage	Analog
20	Regulated Bus #2 Voltage	Analog
21	Power Amplifiers Bus #1 Voltage	Analog
22	Power Amplifiers Bus #2 Voltage	Analog
23	Battery #1 Conditioner Voltage	Analog
24	Battery #2 Conditioner Voltage	Analog
25	Command Subsystem #1 Voltage	Analog
26	Command Subsystem #2 Voltage	Analog
27	Battery #1 Main Charger ON/OFF Status	Discrete
28	Battery #2 Main Charge ON/OFF Status	Discrete
29	Battery #1 Trickle Charger ON/OFF Status	Discrete
30	Battery #2 Trickle Charger ON/OFF Status	Discrete
31	Battery #1 Connect/Disconnect Status	Discrete
32	Battery #2 Connect/Disconnect Status	Discrete
33	Regulated +30.5 v Bus #1/#2 Connect/Disconnect Status	Discrete
34	Regulated +25 v Bus #1/#2 Connect/Disconnect Status	Discrete
35	Solar Array Bus Connect/Disconnect Status	Discrete
36	Regulator #1 ON/OFF Status	Discrete
37	Regulator #2 ON/OFF Status	Discrete
38	Regulator #3 ON/OFF Status	Discrete
39	Regulator #4 ON/OFF Status	Discrete
40	Regulator #5 ON/OFF Status	Discrete

Table 14.6-1. Power Subsystem Telemetry Monitors (Cont 'd.)

No.	Monitor	Type
41	High Resolution Camera Experiment ON/OFF Status	Discrete
42	Radiometer Experiment ON/OFF Status	Discrete
43	Environmental Measurements Experiment ON/OFF Status	Discrete
44	Laser Experiment ON/OFF Status	Discrete
45	Radio Frequency Interference Experiment ON/OFF Status	Discrete
46	Time and Frequency Dispersion Experiment ON/OFF Status	Discrete
47	MM Wave (20 GHz) Experiment ON/OFF Status	Discrete
48	MM Wave (30 GHz) Experiment ON/OFF Status	Discrete

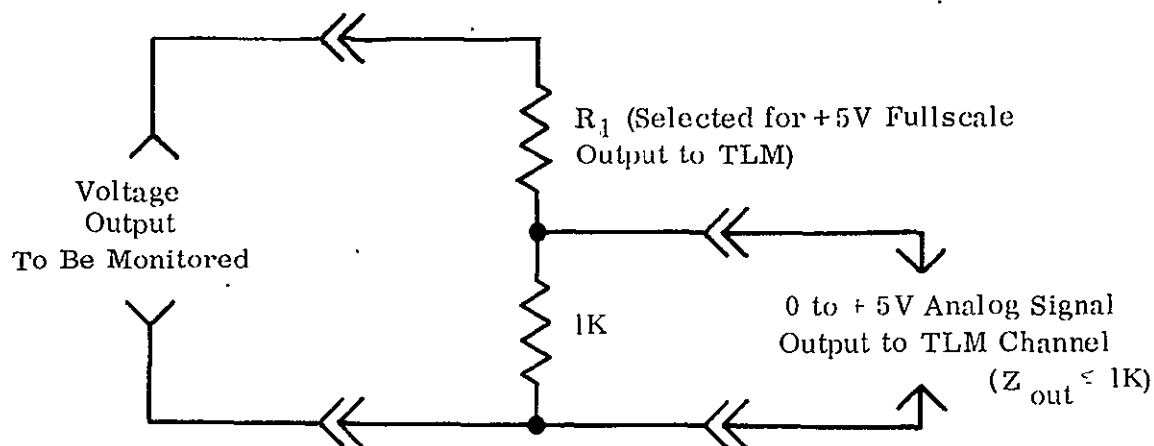


Figure 2.5-1. Typical Analog Voltage Monitor Circuit

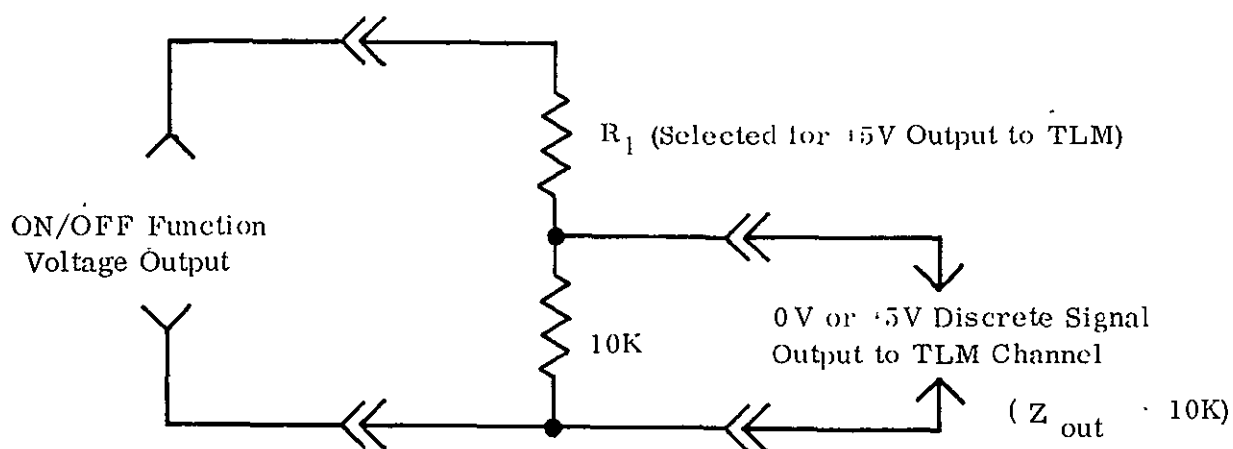


Figure 2.5-2. Typical Discrete Monitor Circuit

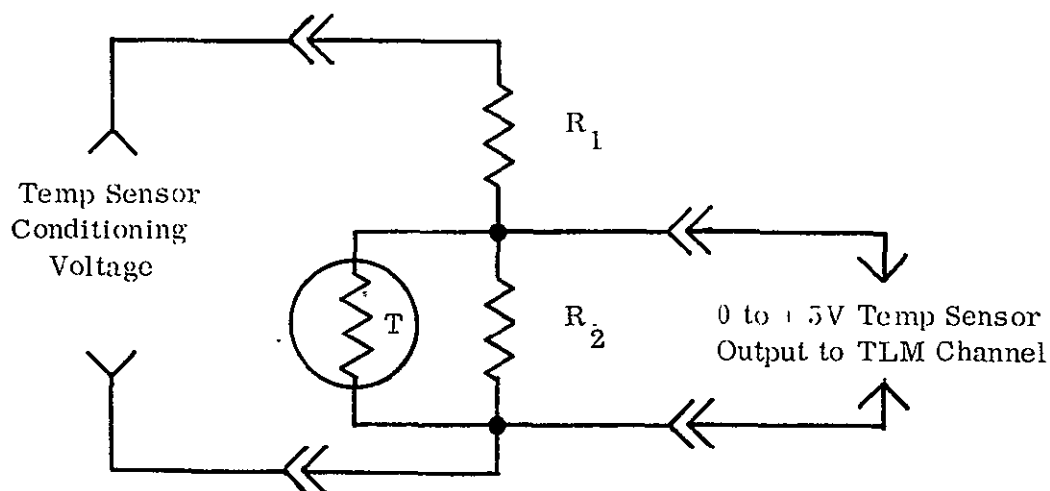


Figure 14.6-3. Typical Temperature Monitor Circuit

SECTION XV

HARNESS SUBSYSTEM

15.1

INTRODUCTION

The ATS Harness Subsystem is defined to include all electrical interconnecting cables that interface with individual equipment modules. These cables may be a portion of an individual subsystem or a connection between subsystems. Encompassing all spacecraft cabling within the purview of one subsystem has allowed the standardization of cable design and interconnecting guidelines. In this way black box interface coordination and interconnecting wiring techniques for the entire spacecraft are regulated by the Harness Subsystem development, and vehicle integration becomes an inherent process within this development. The product of this single effort is a cable network that will be uniform in appearance, parts usage, and electrical characteristics.

Associated with harness development is the responsibility of maintaining system electrical characteristics compatible with component operation. This includes meeting electromagnetic requirements by incorporating twisting, shielding, and grounding guidelines; dictating power line voltage drop limits; imposing current and voltage restrictions for wire and connector pins; providing test connections; meeting pyrotechnic design restrictions; maintaining signal characteristics. These design goals are fulfilled by imposing system guidelines. This control, utilized in the development of the Harness Subsystem, ensures that individual component performance is not compromised by the system integration process, but also commands that subsystem interconnection be regulated by system design. Each of the specific wiring guidelines is discussed in the ensuing paragraphs of this section.

The integration of the spacecraft equipments into the electronics system is a crucial point in spacecraft buildup which must not be impeded by harness redesign. Therefore, the importance of completing a thoroughly developed Harness Subsystem early in the program is emphasized by FHC. The approach is to force system harness design on each subsystem component by creating extensive documentation to identify the pre-designed system. This approach necessitates the systematic organization of component interfaces to ensure economical wiring. The telemetry, command and power interfaces present the major areas where organization is required to identify how an individual component will interface with the respective subsystems. This interface control minimizes system interfacing exterior to components, forcing compatibility to be built into each rather than planning on harness adaption.

The Harness Subsystem was derived from basic philosophies originated to ensure ATS Spacecraft performance without degradation. These philosophies have been reduced to a set of specific working rules contained in the FHC Spacecraft Wiring, Grounding Techniques and Components Design Implementation (685-SD-01) section of the ATS F&G Design Manual (ref. appendix A). Included in the guidelines are the following topics:

- Redundant Wiring
- Isolation Between Redundant Wiring
- Return Line Implementation
- Twisted Wire Usage
- Shielding and Shield Termination
- Grounding and Isolation of Returns
- Power Line Voltage Drop
- General Wire Sizing
- Connector Pin Current Derating

These guidelines along with specific requirements for individual components form the basis for implementation of the Harness Subsystem design. Presented in the following sections are those wiring and interfacing methods which have been imposed on system design.

15.1.1 REDUNDANT WIRING

In general, the system approach to redundant spacecraft wiring is concerned with maintaining the reliability achieved within each subsystem. Thus, where redundant components are implemented (e.g., in the Telemetry Subsystem) there will be completely independent-isolated wiring to each unit. To increase spacecraft reliability, additional redundant wiring to any single unit must be considered. The evaluation of additional redundant wiring must be made from the following criteria:

- Delta change in the reliability of the specific function
- Category of failure should the single line fail
- Increase in weight for redundancy
- Increase in cost for adding redundant wiring and isolation

Each class of wiring must be evaluated in light of these criteria to determine if the added line failure protection is required.

15.1.1.1 Redundant Power Lines

The wire sizing to each component will generally be a function of the power requirements of each unit and a maximum limit on power line voltage drop of one percent. However, minimum size will be limited to #22 and maximum size to #16. The latter is imposed by the unavailability of qualified large-pin connectors. Redundancy for power lines will be implemented by providing a minimum of two twisted pairs to each specified component. Each of the pairs will satisfy the one percent criterion independently, i.e., with the alternate pair inactive. Thus, redundancy of power lines is defined as enough conductor capability to handle twice the power requirements of the particular component. The guidelines for power line redundancy are:

- Each component shall provide twice the number of power pins required.
- Redundancy of a power line shall include return redundancy.
- All power subsystem interconnecting power lines shall be redundant.
- Components that are not redundant in system design shall be tied with one power line to one of the two spacecraft power busses - no redundancy.
- Critical components that are not redundant in system design, but which have isolated power input pins, shall be tied with one power line to each of two spacecraft power busses - isolated redundancy.
- Units redundant in system design shall have one line but each shall be tied to a different power buss.
- Experiment LIC's shall be connected with one power line to one of the two spacecraft power busses - no redundancy.
- Experiments shall be connected to LIC's with one power line.
- Pyrotechnic power lines shall be redundant.

15.1.1.2 Redundant Telemetry and Command Lines

The system design includes two redundant Data Acquisition Control Units and two redundant Command Decoder/Distributors. Thus, redundant wiring for each telemetry monitor and command is required to allow each of these units to be cabled independently, in keeping with the FHC general philosophy concerning redundant units.

15.1.1.3 Subsystem's Redundant Wiring

The guideline for determining redundancy of the various subsystem's wiring is to provide such wiring where a single line failure would cause a category I failure.

15.1.2 SHIELDED WIRE USAGE

The basic philosophy behind the use of shielded conductors is to protect the low-level shielded function from radiated interference and contain high frequency radiating sources. Thus, the implementation of shielded wire depends upon the function to be protected. The general shielding philosophies followed for the Harness Subsystem design are based upon general engineering practice and the following system design. Telemetry monitor lines are shielded to protect the low level functions from extraneous noise that would affect a particular function accuracy and that could adversely affect the operation of the Data Acquisition Control Units. The command lines are not shielded because the command subsystem design incorporates a minus five volt back bias when no command is present at the digital device.

15.1.3 TWISTED WIRE USAGE

The limitation of return lines necessarily limits the implementation of an extensive twisted pair wiring philosophy. The great reduction in magnetic field, which is realized by twisting two wires carrying equal and opposite current, is recognized, but requirements for this control have been limited. The magnetic field created by the current flow in a single wire becomes a problem in two areas. It may induce extraneous current flow on adjacent conductors and it may have an adverse effect on magnetic field measuring equipment. The first problem is primarily associated with relatively large magnetic fields set up around a conductor (i.e., large current). Thus, the solution to such a problem is to twist conductors where large current is expected. The second problem is concerned with minimizing the magnetic field in the field of view of the measurement device, the Environmental Measurement Experiment located on the reflector support hub.

The use of twisted pair for all direct current power lines has been incorporated in the harness design. This deals effectively with the first problem concerning interference with the functions carried on adjacent conductors. In addition, should three-phase alternating circuit be routed in this spacecraft harness, it will be cabled on triple twisted pair. General telemetry command, and low level signals are not to be wired as twisted pair.

15.1.4 SYSTEM GROUNDING

The effort to establish grounding control for the ATS F&G system integration is based upon isolation of individual types of grounds from one another and the ideal goal of a Single Point Ground system. Grounds have been classified into four categories; system power return, secondary power return, signal return, and chassis or case ground. Four possible points of departure from the ground isolation scheme have been considered.

- Spacecraft System Power Return Common with Signal Return
- Secondary Power Return Common with Signal Return

15.1.4 SYSTEM GROUNDING (cont.)

- Power Return Common with Chassis Ground
- Signal Return Common with Chassis Ground

The requirements of the communications equipment and those of particular qualified units which demand connection of different ground types preclude the implementation of the ideal ground scheme. A variation of the Signal Point Ground is a system which allows one component assembly with common grounds. This forces the selection of the ground point location. Although this plan requires cautious approach to lead lengths and parallel hook-up, the problem of ground loops is avoided.

The primary concern for the effects of common grounds is in the highly susceptible components and circuitry rather than the particular high power transmitters which usually force the common grounds. Therefore, FHC proposes to accept common grounds in transmitters and communication subsystem amplifiers. If the high power users violation of the ground plan presents a grounding problem, changes in isolation techniques or protective filtering will be incorporated in the spacecraft design.

The envisioned system has all power returns terminated at a central common point located adjacent to the Power Control Unit. This would provide a central point at which each component's power return lead would terminate, but would not require the added connectors to route all returns into the Power Control Unit. The central return terminal would be cabled into Power Control Unit on an adequate number of lines. This method displays the following attributes:

- Permits proximate termination of Hi and Lo sides of any set of twisted power lines
- Permits connection of the power return to chassis at a selected point in the system
- Guarantees that no single long cable run will carry added currents
- Minimizes the equipment necessary to buss returns
- Minimizes connectors on the PCU
- Is compatible with complete power distribution and control at the PCU

15.1.5 GENERAL WIRE SIZING

The Harness design includes recognition of the need to minimize the voltage drop on each component power line. In this light a maximum allowable voltage drop has been set at one per cent (1%) of the line voltage. The conductor gauge selected for the general areas of telemetry lines, command lines, and power lines are listed below. The basic restrictions on these selections are:

15.1.5 GENERAL WIRE SIZING (cont.)

- Minimum wire size of AWG 24 - limited by system requirements and connector design
- Maximum wire size of AWG 16 - limited by available qualified connector selection. (heavy currents to be carried by multiple conductors)
- Telemetry lines - AWG 24 wire
- Command lines - AWG 24 wire
- Power lines - Determined from voltage drop, due to current carried
- Subsystem interconnection - AWG 24 wire unless a larger conductor is required, in which case the conductor will be AWG 22, 20, or 16.

15.1.6 CONNECTOR PIN CURRENT DERATING

System operational performance depends heavily upon the dependability of each connector used in the spacecraft. Therefore, connector current limits have been derated. The following limits have been imposed on connector contact maximum current design for the ATS F&G spacecraft:

Non-hermetic sealed connectors shall be derated as follows:

- 60% of maximum contact current rating if 25% or fewer pins are used.
- 40% of maximum contact current rating if more than 25% of the pins are used.

Hermetic sealed connectors shall be derated as follows:

- 40% of maximum contact current rating if 25% or fewer pins are used.
- 30% of maximum contact current rating if more than 25% of the pins are used.

15.2 HARNESS OPERATIONAL REQUIREMENTS

The Harness Subsystem responsibilities include the additional control of various system operational capabilities. These requirements serve to allow integrated system performance and system accessibility.

15.2.1 BATTERY DISCONNECT

The system requirements stipulate that a manual battery disconnect be provided. This becomes an integration requirement and is met by a harness design that allows the ground lines of each battery harness to be broken, electrically separating the battery from the spacecraft system. This is accomplished by removing a readily accessible jumper plug located on the EVM-exterior harness interface connector plate (ref. Section III).

15.2.2 PYROTECHNICS SYSTEM

The pyrotechnic design requires special harnessing which will be implemented as individual cables. These cables will be coordinated with pyrotechnic design guidelines and safeguards. The cabling will be completely redundant, providing power lines from the battery system to the Squib Interface Circuits, on to the explosive device, and command lines to the Squib Interface Circuits. Shielded twisted pair conductors will be used exclusively and the routing of this cable will be physically separated from other cabling. This harnessing will provide lines to the Titan IIC and adapter separation mechanism, the solar array release and deployment, solar paddle boom-hinge release, reflector deployment, and Commandable Gravity Gradient thermal battery activation and separation system. The four transtage separation wiring cables are redundant backup for the Titan separation system.

15.2.3 UMBILICAL INTERFACE

The Harness Subsystem responsibility includes the coordination between individual subsystems and ground support efforts that will establish umbilical interface requirements. Currently defined as a definite requirement is ground power to operate the system prior to launch. This umbilical connection is made with power lines routed through the Titan launch vehicle. The location of the umbilical connector on the EVM is discussed in Section III. The separation of these lines is by means of a lanyard pull, fly-away connector disconnected at Titan-spacecraft separation.

15.3 HARNESS SUBSYSTEM CONFIGURATION

The physical description of the ATS Harness Subsystem includes the presentation of the basic harness configuration, the philosophies underlying this technique, the parts used in this system, its maintainability, and weight. This description is presented to demonstrate harness compatibility with the spacecraft structure and the conformance with the integration philosophy that evolved with the modularized Earth Viewing Module (EVM).

The spacecraft requires an interconnecting harness system for the EVM equipments, Orbit Control Propellant Jets located on the reflector support truss, reflector support hub mounted equipments, solar panels, and VHF antennas. The necessity for establishing a clean interface between EVM and external cabling has been established from design and test requirements. The design requirements stipulate that the selection of harness parts for that cabling exterior to the EVM be given

special consideration, and the test requirements propose that the EVM be integrated and partially tested without the support truss. Therefore, the harness design includes separate cable systems for the EVM and external harnesses. The following description is divided into these two areas.

The transformation from the unified circular EVM to a modular rectangle has brought forth the extensive effort to segregate each spacecraft subsystem into a separable structural section. The primary purpose of this design being the remote parallel integration and test at the subsystem level and the consequential system integration simplification. This approach necessarily involves the utilization of separate harness networks for each spacecraft subsystem. This technique has been adopted to enhance the modular concept and the separation of structural sections after harness installation. Further discussion is presented in the succeeding paragraphs.

15.3.1 GENERAL EARTH VIEWING MODULE HARNESS LAYOUT

The division of the EVM harness into separate networks for the Attitude Control Subsystem, Communications Subsystem (including RFI, T&FD, and Radio Beacon communications experiments), and the Experiment, Power and Telemetry and Command Subsystems has been accomplished by utilizing approximate locations for subsystem components and stipulating subsystem level interfaces. Each harness network is comprised of a primary cable and smaller secondary cable. The interfaces between the various subsystem are made through dedicated connectors on certain components (e.g., the Data Acquisition Units of the T&C Subsystem have specific connectors allocated exclusively to the Attitude Control Subsystem) and bulkhead connectors where the harness is broken at a point other than the component interface (e.g., the intermodule RFI shield).

The use of secondary cables between several components has been incorporated to promote direct cabling, maintainability, and parallel construction, but is limited in use by weight and surface area considerations. The spacecraft RF cables will be manufactured as individual cables and installed with the primary harness where appropriate. This allows complete flexibility in RF cable removal and installation.

Figure 15.3-1 shows the EVM before integration of the separate structural modules. The EVM harnessing has been divided to allow this separation after cable installation. This approach allows flexibility in integration, test, and maintainability. The EVM detail harnessing is discussed in the following paragraphs.

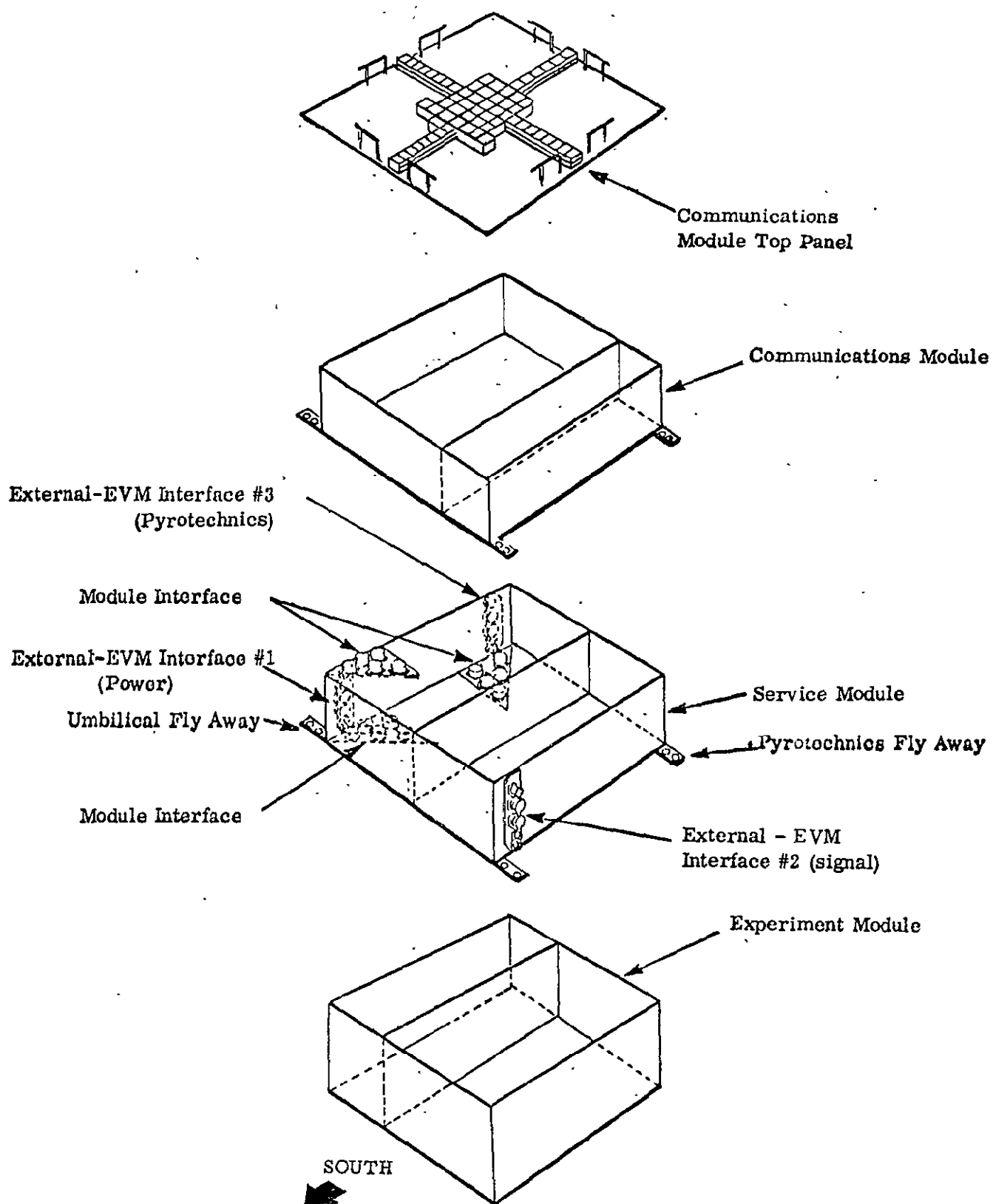


Figure 15.3-1 Exploded EVM - Prior to Harness Installation

The experiment module is the lower section of the EVM, as depicted in Figure 15.3-2. The electronic equipments and detailed harness located in this module are called out in Figure 15.3-3, EVM Experiment Module Harness Layout. Generally, these equipments are categorized into Experiments, Power Subsystem Units, Attitude Control Subsystem units, and the Interferometer units. The cabling within this module has been divided into segments which maximize parallel construction of the cabling and allow separate subsystem testing. Shown in Figure 15.3-2 are each of the harnesses separated from the equipment of this level. The first harness is the cabling for experiments and power equipments in this module. This harness (A445A6) is clamped to the upper edge of this module and utilizes the center heat pipe wall as support for the interconnections between the north and south mounted equipment. The second depicts the ACS harness (A445A7) in this module. This harness follows the routing shape of the experiment harness, but has been completely segregated to allow complete ACS integration and test without interfering with the remaining spacecraft development. The third harness of this module, shown at level number three, is the ACS Earth Sensor Electronics and Sensor head interconnection (A445A11). This harness, along with the interferometer cabling (A445A8, A9, A10) shown at the fourth level are secondary cables provided to allow direct wiring and enhance parallel construction.

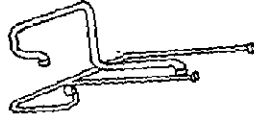
The interconnection between the experiment and service modules is made through interface connectors with the receptacles hard mounted on plates located at the lower south-west and north-west corners of the service module. This interface consists of ACS interface connectors, power interface connectors, and the experiment telemetry, command and signal connectors.

15.3.2.1 ACS Harness

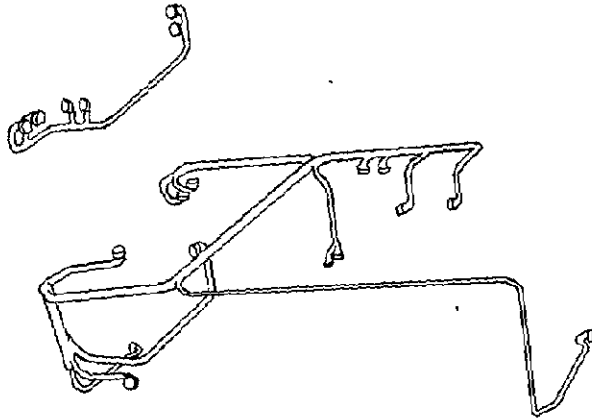
The Attitude Control Subsystem Harness A445A7 (ref. Figure 15.3-2) located in the experiment module provides subsystem interconnection, power lines, and telemetry and command lines to the following ACS units:

A151	Polaris Sensor #1
A152	Polaris Sensor #2
A153	Power Converter Signal Conditioner (PS #1)
A154	Power Converter Signal Conditioner (PS #2)
A123	Fine Sun Sensor (East)
A124	Fine Sun Sensor (West)
A178	Earth Sensor Electronics Assembly

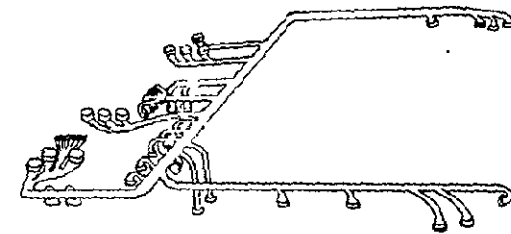
Level 4
Interferometer Harness
A445A8, A9, A10



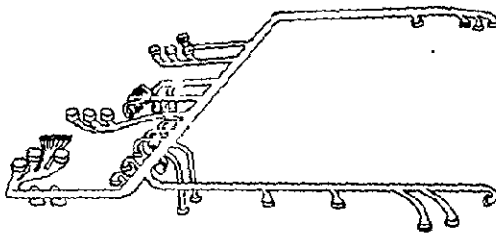
Level 3
Earth Sensor
ACS Harness
A445A11



Level 2
ACS Harness
A445A7



Level 1
Power, Experiment
Harness
A445A6



Experiment Module

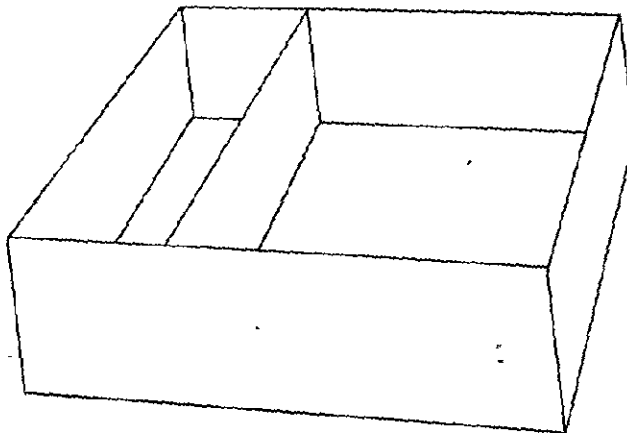


Figure 15.3-2. Experiment Module Harnessing

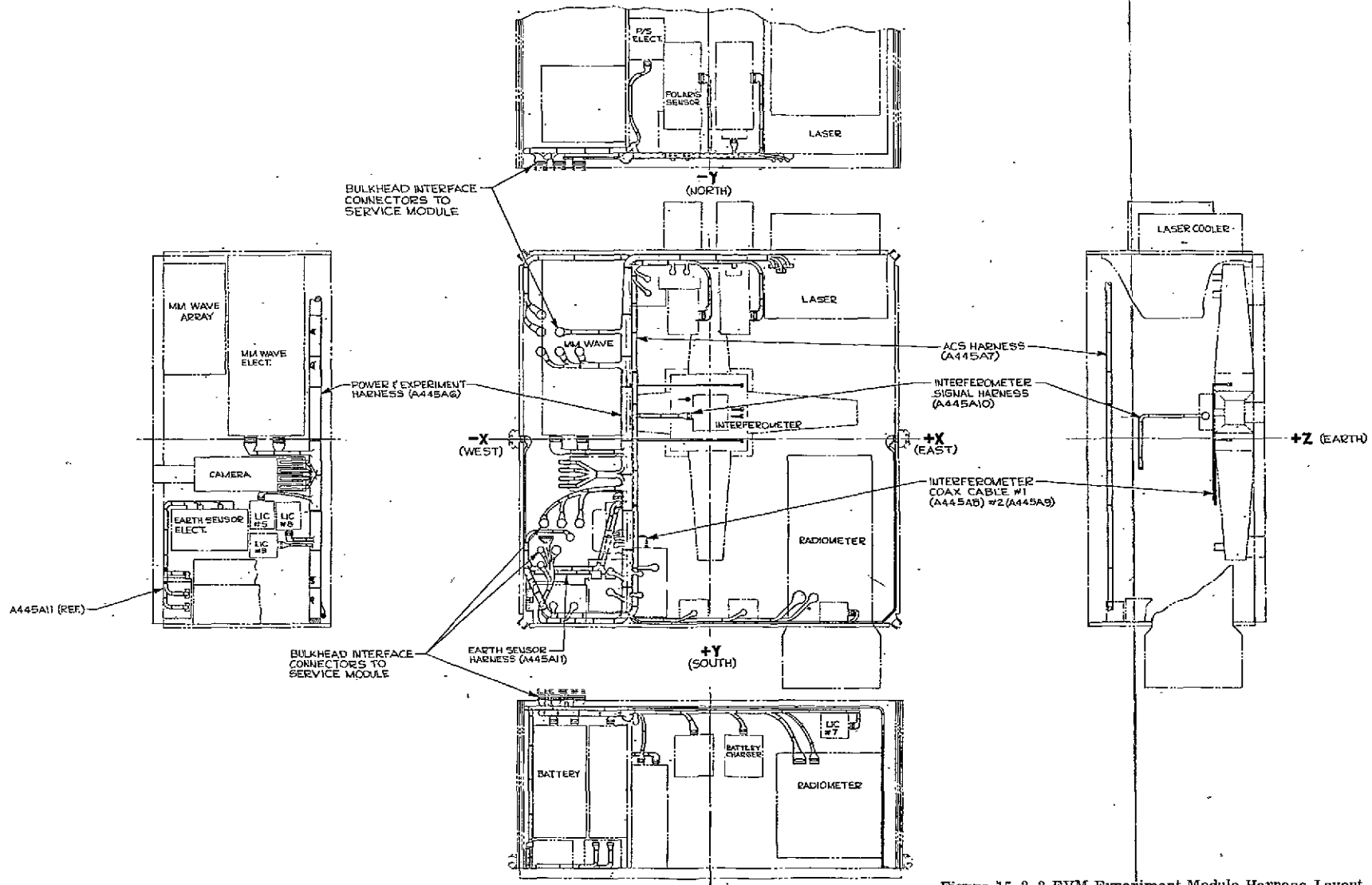


Figure 15.3-3.EVM Experiment Module Harness Layout

15.3.2.1 ACS Harness (cont.)

This harness has approximately 210 individual interconnecting wires of which approximately 140 interface with the service module harnessing at the inter-module connector plates identified in Figure 15.3-2. This harness will be constructed from the Raychem Specification 44 type wire.

15.3.2.2 Experiment Harness

The Experiment Subsystem Harness A445A6 located entirely within the experiment module provides experiment signal, power, telemetry, and command cabling and interfaces with several of the Power Subsystem components. The following units interface with this harness:

A731	High Resolution Camera
A761	Laser Electronics
A771	Millimeter Wave Electronics #1
A772	Millimeter Wave Electronics #2
A800	Radiometer
A418	LIC #1 (Laser)
A422	LIC #5 (Camera)
A425	LIC #7 (Radiometer)
A426	LIC #8 (MMW K _u)
A427	LIC #9 (MMW K _a)
A401	Battery #1
A402	Battery #2
A406	Battery Charger #1
A407	Battery Charger #2
A272	Interferometer Electronics

The total number of conductors in this harness is approximately 550 with 480 of these terminating at the intermodule connector plates. In addition, miniature coaxial cables are routed with this cabling and pass through the intermodule connector plate. These cables connect various experiments and the communication subsystem.

15.3.2.3 ACS Harness (Earth Sensor)

This small Attitude Control Harness interconnects the Pitch Earth Sensor Head Assembly A176 and the Roll Earth Sensor Head Assembly A177 with the Earth Sensor Electronics Assembly A178 (Ref. Figure 15.3-2). This harness is a small interconnecting cable between associated equipments (approximately 50 wires) which FHC has isolated as a separate cable to contribute to the modularized cabling approach. This cable does not interface through an intermodule connector plate.

15.3.2.4 Interferometer Harness

The interferometer array (A276) and electronics (A272) assemblies are interconnected by three small cable assemblies; one low voltage cable (A445A10) and two coaxial cables (A445A8 and A445A9). These assemblies are point to point hook-up and consequently are segregated from the Experiment Harness which interconnects with the Interferometer Electronics Assembly.

15.3.2.5 Harness Segregation

Additional harnesses may evolve within this Experiment Module. Investigation into the feasibility of separating the Polaris Sensors interconnection with its electronics and of separating the Battery and Charger cabling from the main harnesses is being conducted. The primary concern in the effort, is associated with available cable tie down surface and dividing each component interface into discrete segments to allow point to point wiring.

15.3.3 SERVICE MODULE LAYOUT

The Service Module is the middle structural section of the EVM, as shown in Figure 15.3-1. The specific equipments and harnessing located within this module are shown in Figure 15.3-4. These equipments may be grouped into Power Subsystem units, Telemetry and Command Subsystem units, and the Attitude Control Subsystem units. The Propulsion System electrical components suspended from this module will also be harnessed as a part of the ACS equipment. Again the cabling in this module has been divided into segments that comply with design objectives. Shown in Figure 15.3-5 is an exploded view of the basic harnesses. The cabling in this module is clamped to the upper edge of the heat pipe wall and north-south faces. Where necessary, the cable legs are supported by fiberglass raceways.

The raceway has the following advantages:

- It frees the heat pipe lined surfaces from all cabling in the high density service module
- It provides an unobstructed harnessing path
- It is designed to position the harness for specific equipment so connections are made without strain on the cable

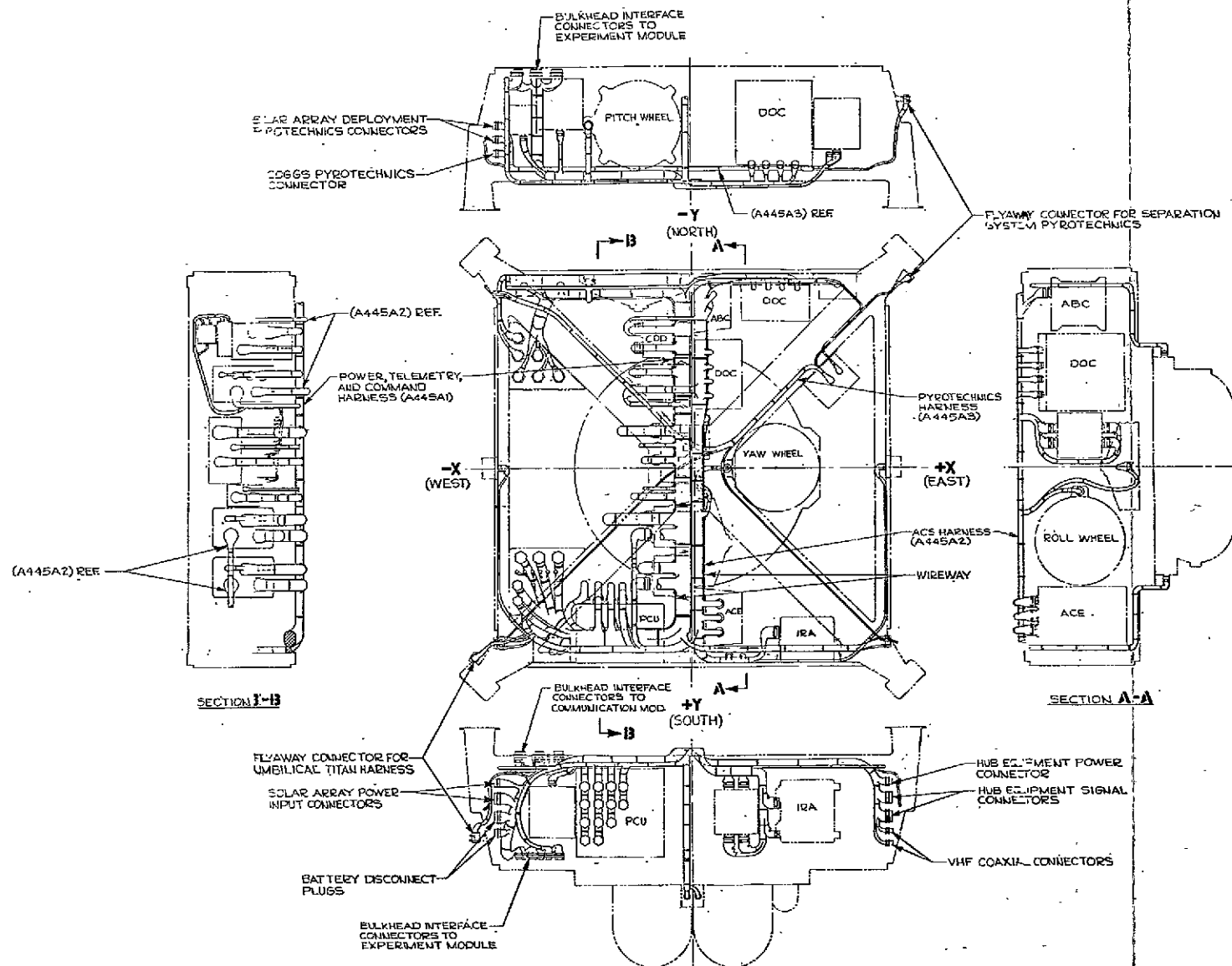
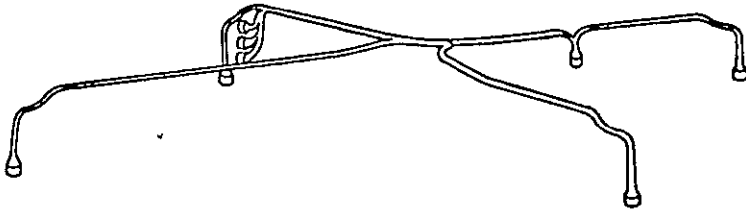


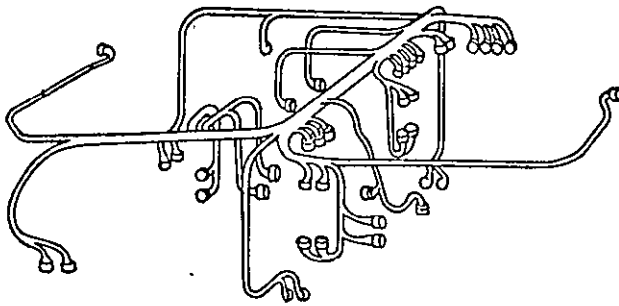
Figure 15.3-4 EVM Service Module Harness Layout

FOLDOUT FRAME 2

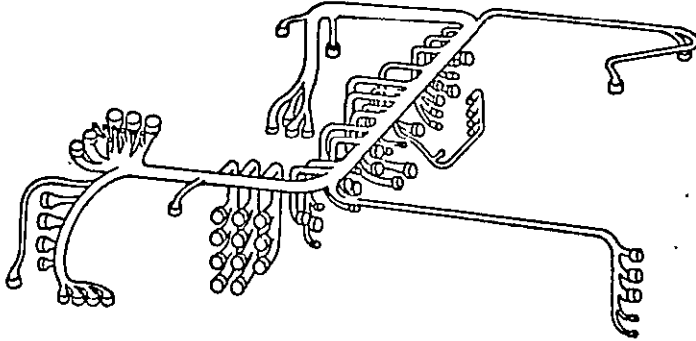
FOLDOUT FRAME 3



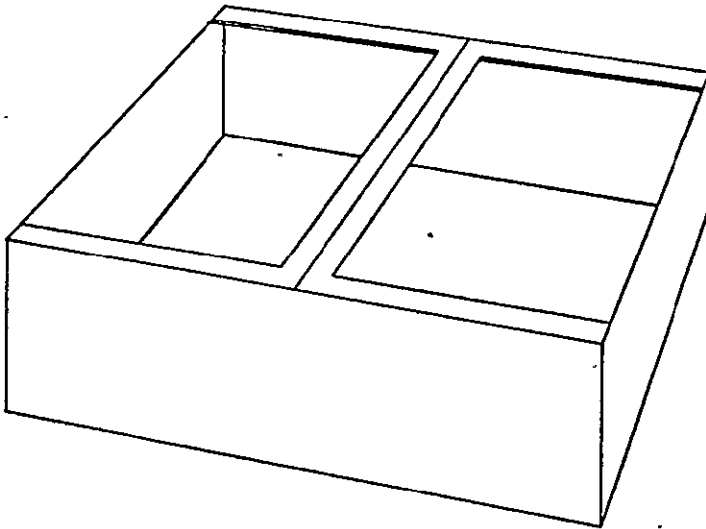
Level 3
Pyrotechnics Harness
A445A3



Level 2
ACS Harness
A445A2



Level 1
Power, T&C Harness
A445A1



EVM - Service Module

Figure 15.3-5 Service Module Harnessing

15.3.3 SERVICE MODULE LAYOUT (cont.)

- It simplifies installation (clamping large harnesses to the fiberglass trough rather than the honeycomb panels)

The first level depicts the harness that connects the Power Subsystem Units and Telemetry and Command Subsystem units into the system. This harness (A445A1), also interfaces with the exterior harnessing, umbilical, communication module, and the experiment module. The second level shows the Attitude Control Harness (A445A2) associated with the service module. The ACS harness assembly interconnects the various subsystem components in this module, the APS electronics, and interfaces with the exterior and experiment module harnesses to interconnect remote ACS components. The third level shows the squib pyrotechnics harness (A445A3). The overall cable layout of this module displays the high conductor density associated with these equipments, and emphasizes the importance of an exact layout design that strives to simplify manufacturing.

15.3.3.1 Power, Telemetry, and Command Harness

This primary service module harness (A445A1) (ref. Figure 15.3-5), interconnects the various Power Subsystem units, interconnects the T&C Subsystem units, and routes power, telemetry and command lines to each of the four interfaces; exterior, umbilical, communications module, and experiment module. The following units are connected to the system by this assembly:

A416	Power Control Unit
A413	Regulator #3
A415	Regulator #4
A417	Regulator #5
A421	LIC #4 (EME)
A201	Data Acquisition and Control Unit #1
A202	Data Acquisition and Control Unit #2
A205	Spacecraft Clock
A211	VHF Transmitter #1 (OMNI)
A212	VHF Transmitter #2 (OMNI)
A216	VHF Diplexer #1
A217	VHF Diplexer #2
A231	VHF Receiver #1
A323	VHF Receiver #2

15.3.3.1 Power, Telemetry and Command Harness (cont.)

A236	Command Decoder/Distributor #1
A237	Command Decoder/Distributor #2
A241	Data Switching Unit
	Umbilical Interface Connector

This harness is comprised of approximately 1650 conductors arranged in shielded or unshielded and twisted or untwisted configurations. Approximately 200 wires interface with the exterior harness, 480 with the experiment module, 285 with the communications module, and 10 with the umbilical. This harness will be constructed from the Raychem Corporation Spec. 44 wire.

15.3.3.2 ACS Harness

The Attitude Control Harness (A445A2) (Ref. 15.3-5) for this module interconnects the majority of the subsystem located in the east half of the compartment and routes power, telemetry and command lines for the entire ACS subsystem. The interface between the three primary functions cabled independently in this module is made by utilizing dedicated connectors on the distribution units of the power and T&C subsystems. All power ACS interfacing is done through specific connectors on the Power Control Unit exclusively allocated to the ACS power interconnections. These power lines are routed within the ACS Harness (A445A2) to specific subsystem units within the module and through the various ACS interface connectors to the ACS harnesses associated with the various locations (i. e., through the intermodule connector plate to the Experiment Module ACS Harness (A445A7). The same technique, incorporating dedicated connectors, is utilized to interface the ACS with the T&C Subsystem. In this manner, the entire Attitude Control Subsystem is interconnected with a single set of cables. Further, the interface with other subsystems is made at the dedicated connectors which mate with the service subsystems' components, providing a clean interface for subsystem test. The units connected by this ACS harness are:

A108	Orbit Control Jet Nozzle Assembly #1 (E-W)
A109	Orbit Control Jet Nozzle Assembly #2 (N-S)
A113	Regulated Pressure Feed Assembly #1
A114	Regulated Pressure Feed Assembly #2
A117	Flow Control Valve Assembly #1
A118	Flow Control Valve Assembly #2
A120	Flow Control Electronics #1
A212	Flow Control Electronics #2
A133	Coarse Sun Sensor (+X)

A134	Coarse Sun Sensor (-X)
A141	Sun Sensor Electronics #1
A142	Sun Sensor Electronics #2
A161	Inertia Wheel Pitch
A162	Inertia Wheel Yaw
A163	Inertia Wheel Roll
A169	3-Axis Inertia Reference Assembly
A185	Digital Operational Controller #1
A186	Digital Operational Controller #2
A188	Analog Backup Controller
A190	Actuator Control Electronics Module Interfaces Exterior Interface

The interconnection with the attitude propulsion associated electronics is made directly through this harness (i.e., DOC-Actuator Control-Flow control wiring is entirely within this harness).

The total number conductors in this harness is approximately 750, with 140 interfacing with the telemetry units, 190 with the command decoders, 90 with the Power Control Unit and 140 with the external and intermodule interfaces. This harness will be constructed from the Raychem Corporation Spec 44 type wire, except for the Orbit Control Jet Heater lines which will be Kapton Polymide Insulated Wire.

15.3.3.3 EVM Pyrotechnics Harness

The deployment and separation harnessing (A445A3) routed within the EVM interfaces with the squib interface circuits unit (A450) and routes lines to the deployment pyrotechnics interface connector plate and through the holes in the corners of the EVM to the lanyard pull fly-away connector and to the four separation pyrotechnic devices. The holes through the EVM skin are utilized to minimize connector interfaces between the squib control unit and explosive devices. The diameter of these holes is minimized by semi-circular cut outs in both the skin and the structure, thus allowing cable installation without pulling the connector through the hole. This harness layout is pictured in Figure 15.3-5. The total number of conductors in this system is 70.

15.3.3.4 Umbilical Interface

The umbilical connector interfaces with the Titan IIIC harnessing and provides ground power to the ATS spacecraft prior to launch. This wiring is part of the service module power harnessing (A445A1). The cable leg from this main harness is routed through a hole in the east skin to a horizontal connector plate to

which the umbilical receptacles mounted. The connector is mounted so that the mating face is downward toward the Titan. This allows vertical separation of the fly-away connector plug, attached to the adapter structure by a lanyard pull. The cable from this plug is routed down the adapter to the Titan. The umbilical fly-away connector receptacle is at the south-west corner of the spacecraft.

15.3.4 COMMUNICATIONS MODULE LAYOUT

The Communications module is the upper segment of the EVM, as shown in Figure 15.3-1. The harnessing in this module interconnects the equipment as identified in Figure 15.3-6 and the antenna feed network components which are located primarily on the underside of the module top panel. The division of this single harness into small harnesses is not required for test reasons, but the economies of construction may be realized by multiple harness design. Therefore, the RF lines are designed and built as separate cables. Additional segregation of harnesses will be made as point to point wiring is more exactly defined. The primary harness will be clamped above the equipment mounted on the north-south faces and interconnecting the heat pipe web. The total number of wires in this module is approximately 420. These interconnect the following:

A320	Translator
A340	X-Band TWTA 1 & 2
A331	Synthesizer #1
A332	Synthesizer #2
A310	S-Band Monopulse and Preamp
A310	S/L-Band Filter
A330	S-Band Power Amplifier
A330	L-Band Power Amplifier
A350	UHF Power Amplifier
A781	Time & Frequency Dispersion Electronics
A741	Radio Frequency Interference Electronics
A791	Radio Beacon Electronics
A411	Power Regular #1
A412	Power Regular #2
A419	Load Interface Circuit #2 (RFI)
A420	Load Interface Circuit #3 (T&FD)
A213	VHF Transmitter #3
A214	VHF Transmitter #4
A218	VHF Diplexer #3

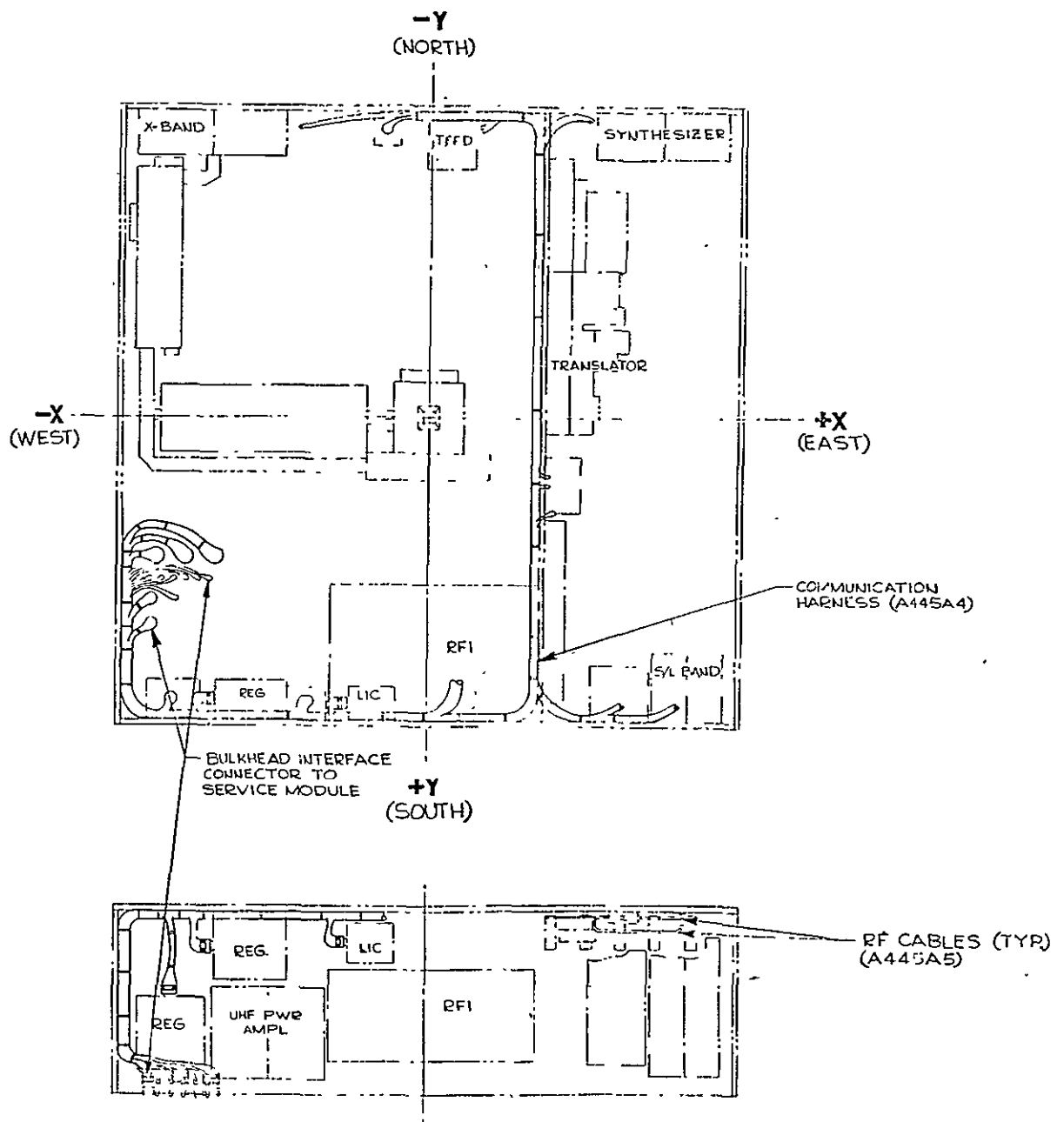


Figure 15.3-6 EVM Communication Module Harness Layout

A222 VHF Relay
A233 VHF Receiver #3

15.3.5 MODULE INTERFACE

The electrical interconnections between the three modules are made through interface bulkhead type connectors. These connectors will be circular bayonet lock connectors. The interface plates, to which the connector receptacles are mounted, are located at the upper and lower edges of the service module, (ref. Figure 15.3-1). Connect-disconnect is accomplished by reaching the connector plugs that mate at the interface plates. The quick disconnect of these two interfaces allows the three modules to be separated without the removal of harnessing.

The connectors used at the communications-service module interface include RFI filter pins. This line filtering has been incorporated to uphold the RFI shield protection obtained by the RFI screen installed between the communications and service modules.

15.3.6 EXTERNAL HARNESS LAYOUT

The ATS spacecraft incorporates harnessing exterior to the main equipment module. This external harnessing services those functions located on the reflector support truss, reflector hub, and solar array deployment arms. These three physical locations are also ideal functional divisions. The cabling to each area is constructed as an individual discrete function harness. The philosophies behind this approach are to retain the flexibility to modify harness runs, segregate subsystems, promote parallel construction, and allow test of individual functions.

The cabling to each function will be clamped to one of the reflector support truss members. This allows separation of functions where desired. Cables that go to hub equipment will pass through the central section of the hub and mate with the appropriate equipments. Cabling to the solar array and omni VHF antenna will pass through the center of the hub and extend to the deployment arms with a short service loop included to accommodate deployment arm movement.

The external cabling interfaces with the EVM at hard mounted bulkhead connectors located on the east and west faces of the module. These interfaces are made at selected points near the support truss attachment points. The interface connections maintain the separation of function groupings.

The individual exterior harnesses that interconnect various remote functions with the EVM are shown in Figure 25.3-7. All external harnesses except the VHF antenna will be constructed from Kapton Polymide Insulated wire. The interface with the EVM is made at three interface plates through which the receptacle connectors of the EVM harness are mounted. One sample of an external interface plate is shown in Figure 15.3-8.

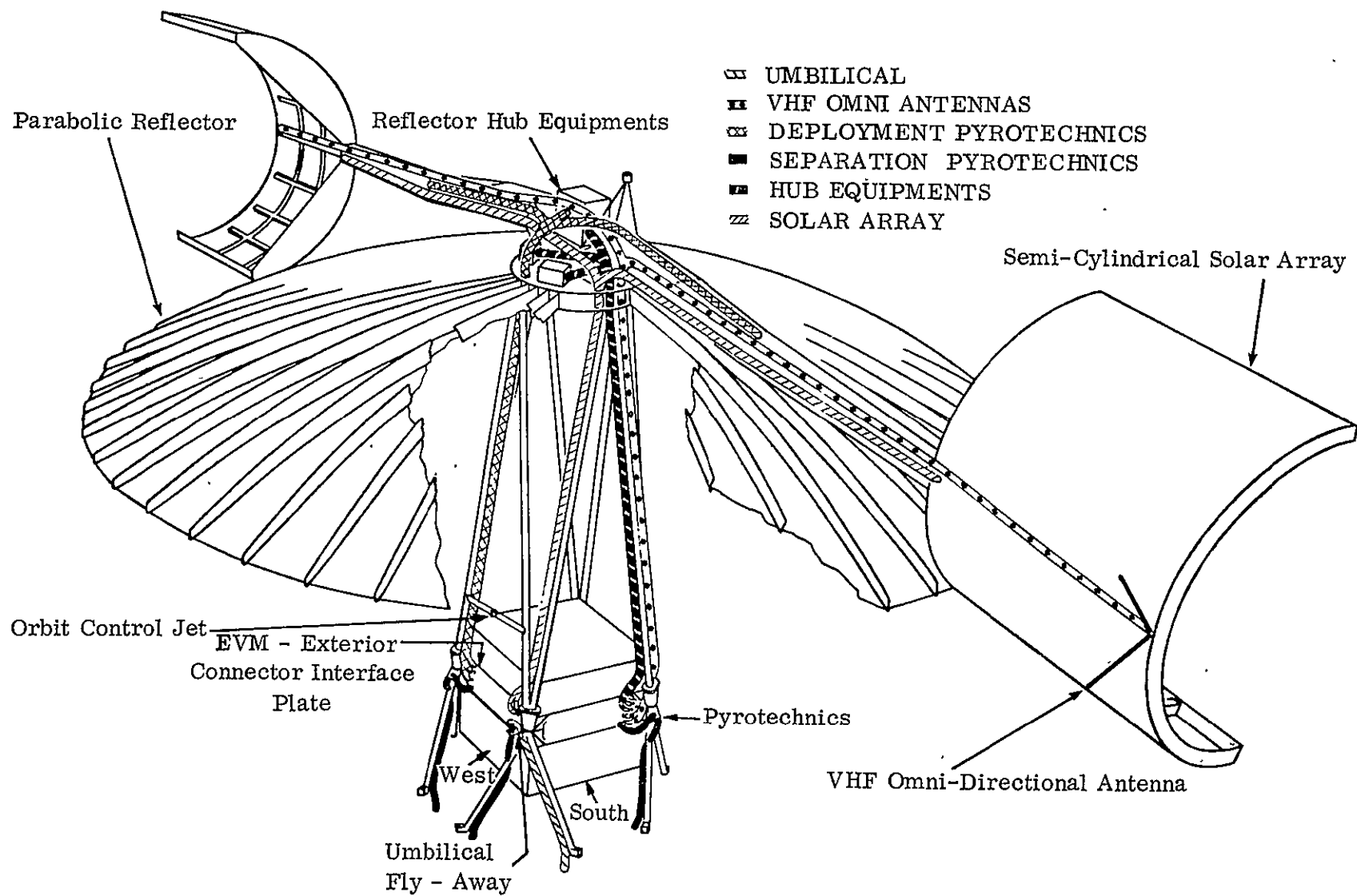


Figure 15.3-7 ATS-F Spacecraft Configuration and External Harness

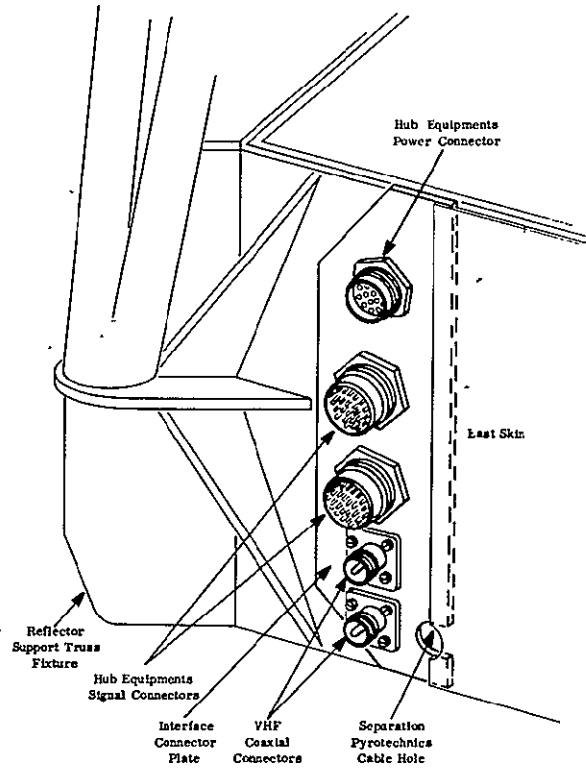


Figure 15.3-8 EVM Interface

15.3.6.1

Solar Array Power Harness

The solar array interconnection with the EVM is made through a single harness which mates with each solar panel through two connectors. The cable legs are clamped to the respective deployment arm, with appropriate service loops provided at the two moveable joints in each arm. The cable legs from each panel are combined into one bundle at the hub. This cable is routed down the south-west member of the reflector support truss and clamped to the deployment arms and truss member. This run is approximately forty feet and will consist of sixteen twisted pair conductors for power and four twisted pairs for the thermal measuring device on the arrays.

15.3.6.2

VHF OMNI Antenna Cables

The two sets of omnidirection VHF antennas are connected to the Telemetry and Command Subsystem by two coaxial cables that interface at the EVM south-east interface plate. These two cables are laced together and clamped with the hub equipments harness to the truss member. This cable run is approximately forty-five feet.

15.3.6.3

Hub Equipments Harness

The equipment mounted on the parabolic reflector hub is connected to the EVM by one harness which routes power and signal lines to the hub equipments from the south-east interface plate. At the interface plate the power and signal leads terminate in different connectors to avoid crosstalk. This cable is routed along the south-east truss member, and through the center of the hub to the respective equipments. This harness contains approximately 135 conductors and interfaces with the following equipments:

A821	EME Electronics
A832	Commandable Gravity Gradient
A143	Sun Sensor Signal Amplifier
A135	Coarse Sun Sensor (+x, -y)
A136	Coarse Sun Sensor (-x, +y)

15.3.6.4

Deployment Pyrotechnics Harness

The deployment pyrotechnics harness is routed from the north-west interface connector plate up the north-west member of the truss to the reflector and solar panel arm deployment and Commandable Gravity Gradient separation systems. This harness includes approximately 20 conductors.

15.3.6.5

Separation Pyrotechnics - Titan Interface Harness

Interfacing at each of the four separation fly-away connectors is accomplished by Titan-Spacecraft interconnecting harness. This harness routes Titan signals and power and directly ties the spacecraft backup system, routed through the

fly-away connector, to the separation devices. This harness contains approximately 30 conductors.

15.4 HARNESS COMPONENT PARTS

Major parts selection trade-off studies are contained in section VI with only the final choice being discussed here. The parts represented within these paragraphs display the implementation FHC will use on the ATS F&G spacecrafts. This discussion includes descriptions of hardware in the following categories:

- Connectors
- Wire
- Terminal functions
- Ties and clamps
- Shield terminations
- Shrinkable molded parts

15.4.1 CONNECTORS

The field of connector types chosen for the ATS program includes three families; rectangular, hi density circular, and standard MIL-C-26482 circular. The objective of this choice was to standardize the type of connector used on the spacecraft, but retain a variety sufficient for the various packaging needs and connector usage. The connectors chosen are:

- Bendix JT - Pancake Series -- new, qualified, hi-density, circular connector.
- ITT Cannon D-Subminiature Series -- a widely used, qualified, rectangular connector.
- ITT Cannon PV-Series -- the hi-rel, qualified version of the popular MIL-C-26482 connector.

Within these three families the major needs of the spacecraft have been met, yet component designers retain the freedom to choose which type is suited for their design. Exceptions to this standardization will exist on several qualified units that display different connector types, but even these for the most part will mate with the PV-Series and allow the harness to remain nearly standardized.

15.4.1.1 Bendix JT-Series

This thin profile, light weight, hi-density connector will be used extensively on the ATS spacecrafts. The primary advantages are:

- Lighter weight than MIL-C-26482
- Additional pin size for AWG 24 wire

- Hi-density pin arrangements -- up to 128 contacts in a 24 shell
- Space saving profile

The significant features of this connector are listed below:

Military Specification	-- MIL-C-38999
Locking Means	-- Bayonet
Keying	-- 5 Key/Keyway Polarization
Shell Sizes	-- 8, 10, 12, 14, 16, 18, 20, 22, 24
Pin Arrangements	-- Reference Figure 3.3-2
Pin Sizes	-- 16, 20, 22, 22M
Pin Types	-- Crimp, solder (crimp has rear insert-release)
Shell Material	-- Extruded or machined bar stock aluminum
Pin Material	-- Copper alloy
Insert Material	-- Epoxy compound reinforced with glass fibers
Shell Material	-- Iridite 14-2
Pin Finish	-- 50 or 100 micro-inches of gold

15.4.1.2

ITT Cannon D-Subminiature

This rectangular connector has been widely used on space programs. Its shape and pin arrangements make it a favorite of packaging designers. The gold finish will be utilized on the ATS F&G spacecrafts. The significant features of this connector are listed below:

Specification	-- GSFC-S-323-P-10
Locking Means	-- Screw-lock assemblies
Keying	-- Keystone cornered shell
Shell sizes	-- A, B, C, D, E, (ref. Figure 3.3-6)
Pin Arrangements	-- Ref. Figure 3.3-6
Pin Sizes	-- #20 and coaxial inserts
Pin Types	-- Solder and crimp (rear insert release)
Shell Material	-- Soft rolled brass
Pin Material	-- Copper alloy

Insert Material	-- Glass filled diallyl phthalate
Shell Finish	-- Gold plate (.0001) over copper plate (.00005)
Pin Finish	-- Gold plate (.0001) over copper plate (.00005)

15.4.1.3 ITT Cannon PV-Series

This hi-rel version of the standard MIL-C-26482 connector incorporate crimp pins with a rear-insertion-release mechanism that utilizes a plastic removal tool. This connector is available with a black anodized finish particularly suited to space usage and is the space replacement for the MIL-C-26482 connector. The significant features of this connector are listed below:

Specification	-- NAS 1599
Locking Means	-- Bayonet
Keying	-- 5 key polarization/3 point lock
Shell Sizes	-- 8, 10, 12, 14, 16, 18, 20, 22, 24
Pin Arrangements	-- Reference Figure 3.3 5
Pin Sizes	-- 16, 20
Pin Types	-- Crimp (rear release)
Shell Material	-- Aluminum alloy
Pin Material	-- Copper alloy
Insert Material	-- Thermosetting plastic & silicone elastomer
Shell Finish	-- Environmental resistant anodized finish
Pin Finish	-- Gold plate

15.4.2 WIRE

Two types of wire insulations have been chosen for the ATS F&G spacecrafts. For use within thermally controlled areas Raychem specification 44 Outer Space Cable will be used. For cabling exterior to the thermally controlled areas a Kapton Polyimide insulated wire will be used. The Kapton type wire has been chosen for the special usage to insure excellent electrical performance over a wider temperature range, through temperature cycling, and after deployment flexing; the areas where the Raychem wire is marginal. It was not used throughout the spacecraft because of manufacturing difficulties that arise with this configuration.

15.4.2.1

Raychem Corporation Specification 44 Wire and Cable

This wire is manufactured specifically for space application. The primary advantages include light weight, high radiation resistance, low outgassing, small diameter, and easy manufacturing. This wire has the following characteristics:

- Temperature rating -- -65°C to $+135^{\circ}\text{C}$
- Voltage rating -- 600 volts (rms)
- Cable Configurations -- Shielded and unshielded, single and multiconductor jacketed types
- Weight and OD for:

	<u>Weight</u>	<u>OD of wire</u>
#24 single conductor	-- 3.0 lbs/1000'	.046 inches
#24 single conductor shielded	-- 3.6 lbs/1000'	.061 inches
#20 twisted pair	-- 13.3 lbs/1000'	.133 inches
- Radiation Resistance -- 500 megarads
- Primary Insulation -- Radiation Crosslinked Extruded Polyalkene
- Jacket -- Radiation Crosslinked Extruded Polyvinylidene Fluoride
- Color -- White with identifying striping
- Shield -- Flat Braid -- 90% coverage -- tin coated copper
- Military Specification -- MIL-W-81044
- Conductor -- Tin coated copper

15.4.2.2

Kapton Polymide Insulated Wire and Cable

This insulation displays primarily the same characteristics as the Raychem wire but has a larger temperature range, performs well under scrape and flexibility testing and maintains its outgassing and radiation resistance parameters under temperature cycling. It does present manufacturing problems in stripping and soldering. The shielded configurations are difficult to strip and the silver coated conductor (required for this insulation) presents wicking of solder problems. The following characteristics pertain to this wire:

- Temperature -- -270°C to $+200^{\circ}\text{C}$
- Voltage rating -- 600 volts (rms)

- Cable Configurations -- Shielded and unshielded, single and multiconductor configurations
- Weight and OD for:

	<u>Weight</u>	<u>OD for wire</u>
#24 Single Conductor	-- 2.3 lbs/1000'	.043 inches
#24 Single Conductor Shielded	-- 6.7 lbs/1000'	.076 inches
- Radiation Resistance -- 10^{10} rads
- Primary Insulation -- Kapton Polyimide
- Jacket -- Fep Dispersion
- Color -- Gold
- Shield -- Round Braid -- 85% Coverage
Silver
- Conductor -- Silver Coated Copper
- Military Specifications - MIL-W-81381

15.4.3 TERMINAL JUNCTIONS

The need for terminal strips and junction boxes has been recognized in the areas of return line bussing and grounding systems. To accomplish these various tasks FHC proposes the use of the Deutsch Company Terminal Junctions consisting of modularized building block junctions.

These lightweight Terminal Junctions consist of a number of modules assembled in a frame and connected by crimp type insertion pins. The modules are available in a wide variety of bussing configurations and pin sizes, and may be combined in any one of a number of frame lengths. The crimp type pin contacts are available in sizes 20, 16, and 12 and are designed to be geometrically similar to the NAS 1599 contact. These pins utilize an expendable plastic tool for insertion and removal.

15.4.4 HARNESS WEIGHT ANALYSIS

The ATS Spacecraft Harness Subsystem weight analysis was conducted to provide a total weight and a weight breakdown by functional grouping (e.g. telemetry lines or power lines). The estimate for each area was established from the total length of wire allocated to that function. The total length was arrived at by multiplying the average run length by the total number of runs. To upgrade this technique external cabling and associated long runs were treated separately. Table 15.4-1 shows the harness weight breakdown and the total weight.

Table 15.4-1 ATS Harness Weight Analysis

FUNCTION CONSIDERED	WEIGHT ALLOCATED	RUN LENGTH	NUMBER of WIRES	TOTAL WIRE REQUIRED	WIRE TYPE	WEIGHT/1000 FT.	
Telemetry Monitor Lines	27.0 lbs	5 ft average	1500	7500 ft	24 AWG Single Shield Jacketed	3.6 lbs	EVM Harnessing (84.0 lbs.)
Command Lines (system)	9.0 lbs	5 ft average	900	4500 ft	24 AWG Single Conductor Jacketed	2.0 lbs	
Power Lines (system hookup)	4.5 lbs	5 ft average	70 unshielded	350 ft	20 AWG Twisted Pair Jacketed	11.2 lbs	
			10 shielded	100 ft	20 AWG Twisted Shielded Pair Jacketed	13.7 lbs	
Attitude Control Subsystem Interconnection	6.5 lbs	4 ft average	550	2200 ft	22 AWG Single Conductor Jacketed	3.0 lbs	
Telemetry and Command Subsystem Interconnection	3.0 lbs	4 ft average	150	600 ft	22 AWG Single Conductor Jacketed	3.0 lbs	
			21	100 ft	Coax	11.2 lbs	
Communications Subsystem Interconnection	9.0 lbs	5 ft average	350	1750 ft	22 AWG Single Conductor Shielded	4.6 lbs	
				200 ft	Coax	11.2 lbs	
Power Subsystem Interconnection	3.0 lbs	4 ft average	10 unshielded	225 ft	20 AWG Twisted Pair Jacketed	11.0 lbs	External Harnessing (24.0 lbs.)
			15 shielded	75 ft	20 AWG Twisted Shielded Pair Jacketed	13.7 lbs	
Pyrotechnics Lines	2.0 lbs	4 ft average	70	280 ft	20 AWG Twisted Shielded Pair Jacketed	13.7 lbs	
Solar Array Lines	7.5 lbs	40 ft average	16-power	640 ft	20 AWG Twisted Pair Jacketed	13.7 lbs	
			4-Telemetry	160 ft	21 AWG Twisted Shielded Pair Jacketed	7.4 lbs	
EME Lines	2.5 lbs	20 ft	36	700 ft	24 AWG Single Conductor Jacketed	2.0 lbs	
			8	120 ft	20 AWG Twisted Shielded Pair Jacketed	13.7 lbs	
			2	40 ft	Miniature Coax	11.0 lbs	
Commandable Gravity Gradient Lines	4.0 lbs	20 ft	40	800 ft	24 AWG Single Shielded Jacketed	3.6 lbs	
			4	80 ft	20 AWG Twisted Shielded Pair Jacketed	13.7 lbs	
Sun Sensor and Amplifier	1.0 lbs	20 ft	4	80 ft	22 AWG Twisted Shielded Pair Jacketed	10.0 lbs	Harness Connectors & Accessories
			6	100 ft	24 AWG Single Shielded Jacketed	3.6 lbs	
VHR-OMNI Antenna Lines	4.0 lbs	45 ft	2	90 ft	Coax	41.0 lbs	
Deployment Pyrotechnics	5.0 lbs	30 ft	15	450 ft	22 Twisted Shielded Pair Jacketed	10.0 lbs	
Connectors and Accessories	14.0 lbs	----	----	----	-----	---	

TOTAL Harness Weight = 102.0 lbs

SECTION XVI

RADIO BEACON EXPERIMENT

16.1 INTRODUCTION

The Radio Beacon Experiment consists of a low-power, three-frequency transmitter and an array of 1/2 V-beam antennas. The transmitter is mounted in the upper box of the EVM with the antennas mounted on the west face, as shown in Figure 16.1-1. Both the transmitter and antennas will be integrated with the communications subsystem (located in the upper box) before being married to the spacecraft.

A block diagram of the proposed transmitter is shown in Figure 16.1-2. The three basic frequencies, 40, 140, and 360 MHz are derived from a single 20 MHz crystal oscillator. Both the 40 and 140 MHz carriers are 100% AM modulated with a 1 MHz signal generated from a separate oscillator. In addition, the 40 and 360 MHz carriers are modulated with a 100 kHz signal which is derived from the 1 MHz oscillator. Using only two oscillators insures that the carriers are phase coherent as are the subcarriers.

The requirements to maintain the 40 MHz output power constant to ± 0.1 dB requires that a feedback loop be used with an AGC amplifier. Because of the amplitude envelope of the carrier, a thermistor bridge is used for measuring output power. The ± 0.1 dB requires that temperature variations in the thermistor be eliminated. A constant temperature environment is maintained through the use of an extremely light weight simple oven which has already been space qualified.

The antenna design is significantly constrained by the presence of other experiments already contained in the EVM. The Radiometer, located in the earth viewing side of the EVM prevents any protrusions below the bottom of the EVM. Likewise, any protrusions into the top surface of the EVM cause interaction with the prime focus feed. Sun and star sensors located on the sides of the EVM requires that extreme caution be used to avoid blocking these sensors. Because of these constraints, it was necessary to compromise on the Radio Beacon antenna gains and provide a slight increase in transmitter power to maintain the required ERP.

The details of the transmitter and antenna design are discussed in Section 2.

A summary of the RF performance is contained in Table 16.1-1.

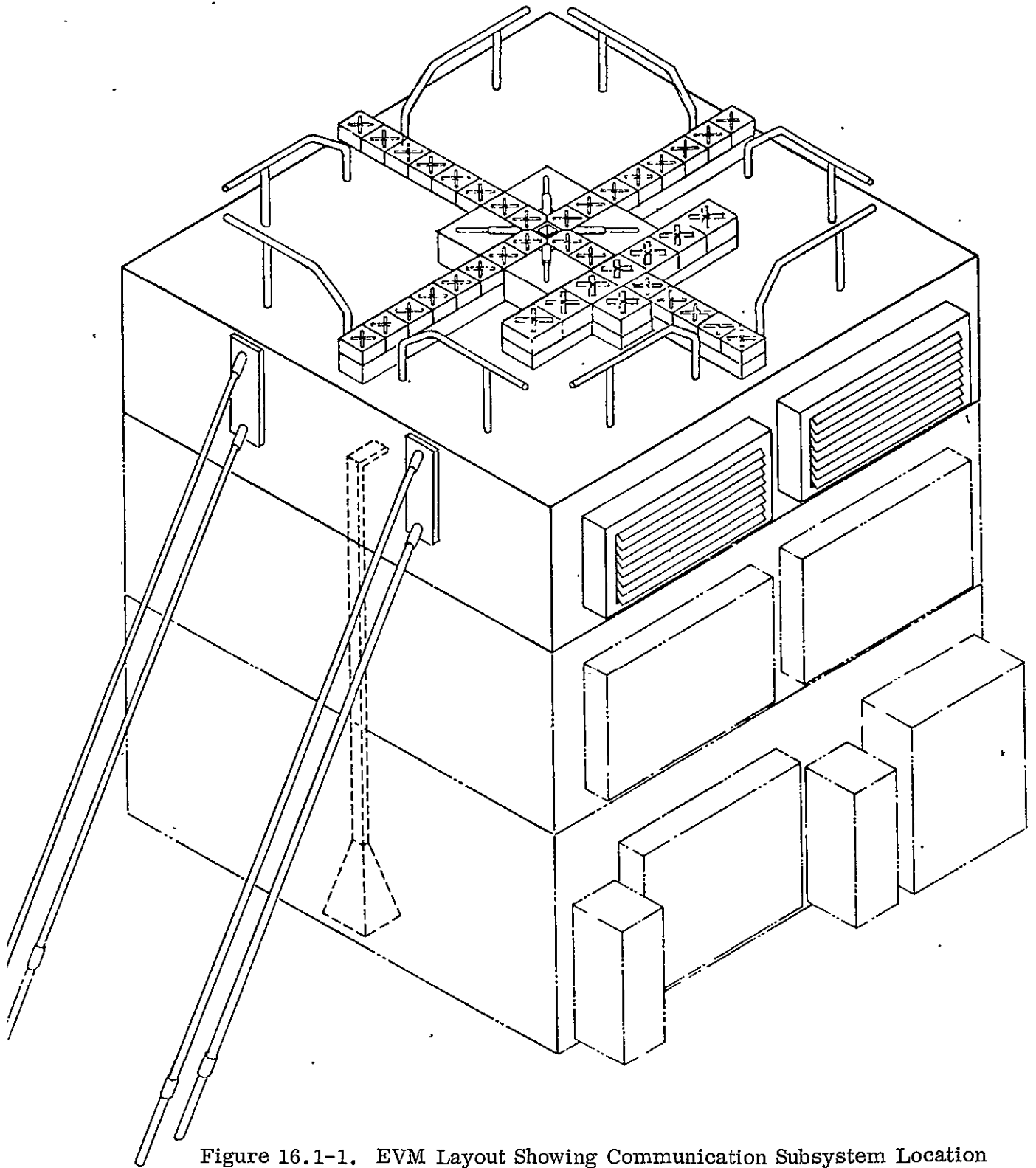


Figure 16.1-1. EVM Layout Showing Communication Subsystem Location

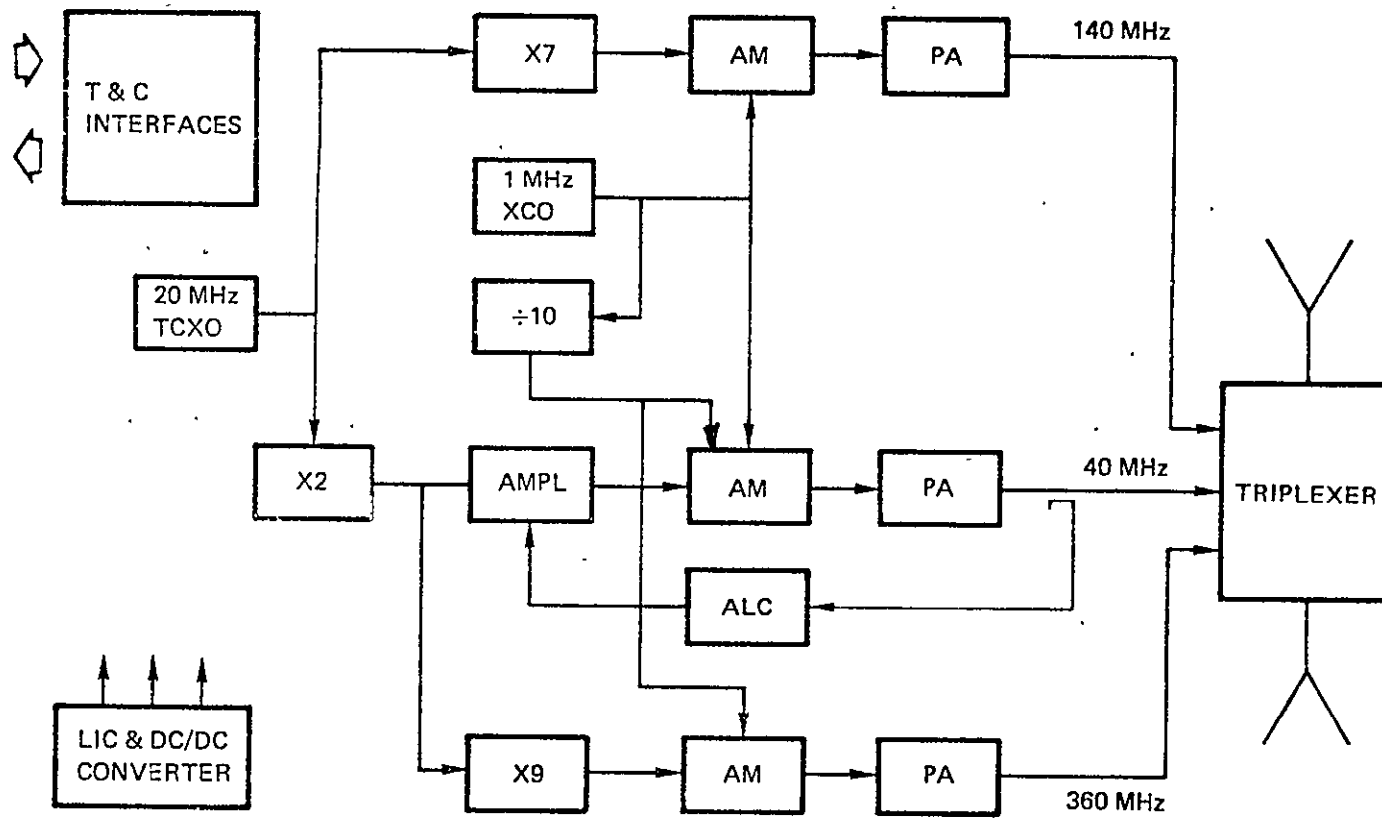


Figure 16.1-2. Radio Beacon Experiment Functional Block Diagram

Table 16.1-1. Summary of Electrical Characteristics

PARAMETER	VALUE
Nominal ERP Outputs:	
40 MHz Carrier	1600 mw
100 KHz sidebands (on 40 MHz)	200 mw
1 MHz sidebands (on 40 MHz)	200 mw
140 MHz Carrier	1600 mw
1 MHz sidebands (on 140 MHz)	800 mw
360 MHz Carrier	2000 mw
100 KHz sideband (on 360 MHz)	1000 mw
Antenna Gains, Nominal:	
At 40 MHz & S/B	2.2 dB
At 140 MHz & S/B	7.0 dB
At 360 MHz & S/B	10.5 dB
Equivalent Transmitter outputs at antenna input	
40 MHz (w S/B)	1500 mw $\pm$ 0.1 dB
140 MHz (w S/B)	540 mw $\pm$ 2 dB
360 MHz (w S/B)	300 mw $\pm$ 2 dB
1 MHz Modulation (on 40 and 140 MHz carriers) phase	
Settability	5°
Stability	$\pm 1^\circ$
100 KHz modulation (on 40 and 360 MHz carriers) phase	
Settability	5°
Stability	$\pm 1^\circ$
100 KHz Modulation Relative to 1 MHz ± 10 (on 40 MHz carrier) phase	
Settability	5°
Stability	$\pm 1^\circ$
Frequency Stability vs. Temperature	± 2 parts in 10^7
Oscillator aging (pre-aged)/2 year mission	± 2 part in 10^7
Prime Power Requirements	10.65 watts maximum
Qualification Temperature Range	-11° C to +53° C
Package Size (excluding antenna)	211 cu. in. (14" x 5.2" x 2.9")
Transmitter Package Weight (excluding antenna)	8.7 lbs.
Antenna, cable and associated hardware weight.	1.7 lbs.

16.2**FUNCTIONAL REQUIREMENTS**

The objective of the Radio Beacon Experiment is to provide scientific data related to the properties and behavior of the exosphere and ionosphere through measurement of the dispersion of radio signals in those propagation media. In particular, the following investigations are planned:

- Ionospheric electron content as a function of ionospheric storms. Rates of production loss and transport of ionization in the F region of the ionosphere. Study of the effects of eclipse and gravity.
- Exospheric electron content as a function of solar/geomagnetic activity and diurnal and seasonal changes.
- Measurement of ionospheric scintillations (amplitude and phase).
- Ionospheric absorption measurements.

In order to accomplish these measurements, the following parameters are to be measured from a geostationary radio beacon at 6.6 earth-radii emitting three plane polarized, modulated carrier signals to many earth receiving stations.

- Rotation of plane polarized waves due to interaction with earth's magnetic field and ionosphere (Faraday effect).
- Phase delay of modulated signals due to interaction with free electrons (dispersion).
- Absorption of signal strength due to electron content.

These three parameters, with their fundamental relationships, are described in Table 16.2-1.

Frequency Choice Rationale

Since Faraday rotation, modulation phase delay, and ionic absorption measurements are all inversely proportional to the square of the carrier frequency, the chosen carrier frequencies of 40/140/360 MHz provide a basis for fine/medium/coarse measurements, respectively. (The 360 MHz carrier is essentially unaffected by Faraday rotation, dispersion, ionic attenuation or refraction.)

The fine measurements will provide the required accuracy, while the coarse measurements are used to resolve ambiguity.

Table 16.2-1. Measurement Parameter Fundamental Relationships*

NO.	PARAMETERS AND DEFINITIVES
1	Faraday rotation (Ω) Ω = number of polarization rotations (360°) = (constant) (I_t/f^2) rotations
2	Modulation phase delay ($\Delta\phi$) $\Delta\phi$ = modulation phase delay caused by ionospheric dispersion = (constant) ($f_m I_t/f^2$) cycles
3	Absorption of signal strength (dB) ΔL = Absorption due to total electron content = (constant) (I/f^2) dB
where: I_t = total electron content along the signal path (electrons) I_i = ionospheric content of I_t I_e = exospheric content of $I_t = I_t - I_i$ f = carrier frequency f_m = modulation frequency (AM or narrow-band FM)	
*Refer to ITSAESSA report, "A Radio Beacon For A Geostationary Satellite", Hargreaves, Fitz, Schiffmacher, Boulder, Colorado, August 28, 1967.	

The selection of the 40 MHz transmission as the low frequency was based primarily on suitable bands existing in the radio spectrum and in keeping with the preferred frequency ratio of 1:3:9 (40/140/360 MHz).

The modulation phase delay will be directly proportional to the modulation frequency. The 100 kHz modulation on the 360 MHz carrier provides a stable reference for all modulation phase delay measurements.

The 1 MHz modulation on the 140 MHz carrier provides coarse phase delay measurements, while the 100 kHz and 1 MHz modulation on the 40 MHz carrier provide coarse and fine measurements, respectively.

Differential refraction (and hence differential path length/electron content) between the three carrier frequencies will be indicated in the measured data when the data sets deviate from the expected inverse frequency squared relationship.

16.3 RF DESIGN
16.3.1 TRANSMITTER
16.3.1.1 Detailed Description Of Operation

The detailed block diagram of the Radio Beacon Transmitter is shown in Figure 16.3.1. The heart of the transmitter is the 20 MHz crystal-controlled oscillator. The oscillator is temperature compensated with the crystal pre-aged for optimum stability. Design details and typical data for this oscillator are presented in Paragraph 16.3.1.1.2.

The 20 MHz signal is buffer-amplified and is then split by means of a miniature hybrid power divider. One portion of the drive is applied to the X7 transistor multiplier which sets up the 140 MHz output while the other part drives a X2 transistor multiplier from which two 40 MHz signals are obtained. One of these 40 MHz signals is applied to a X9 two-stage transistor multiplier which generates the 360 MHz output, while the other 40 MHz drive is already at the desired frequency and is now processed through the modulator and a two-stage power amplifier.

After the appropriate multiplications have been performed, the signals enter the three transmitter chains which are similar in design philosophy. A transistor amplitude modulator is used in each chain. For best linearity the modulators are operated at relatively low levels so that at their outputs about 0 dBm signal power levels exist.

Each power amplifier consists of two stages. The first stage is a simple common-emitter driver operating in a class A mode. The second stage, however, is a push-pull class B configuration. This approach avoids imparting any distortion to the transmitted AM and simultaneously reduces the required supply power.

Fortunately the output powers required of the transmitter are lower at the higher frequencies, which makes the design effort a little easier.

All the modulators in the transmitters are driven by signals derived from a common oscillator. This oscillator is crystal-controlled and is operating at 1 MHz. After buffer-amplification, the 1 MHz modulating signal is split into two parts in a hybrid divider with one arm driving a digital $\div 10$ divider and the other arm being split again in a miniature hybrid to provide 1 MHz modulation for the 140 MHz and 40 MHz carriers. The tunable phase shifter in the path leading to the 140 MHz modulator allows the 1 MHz modulation envelopes on 40 MHz and 140 MHz carriers to be phased relative to each other to within a few degrees and also provides a convenient spot for temperature compensating the modulation phases so that they stay fixed to within ± 1 degree relative to each other. At the output of the $\div 10$ frequency divider, the 100 kHz signal is processed through a phase shifter and is then split in a miniature power divider. One portion of 100 kHz drive is then combined with a 1 MHz signal in a power combiner prior to being applied to the 40 MHz modulator while the other part is again phase shifted and is then modulated onto the 360 MHz carrier. This phase shifter is necessary in order to phase the 100 kHz envelopes on the 40 MHz and 360 MHz carriers relative to each other.

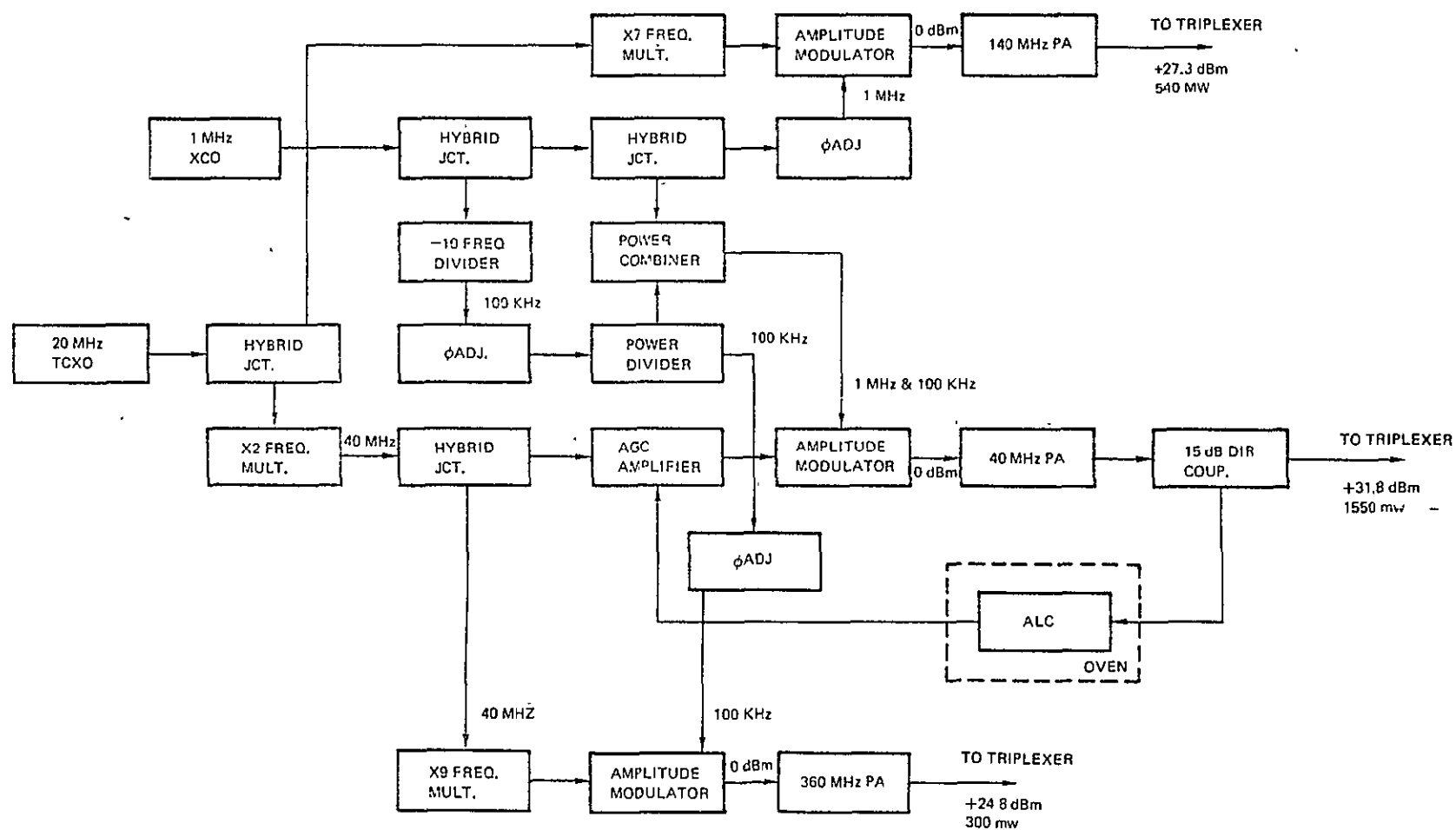


Figure 16.3-1. Radio Beacon Transmitter Detail Block Diagram

The adjustable phase shifter at the +10 divider output is used for phasing the 100 kHz and 1 MHz modulating frequencies on the 40 MHz carrier. The specifications call for these modulating signals to be settable within ± 5 degrees and stable to ± 1 degree relative to the 100 kHz envelope.

All of the power dividers and combiners used in the modulator chain are miniature hybrid types providing good isolation between ports and making phase adjustments quite independent of each other. For further discussion of modulation phase stabilities see Paragraph 16.3.1.1.1

In order to maintain the 40 MHz output power level constant to within ± 0.1 dB, it is necessary to use a high gain ALC (automatic level control) loop. The loop operates in the following manner. A 15 dB miniature directional coupler is used to provide a sample signal to drive one arm of the thermistor bridge; the other arm of the bridge sees only a fixed dc bias. A stable differential amplifier is used to amplify the minute dc variations due to the RF drive changes; the output of the amplifier controls the gain of a 40 MHz amplifier stage, thus closing the feedback control loop. Design details and test data pertaining to the ALC loop are contained in Paragraph 16.3.1.1.3.

The three outputs, thus generated, are subsequently combined in a triplexer and proceed on to the V-beam antenna, whose design is presented in Section 16.3.3

16.3.1.1.1 MODULATION PHASE STABILITY

The modulation stabilities required of the Radio Beacon transmitter are quite stringent. For convenience, the section of specifications dealing with modulation phase stability is reproduced here verbatim.

- a. 1 MHz. The 1 MHz modulation phase of the 140 MHz carrier, measured with a wide-band demodulator at the antenna terminals, shall not vary more than 2° p-to-p from the 1 MHz modulation recovered from the 40 MHz carrier by a wide-band demodulator. A constant difference shall be allowed but shall preferably be $\leq 5^\circ$ and shall be known to $\pm 1^\circ$.
- b. 100 kHz. The 100 kHz modulation phase on the 40 and 360 MHz carriers recovered with a wideband demodulator at the antenna terminals shall not show a difference of more than 2° p-to-p. The 100 kHz modulation phase on the 40 MHz carrier shall not show a variation of more than 2° p-to-p, measured at 100 kHz, when compared with the 1 MHz recovered by a wideband demodulator from the 40 MHz carrier. A constant difference shall be allowed but shall preferably be $\leq 5^\circ$ and should be known to $\pm 1^\circ$.

As can be observed, the phase comparisons of interest are of three kinds:

- a. Comparison of 1 MHz modulations
- b. Comparison of 100 kHz modulations
- c. Comparison of 1 MHz modulation to 100 kHz modulation

Furthermore, two basic requirements are placed on each: better than 5° setability and better than $\pm 1^\circ$ stability.

The modulation phase stability is a function of two main contributors:

- Group delay variations in the RF stages caused by detuning due to temperature variations.
- Any phase discrepancies that may enter between the modulation oscillator and the individual modulators.

Which of the two contributors is the dominant one depends on the frequencies involved. Thus, it should be apparent that the smaller the ratio between the carrier frequency and the modulation frequency the more aggravating the group delay variations will become. This is due to the practical limitations imposed on the amplifier bandwidth which must be maintained relatively narrow for best efficiency. Clearly the worst situation occurs in connection with the 1 MHz modulation on the 40 MHz carrier.

16.3.1.1.2 FREQUENCY STABILITY

The heart of the system is a precision, temperature compensated crystal oscillator. During the study effort, the oscillator has not been breadboarded because a unit answering these exact requirements has previously been developed at Philco-Ford for a military satellite program. This satellite program, which has been in operation since June 1966, consists of 27 communication satellites. Thus it has already been in operation for over three years, whereas the Radio Beacon is expected to be in operation for only two years. During these three years, the temperature compensated crystal oscillators in these satellites have performed most satisfactory and all performance specification have been fulfilled.

By careful choice of crystal cut and temperature compensation, temperature stability of $\pm 2 \times 10^{-7}$ was obtained over the range 5°C to $+40^{\circ}\text{C}$. In the IDCSP transponder oscillator the one year stability has been held to less than $\pm 1 \times 10^{-6}$ (by pre-aging the crystals) and the short term stability to $1 \times 10^{-10} / .25 \text{ sec}$ (typically $0.5 \times 10^{-10} / .25 \text{ sec}$). To meet the ATS F/G Radio Beacon requirements of $\pm 1 \times 10^{-6} / 2 \text{ years}$ for temperature and aging, at least 500 hours of pre-aging will be required to be sure that all the crystals will drift less than 2 parts in 10^7 per year.

Life tests on the IDCSP oscillator here at Philco-Ford have shown that without any specific selection or pre-aging of the crystals used, the frequency stability over 5 years has averaged less than 4 parts in 10^7 /year. With pre-aging at an elevated temperature, the crystal aging rate is reduced by at least one half, becoming 2 parts in $10^7/2 \text{ years}$ for the proposed oscillator. This would give a frequency stability of 4 parts in $10^7/2 \text{ years}$ for the proposed oscillator.

Studies* have shown that crystals having positive drift rates of less than 2.0 ppm (about 40 Hz) in a five-year period can be selected on the basis of information gained during the 500 hour pre-aging tests at 85°C . Offsetting the original operating frequency 2 parts in 10^7 below the assigned operating frequency allows the oscillator to drift upward toward the assigned frequency during an initial period of operation and upward past the assigned frequency during the remaining year of operation. This limits the maximum deviation from aging drift to ± 2 parts in 10^7 during the two year period. (NOTE: Well designed crystal oscillators with AT crystals will age only in a positive direction).

* "Quartz Crystal Life Test Data", by Frank Wolf and George Bistline, Internal Report of McCoy-Electronics, Mt. Holly Springs, Pennsylvania.

The master oscillator is a highly-stable temperature compensated crystal oscillator. The oscillator unit includes a buffer amplifier to reduce the effects of load variations on the oscillator. A large C/L ratio is used to minimize the effect of variations of stray capacitance. It is also necessary to closely control the mechanical layout because of stray capacitance. A series-operated third-overtone crystal is used to allow for temperature compensation over a wide temperature range. Variations of oscillator frequency with temperature have been compensated by a thermistor-diode-capacity network which senses the temperature and retunes the circuit to off-set the frequency shift.

The performance specifications of the oscillator is summarized in Table 16.3-1. Typical frequency stability with temperature for a flight unit oscillator is shown in Figure 16.3-2.

16.3.1.1.3 POWER LEVEL STABILITY

One of the more difficult requirements of the Radio Beacon Transmitter design is the need to maintain the 40 MHz output power level constant to within ± 0.1 dB at the input to the antenna. An added complication is the two-tone amplitude modulation present at the output.

Simultaneous modulation by 100 kHz and 1 MHz tones is employed. Even though each tone is modulating the output with an index of only 50%, the fact that there are two tones means that the instantaneous envelope excursions will reach an index equivalent to 100% or double the unmodulated carrier voltage envelope. The peak power developed will be 4 times the carrier power, while the total average power will be only 1.25 times the carrier power. These considerations are important when selecting the optimum detection scheme for the power tracking loop.

At least two basic methods for power detection include: a) a true power measurement utilizing a heating element, e.g., a thermistor and b) a voltage envelope detector using a diode rectifier approach. Since amplitude modulation is the preferred form of modulation (as implied in the specifications) it is felt that a true power measurement is the preferred method for the Radio Beacon for two reasons:

- a. The output accuracy is easier to maintain because it is independent of the instantaneous envelope excursions.
- b. The system has greater flexibility in that either one or both of the modulating tones can be commanded OFF with the result that the output carrier power would increase by at most a factor of 1.25, which the output power amplifiers can easily handle. On the other hand, if modulation is disabled with a diode in the feedback loop, the AGC would shift the operating point until the output power is 4 times the nominal carrier level, which could cause excessive component stress.

In the event that the Radio Beacon subcarriers are to be phase modulated, the thermistor bridge can probably be replaced with an RF peak diode detector since phase modulation provides constant envelope.

Table 16.3-1. Master Oscillator Performance Characteristics

Parameter	SPECIFICATIONS
Frequency vs Temperature (Qualification Test)	Freq = ± 1 part in 10^6 for a temperature range of -11°C to 53°C.
Frequency vs Temperature (Acceptance Test)	Freq = ± 2 parts in 10^7 for a temperature range of 5°C to 38°C.
Frequency accuracy at 19°C	± 1 part in 10^7
Power Output	0.2 mw delivered to a 50 Ω load.
Short term stability measured at 20 MHz	1 part in 10^{-10} for 0.25 sec average time.
Power Consumption at 25°C	200 mw, maximum
Supply Voltage Variation effect on Frequency at 25°C ($V_s \pm 10\%$)	Less than 3 parts in 10^7 measured at 25°C.
Size	2.43 x 1.63 x .8 or 3.17 cu in.
Weight	2.5 oz.

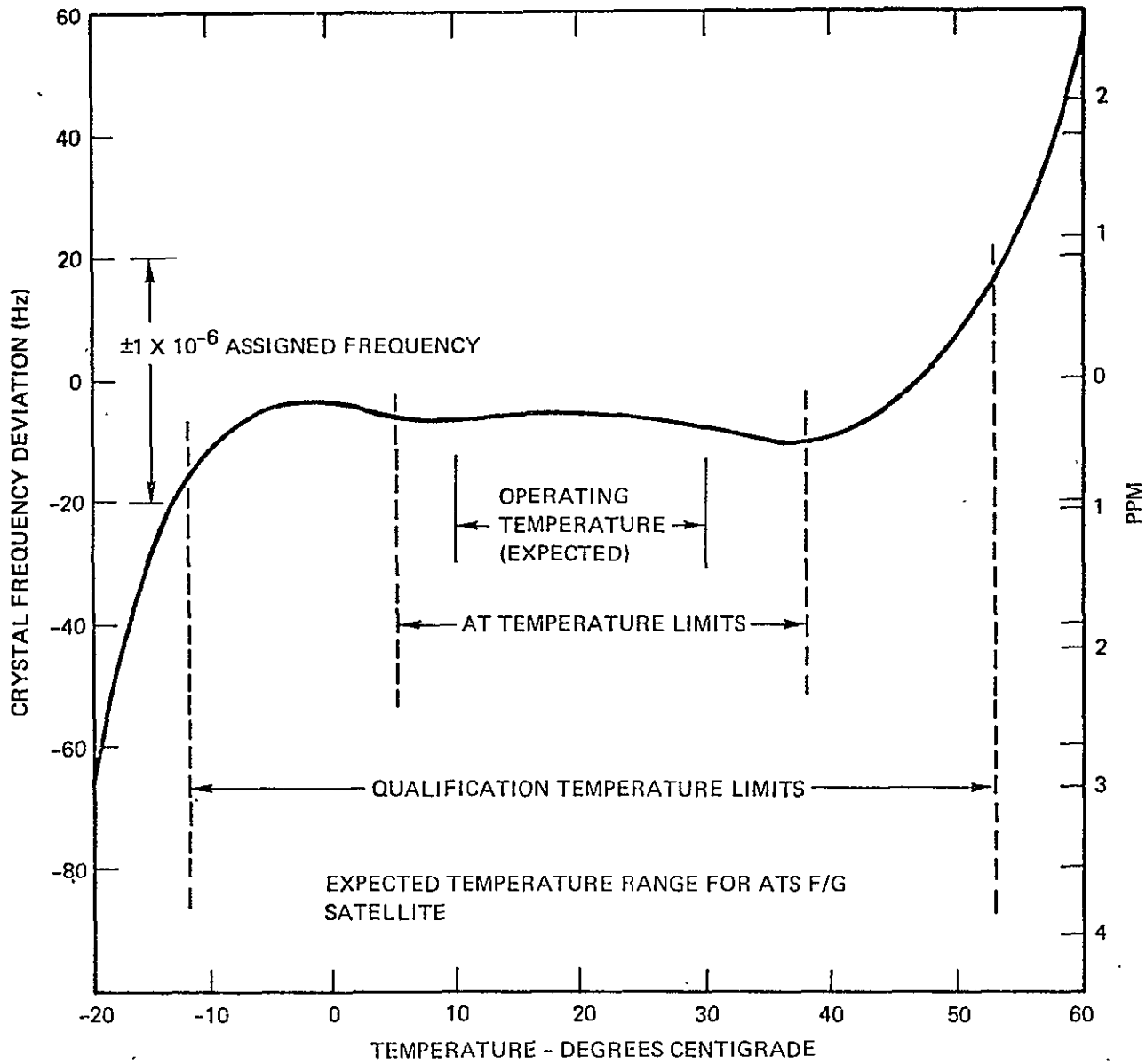


Figure 16.3-2. Frequency Deviation vs Temperature Curve

The present design, therefore, is built around a thermistor controlled bridge in the ALC feedback network. The bridge is designed in a balanced configuration utilizing two thermistors. A sample of the 40 MHz signal drives one arm of the bridge while the other arm receives a commensurately larger dc bias current to maintain both thermistors at the same operating temperature. The dc voltages developed across the thermistors are applied to a differential amplifier which controls the AGC stage in the 40-MHz amplifier, thus closing the control loop.

The thermistors are maintained in a proportionally controlled oven because the thermistor operating point is a function of three items:

- a. The dc bias current
- b. The RF power level
- c. The ambient temperature

The dc voltage across the thermistor varies as a function of changes in any of the above contributors; this value in turn changes the thermistor temperature and the bridge sensitivity, which is undesirable. The oven does not have to be very tightly controlled because the thermistors are procured in well matched pairs that track within a fraction of a percent over the temperature range of 10 to 20 centigrade degrees.

16.3.2 ANTENNA DESIGN

The design of the Radio Beacon Antenna was primarily constrained by the physical restrictions imposed by the other experiments. Protrusions into the bottom plane of the EVM are prevented by the presence of the Interferometer. Protrusions into the top plane of the EVM cause interacting with the prime focus feed. This leaves only the sides of the EVM, which are covered with sensors and thermal louvers. Because of these restrictions, compromises in antenna performance must be accepted if a relatively simple antenna is to be used.

The design proposed for the Radio Beacon Experiment is an array of four 6-foot rods forming a broadside array of two quadrature arrays of half V-beam. (See Figure 16.1-1).

At 360 MHz, the elements are 2.01 and 1.7 wavelengths long; at 40 MHz, the elements are 0.78 and 0.665 wavelengths long; and at 40 MHz, the element length becomes 0.224 and 0.190 wavelengths long. Therefore, at 360 MHz and 140 MHz, the expected performance would be that of a V-beam (the half V-beam uses the ground plane of the EVM to form the other half), while at 40 MHz, the expected performance becomes that of a dipole 0.06λ in front of a 1.32λ parabola.

The antenna consists of two half V-beam antennas spaced $\lambda/4$, one in front of the other, and fed in phase quadrature. This approach was necessary to increase gain. The included angle between the elements and the box was reduced to 35 degrees in order to eliminate beam squint at 360 MHz, and the elements were shortened to avoid extending beyond the face of the EVM. Both of these changes tend to reduce gain, but this loss is made up by the quadrature array gain factor. The termination resistor at the end of each element absorbs any reflected energy and will

be adjusted for best front-to-back ratio at 360 MHz in order to minimize interaction with the parabola.

16.3.3 TRIPLEXER DESIGN DESCRIPTION

A triplexer is used to feed the half V-beam antenna array from three transmitters. The triplexer must provide 20 dB of isolation between transmitters. The following paragraphs discuss the design of a triplexer to meet the following specifications:

- a. Isolation: 20 dB
- b. Insertion Loss: 0.46 dB max. 40 MHz
0.44 dB max. 140 MHz
0.41 dB max. 360 MHz
- c. Qualifying Temperature: -11°C to +53°C
- d. Amplitude Stability: 0.02 dB at 40 MHz

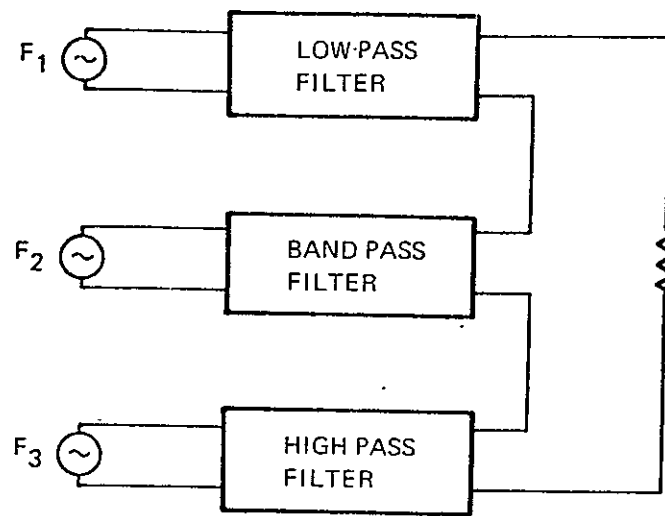
Two configurations are commonly used for this type circuit; a parallel correction or a series connection of filters as shown in Figure 16.3-3. Of these two possibilities, the parallel configuration was chosen for this application because it offers lower insertion loss. However, it has the disadvantage of not being able to use conventional filters.

Conventional filters present a very low shunt reactance to achieve rejection of the stop band frequencies. The series configuration utilizes this low shunt reactance to deliver power to the load; however, in the case of the parallel configuration, a low shunt reactance would short circuit the outputs of the transmitters. For example, the filter used to protect the 40-MHz transmitter from the 140 MHz and 360-MHz transmitters would damage the 140-MHz and the 360-MHz transmitters. Since in the series configuration, the signal must pass through at least three resonant circuits each with its loss, the parallel configuration was chosen. Filters must be designed which present a very high reactance to the stop-band frequencies.

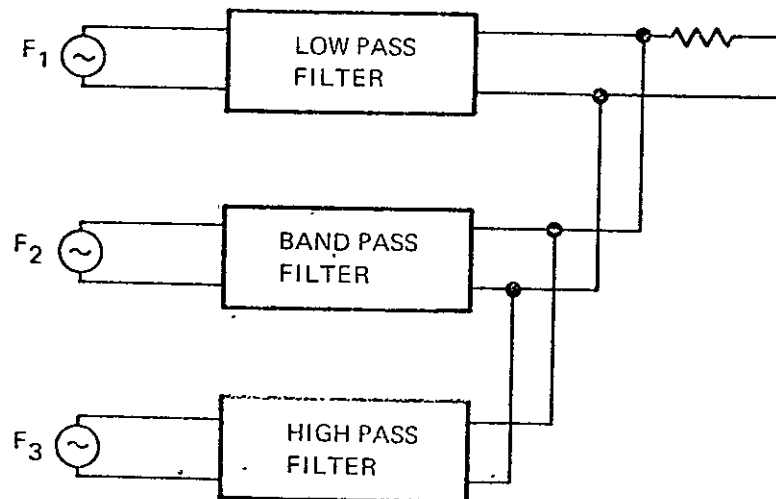
Three techniques were considered for constructing the triplexer:

1. Coaxial
2. Stripline
3. Lumped elements

The 40-MHz coaxial components are entirely too large for satellite use. Stripline, on the other hand, is very compact but useful inductance values could not be obtained for these frequencies. For example, a 0.010-inch copper strip a quarter wavelength long at the 360 MHz is equivalent to an inductance value of only 0.05 microhenry. This leaves lumped element components as the most feasible design approach.



a) SERIES CONNECTED TRIPLEXER



b) PARALLEL CONNECTED TRIPLEXER

Figure 16.3-3. Series and Parallel Triplexer Configuration

The 20 dB of isolation between transmitters can be achieved with simple series resonant circuits. Because the circuit element values will change with temperature, the filters must have a broad enough passband to prevent the passband from drifting off the signal frequency. Ten percent bandwidth will provide enough bandwidth for well designed components. Negative temperature capacitors can also be incorporated to help maintain amplitude stability.

16.4 PHYSICAL DESCRIPTION

16.4.1 LAYOUT

The Radio Beacon will be mounted in the upper box of the EVM along with the communications transponder. The location of the Radio Beacon is shown in Figure 16.4-1. Because of its relatively low power requirements, it is mounted on the inner wall of the EVM. The antennas are mounted on the west side of the EVM and are connected to the transponder by equal length RF71 double shield coaxial cable.

16.4.2 COMPONENT AND ASSEMBLIES

16.4.2.1 Transmitter Design

The mechanical design of the ATS Radio Beacon Experiment has been analyzed in order to arrive at a proposed design which will be reliable, producible, economic, and consistent with the contract requirements with respect to weight, size, environmental maintainability, shielding and electrical performance. A sketch of the proposed design is shown in Figure 16.4.2.

The design techniques used include printed circuit modules, potted magnetics (coils, transformers and chokes) modules, metal frame modules, connector modules and strip lines. Economy of design has been achieved with this modular design approach. Each module represents a functional portion of the total circuit and may be tested, operated, or replaced separately.

During the design phase of the program, careful attention is given to the layout of each module. The circuit designer and the product design engineer work closely with the designer to assure that the physical configuration of the equipment will enhance electrical performance. As a general practice, all lead lengths and interconnections are kept to a minimum, coils are located so as to minimize mutual inductance, and the capacitance effects of walls and covers are considered.

16.4.2.2 ANTENNA DESIGN

The radio beacon antenna system consists of two arrays of two elements each. These elements are mounted on the -X (west) wall of the communications section of the EVM. The arrays of elements are equally spaced from the X axis centerline and are 22.5 inches apart. The elements in the array are 8.05 inches apart and intersect the EVM wall at a 35° angle. One of the elements is 66 inches long and the other is 56 inches long, and do not project beyond the interferometer surface of the EVM. Figure 16.1-1 illustrates this arrangement.

The antennas are in the field of view of the yaw coarse sun sensors. In order to bring the shading, created by the elements, to acceptable limits, the elements are fabricated from .072 stainless steel rod. The elements are terminated in aluminum fittings which are mounted on dielectric blocks which are secured to the EVM wall. A stripline interconnects the long and short elements of an array and electrical connection to the transmitter cables is accomplished with the TNC connectors.

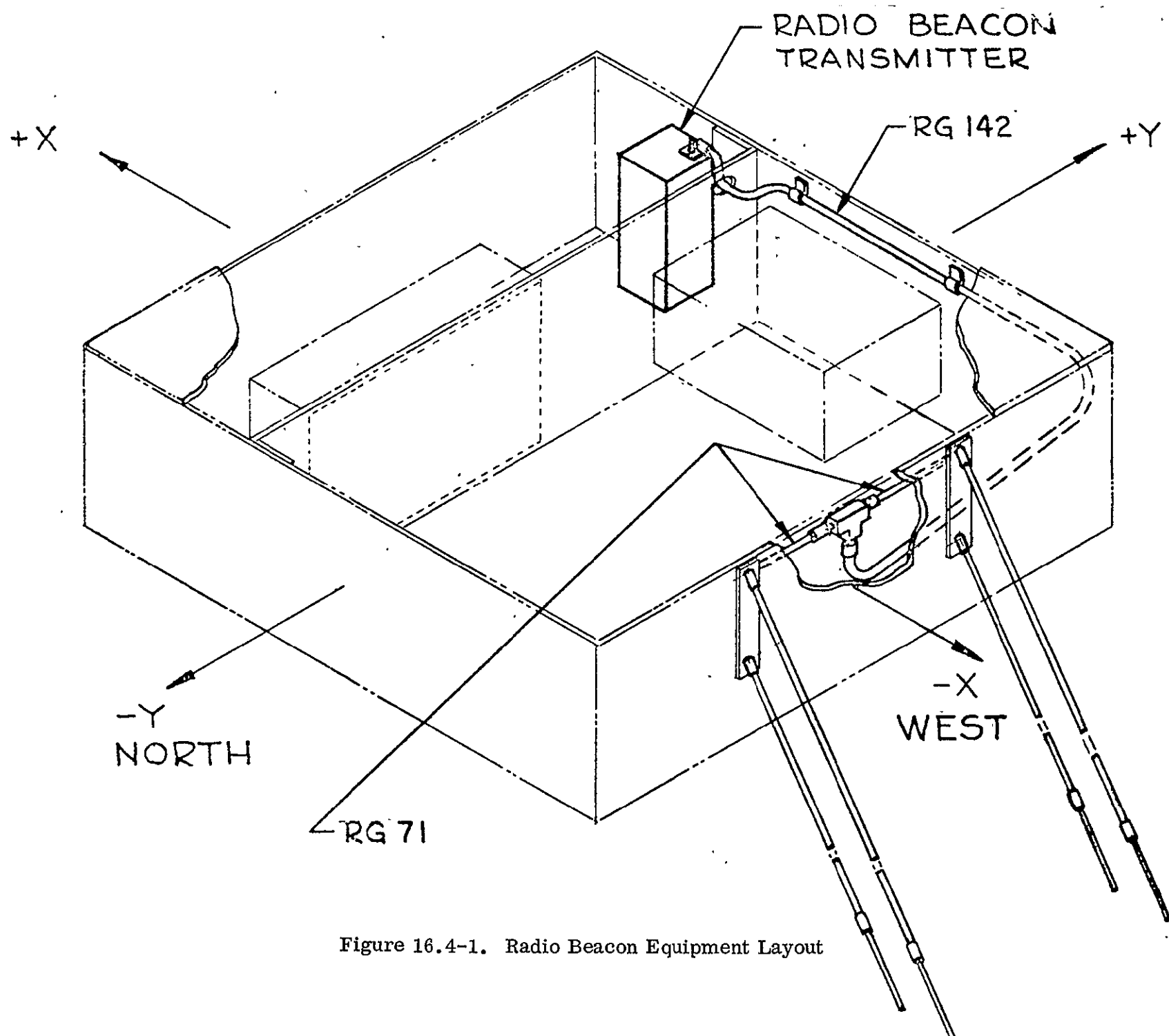


Figure 16.4-1. Radio Beacon Equipment Layout

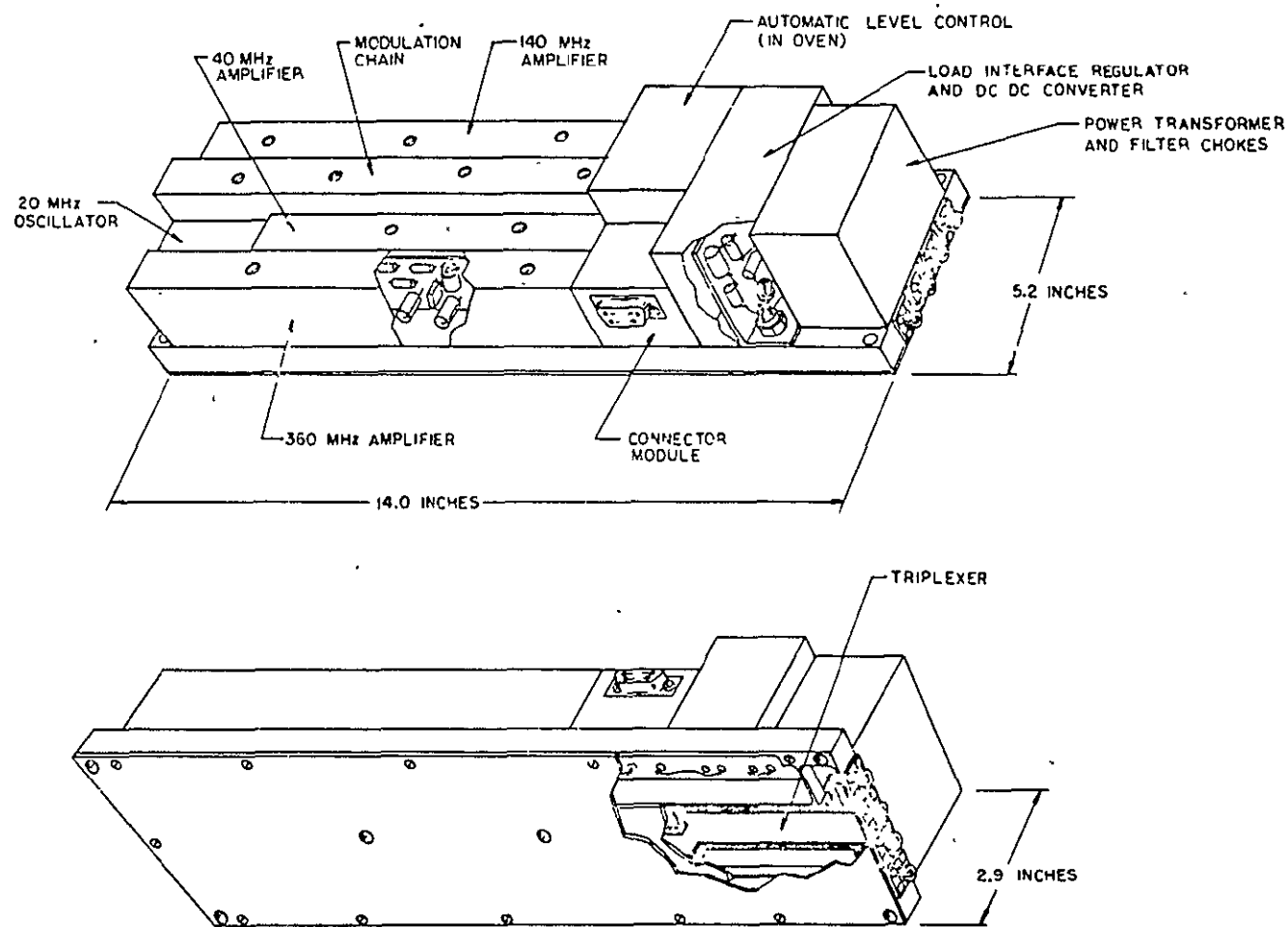


Figure 16.4-2. Transmitter Layout

In the operational position, the antennas extend beyond the fairing dynamic envelope. To stow the antennas for launch, the antenna tips are inserted in loose, capped, teflon lined tubes. The antennas are bent into position and slide out of the retaining tubes during separation of the spacecraft from the Titan III transtage.

16.4.3 WEIGHT ALLOCATION

The Radio Beacon as proposed will have a maximum weight of 10.5 lb. The weight of the individual module, chassis and associated parts are tabulated in Table 16.4.1. The weight shown are engineering estimates which are based on previous similar designs.

16.4.4 POWER ALLOCATION

The Radio Beacon's DC power consumption is 10.0 W under normal operation. The power requirements for the transmitter in detail are tabulated in Table 16.4-2.

Table 16.4-1. ATS Beacon Weight Table

COMPONENT	WEIGHT (LB)
Automatic Level Control (Oven)	0.25
20 MHz Oscillator	0.08
360 MHz Amplifier	0.82
40 MHz Amplifier	0.75
Modulation Chain	0.94
140 MHz Amplifier	0.82
Connector Module	0.38
Power Supply & Regulator	1.18
Chassis	1.26
Cover	0.66
Triplexer	0.40
Wire	0.50
Hardware	0.60
Connectors	0.08
Coaxial Cables & Fittings	0.5
Antenna Elements	0.8
Attachment Hardware	0.4
Total for Radio Beacon	10.42 lb

Table 16.4-2. DC Power Requirements

COMPONENT	DC DRAIN
360-MHz Power Amplifier and Driver	720 mW
140-MHz Power Amplifier and Driver	1235 mW
40-MHz Power Amplifier and Driver	3100 mW
ALC { Bridge Oven dc Amplifier AGC Amplifier	1050 mW
Modulators (3 ea.)	200 mW
Multipliers (x2, x7, x9)	520 mW
÷10 Divider	275 mW
Oscillators (1 MHz & 20 MHz)	300 mW
Command Circuit Requirements	50 mW
TLM Signal Conditioner Circuit Requirements	280 mW
SUBTOTAL	7.730 W
DC/DC Converter ($\eta = 85\%$)	1.37 W
Load Interface Circuit	.812 W
Total Power	9.912 $\approx$ 10.0 W
COMMAND CIRCUIT "ON" "OFF" SURGE POWER	400 mW/50 msec

SECTION XVII

GROUND SUPPORT EQUIPMENT

17.1 INTRODUCTION AND SUMMARY

The test equipment required to conduct the ATS F&G Integrated Test Program is divided into two major classifications, Aerospace Ground Equipment (AGE) and Bench Test Equipment (BTE). AGE is defined as that equipment which is used to conduct tests at the subsystem and spacecraft system test levels. BTE is defined as that equipment which is used to conduct tests below the subsystem level, i.e., component, assembly, and subassembly levels. The AGE and BTE may be further divided into two major categories, electrical AGE/BTE and mechanical AGE/BTE. Paragraph 17.2.1 presents a description of the electrical AGE/BTE necessary to conduct the Integrated Test Program. Mechanical AGE is presented in Paragraph 17.2.2.

17.1.1 DESIGN GUIDELINE

The basic guidelines used in establishing design concepts for the AGE/BTE subsystem are summarized as follows:

- Provide the minimum amount of equipment in terms of cost and complexity consistent with adequately supporting the Integrated Test Program.
- Utilize "building block" concept to greatest extent possible, whereby, a BTE test set is essentially identical to a module/panel in the subsystem AGE, and a subsystem AGE panel/console is essentially identical to a rack in the spacecraft system test complex (STC).
- ATS "peculiar" BTE will be limited to the greatest extent feasible by maximum utilization of capital commercial equipment and test harnesses, except where test/schedule/safety considerations dictate otherwise.
- All test equipment will be designed and built to factory test equipment (FTE) drawings and specifications (best commercial practices). Material and parts will be of good commercial quality, of the lightest practical weight, cost effective, suitable for the purpose, readily available and offer reasonable reliability.

All AGE/BTE will be designed and built in accordance with the provisions specified in FHC-685-GR-0000 General Specification for Ground Support Equipment ATS F&G.

Design and fabrication of the AGE/BTE will be primarily performed by the initial user of the equipment. Honeywell, Inc. will be responsible for the Attitude Control Subsystem (ACS), IBM for the Telemetry and Command (T&C) and Interferometer, Philco-Ford for the Communications and Fairchild Hiller for the Power Subsystem (PS) and Spacecraft AGE/BTE.

17.2 SPACECRAFT AGE

17.2.1 SPACECRAFT ELECTRICAL AGE SYSTEM TEST COMPLEX

Spacecraft level electrical AGE consists of the equipment required to monitor and control the prototype and flight spacecraft systems during the Integrated Test Program. This equipment is assembled and integrated in order to establish a system test complex (STC) (Figure 17.2-1) which serves as the focal point and single control source for the conduct of all spacecraft level tests.

Mechanical design of the STC is based on a modular concept. Each module is an independently transportable unit mechanically interconnected to the other modules of the STC by a common floor. Electrical interconnections are made through cables in raceways contained in the floor. Each module also contains a patch panel which will provides access to any of the recording and/or readout equipment contained in the STC. With this approach, redundant equipment is minimized.

Electrical design of the STC is based on an end-to-end spacecraft system test concept. Spacecraft test input and control will be through the command system, communications system, and stimulation of the various on-board sensors. Functional verification of the spacecraft operation will be through telemetry readout and/or down link RF transmissions. Telemetry readout will be accomplished through selected word display, analog recording, printout of selected subcoms and/or the entire bit stream. RF transmissions will be received, demodulated, and the data displayed on an oscilloscope, X-Y recorder, stripchart recorder, or magnetic tape as required.

The Telemetry and Command portion of the STC will provide the major portion of the spacecraft control and monitoring. It will contain a command subsystem in both open loop and hardline transmission modes. Telemetry transmissions from the spacecraft will be received at the STC, demodulated, and the demodulated signals routed to a computer for decommutation, processing, and distribution to the desired display and/or recording device.

Testing of the ACS at the spacecraft level will consist of verifying the functional performance of operational modes within the practical constraints of the test environment. Mode simulation will be provided through stimulation of the various on-board sensors, direct comparison of the DOC functions against the STC computer and through the command link. The ACS portion of the STC will contain sun stimulators, a polaris simulator, an earth simulator, and a computer.

Table 17.1-1. Summary Listing Of Electrical AGE And BTE

<u>Test Equipment</u>	<u>AGE/BTE</u>	<u>Supplier</u>
System Test Complex	AGE	FHC
ACS Test Station	AGE	HI
IRA Test Station	BTE	HI
IRA Card Test Station	BTE	HI
DOC Test Station	BTE	HI
Memory Test Station	BTE	HI
Subassembly Test Station	BTE	HI
ABC Test Station	BTE	HI
Actuator Control Test Station	BTE	HI
Inertia Wheel Test Station	BTE	HI
Earth Sensor Test Station	BTE	HI
Star Tracker Test Station	BTE	HI
Sun Sensor Test Station	BTE	HI
Sun Sensor Electronics	BTE	HI
Test Station		
Cable Harness Test Station	BTE	HI
Auxiliary Propulsion Test	BTE	HI
Station		
Gravity Gradient Test Station	BTE	HI
Trust Measurement Test	BTE	HI
Station		
T&C Sybsystem	AGE	IBM
DACU Test Set	BTE	IBM
CDD Test Set	BTE	IBM
Spacecraft Clock Test Set	BTE	IBM
DSU	BTE	IBM
Transmitter Test Set	BTE	IBM
Receiver Test Set	BTE	IBM
Low Pass Filter Test Set	BTE	IBM
Diplexer Test Set	BTE	IBM
Antenna Test Set	BTE	IBM
Interferometer Test Set	BTE	IBM
Communication Test Set	AGE	P/F
Prime Focus Feed Test Set	BTE	P/F
Test Parabola and Positioner	BTE	P/F
Standard Gain Horns	BTE	P/F
Antenna Coupler	AGE	P/F
Power Subsystem		
SMR Test Set	BTE	FHC
Battery Charger Test Set	BTE	FHC
LIC Test Set	BTE	FHC
Battery Test Set	BTE	FHC
Solar Module Test Set	BTE	FHC

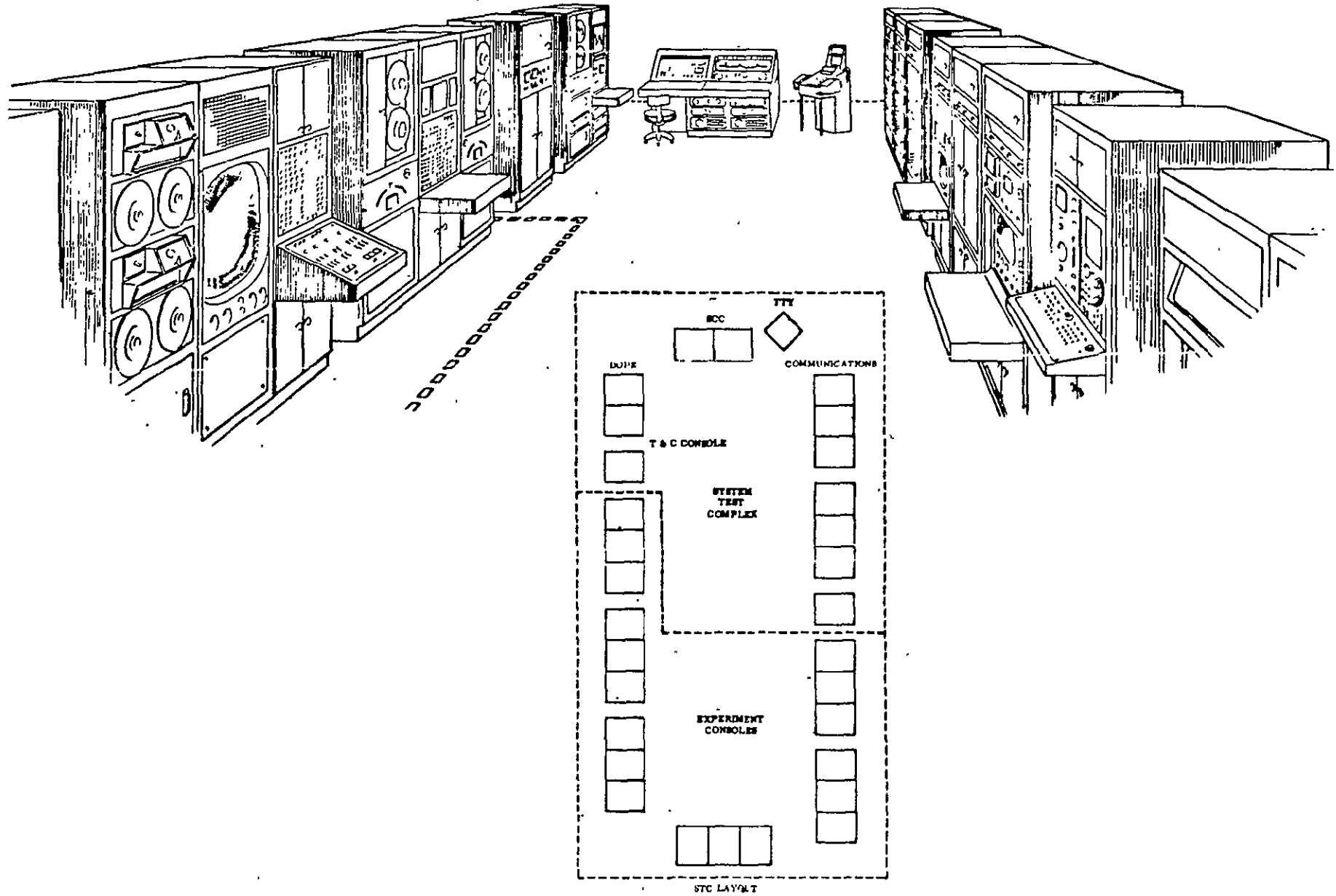


Figure 17.2-1. System Test Complex (Sheet 1 of 2)

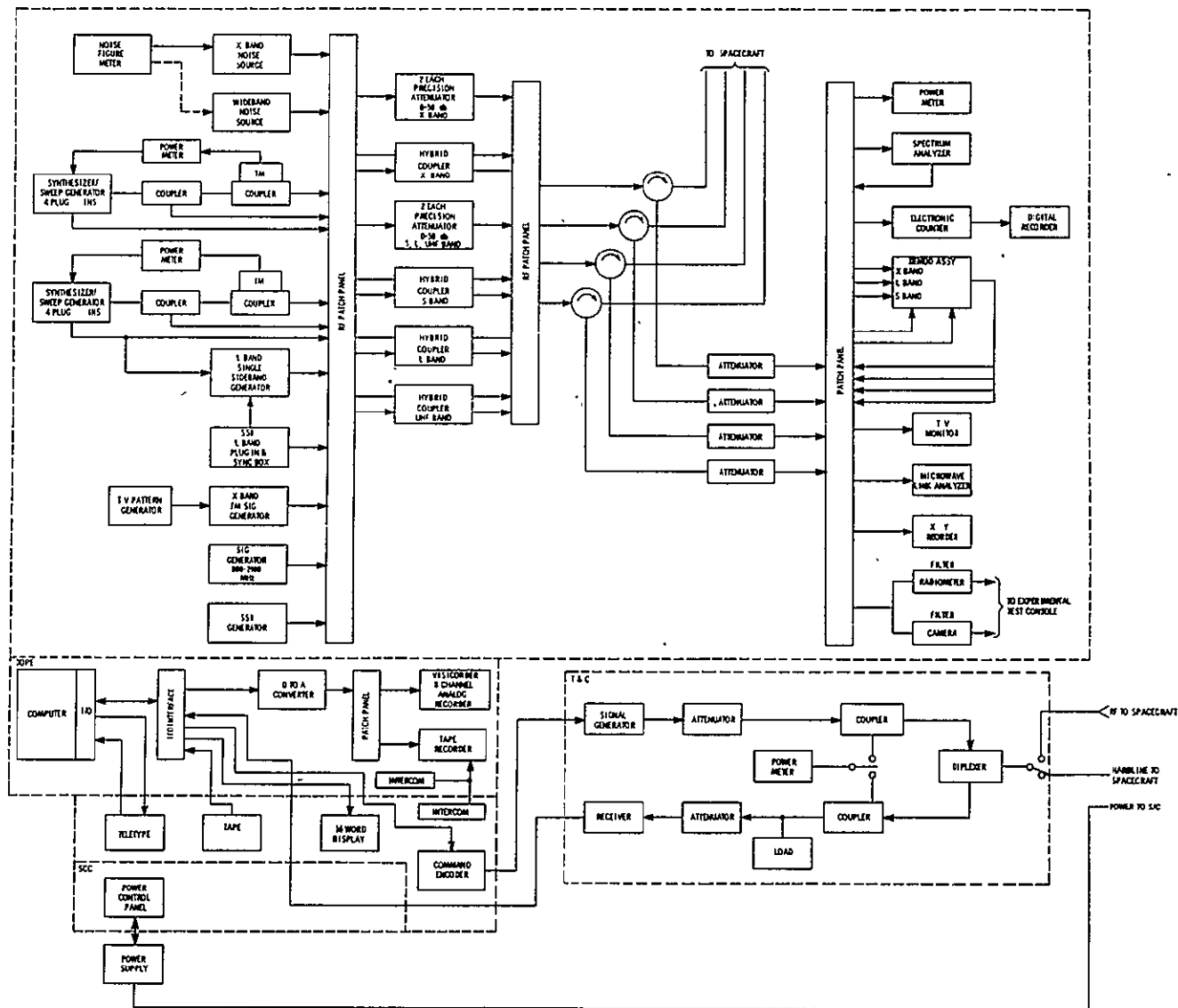


Figure 17.2-1. System Test Complex (Sheet 2 of 2)

The Power Subsystem portion of the STC will consist of a power supply and a power control panel for On-Off Control and Monitoring. The console will have the capability of remote operation when physically removed from the rest of the STC. This feature eliminates the requirement for long power cable runs during environmental testing and launch operations. Spacecraft power will be provided through the spacecraft umbilical connector and solar panel power interface.

All tests of the communications subsystem will be RF closed loop with the up-link signal originating from the STC, transmitted to the spacecraft through an antenna coupler and processed by the communications subsystem. Verification of performance will be via telemetry readout and/or analysis of down-link data transmission from the spacecraft via the antenna coupler to the STC.

The baseline spacecraft functional test requirements were established and the STC/spacecraft hardware relationships to their elements defined. Listed below are the STC functional areas and a summary description:

- Test Control

Spacecraft test control will be accomplished through the spacecraft Telemetry and Command Subsystem at the STC Spacecraft Control Console (SCC).

- Data Acquisition and Processing

Spacecraft telemetry will provide the primary source of spacecraft test data. The STC automatic data acquisition and processing system (Telemetry Processor) will receive, process, and distribute this data to the desired readout and/or recording device. All command, telemetry and intercom voice signals are automatically recorded on magnetic tape during all testing.

- Test Interface

The STC interfaces with the spacecraft through the telemetry and command subsystem, communications subsystem and power subsystem. The following equipment provides the STC/spacecraft interface:

- (a) Telemetry and Command Console

- (b) Communication Consoles

- (c) Power Console

- Specialized Test Functions

Specialized test functions consist of the various simulators required for sensor stimulation, experiment AGE, specialized cabling and test aids.

The complete STC consists of the following consoles:

- Spacecraft Control Console
- Data Operational Processing Equipment
- Telemetry and Command Console
- Communication Consoles (7)

Descriptions of the major consoles which comprise the STC are presented in the following paragraphs.

17.2.1.1 Spacecraft Control Console

The Spacecraft Control Console (SCC) will be the control center during all spacecraft tests. The following equipment is contained within this console:

- Command Encoder
- Teletypewriter
- Tape Reader
- Display Panel
- Intercom
- Functional

17.2.1.2 Data Operational Processing Equipment

Telemetry data is processed and controlled by the Data Operational Processing Equipment (DOPE). This equipment is mounted in a double bay console containing the following major items:

- Computer
- Telemetry Interface Panel
- D to A Converter
- Strip Chart Recorder
- Tape Recorder

17.2.1.2.1 Special Purpose System Versus General Purpose Computer

A study of whether the STC should include a special purpose or general purpose capability for data acquisition and processing was conducted with cost and system effectiveness as primary criterion. FHC has selected a general purpose capability as best suited to the need of the ATS F&G test program for the following reasons:

- (a) Increased flexibility when compared to special purpose systems
- (b) Affords semi-automatic operation, thereby minimizing judgment decisions and manual test sequencing, thus reducing the number of personnel required to conduct a given test

- (c) Human error is less likely to occur since fewer operator actions are required to set up, initiate, conduct, and evaluate a given test
- (d) Test time will be reduced, thereby resulting in schedule and personnel man hour savings
- (e) A single general purpose processor arrangement will accommodate many test setups by a simple change of software
- (f) The general purpose capability can be used for STC self test and validation prior to test
- (g) Selection of the proper computer hardware will allow an analog to digital tape conversion capability to be included at no additional cost
- (h) Centralized test control will be easier to implement
- (i) Reliability of off-the-shelf machines is excellent
- (j) Readily responsive to changing test requirements by software

17.2.1.3

T&C Console

The T&C console will provide communications between SSC and the ATS spacecraft, it will also be used to check out the T&C subsystem during the system test.

The T&C console will be packaged into a nineteen inch equipment rack. Individual RF components, such as the RF power meter, RF couplers, RF dummy loads, pick-up antennas, terminators, and attenuators will be mounted into a special chassis and front panel called the RF test panel. Connectors will be provided for interface connections with other items of AGE which are part of the system test complex. The components contained in the console are as follows:

(a)	Synchronizer	HP 8708A	1
(b)	Frequency Counter	HP 52452/5255A	1
(c)	RF Signal Generator	HP 608F	1
(d)	RF Tuner/Receiver/ Demodulator	Vitro RFT 100A/1037G/101G	2
(e)	Interconnection Cables (Internal)	IBM	
(f)	RF Test Panel	IBM	
	1. 50Ω Terminator	To be determined	1
	2. Attenuators	Texsan, Inc., RA-54	3
	3. RF Power Meter	Unknown	1

4.	Dummy Loads .	Bird 8053	2
5.	RF Couplers	To be determined	3

17.2.1.4 Communications Consoles

The communications consoles will house the simulators, detectors, processing and recording equipment necessary to test the communications subsystem in all modes of operation. The equipments will be items of subsystem AGE integrated with commercial and special test equipments in a flexible arrangement peculiarly suited to system test requirements. Special test equipment is employed only when commercial equipment is not available for a specific requirement.

The equipment is mounted in seven (7) bays of equipment racks with 70 inches of panel space each. The largest assembly will not exceed three bays for ease of handling and shipping. All commercial and unique equipments will be rack-mounted and connected by the necessary coaxial cabling, waveguide, and multiconductor cables. External coax cables and waveguide will be provided for connections between the antenna couplers and the Communications Test Set. External cables will also be provided for connection between the interface panel and the transponder.

17.2.1.5 Power Module

Spacecraft power for ground test will be supplied from the Power Control Module. The Power Control Module contains a DC power supply capable of delivering 15 amps at 100 V. Output of the console will be passed through blocking diodes to prevent reverse polarity power from reaching the spacecraft. Output circuits breakers will be provided for protection against overload. Control of the power will be through the Power Monitor Panel. The Power Monitor Panel will contain the master power switch for the spacecraft power supply the input voltage control, current and voltage monitor meters. This panel will be located in the Spacecraft Control Console such that it is always in easy reach of the test conductors.

17.2.2 MECHANICAL AGE

17.2.2.1 Introduction

The mechanical AGE includes all items of mechanical equipment required throughout the program for handling, testing, and transporting the ATS spacecraft and its components, subsystems, and systems. This section describes the mechanical AGE necessary to conduct the Integrated Test Program.

17.2.2.2 Design Guidelines

The base guidelines used in establishing design concepts for the mechanical AGE are summarized as follows:

- Provide the minimum amount of equipment in terms of cost and design simplicity consistent with adequately supporting the test program.

Table 17.2-2. Summary of Mechanical AGE

<u>Description</u>	<u>Tool Number</u>	<u>Supplier</u>
S/C Access Platform	ATST 1019	FHC
S/C Transport Container	ATST 1072	FHC
Reflector Transport Container		LSMC
Subsystem/Component Transport Container		P-F, IBM, Honeywell
Solar Array Transport Dolly	ATST 1005	FHC
S/C - EVM Dolly	ATST 1002	FHC
Hub Dolly	ATST 1007	FHC
Reflector Dolly	ATST 1006	LSMC
S/C Lifting Slings	ATST 1979	FHC
Reflector Lift Sling		LSMC
EVM/Hub Magnetic Test Fixture	ATST 1058	FHC
EVM Hub Magnetic Test Dolly	ATST 1078	FHC
Solar Boom Deployment Fixture	ATST 1057	FHC
S/C Separation Test Fixture	ATST 1086	FHC
S/C Vibration Fixtures	ATST 1083	FHC
	ATST 1063	FHC
	ATST 1064	FHC
S/C Mass Properties Fixture	ATST 1062	FHC
Hub Alignment Fixture	ATST 1053	FHC
Hub Alignment Spider	ATST 1067	FHC
Reflector Alignment Spider		LSMC
Component/Subsystem Vibration Fixture		P-F, IBM, Honeywell
Holding/Support Fixture For Reflector		LSMC

Table 17.2-2. Summary of Mechanical AGE (continued)

<u>Description</u>	<u>Tool Number</u>	<u>Supplier</u>
Reflector Torque Measurement Fixture		LSMC
Precision Turntable Alignment	ATST 1082	FHC
Theodelite & Misc. Optics	ATST 1012	FHC
Vertical Tooling Bar	ATST 1012	FHC
EVM Blower Cooling Cart	ATST 1068	FHC
Louver Protection Covers	ATST 1069	FHC
Solar Array Protection Covers		FHC
Sensor Protective Covers		P-F, IBM, Honeywell
Louver Calibration Fixtures	ATST 1052	FHC
Thruster Nozzel Protection Covers		Honeywell
Whiffle Tree Tension Fixture	ATST 1055	FHC
Contour Measurement Fixture		LSMC
Reflector Support Fixture		LSMC

- Utilization of previous techniques where commonality can be demonstrated between ATS F&G requirements and other previous or current NASA programs.
- Provide reliable systems from a maintenance and safety standpoint to both spacecraft and personnel.

A summary listing of mechanical AGE is presented in Table 8.2.2. A description of the major pieces of mechanical AGE is presented in the following paragraphs. The design and fabrication of the mechanical AGE is to be accomplished by the initial user of the equipment. Delivery of FHC is dependent on the specific need for the mechanical AGE during the conduct of tests at FHC. For example, a reflector torque measurement fixture will be used at LSMC during deployment testing at the subsystem level. This fixture will remain at LSMC during the Phase D effort and is not required for use at FHC for reflector deployment since deployment is accomplished at the spacecraft level only.

17.2.2.2 Transportation of the S/C to the launch site will be in a sealed container mounted on a low-bed trailer. The container-trailer configuration will provide sufficient protection from the shipping environment to enable the fully-assembled spacecraft to be delivered directly to the pad. Since the container width is 11 feet, transportation procedure for a wide load must be used. The 11-foot dimension is dictated by the diameter of the S/C adapter. By eliminating the adapter, the shipping container width can be reduced to approximately 10 feet. No specific advantage can be seen at this time to reduce the container width to 10 feet since wide load procedures would still be required. The container will consist of an inner, shock and vibration isolated, supporting frame and an outer fixed frame and cover.

Seals will be used to permit internal use of clean, dry nitrogen, under pressure, to protect the S/C from the corrosive effects of the uncontrolled ambient atmosphere and its attendant contaminants.

17.2.2.2.1 S/C Transportation Vehicle

The S/C will be transported, in the "all up" configuration in a shipping container design for that purpose.

Over the road transportation will be furnished by a trailer with an air-ride suspension and a tractor with an air-ride drive axle suspension.

Tests performed for United Van Lines, for the purpose of evaluating the road shock and steady, continuous vibration effects on cargo damage, indicate that the damping of vibrations in excess of one Hz, and limitation of peak accelerations to less than one "g" are readily obtained by a combination of tandem air-ride trailer and air-ride drive axle suspension of the tractor. Additional improvement, in reduction of acceleration amplitudes, can be obtained by using an air-ride fifth wheel. Results of the tests, based on peak acceleration levels, are shown in Table 17.2.3.

The tests were performed over nine separate 1,000 foot stretches of highway at speeds of 30, 40 and 50 miles per hour. All possible combinations of four-load configurations were tested at the various speeds to obtain 540 acceleration

Table 17.2-3. Summary of Road Vibration Effects

Equipment Combination	Maximum Recorded Levels			
	Acceleration-g's		Acceleration Intensity-g's/sec.	
	Location on Trailer		Location on Trailer	
	Front	Rear	Front	Rear
Steel Tractor Suspension Steel Trailer Suspension Conventional Fifth Wheel	2.25 g's	2.25 g's	1.78 g's/sec.	2.95 g's/sec
Steel Tractor Suspension Air Ride Trailer Suspension Conventional Fifth Wheel	1.75 g's	0.88 g's	0.71 g's/sec	0.88 g's/sec.
Steel Tractor Suspension Air Ride Trailer Suspension Air Ride Fifth Wheel	0.69 g's	0.72 g's	0.57 g's/sec.	0.84 g's/sec.
Air Ride Drive Axle Tractor Suspension Air Ride Trailer Suspension Conventional Fifth Wheel	0.38 g's	0.60 g's	0.17 g's/sec.	0.28 g's/sec.
Air Ride Drive Axle Tractor Suspension Air Ride Trailer Suspension Air Ride Fifth Wheel	0.37 g's	0.48 g's	0.27 g's/sec.	0.22 g's/sec.

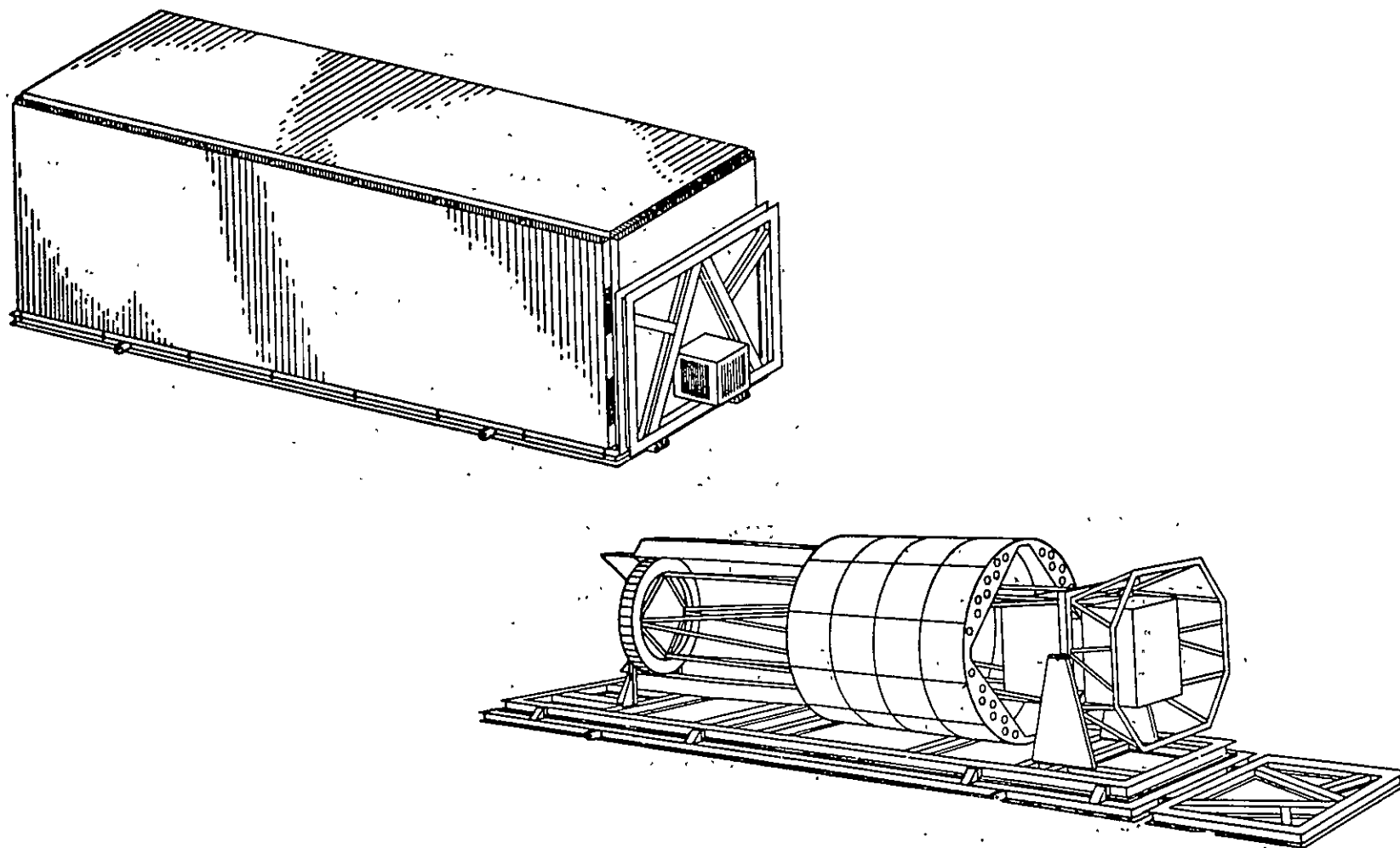


Figure 17.2-2. Spacecraft Shipping Container

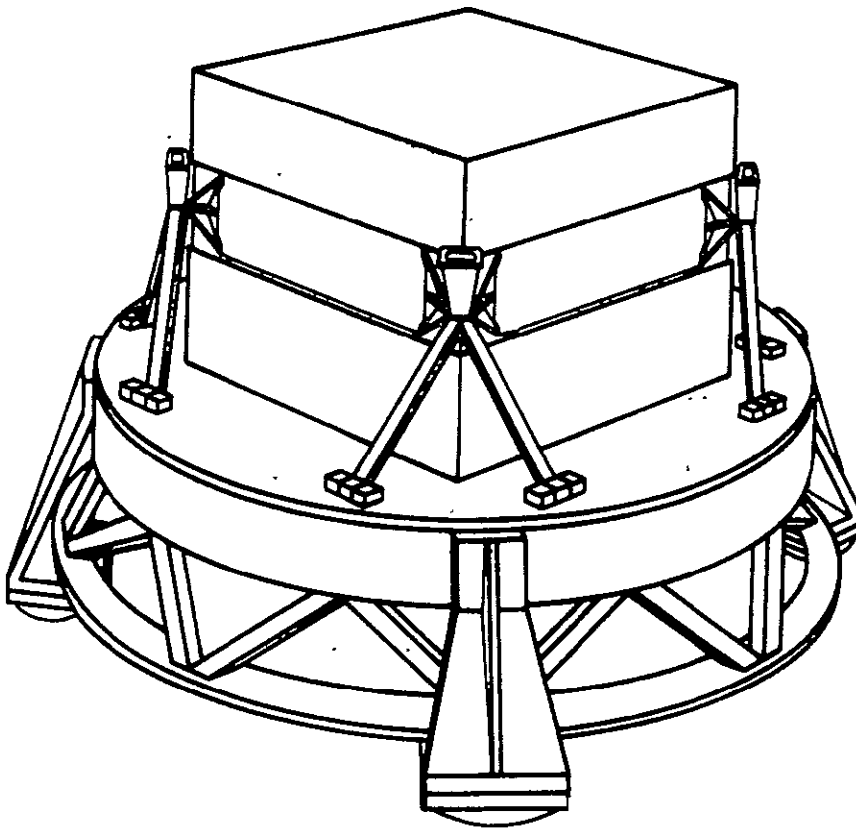


Figure 17.2-3. Spacecraft Holding Fixture

time histories for evaluation. Longitudinal and lateral accelerations, measured at the start of the tests, were of such low magnitudes that such measurements were discontinued and were not included in the evaluation.

At 50 miles per hour, peak accelerations were noted 17 out of 20 times, where at 30 miles per hour minimum levels were noted 18 out of 20 times. In general, high accelerations and steadier vibration patterns were noted at the higher speeds.

Basically, the results of these tests indicate that movement of the S/C to the launch site can be accomplished at speeds of up to 50 miles per hour over good highways with resulting peak accelerations below one "g" and with steady vibrations below one Hz.

Use of a monitoring and alarm system will indicate the necessity for reduced speed whenever road conditions cause accelerations or vibrations in excess of the predetermined acceptable conditions.

Further insurance against damage in transportation will be provided by supporting the S/C in a shock mounted frame with replaceable, collapsible snubbers. A built-in alarm will indicate that the snubber has collapsed and that replacement is required.

17.2.2.2.2 Supporting Frame

As shown in Figure 17.2-2, the supporting frame will consist of two sections. One, consisting of welded structural steel sections, is fixed and forms the basic support. The other section, consisting of welded and bolted members, is attached to the lower section through shock mounts.

17.2.2.2.3 Spacecraft Holding Fixture

The S/C holding fixture (Figure 17.2-3) used to support and transport the S/C within the test area, will consist of the wheeled dolly used for assembly of the S/C. It will hold the S/C and the Z-axis vertical and will be designed to meet the requirements for assembly operations and for support during the various tests, such as the thermal vacuum test.

An adapter ring, fastened to the transtage adapter, will be used for the all-up S/C at all times until mating with the transtage is accomplished. This ring will be fastened to the S/C holding fixture.

17.2.2.2.4 Truss Holding Fixture

A holding fixture (Figure 17.2-4) consisting of structures similar to the hub and EVM will be used to support the ends of the truss for transportation and test. Provision will be made for attachment of hoisting cables to lift the truss in the horizontal or vertical positions.

The simulated EVM will be fastened to a fixture for testing the truss under static load conditions.

Movement of the truss about the test area will be by a wheeled dolly used by the manufacturing department during the assembly stage.

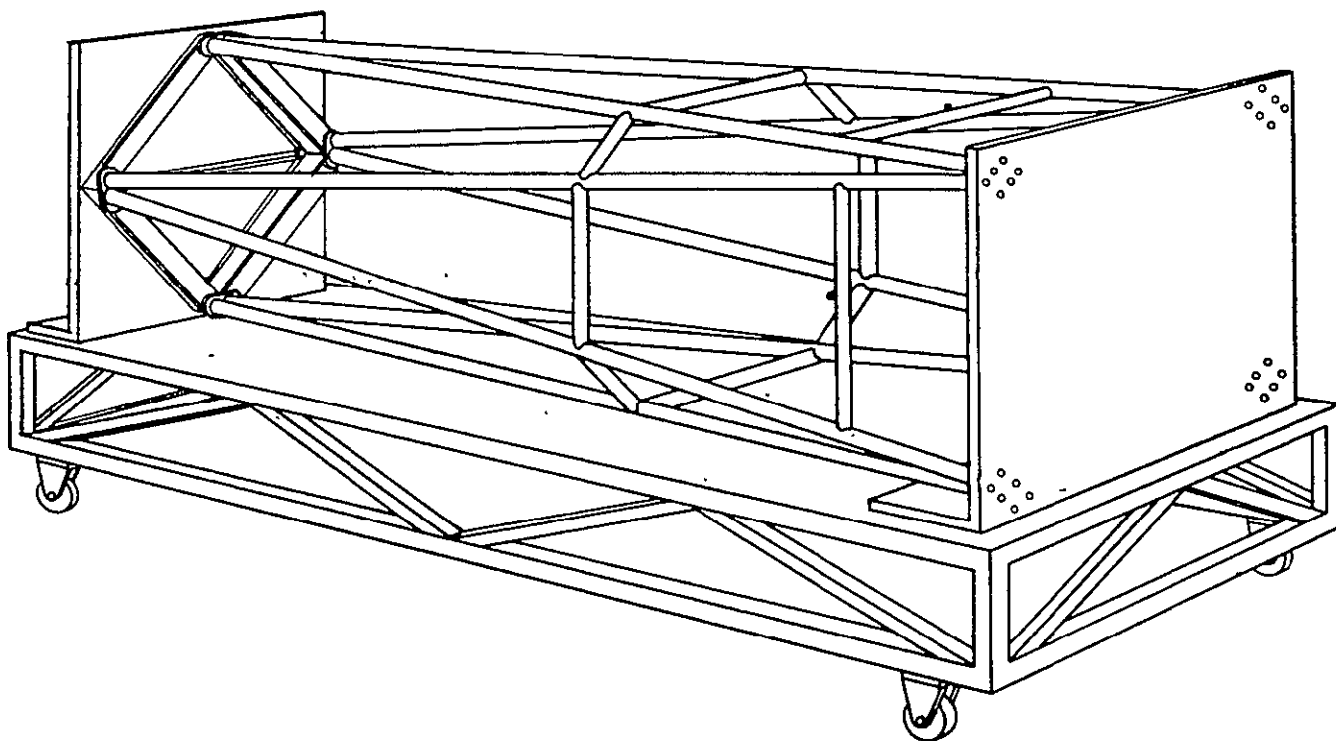


Figure 17.2-4. Truss Holding Fixture

For transportation, the truss may be supported and fastened down in either the vertical or horizontal positions at the simulated hub and/or EVM.

17.3 ATTITUDE CONTROL SUBSYSTEM AGE/BTE

Design and fabrication of the Attitude Control Subsystem (ACS) is conducted in the Aerospace Defense Group under the direction of the Aerospace Support Equipment (ASE) Department of the Honeywell Aerospace Division--Minneapolis Division.

ACS AGE/BTE is made up of stations consisting of one, two, or three bay consoles mounted on casters which accept standard 19-inch width panels. The required test stations are as follows and are discussed in the following paragraphs:

- ACS Test Station (AGE)
- ACS Spacecraft Test Console Simulators (AGE)
- IRA Test Station (BTE)
- IRA Card Test Station (BTE)
- DOC Test Station (BTE)
- Subassembly Test Station (BTE)
- DOC Memory Test Station (BTE)
- ABC Test Station (BTE)
- ACE Test Station (BTE)
- Inertial Wheel Test Station (BTE)
- Earth Sensor Test Station (BTE)
- Star Tracker Test Station (BTE)
- Sun Sensor Test Station (BTE)
- Sun Sensor Electronics Test Stations
- Cable Harness Test Stations (BTE)

17.3.1 ACS TEST STATION

The ACS Test Station contains the necessary power supplies, input and output simulations, signal generation, test application control and measurement instrumentation required to perform the tests of the Attitude Control Subsystem and associated software. The test station will have the capability to test the ACS in the following acquisition modes:

- Rate Damping
- Sun Acquisition
- Earth Acquisition
- Star Acquisition
- Orbital Adjustment

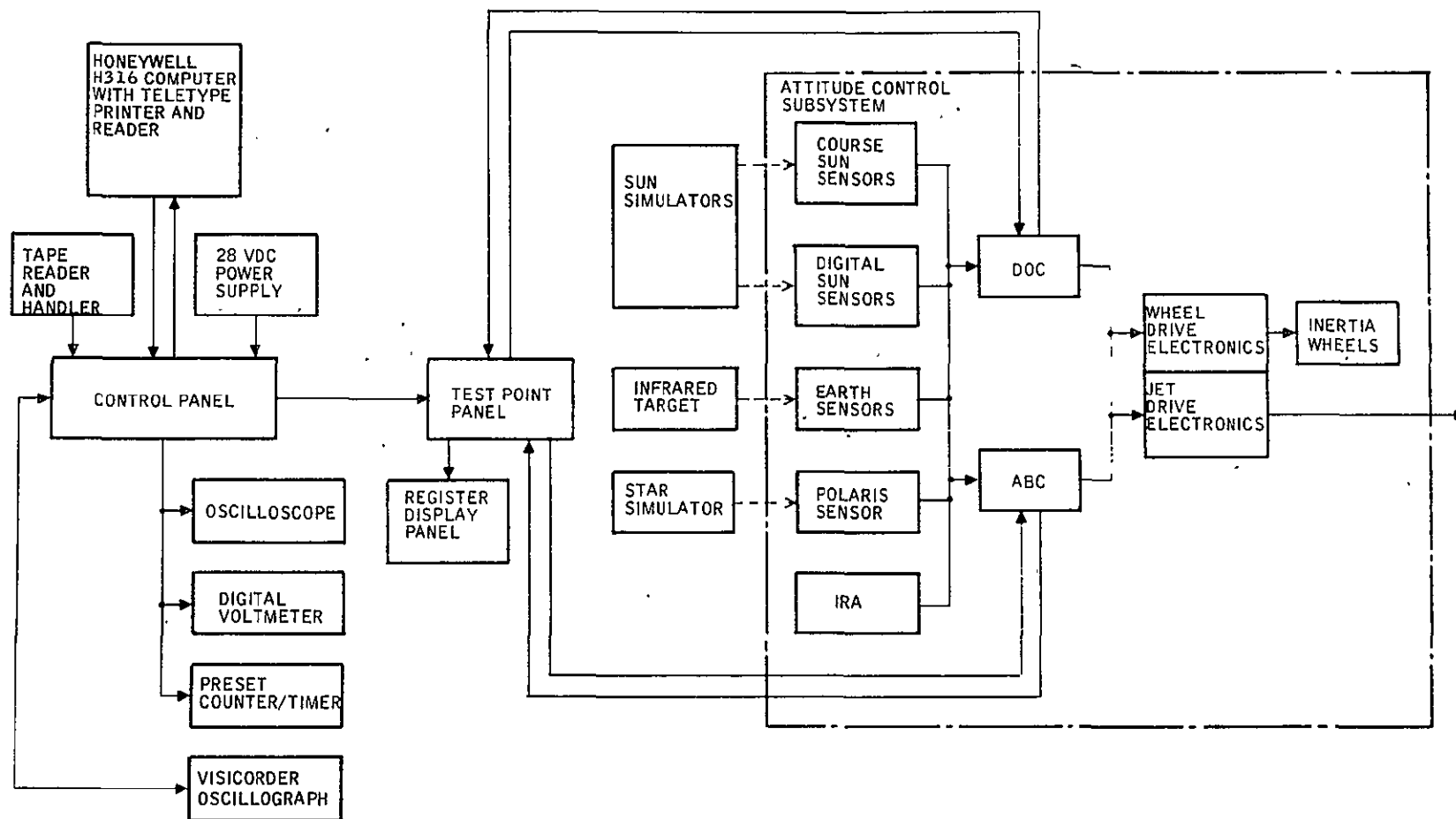


Figure 17.3-1. Attitude Control Sybsystem Test Station Functional Block Diagram

The station consists of a two-bay console containing the following equipment:

- Special Purpose Panels
- Earth Simulator
- Star Simulator
- Sun Simulator
- Commerical Equipment

A functional block diagram of the test station is shown in Figure 17.3-1. The ACS Test Station assemblies are discussed in the following paragraphs.

17.3.1.1 Special Purpose Panels

- The control panel provides the switching control for power, performance mode, instrument select, monitoring functions, input stimuli, and test load simulation.
- The test point panel contains all of the test points that are available from the Attitude Control Subsystem. It provides for test monitoring and trouble shooting as required.
- The display panel provides for visual displays of parallel output monitoring points.
- The power distribution panel provides the power distribution control of the test station.

17.3.1.2 Earth Simulator

Earth simulation for stimulation of the earth sensors at the spacecraft level is provided by the Spacecraft Earth Simulator (SES). The simulation of the earth is accomplished with a heated target disc, the shape of which will approximate the earth shape as viewed from space. The background (space) target provides a temperature differential to the earth sensor.

Earth/Space temperature differential is obtained by applying 115 vac to the earth target via a target temperature controller and comparing said earth target temperature with the "background" (space simulation) temperature. Facility for measuring the target and "space" temperature will be by temperature sensing bulbs dispersed on both the earth target and the "background" to provide a rms temperature for each of the above items to a temperature monitoring panel. The temperature monitoring panel and the two temperature controllers (right and left earth targets) will be included in the Spacecraft Control Console.

17.3.1.3 Polaris Simulator

The Star Simulators are required to provide simulated stars to the Star Trackers. The simulators will consist of a Polaris OSE (Operational Support Equipment) Hood and a Polaris Star Simulator. The OSE Hood will mount on the Star Tracker and will provide five (5) collimated light sources simulating different cone angle positions of the Star Polaris with a means to move same through the field of view of the Polaris Sensor.

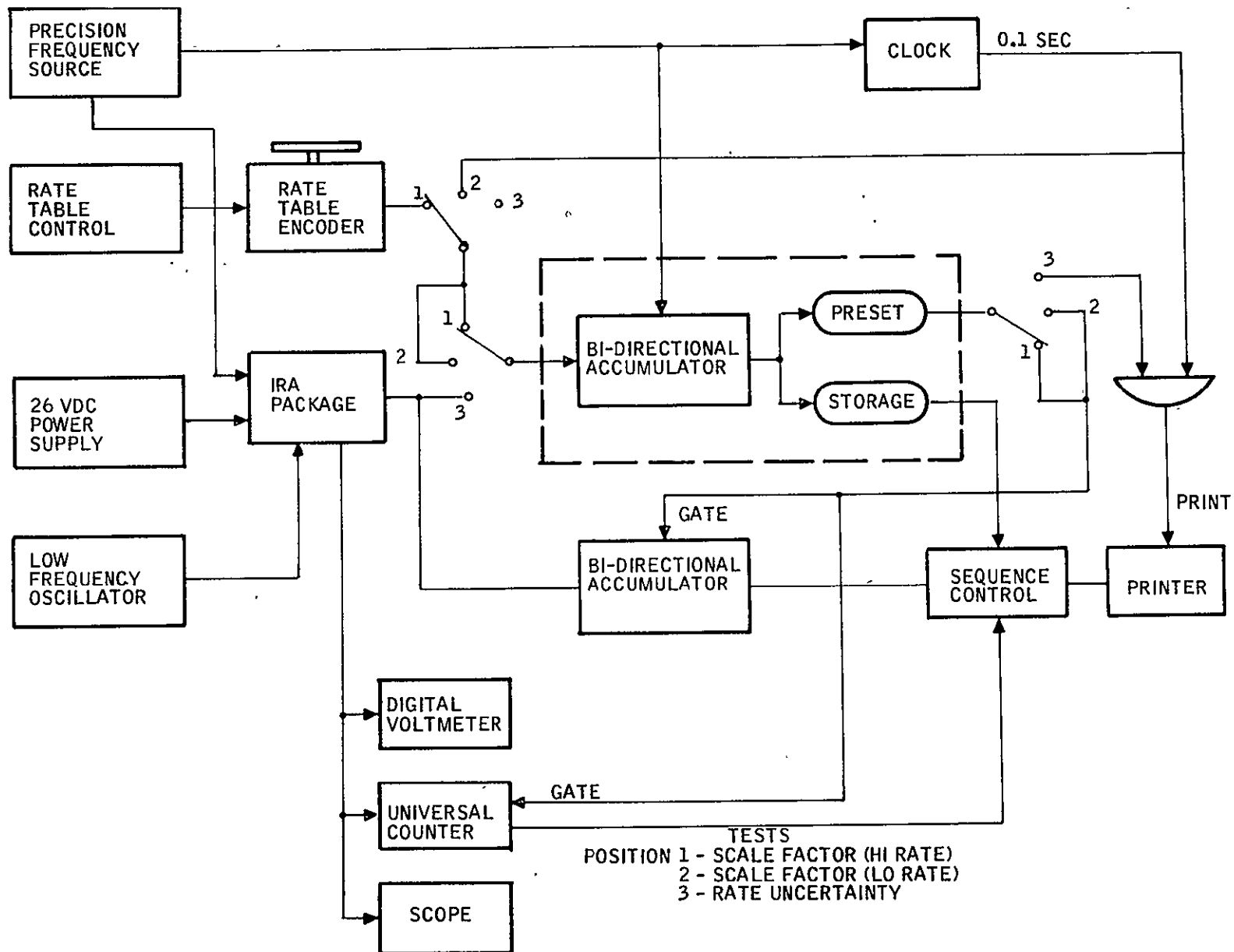


Figure 17.3-2. IRA Test Station Block Diagram

17.3.1.4 Sun Simulator

The Sun Simulator is required to provide a simulated sun to the Sun Sensors. The Sun Simulator will be a collimated light source used to check the sun sensors functions and the sun sensor rejection circuitry within the Earth Sensor. An iodine-quartz lamp will supporting condensing optics and a mask with a predetermined aperture fabricated into an assembly to simulate the sun.

17.3.2 INERTIAL REFERENCE ASSEMBLY TEST STATION

The Inertial Reference Assembly (IRA) Test Station will be a manually operated three-bay console with attendant auxiliary equipment assemblies of a rate table, dividing head, precision holding fixture and interconnecting cabling. The console contains the necessary power supplies, input and output simulations, signal generation, test application control and measurement instrumentation required to perform the performance tests of the IRA. The test station has the capability to perform the following tests. The IRA functional block diagram is shown in Figure 17.3-2.

- IRA electrical interface compatibility
- IRA excitation
- IRA Power
- Internal timing reference clock
- IRA sensing polarity
- G-insensitive drifts
- G^2 sensitive drift
- Scale factor linearity
- Input axis alignment
- Frequency response
- Telemetry

The stations which will be used were built for the P-95 IRA. The following is a description of the usage of the special purpose panels:

- The Control Panel provides the switching control for power, mode, instrument select, test monitoring functions, test load simulation, and input stimuli. Overheat fail-safe logic provides power shutdown in event of gyro overheat. An elapsed time indicator provides a monitor of the IRA device monitoring time. Provisions for a self test of the timing system will be provided.
- The Test Point Panel contains all of the test points that are available from the IRA. It provides for test monitoring and trouble shooting, as required, and contains the logic and relays required to remote program the test signals for the various tests.

- The Digital Clock that provides the clock timing and reference pulses to the IRA. It also interfaces with the preset controller to provide a precision timing reference to the counters which will ensure that the counters start and stop sampling at the precise timing.
- The Power Distribution Panel provides the power distribution control of the test station.
- The Rate Table Control allows the Rate Table to be operated from the test console.
- The precision holding fixture provides for accurate axis alignment of the IRA to the test axis of the rate table or dividing head. The fixture is a cube type casting with accurately machined reference surfaces. Parallelism and perpendicularity of the reference surface will be less than 0.5 minute of arc. Mounting bolts integral to the fixture provide secure fastening to the test table. Fixture design is similar to that used for the Apollo and MOL programs.
- The Rate Table Assembly provides for mechanical rate inputs to the IRA as required over the rate range of 0.02 deg/sec to approximately 40 deg/sec. The table has a direct drive DC torque motor which is controlled from the IRA test console. Bi-directional encoder outputs are provided for angular displacement in terms of pulses/degree. Encoder scale factor is 1000 pulses/degree, or 360,000 pulses/revolution. The encoder output interfaces with the test station preset counter and forms an integral part of the gyro rate measurement test circuit. The assembly, designed and manufactured by Honeywell, consists of rate table, stable mount, mount pads, and interconnect cables.
- The Dividing Head Assembly provides for accurate positioning of IRA selected axis with respect to the earth's cardinal reference points and axis. This function provides for obtaining rate inputs as components of earth rate up to earth rate (0.0004 deg/sec), and also provides the positioning required for measurements of g-insensitive and g-sensitive gyro drift rate. Attitude measurement capability is also included. The dividing head assembly consists of dividing head, face plate, stable mount, mount pads, and interconnection cable.
- The Breakout Test Box is used during checkout of the station and for compatibility testing of the IRA. The test box will have the provisions to monitor the IRA or by the test

station completely isolated from the IRA or by the addition of shorting bars, the signals will go to the test point panel in the test station.

17.3.3 DIGITAL OPERATIONAL CONTROLLER TEST STATION

The Digital Operational Controller (DOC) Test Station has the capability to test the DOC in the following operational modes using software program control:

- Control loop computations and logic
- Mode arithmetic and logic
- Experiment data processing and control
- Sensor interface logic and computation
- Command/Telemetry system interface
- Data transfer and control
- Controller self-test

The station will contain a tape reader which will feed a diagnostic program into the unit under test. The unit will then diagnostically test itself. The outputs of the unit (such as analog, discrete, or digital) will be programmed to the inputs of the units, thus allowing the majority of the functions of the unit to be tested with GO NO-GO results. Programs which will be supplied will include a basic operating tape, instruction diagnostic, memory diagnostic, and an I/O diagnostic. The block diagram is shown in Figure 17.3-3. The following is a description of the usage of the special purpose panels:

- The Control Panel provides the capability to perform hardware checkout, program breakpointing and memory content modification. It provides switching for power, load simulation, input stimuli, instrument select and test monitoring functions.
- The Power Distribution panel provides the power distribution control of the test station.
- The Register Display panel provides the visual display of all of the output registers of the unit.
- The Pulse Generator is a special pulse generator source which provides the necessary single or serial pulse trains to the unit under test.
- The Computer Interface Adapter is required to provide the proper interface between the inputs and the outputs of the DOC.

17.3.4 ANALOG BACKUP CONTROLLER TEST STATION

The Analog Backup Controller Test Station (ABC) is a double bay console. The test station has the capability to test the ABC function operational modes as follows:

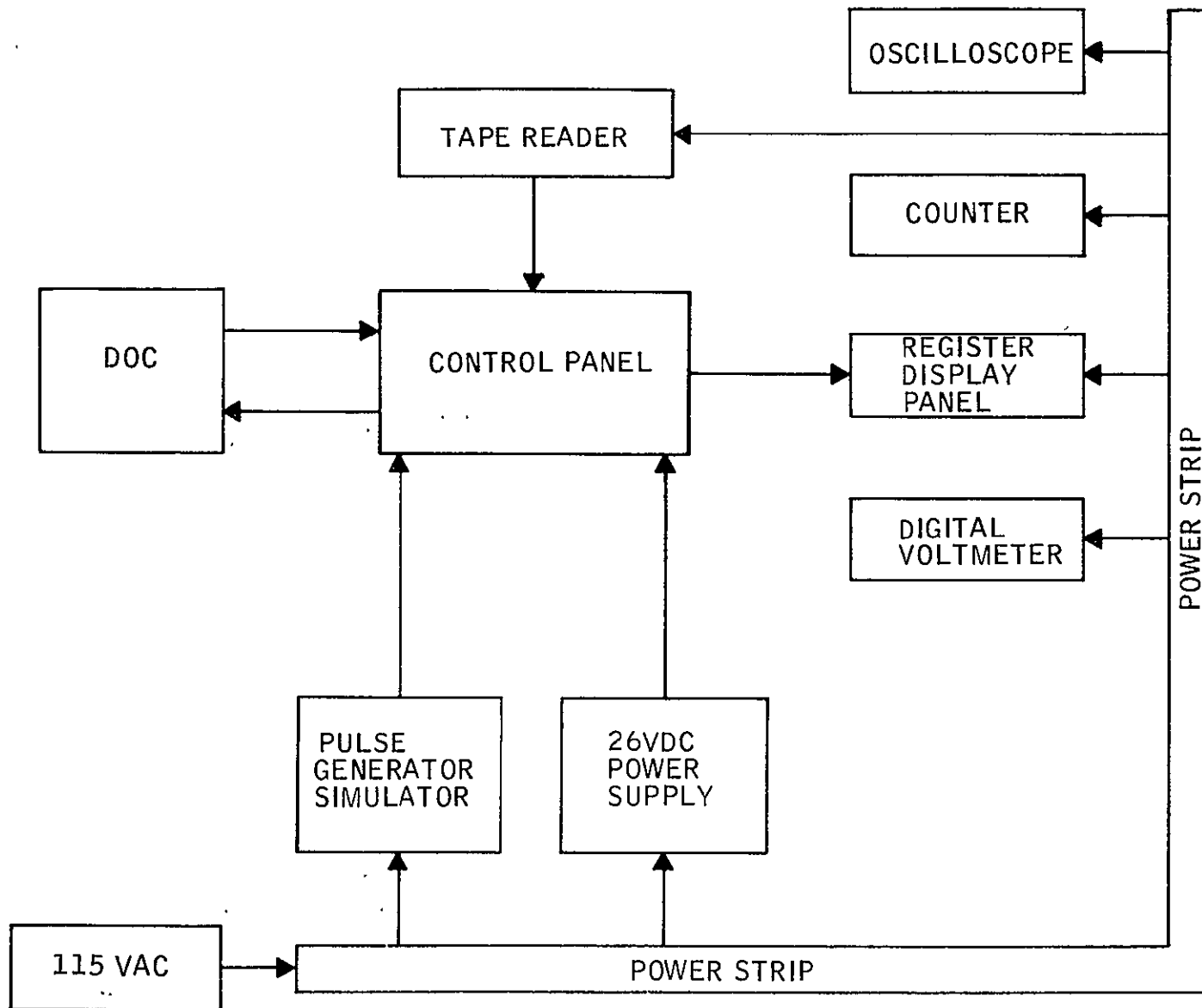


Figure 17.3-3. DOC Test Station Block Diagram

- Sun acquisition
- Earth acquisition
- Polaris acquisition
- Rate damping
- Local hold

The station simulates the inputs of the sun sensor, earth sensor, star sensor, three axis IRA, and the interface of the Inertia Wheels and Auxiliary Propulsion System. The station block diagram is shown in Figure 17.3.4. The following is a description of the usage of the special purpose panels:

- The Control Panel provides the switching control for power, performance mode, instrument select, monitoring functions, input stimuli and test load simulation.
- The Test Point Panel contains all of the test points that are available from the ABC. It provides for test monitoring and trouble shooting as required.
- The Register Display Panel contains lamp drivers which will provide a visual display of the logic level outputs of the ABC.
- The Pulse Generator simulates the digital inputs of the ABC.

17.3.5

ACTUATOR CONTROL ELECTRONICS TEST STATION

The Actuator Control Electronics Test Station is a single bay console capable of performing tests on the propulsion control electronics portion of the Actuator Control Electronics which includes minimum impulse, dead band, hysteresis, pulse and holding power, regulation, stability, noise, and impedance of the various output signals required to control the solenoid valves and heaters. The tests which are performed on the inertial wheel electronics portion of the actuator control electronics assembly includes power, amplification scale factor, phase, phase sequence and the voltage to frequency (V/F) scale factor. The block diagram is shown in Figure 17.3-5. The following is a description of the usage of the special purpose panels:

- The Control Panel provides the switching control for power, instrument select and test monitoring function.
- The Power Distribution Panel controls the power distribution of the test station.

17.3.6

INERTIA WHEEL TEST STATION

The Inertia Wheel Test Station is a single bay test console which contains the necessary power supplies, input-output stimulations, signal generation, test application control, and the measurement instrumentation required to perform the performance tests of the Inertia Wheel. The test station has the capability to perform Torque Speed Characteristics, reaction torque, tachometer output, alignment phasing, starting voltage, input power, and no-load speed tests. The functional block diagram

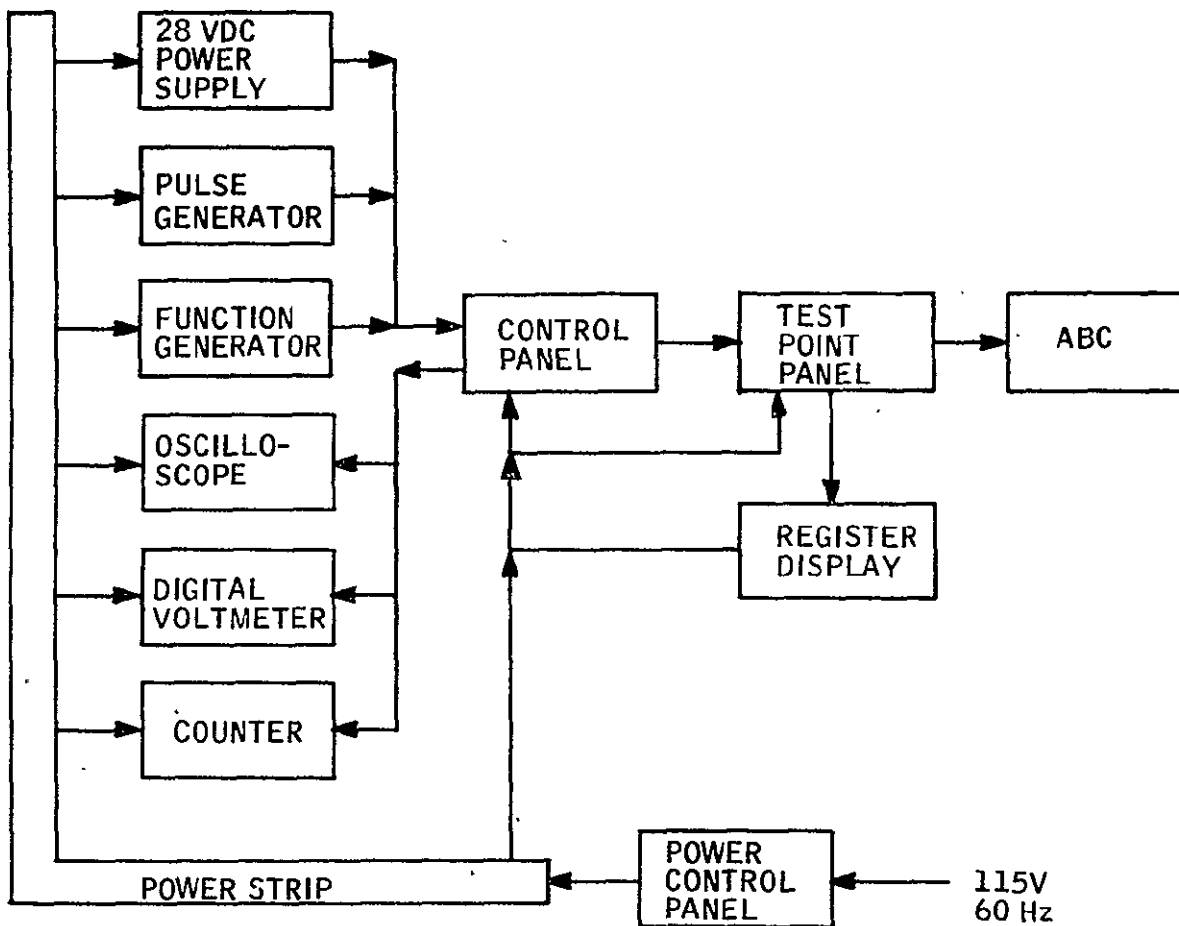


Figure 17.3-4. ABC Test Station Functional Block Diagram

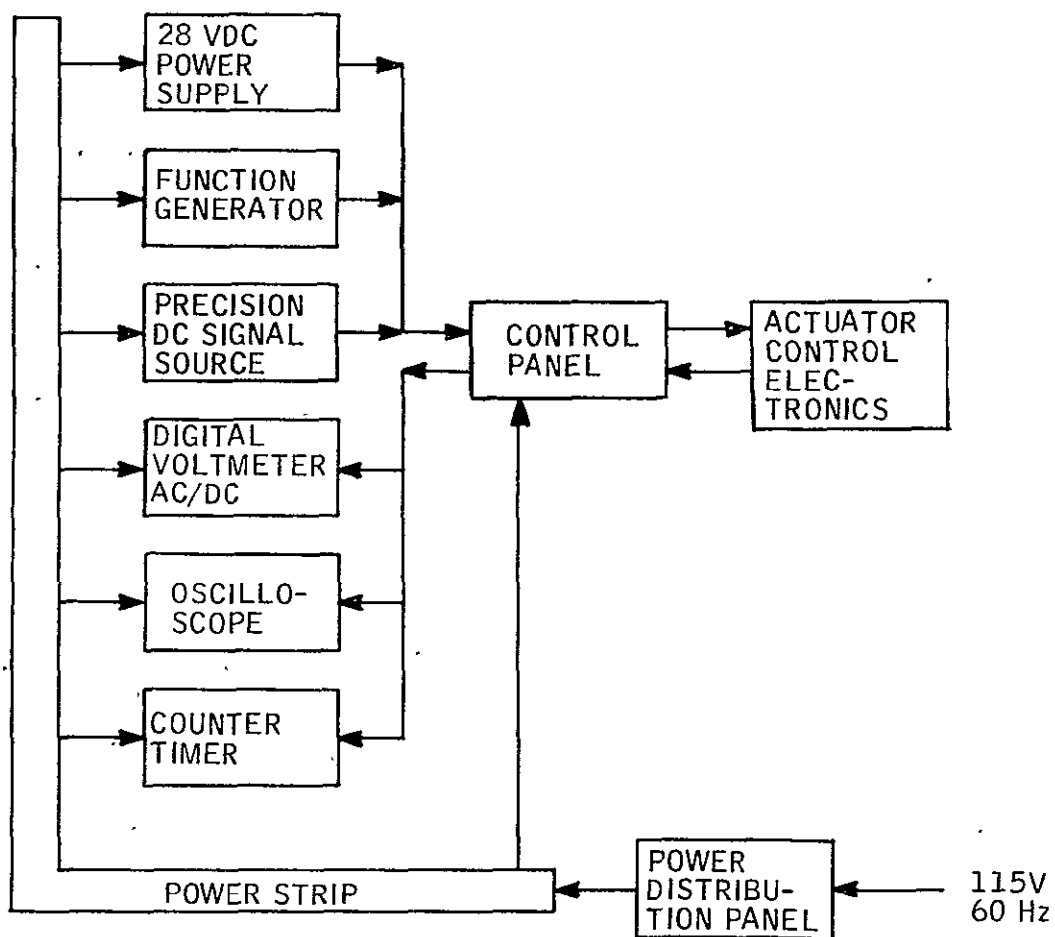


Figure 17.3-5. Actuator Control Electronics Test Station
Functional Block Diagram

is shown in Figure 17.3-6. The following is a description of the usage of the special purpose panels:

- The Control Panel provides the switching control for power, instrument select, and test monitoring functions.
- The Wheel Driver Power Supply is a two phase 400 Hz supply which will drive the inertia wheel from zero speed to maximum speed.
- The Power Distribution Panel provides the power distribution control of the test station.

17.3.7 EARTH SENSOR TEST STATION

The Earth Sensor Test Station is a single bay console with attendant auxiliary equipment assemblies of precision index head and infrared targets. The station contains the necessary power supplies, input-output stimulations, signal generation, test application control and measurement instrumentation required to perform the performance tests of the Earth Sensor Assembly. The tests include static accuracy (null and offset) and linearity, offset mirror position vs command, scan bias position vs command, scan rate and scan linearity, sun interference accuracy as a function of reduced chord length, and acquisition. The functional block diagram is shown in Figure 17.3-7. The following is a description of the usage of the special purpose panels:

- The Control Panel provides for the switching control for power, performance mode, instrument select, monitoring functions, input stimuli, and test load simulation.
- The DOC simulator contains a precision clock frequency source which will simulate the DOC interfaces.
- Earth simulator is required to evaluate the Earth Sensor sun rejection circuitry. The synthetic sun is generated in an auxiliary console by a carbon arc lamp. Cooling of the console is accomplished by a water coil. A close approximation of a true sun spectrum is accomplished by the addition of filters and other optical elements. The energy difference between the 4000°C approximate temperature of the synthetic sun and the 8000°C true sun temperature will be compensated by subtending a greater aperture opening in the sun sensor within the earth sensor system will sweep a larger simulated sun angle and the integrated area of energy will equal the energy of the true sun.
- The Holding Fixture is required to mount the earth sensor on the precision index head.

17.3.8 STAR TRACKER TEST STATION

The Star Tracker Test Station is a single bay console with attendant auxiliary equipment assemblies of precision dividing head, a Star Simulator and a Sun Simulator.

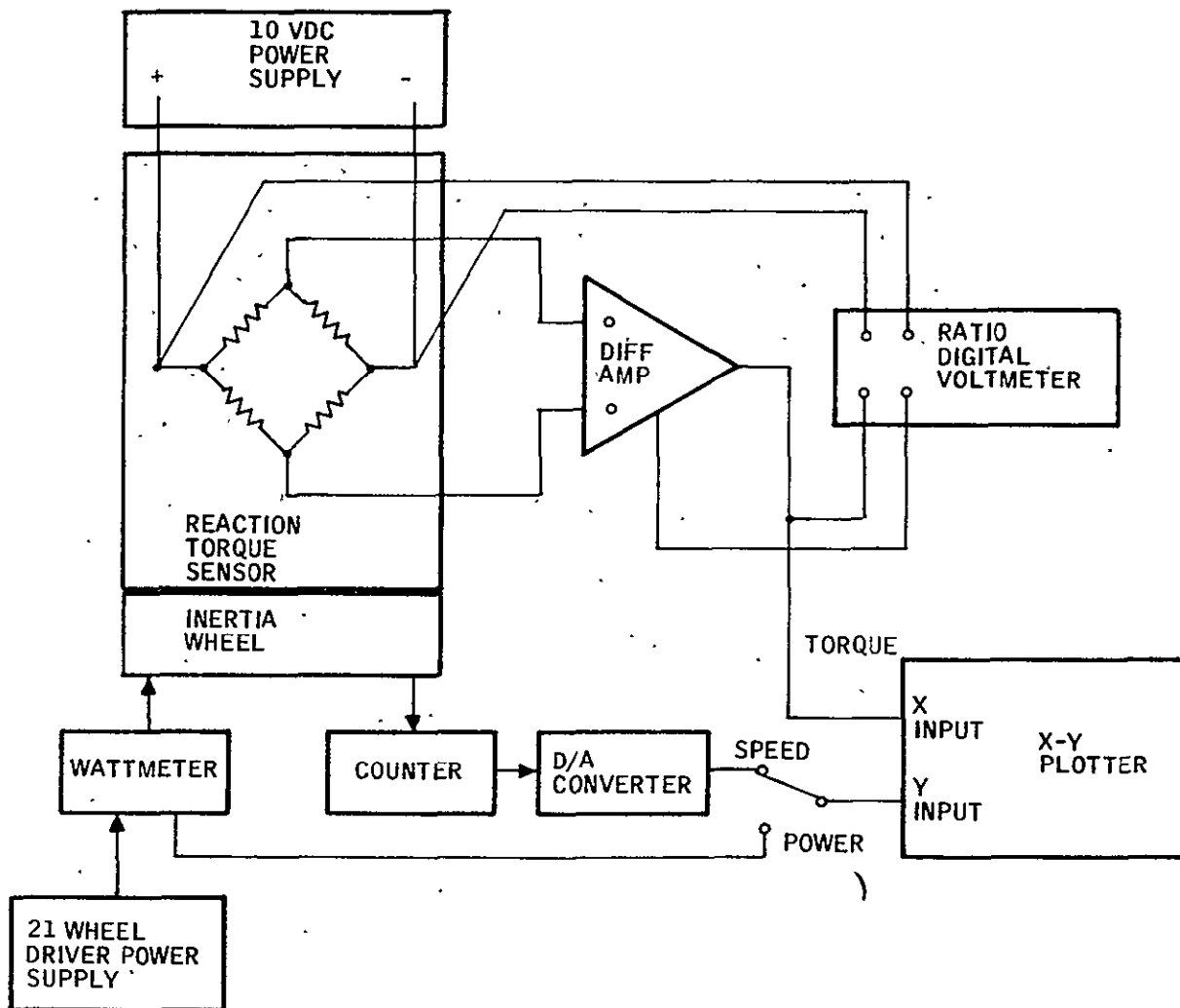


Figure 17.3-6. Inertia Wheel Test Station Functional Block Diagram

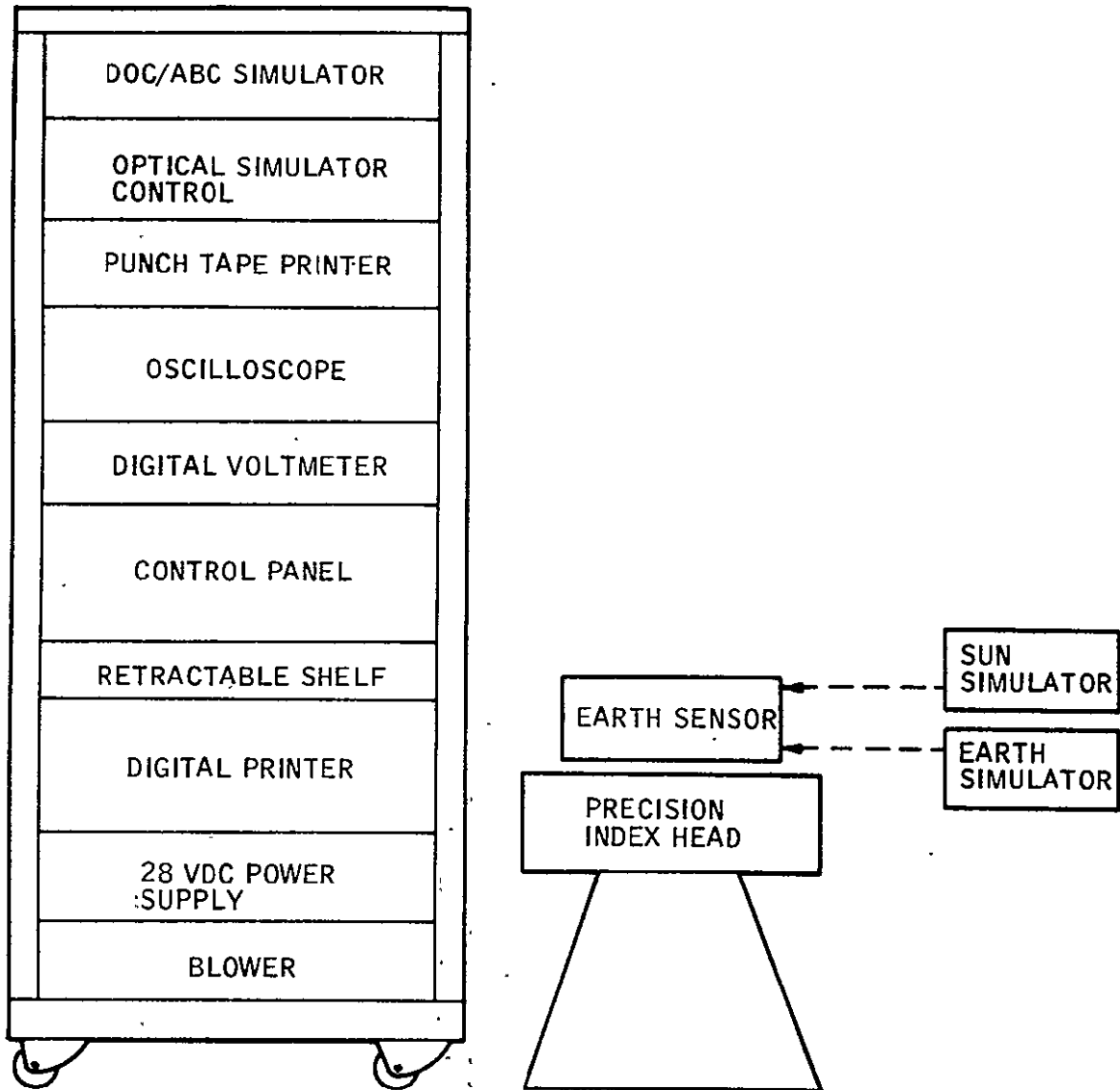


Figure 17.3-7. Earth Sensor Test Station

The Star Tracker Test Station has the capability of performing tests on the Polaris Star Tracker. The tests include input power programming commands, axis readout, sun shutter operation, star acquisition output logic, star brightness measurement output logic, and field of view. The functional block diagram is shown in Figure 17.3-8. The following is a description of the usage of the special purpose panels:

- The control panel provides the switching control for power, performance mode, instrument select, monitoring functions, input stimuli, and test load simulation.
- The Register Display Panel is required to provide a visual display of the two 16-bit parallel registers and the serial 16-bit 16-bit register.
- The Star Simulator provides the simulated star which the Polaris Star Tracker can track. The simulator consists of a Polaris Hood and a Polaris Star Simulator. The Hood mounts on the Star Tracker and provides five (5) collimated light sources simulating different cone angle positions of the Star Polaris with a means to move same through the field of view of the Polaris Sensor. The Polaris Star Simulator is a collimated light source simulating the star Polaris and will be used to determine the field of view and star acquisition of the star tracker.
- The Sun Simulator provides a simulated sun to checkout the Sun Shutter Operation of the Star Tracker.
- The holding fixture is required to mount the Polaris Star Tracker to the precision Dividing Head.

17.3.9 SUN SENSOR TEST STATION

The Sun Sensor Test Station is a single bay console with attendant auxiliary equipment assemblies of a precision dividing head and a sun simulator.

The Sun Sensor Test Station has the capability of performing tests on the Coarse Sun Sensors and the fine Sun Sensors. The tests performed are field of view. The functional block diagram in Figure 17.3-9. The following is a description of the usage of the special purpose panels:

- The control panel provides for the switching control for performance mode, instrument select, and test load simulation.
- The synthetic sun is generated in an auxiliary console by a carbon arc. Cooling of the console is accomplished by a water coil. A close approximation of a true sun spectrum is accomplished by the addition of filters and other optical elements. The energy difference between the 5000°C approximate temperature of the synthetic sun and

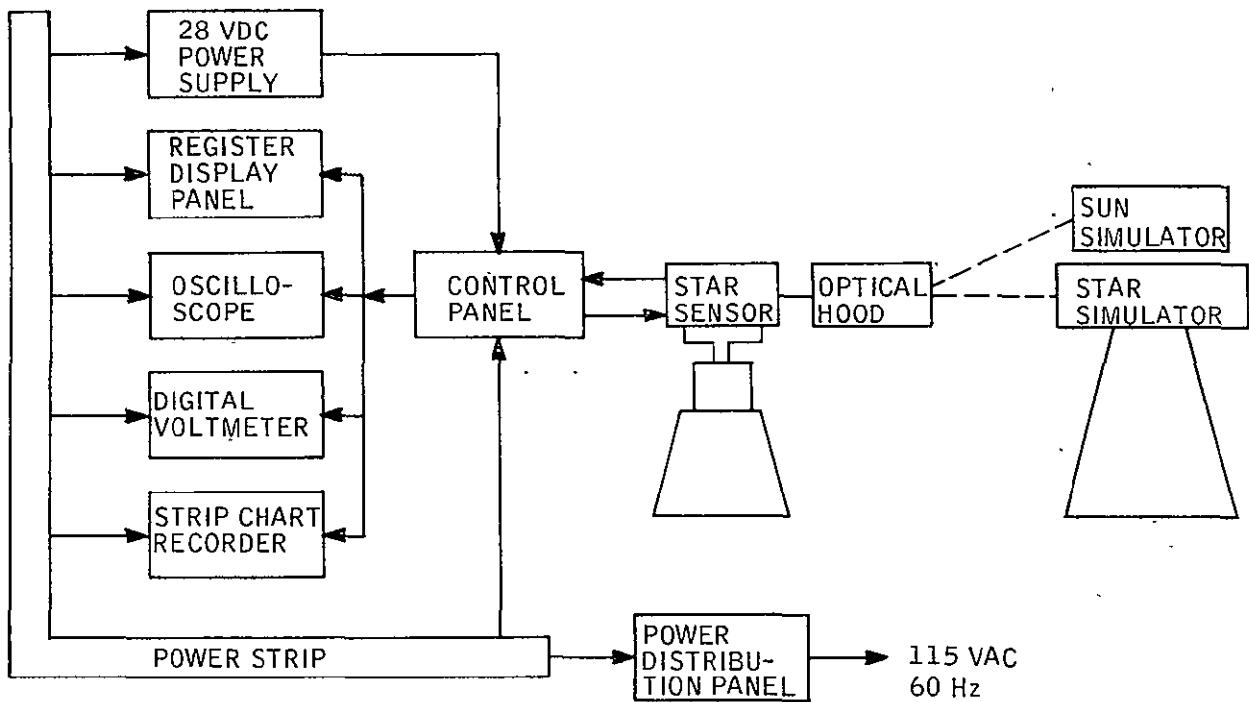


Figure 17.3-8. Star Tracker Test Station Functional Block Diagram

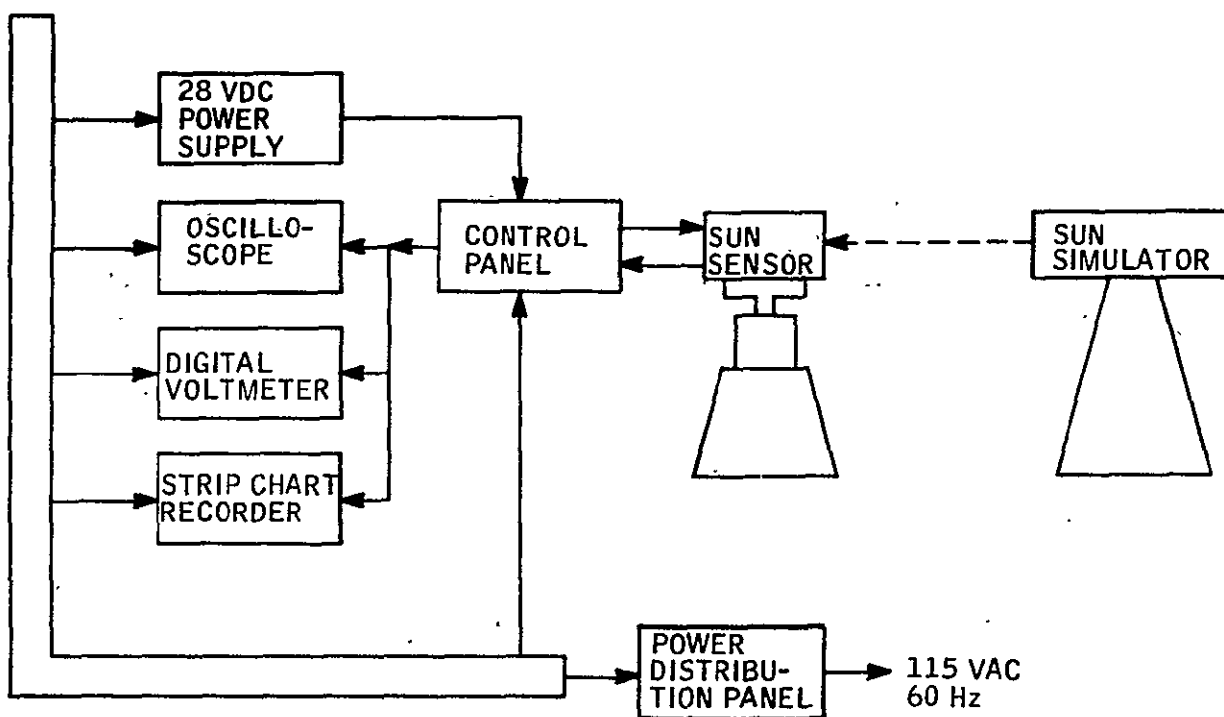


Figure 17.3-9. Sun Sensor Test Station Functional Block Diagram

the 8000°C true sun temperature will be compensated by subtending a greater aperture opening in the simulated sun mask than the true 1/2° sun presents.

17.3.10 SUN SENSOR ELECTRONICS TEST STATION

The Sun Sensor Electronics Test Station is a single-bay console which contains the necessary power supplies, input-output stimulation, signal generation, test application control and measurement instrumentation required to perform the performance tests of the Sun Sensor Electronics. The station simulates the inputs of the course and fine Sun Sensors. The Sun Sensor Electronics consists of three assemblies; two EVM electronics assemblies and one Hub electronics assembly. The electronics assemblies are tested for gain, level detector, and power dissipation. The internal power supply is tested for regulation, ripple and noise. The functional block diagram is shown in Figure 17.3-10. The following is a description of the usage of the special purpose panels:

- The Control Panel provides the switching control for power, performance mode, instrument select, monitoring functions, input stimuli, and test load simulation.
- The Test Point Panel contains all of the test points that are available from the Sun Sensor Electronics. It provides for test monitoring and trouble shooting as required.

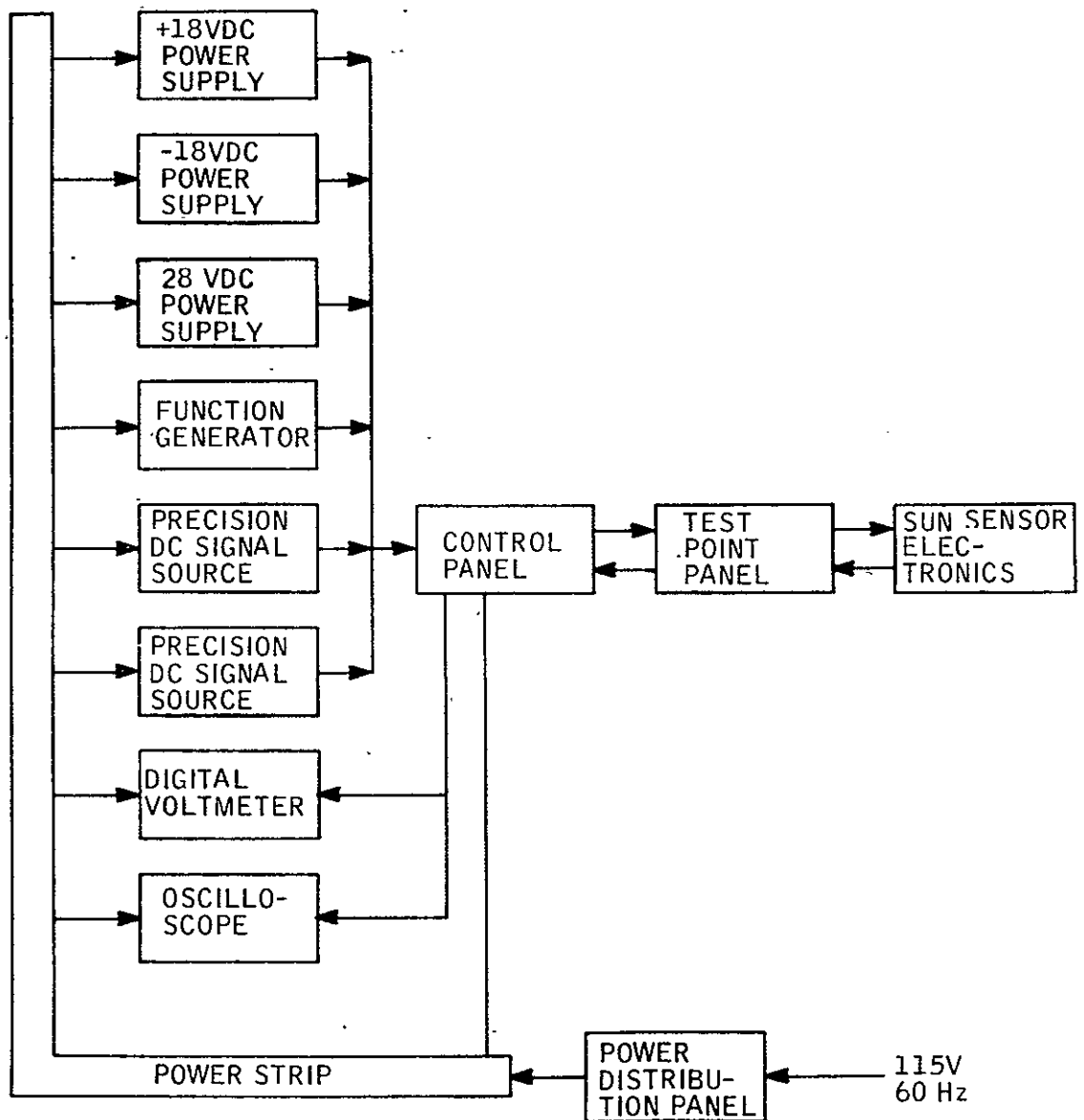


Figure 17.3-10 . Sun Sensor Electronics Test Station Functional Block Diagram

17.4

AUXILIARY PROPULSION SUBSYSTEM AGE/BTE

The AGE/BTE required to support the testing of the Auxiliary Propulsion Subsystem (APS) consists of the following stations:

- APS Test Station (AGE)
- APS Field Equipment (AGE)
- Thrust Measurement Test Station (BTE)

These test stations are described in the following paragraphs.

17.4.1

AUXILIARY PROPULSION SUBSYSTEM TEST STATION

The APS Test Station is a two-bay console which contains the necessary power supplies, input/output stimulations, test application control and measurement instrumentation required to conduct the performance testing of the measurement instrumentation required to conduct the performance testing of the Auxiliary Propulsion Subsystem. Ancillary equipment used in conjunction with the two-bay console, and which will be a part of the overall APS Test Station, includes leak detection equipment, a thrust repeatability fixture, and ammonia distilling and purification equipment. In addition to the test capability, the APS Test Station provides the necessary operator safety through detailed operating instructions, warning devices, safety equipment and protective covering for proper handling of anhydrous ammonia.

The APS Test Station has the capability to perform tests on the Propulsion Subsystem and the Regulated Pressure Feed Subassembly. The tests on the Propulsion Subsystem includes package, pressure proof, regulated pressure flow, and thrust repeatability. The following is a description of the application of the special purpose panels:

- The control panel provides for the switching control for power, instrument select, and test monitoring functions.
- The power distribution panel controls the power distributing throughout the test station and to the ancillary equipment.
- The Ammonia Distillation and Purification equipment provides the capability to reduce the impurities in commercially available ammonia to an acceptable level for use as a fuel in the Propulsion Subsystem. This will include the equipment and procedures for cleaning, baking, flushing and evacuating the spacecraft fuel tanks prior to filling them with purified ammonia.

- o The vacuum pump in conjunction with baking and flushing equipment is used to create a non-contaminating environment within the ammonia fuel tanks prior to charging with ammonia. The vacuum pump also provides a partial vacuum in the nozzle adapter fixtures during thrust repeatability tests.

17.4.2

THRUST MEASUREMENT TEST STATION

The Thrust Measurement Test Station is required to measure the thrust of the Attitude Control thrusters of the Auxiliary Propulsion Subsystem. The Thrust Measurement Test Station consists of a special fixture (see Figure 17.4-2) used in conjunction with the APS Test Station. The fixture provides the mounting and force measurement capability for the AC Thrusters while the APS Test Station provides the power and control for the thrusters. The OC Thrusters are purchased with the thrust measurements already performed.

17.4.2.1

Special Purpose Fixtures

The Thrust Measurement Test Station consists of a special purpose precision pendulum beam balance which is capable of measuring the AC thrust levels of 0.020 lbs and 0.040 lbs. The pendulum beam balance (see Figure 17.4-3) has a self-contained ammonia fuel tank and the necessary gages and pressure regulators to insure the proper gas pressure for correct thruster operation. The amount of deflection of the pendulum balance is a direct measure of the thruster force. The fixture will be calibrated using precision weights. See Figure 17.4-4 for block diagram.

17.5

INTERFEROMETER AGE/BTE

Interferometer testing at the spacecraft level is accomplished through telemetry, command, and internal calibration of the unit. The interferometer BTE (IBTE) is used to perform in-process testing, subassembly testing, qualification testing, and acceptance testing. This equipment is used in IBM, Huntsville manufacturing and test lab areas. Using front panel switches and a minimum of cable changes, all component, in-process, subassembly, qualification, and final acceptance testing are preformed utilizing the same basic test equipment.

The Interferometer Test Console consists of rack mounted commercial equipment, power supplies, interface cabling, and test control panel. The Test Console is portable for maximum utilization and is designed for minimum set-up and calibration requirements. A block diagram of the Interferometer Test Console is shown in Figure 17.5-1. Test cables are provided to mate with each type of module/component to be tested.

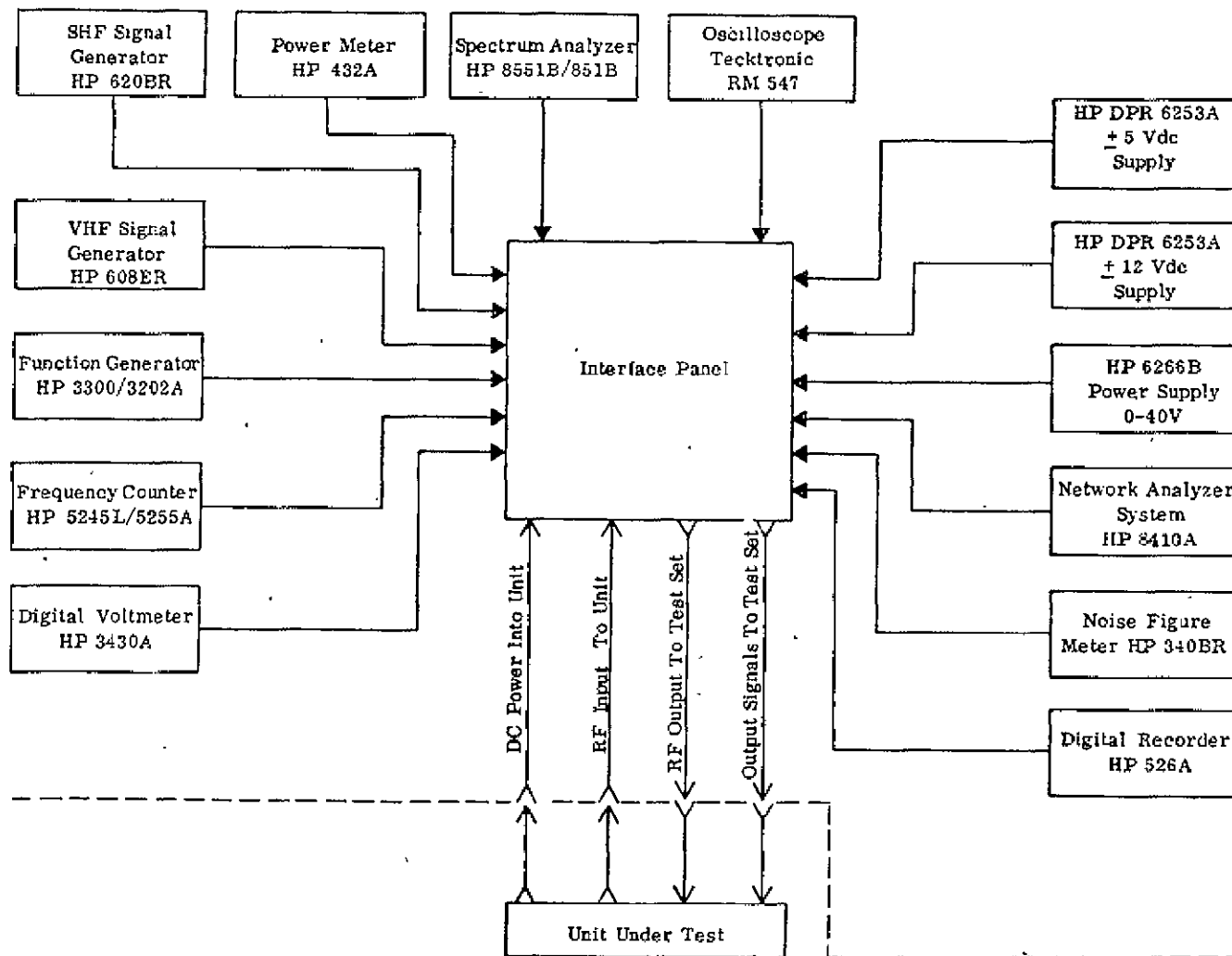


Figure 17.5-1. Block Diagram IBTE

No items of Interferometer AGE are to be delivered to FHC for the purpose of checking out the Interferometer subsystem after installation on the S/C. The primary means for determining whether the Interferometer Subsystem is performing correctly is through readout of the spacecraft telemetry. The Interferometer is commanded into its various modes and the response noted via telemetry. Calibration capability is self contained within the Interferometer Subsystem and is capable of determining the performance of all of the subsystem except for the alignment of antenna array. Firm protective measures will be taken to insure that the antenna array is not subjected to mechanical forces which would destroy final factory calibration.

This paragraph identifies and describes the AGE/BTE required to support the Telemetry and Command (T&C) Subsystem Test Program. No attempt is made to duplicate the information contained in previous sections concerning the test and checkout requirements of the subsystems unless it becomes necessary to describe the test equipment functions.

The T&C Subsystem AGE/BTE consists of four standard equipment racks which house commercial test equipment and three special panels. This console is used to test the T&C components on an individual basis as well as at the integrated subsystem level. Accordingly, it serves both as BTE and AGE.

Table 17.6-1, which lists the T&C Test Console components, identifies the test equipment which is used solely for component testing of BTE items, and test equipment which is used for subsystem testing as AGE items. Items which are used for both component and subsystem testing are identified accordingly. A block diagram of the console is shown in Figure 17.6-1.

Two T&C Test Consoles are required to support the T&C Subsystem Test Program. However, one console could be provided as GFE if IBM acquires NASA's permission to use the following SATURN Test Equipment, (purchased under Contract NAS-8-14000 and presently in-house at IBM, Huntsville), on the ATS program:

- PCM/FM Test Console
- RF Test Console
- Antenna Test Set

The functional details of the Console for subsystem and component tests are discussed in the following paragraphs.

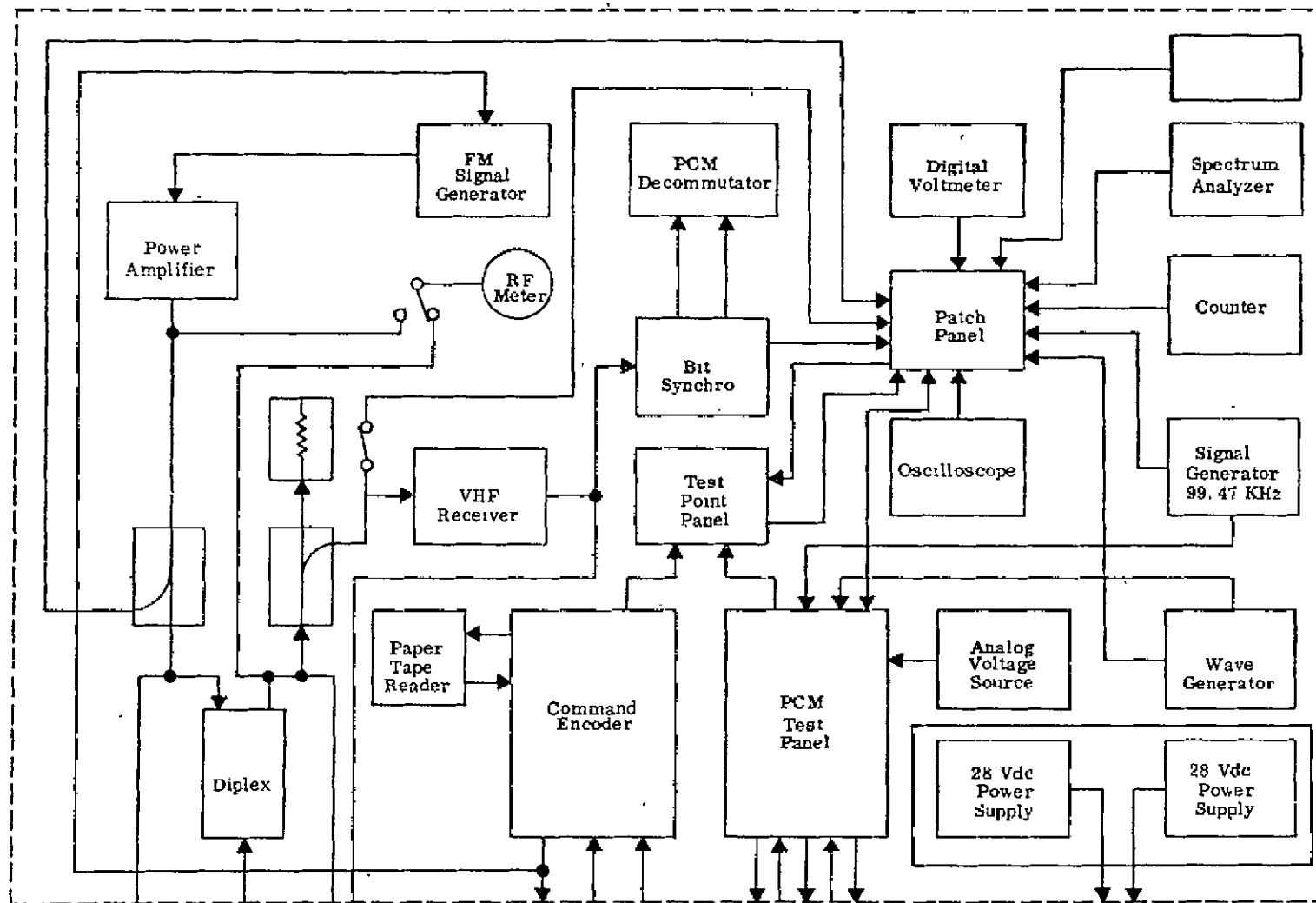


Figure 17.6-1. T&C Test Console Block Diagram

Table 17.6-1. T&C Test Console Equipment List

<u>Item</u>	<u>Vendor or Equivalent</u>	<u>Model or Equivalent</u>	<u>Quantity</u>
Analog Source (c)	Electronic Dev. Co.	VS-HR	1
Function Generator (a)	HP	3300A	1
Plug-In (a)	HP	3301A	1
Bit Rate Synchronizer (c)	EMR	2726	1
Bit Rate Selector (c)	EMR	2727	1
PCM Decom. (c)	EMR	2746	1
Power Supply (c)	HP	6433B	2
DVM (a)	HP	3440A	1
DVM-Plug-In (a)	HP	3444A	1
Electronic Counter (a)	HP	5245L	1
Plug-In (a)	HP	5253B	1
Oscilloscope (Storage) (a)	Tek	R549	1
Plug-In (a)	Tek	Type W	1
Plug-In (a)	Tek	Type M	1
Plug-In (a)	Tek	Type CA	1
VHF Receiver (c)	Vitro	1037G	1
RF Tuner (c)	Vitro	RFT 100A	1
RF Power Meter (a)	HP	432A	1
Thermistor Mount (a)	HP	8478B	1
Signal Generator (c)	HP	608F	1
Synchronizer (c)	HP	8708A	1
Demodulators (c)	Vitro	FSD-101G	1
Spectrum Analyzer (a)	HP	8551B/851B	1
Patch Panel (c)	Trompeter	--	1
Paper Tape Reader (c)	--	--	1
Diplexer (c)	TRF, Inc.	P-158	1

17.6.1 SUBSYSTEM AGE

17.6.1.1 Command Section Description

The command section of the T&C Test Console is comprised of a Paper Tape Reader, a Command Encoder Panel, a VHF Signal Generator and a VHF synchronizer which may be operated in a semi-automatic or manual mode of operation.

In a semi-automatic mode, a Paper Tape Reader will supply the control inputs to the Command Encoder Panel. A sixty-four bit command word and a control response word will be read from the tape. The control response word will be loaded into a Digital Comparator in the Encoder for comparison with a selected Command Decoder/Distributor output. The command word is used in the Command Encoder Panel to control the generation of a composite command signal. The signal consists of a PCM/FSK-AM waveform which is generated in the Command Encoder panel by mixing two to ten tones with an FSK signal which is fifty percent AM modulated by a 128 Hz data clock.

In the manual mode of operation, an operator would manually enter a command word and control response word via switches on the Command Encoder Panel and the Command Encoder applies this composite signal directly the VHF Receiver Input via an FM Signal Generator and Power Amplifier for Subsystem Testing. The output of the Command Decoder/Distributor will be monitored by the test set. The CDD outputs will be loaded with the specified worse case impedance in the Command Encoder Panel and a selected output will be loaded into the Comparator for comparison with the control response word read from the tape.

Commercial equipment located in the test set includes a Digital Voltmeter, an Oscilloscope, and a Spectrum Analyzer which will be used to measure input and output parameters of the Command Decoder/Distributor or the Command Subsystem at the test point panel.

The RF command link will be checked out as follows: Prior to applying power to the command receivers, verification of input VSWR will be ascertained by examination of component record test history. VHF Receiver sensitivity will be verified over TM by readout from the FCM Decommulator.

17.6.1.2 Telemetry Section Description

The Telemetry section of the T&C Test Console is comprised of a PCM Test Panel, an analog voltage source, a Test Point Panel, a Bit Synchronizer, a PCM Decommulator, and a VHF Receiver.

The Telemetry section of the test set is used to test the Data Acquisition and Control Unit (DACU) and Data Switching Unit (DSU), the Spacecraft (S/C) clock and the Telemetry subsystem comprised of a DACU, DSU, S/C Clock, and a VHF Transmitter connected in a flight configuration. For DACU Testing the Telemetry section of the test set will be used to simulate the analog and digital data inputs to the

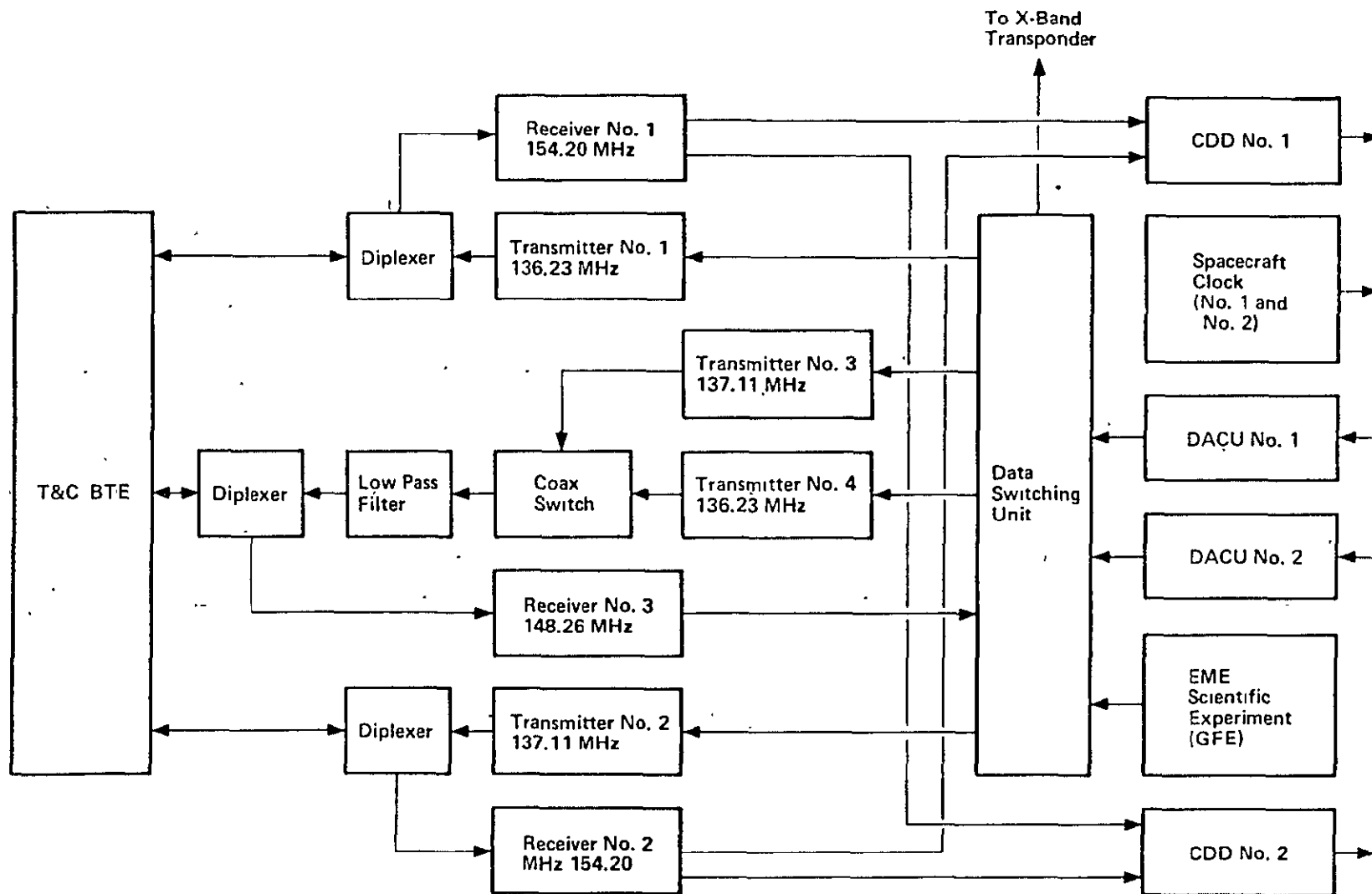


Figure 17.6-2. T&C Subsystem Test Configuration

- Test Parabola and Positioner (BTE)
- Standard Gain Horns (BTE)
- Truss Mock-up (BTE)

These test sets are discussed in the following paragraphs.

17.7.1 COMMUNICATIONS TEST SET

The Communications Test Set provides transponder test capability at three levels of testing, Transponder level, Communications assembly level and Spacecraft level. This approach facilitates comparison of test data at the various test levels and provides economies in test equipment.

The communications test set consists of five main categories of equipment:

- Power sources
- Command and control equipment
- RF Signal generating equipment
- RF Power measuring and monitoring equipment
- General Monitoring and Receiving equipment

A block diagram of the communications test set is shown in Figure 17.7-1.

The equipment will be mounted in seven bays of equipment racks with 70 inches of panel space each. The largest assembly will not exceed three bays for ease of handling and shipping. All commercial and unique equipments will be rack-mounted and connected by the necessary coaxial cabling, waveguide, and multi-conductor cables. External coax cables and waveguide will be provided for connections between the antenna couplers and the Communications Test Set. External cables will also be provided for connection between the interface panel and the transponder.

17.7.1.1 Power Sources

The power sources consist of two DC power supplies, one for each of the spacecraft buses. A ripple generator is used to induce a controlled ripple on either bus for ripple susceptibility tests.

17.8 POWER SUBSYSTEM AGE/BTE

Due to the relative simplicity of the intersubsystem interfaces of the power subsystem, qualification and acceptance subsystem level tests prior to spacecraft integration are not planned, therefore there are no requirements for power subsystem AGE below the spacecraft level.

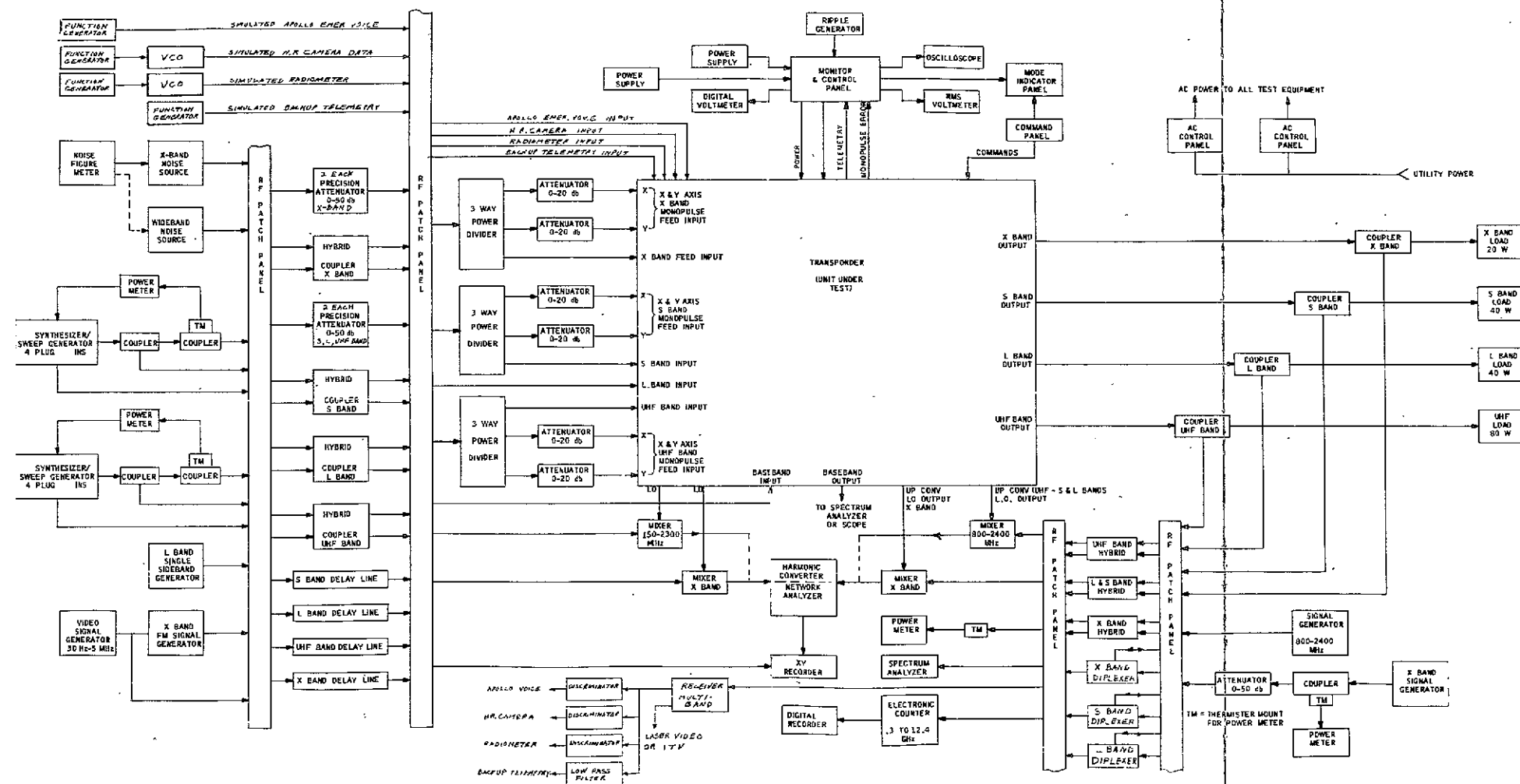


Figure 17.7-1. Communications Test Set Block Diagram

BTE to support the engineering, qualification and acceptance test programs at the component level are classified in two groups, electrical and mechanical.

- Electrical BTE is that equipment which is designed and fabricated primarily to support the electrical performance testing of the component.
- All BTE will be designed to utilize standard commercial (capital) test equipment to the maximum extent possible. The primary purpose of the BTE will be to provide loads, switching functions, mode control, and test point interface to external test equipment.
- Design and fabrication of the BTE will conform to the requirements specified in 685-GR-1000 Applications Technology Satellite Model F&G General Specification For Ground Support Equipment.

The following BTE will be provided for power subsystem component testing:

- Switching Mode Regulator Test Set (280 watt)
- Switching Mode Regulator Test Set (250 watt)
- Battery Charger Test Set
- Load Interface Circuit Test Set (0-15 watt)
- Load Interface Circuit Test Set (15-60 watt)
- Battery Test Set
- Power Control Unit Test Set
- Squib Interface Unit Test Set
- Solar Panel Module Test Set
- Harness FACT Interface Cables

17.8.1 SWITCHING MODE REGULATOR (SMR) TEST SET

The SMR Test Set provides a test interface to each input and output signal and a load to the SMR. Figure 17.8-1 is a schematic diagram of the SMR Test Set. The test set will provide the capability to perform the following tests:

- Efficiency
- Dynamic Regulation
- Overload and short circuit protection
- Overvoltage clamp
- No load loss
- Starting time
- Output Ripple
- Output voltage transient

17.8.2 BATTERY CHARGER

The Battery Charger Test Set provides a test interface to each input and output signal and load to the Battery Charger. Figure 17.8-2 is a simplified schematic of the Test Set. The test set will provide the capability to perform the following tests:

- Efficiency
- Overload and short circuit protection
- No load loss
- Temperature Track
- Output Ripple
- Reverse Current

17.8.3 LOAD INTERFACE CIRCUIT TEST SET

The LIC Test Set provides a test interface to each input and output signal and load to the LIC. The test set provides the capability to perform the following tests:

- Efficiency
- Dynamic Regulation
- Current Limit
- Input Voltage Shutdown
- Output Voltage Shutdown
- On-off Command
- Rejection

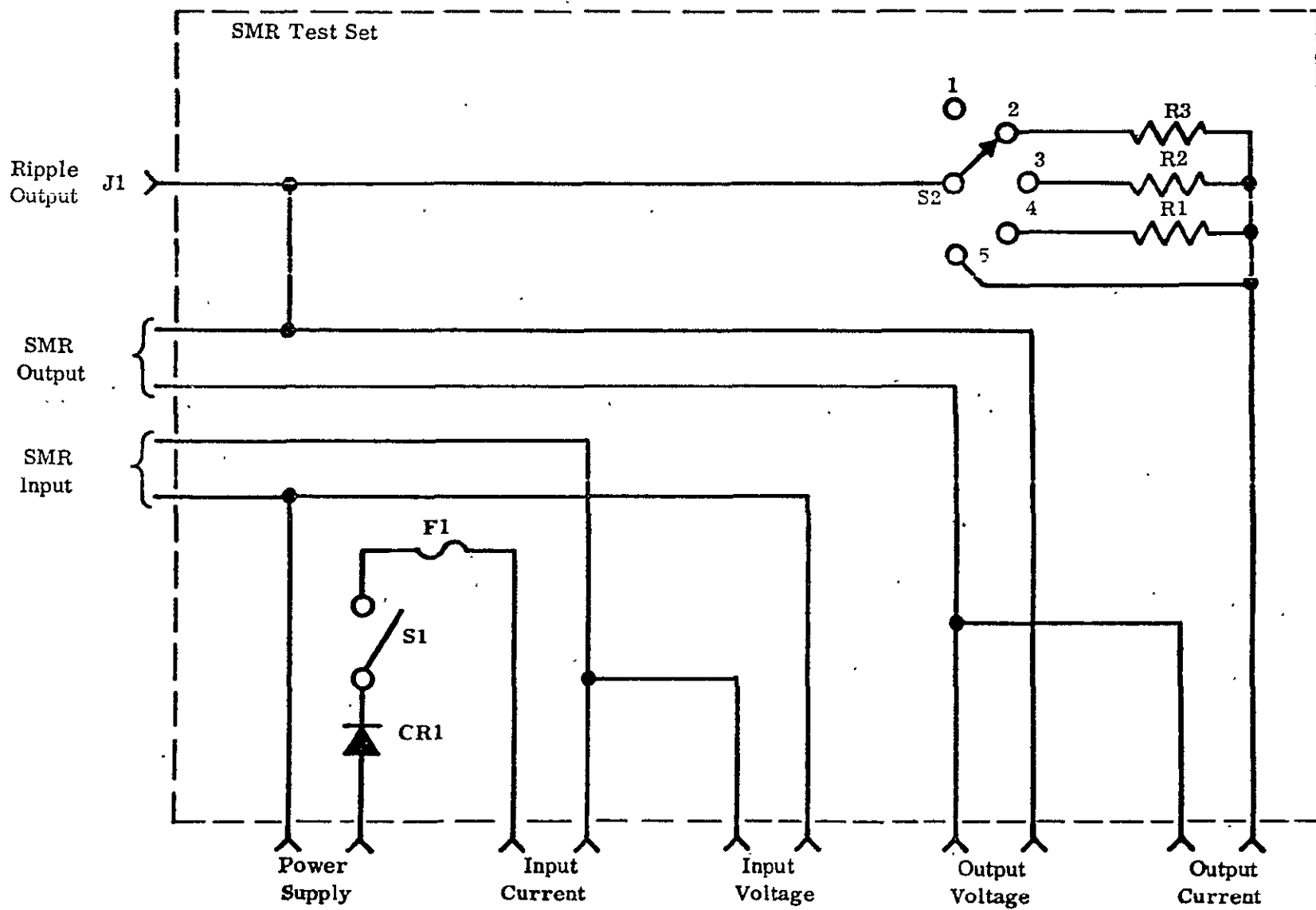


Figure 17.8-1. SMR Test Set Schematic Diagram

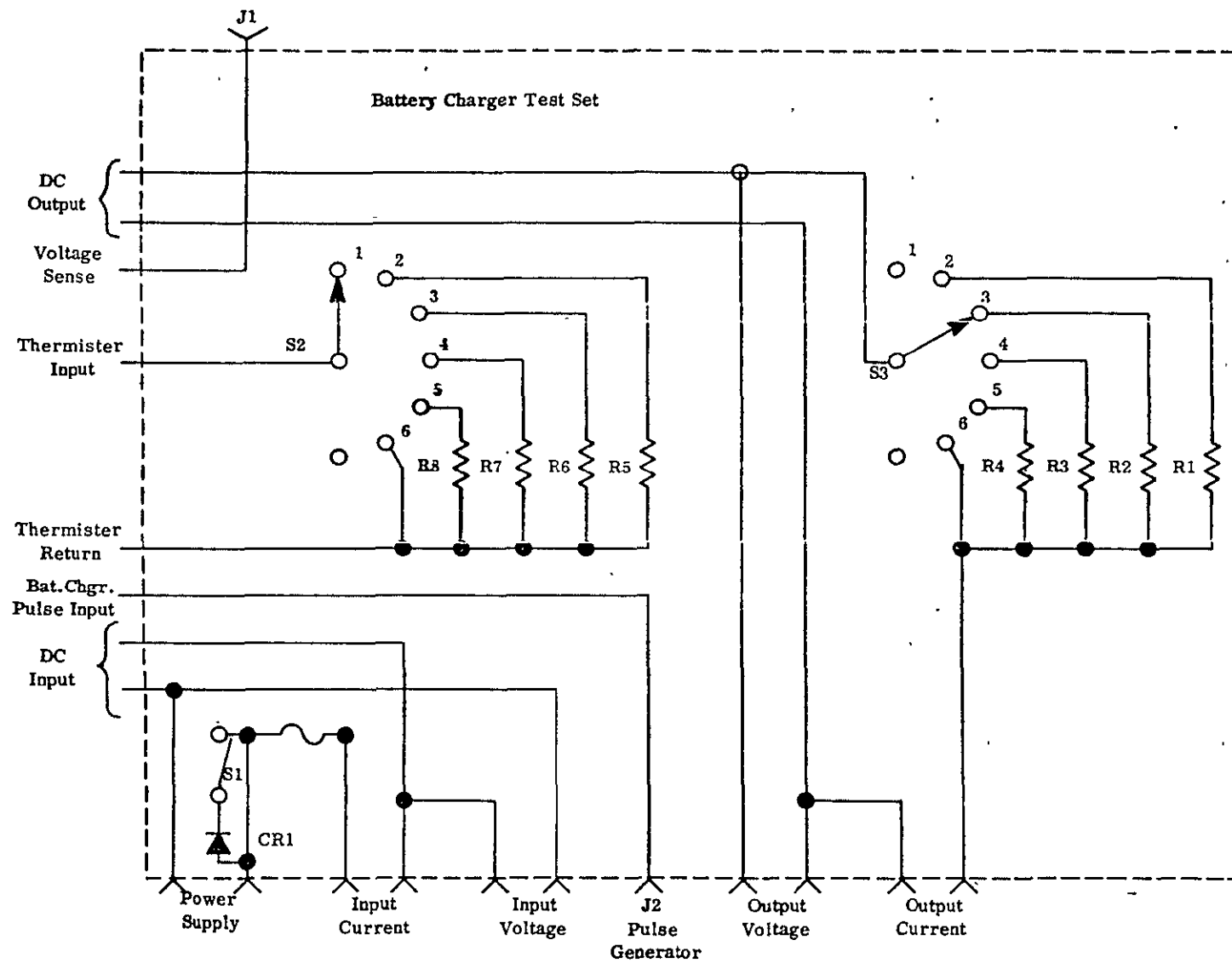


Figure 17.8-2. Battery Charger Test Set Schematic Diagram

17.8.3.1 Physical Description

Figure 17.8-3 is a simplified schematic of the Test Set. The same design will be used for both the 0-15 watt and 15-60 watt LIC except the load resistors will be changed accordingly.

Interconnection of the test set and the LIC is accomplished with a test harness hardwired into the test set at one end and LIC mating connectors on the other. Since all of the LIC connectors are keyed differently, this insures a proper connection. Additionally, with the test set cables hardwired in, the cost of a duplicate set of connectors is deleted.

17.8.4 BATTERY TEST SET

The Battery Test Set provides a test interface to each input and output signal and load to the battery. The test set provides the capability to perform the following tests:

- Individual Cell Voltage
- Total Battery Voltage
- Thermistor Resistance
- Charge-Discharge

Figure 17.8-4 is a simplified schematic of the Test Set.

Interconnection of the test set and the Battery is accomplished with a test harness hardwired to the test set with battery mating connectors attached to the end. Since all of the Battery connectors are keyed differently, proper connection is ensured. Additionally, with the test set cables hardwired in, the cost of a duplicate set of connectors is deleted.

17.8.5 POWER CONTROL UNIT TEST SET

The Power Control Unit (PCU) Test Set provides a test interface to each input and output signal to the PCU. The test set provides the capability to perform the following tests:

- Command and automatic switching
- Battery disconnect
- Battery discharge control
- Telemetry monitor outputs

Figure 17.8-5 is a block diagram of the test setup.

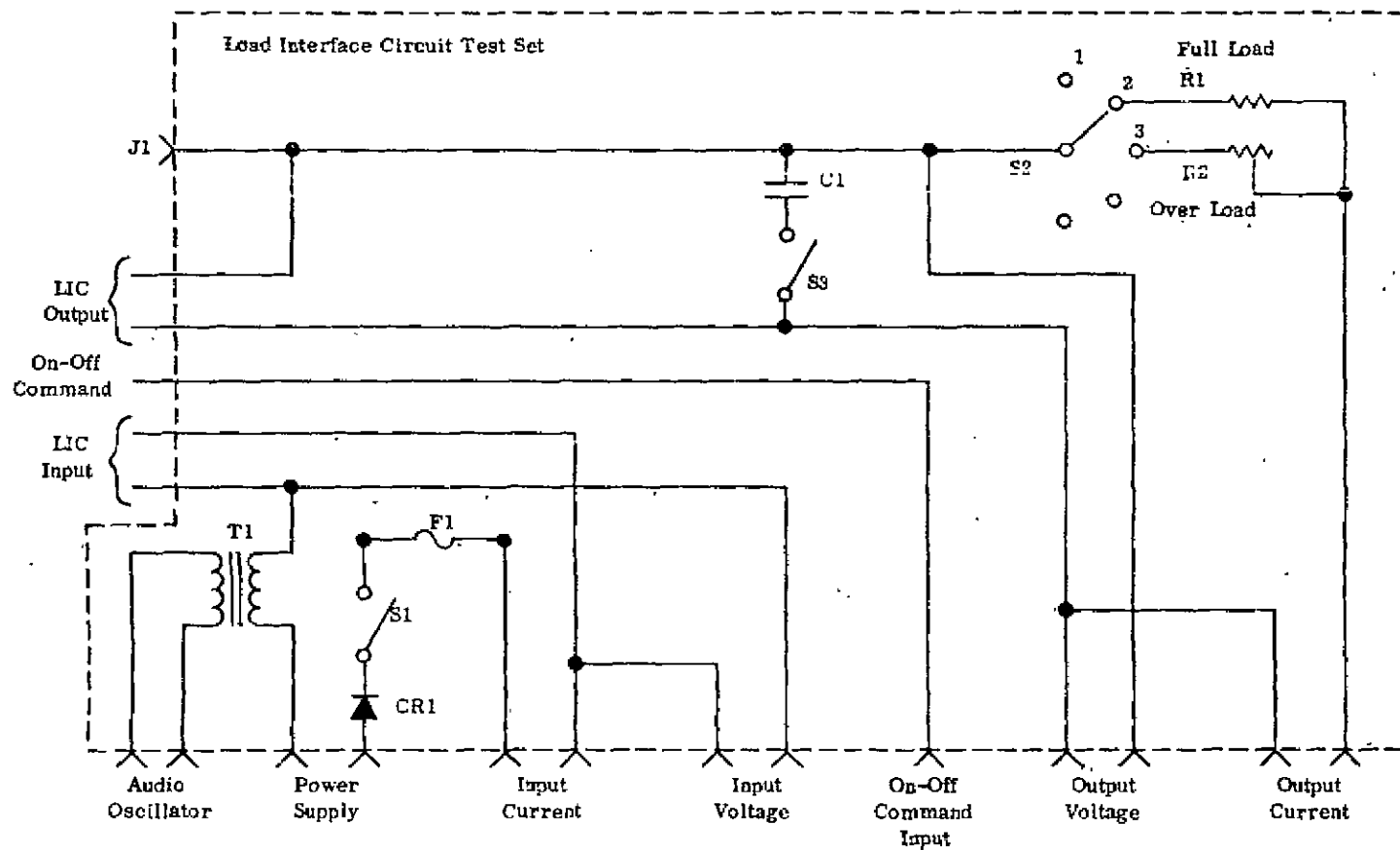


Figure 17.8-3. Load Interface Circuit Test Set Schematic Diagram

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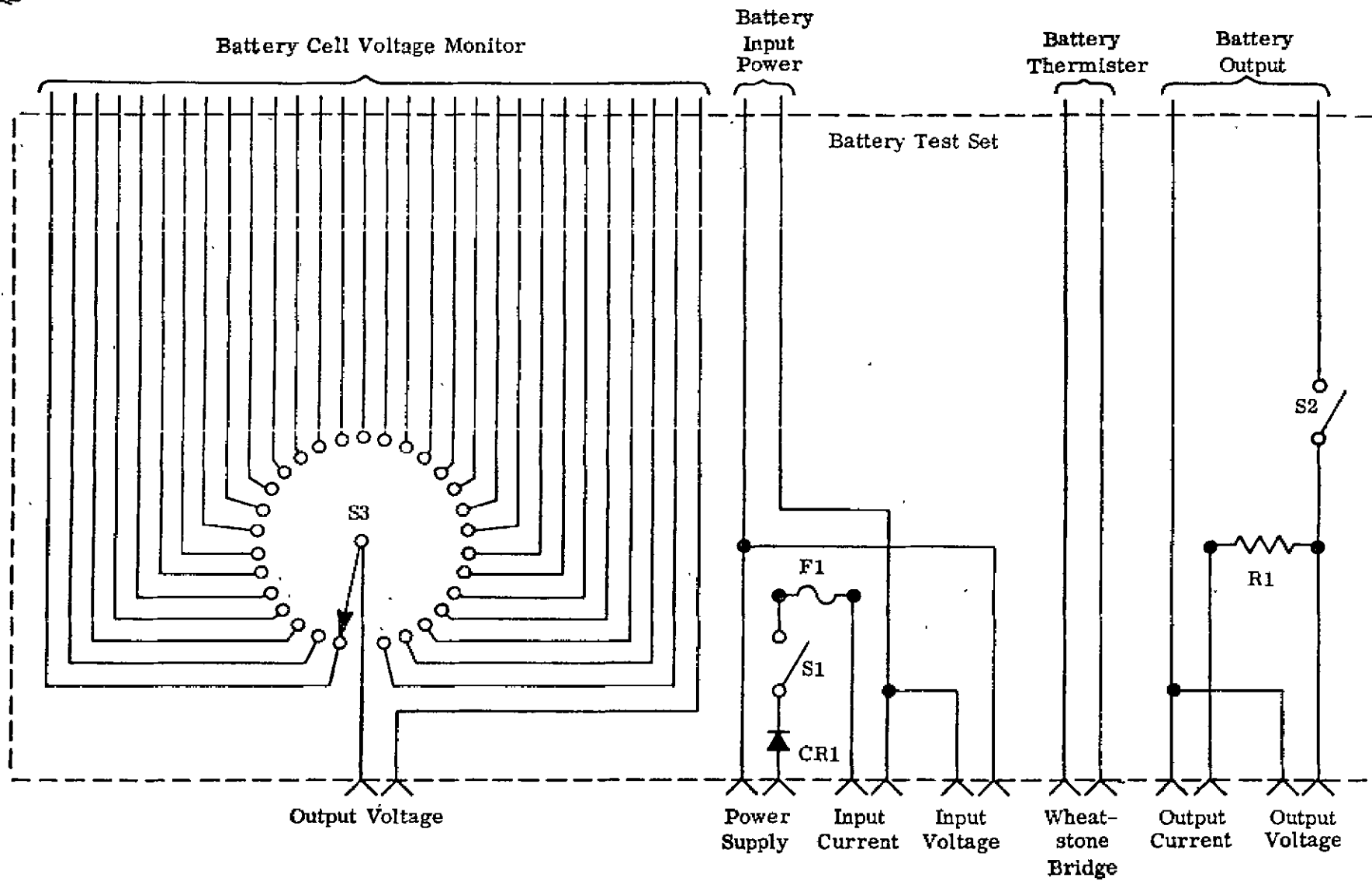


Figure 17.8-4. Battery Test Set Schematic Diagram

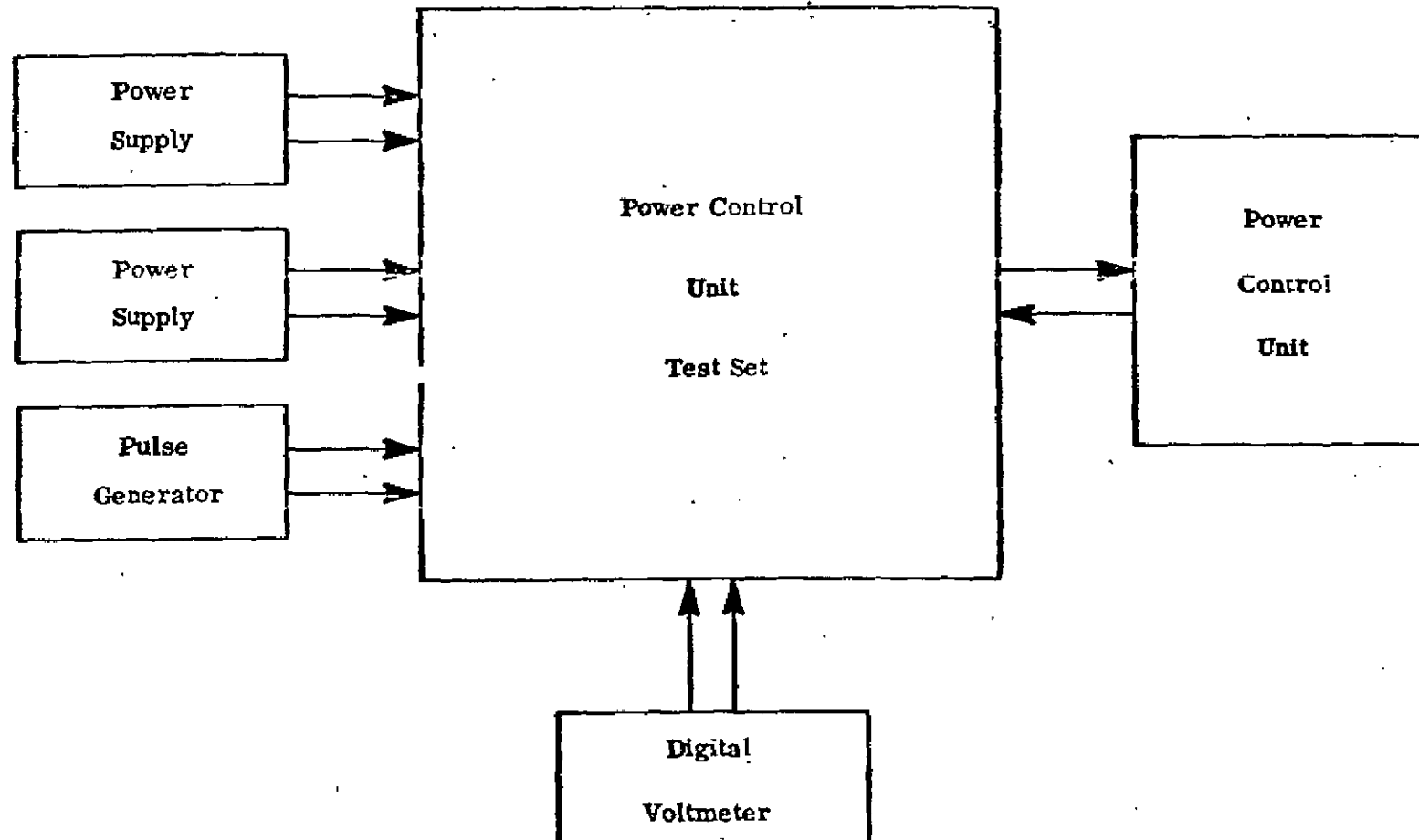


Figure 17.8-5. Power Control Unit Test Set Functional Block Diagram

17.8.5.1

Functional Description

- Inputs - DC power input is supplied by an external power supply connected to the test set at the points provided. Application of power to the PCU is controlled by switch S_1 . Input voltage and current are monitored at provided test points.
- Loads - Output functions of the PCU are loaded as required to simulate operational conditions. In any case where performance is dependent upon operation into a variable load, at least three loading conditions will be verified.
- Signal Inputs - Test points are provided for command inputs to the PCU. Provision will be made to activate each command circuit and function. Command signals are supplied to the test set by an external pulse generator and/or power supply for functions which require a level shift only.

17.9

EXPERIMENT INTERFACE AGE

Experiment interface AGE consists of the Experiment Interface Simulation Unit (EIU). The EIU is single console containing Power, Telemetry and Command Subsystem engineering components, spacecraft experiment loads, a test point panel and a manual command generator. Special features of the EIU are as follows:

- Separate harness for each experiment precluding mispatching.
- Makes maximum use of existing equipment.
- Functional test of the experiment is possible through the actual spacecraft interface with the addition of experiment stimuli/special equipment.
- Experiment tests may be conducted independent of spacecraft operations.

A block diagram of the ISU shown in Figure 7.9-1.

17.9.1

POWER

The EIU power system consist of the following Power Subsystem engineering components.

- Switching Module Regulator (SMR) (1)
- Load Interface Unit (LIC) (2)

Primary power is supplied by the power supply to the SMR. Power is distributed from the SMR to the other spacecraft engineering components incorporated in the EIU and the two experiment LIC's duplicating the spacecraft configuration. Power to the experiments is controlled both by command of the LIC's and a series of front panel switches.

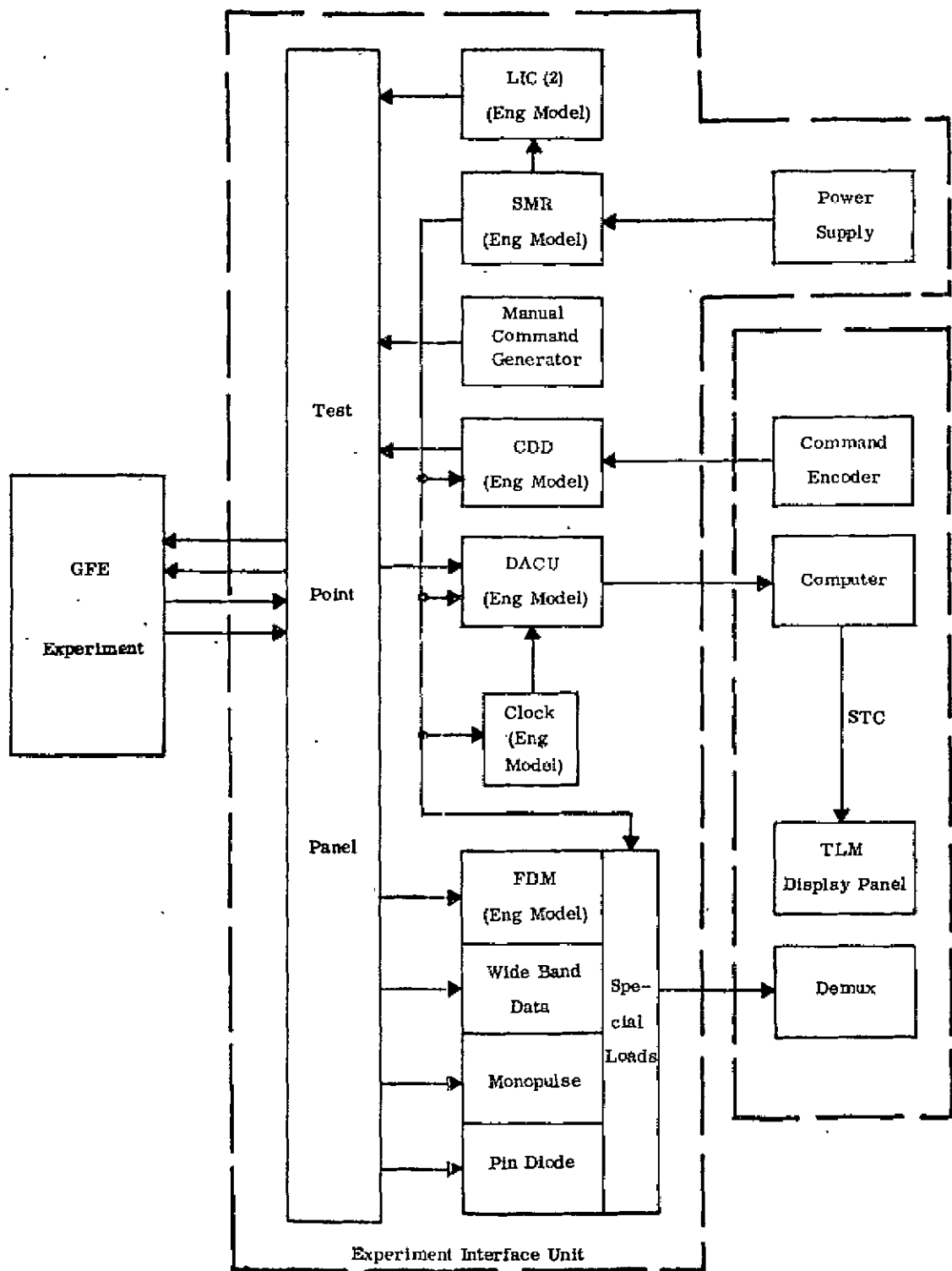


Figure 17.9-1. Experiment Interface Unit Functional Block Diagram

17.9.2 TELEMETRY AND COMMAND

The EIU Telemetry and Command System consists of the following T&C engineering components:

- Command Decoder and Distributor (CDD)
- Data Acquisition and Control Unit (DACU)
- Spacecraft Clock

In addition to the R&C engineering components, a special purpose manual command generator is provided. This unit is required to enable experiment test independent of the System Test Complex (STC).

17.9.2.1 Command Functions

When the EIU is used in conjunction with the STC, the commands are originated at the Spacecraft Control Console Command Encoder. The commands are sent in the same manner as they are sent to the spacecraft except the transmitter and receiver are out of the loop. Commands are received in the CDD and distributed to the proper locations.

Commands to the experiment are also programmed manually by thumbwheel switches at the manual command generator front panel located in the EIU, allowing experiment testing independent of the STC.

17.9.2.2 Telemetry

Experiment telemetry is received by the DACU from the experiment and routed to the Data Operational Processing Equipment in the STC. However, the telemetry may be read out directly from the experiment or the DACU at the test point panel for trouble shooting or for operation independent of the STC.

17.9.3 EXPERIMENT LOADS

Spacecraft experiment loads are provided in experiment load panel. The following special interface loads are provided:

- VHR Camera Wide band ata output 200-800 kHz
- Radiometer Wide band data output 0.1-180 kHz
- Laser Wide band data output 30 Hz- 5 MHz
Monopulse output (time varying dc $\pm$ 10 V)
Wide band data input on TP panel
- MMW Wide band data input on TP panel
- T&FD Pin diode load for modulation
- EME FDM load for PCM data
- RFI RF input/output direct to Exp.

17.9.4 TEST POINT PANEL

A test point panel provides test point access at experiment/spacecraft interfaces. A test point is provided for each wire exiting from the EIU to the experiment. Selected test points are also provided at inter EIU interfaces.

17.10 SPACECRAFT SIMULATORS

Spacecraft simulators will be provided for ATS ground station checkout and personnel training. One each, Telemetry and Command and Transponder simulators will be designed and built as self-contained units. Each simulator will operator from 60 Hertz 115 vac commerical power and contain the necessary circuitry for independent operation and control.

17.10.1 T&C SIMULATOR

FHC will provide one T&C Simulator to be used at the ATS F&G ground stations. The T&C Simulator will be used to functionally checkout the ground station T&C communication links with the F&G model satellites.

The T&C Simulator is packaged n standard nineteen inch equipment racks. The equipment racks are RFI shielded and mounted on casters for mobility. The T&C Simulators operated from a 100-120 vac, 60 Hz, 20 ampere prime power source.

The VHF transmitter, the power supply, the attenuator, and the RF power meter are commercial equipment. The command decoder, the control and display panel, the PCM simulator, and the voice band interface circuitry will be built using IBM Solid Logic Technology, (SLT).

17.10.1.1 T&C Simulators' Capability

The T&C Simulator has the following capability:

- Simulating the ATS F&G decoding function of the unmounted, complex command encoder waveform.
- Displaying the decoded command output.
- Displaying the status of the simulator decoder.
- Generating a PCM serial wavetrain having the same output characteristics as the DACU.
- Integrate into the PCM wavetrain the TM output of the command decoder at the selected word time.
- Wrap the unmodulated voice band output to the VHF downlink transmitter.
- Simulate the VHF downlink by modulating a VHF transmitter with the simulated PCM wavetrain or the buffered voice band signal.

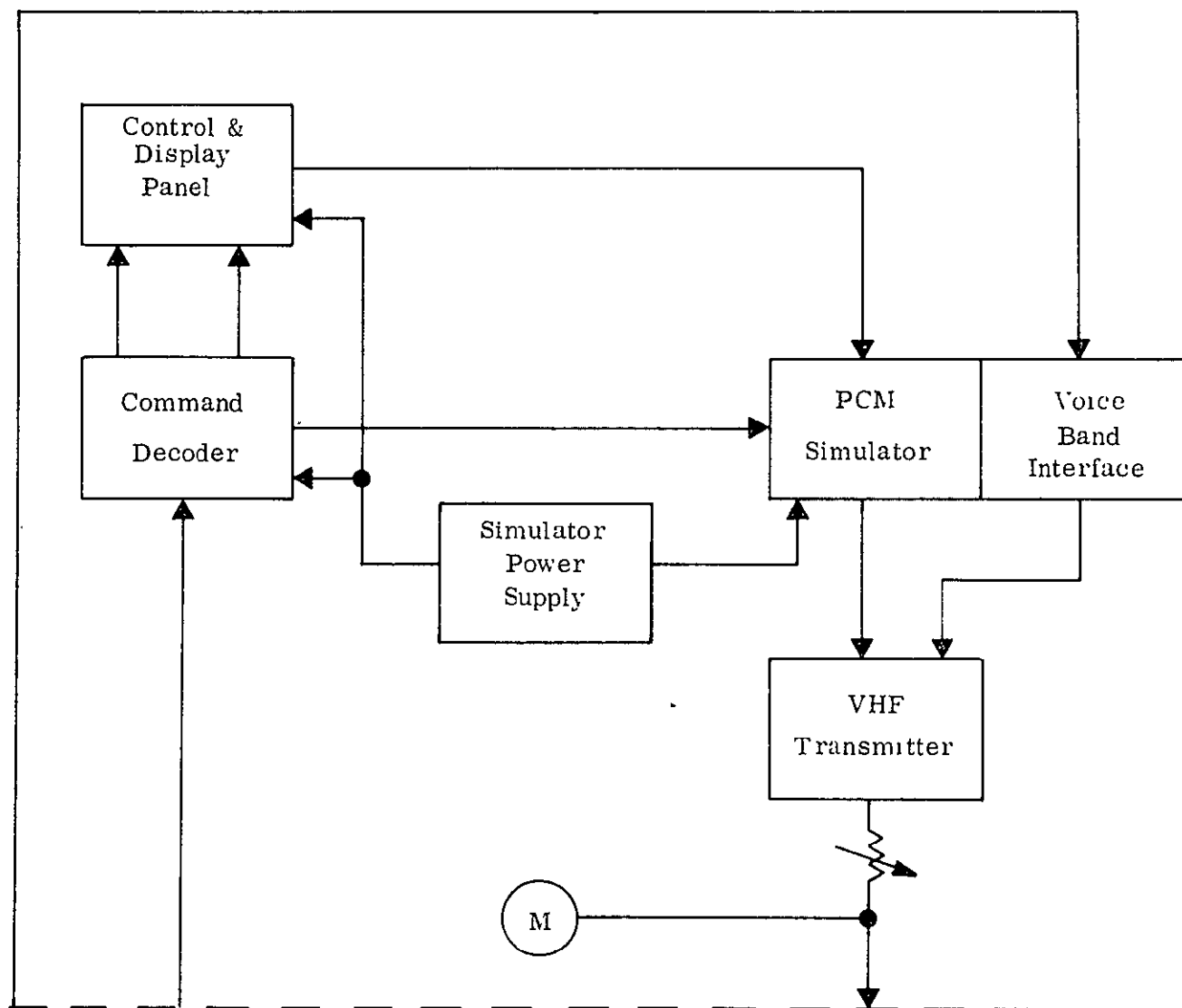


Figure 17.10-1. T&C Simulator Block Diagram

17.10.1.2

Functional Description

A block diagram of the T&C Simulator is shown in Figure 17.10-1. The output of the ground station command encoder will be fed to the T&C Simulator to verify the ground station encoder operation. This signal will be decoded by the simulator command decoder. The format of the command word will be checked, the command address or data word address displayed and the digital data displayed. The portion of the command word which normally is routed to the spacecraft telemetry will be sent to the PCM simulator to be transmitted via the simulator downlink to the ground station. Verification of the command word may either be accomplished by visual comparison, using the control and display panel, or by existing ground station equipment via the simulator downlink.

The PCM simulator will be used to simulate the spacecraft telemetry equipment. The simulator will generate a 400 or 1800 bps, bi-phase pulse train which will be used to modulate the VHF transmitter. The PCM simulator will also integrate into its pulse train the selected outputs of the simulator command decoder. The voice band interface will buffer audio signals received from the ground station and under operator control, using the control and display panel, may be used to modulate the VHF transmitter. The RF output of the VHF transmitter may be monitored and adjusted to the proper output level with the T&C simulator variable attenuator and RF power meter.

17.10.1.3

Detailed Description

17.10.1.3.1

Command Decoder

The T&C Simulator command decoder will be functionally the same as the flight command decoder/distributor, therefore a detailed description will not be given in this section. The construction will be of IBM SLT to enhance maintainability and decrease cost.

17.10.1.3.2

Control and Display Panel

The control and display panel will contain the command displays, the PCM simulator controls, the RF power meter, the RF output attenuator control, and the T&C Simulator prime power control.

The outputs of the command decoder will be routed to the control and display panel. The signals will be buffered, and amplified in order to drive the displays. The following displays will be provided for the command work:

- A discrete lamp to verify that the command word had the proper introduction and conclusion.
- Spacecraft address display which will consist of three octally coded digits to display the spacecraft address.
- A discrete lamp to verify word synchronization.
- Data word address display which will consist of two octally coded numeric displays for displaying the six-bit data word address.

- Discrete command/digital data display which will consist of three octally coded numeric displays for displaying the nine-bit digital word which forms the discrete command address or digital data word.
- Mode indicators will consist of five discrete indicators to display the mode of operation for the decoder. The five indicators will consist of: Hold, Execute, Normal, S/S, and OFF.
- S/S display which will indicate which SAMOC/SAPPSAC tone is being commanded and its duration.

The control and display panel will also provide the necessary controls for the PCM Simulator. These controls will include the following switches:

- Bit rate switch - will control the output bit rate of the PCM simulator. Two positions will be provided: 400 bps and 1800 bps.
- Common word switches - nine toggle switches which will provide the nine bits of data which will appear on all channels except those set aside for either decoder output data and special word data.
- Special word switches - nine toggle switches will provide the nine bits of data which will appear at only the selected word time.
- Special word location switches - eight toggle switches will be provided to specify the word location in the frame. The frame will be specified by the frame select switches of which there will be four toggle switches.

The control and display panel will also provide a switch to control the prime power to the T&C Simulator.

17.10.1.3.3 PCM Simulator

The PCM Simulator provides the capability of generating a simulated bi-phase PCM pulse train. This pulse train will integrate the special data word, frame sync code, the decode output words at the designated word times with a recurring common nine-bit word.

The PCM Simulator will contain a free running clock capable of generating a 3600 or 15,200 bps signal, depending on the control panel setting. This clock will be divided by nine to give bit time, by 128 to give word time, by sixteen to give minor frame time. The decoded output of these divider circuits will be used to serially gate the contents of the special word, decoder TM, sync word, common word registers into the bit stream.

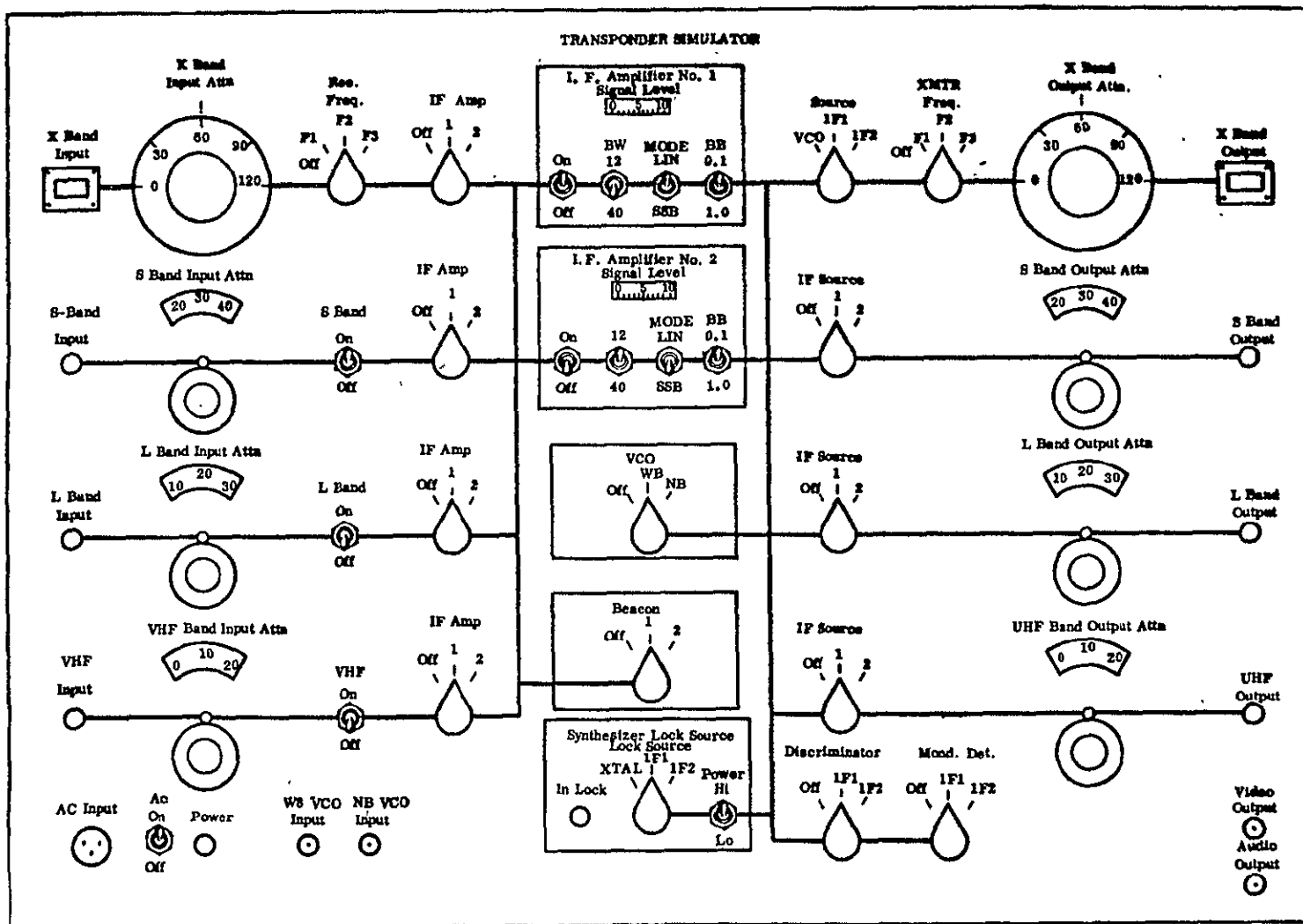


Figure 17.10-2. Transponder Simulator

17.10.2 TRANSPONDER SIMULATOR

The Transponder Simulator accurately simulates the characteristics of the Communications Transponder used in the ATS F&G Satellite.

The Transponder Simulator is a portable, self-contained ground test unit consisting of spacecraft engineering components augmented by control and display units. The components are housed in an RFI shielded enclosure. The RF interfaces shall be RF connectors on the front of the enclosure. An illustration of the front panel configuration is shown in Figure 17.10.2

17.10.2.1 Transponder Simulator Capabilities

The Simulator provides the RF signals required to test and calibrate ground stations associated with the ATS F&G Satellite. The Transponder Simulator provides complete simulation of all RF interfaces with ground link terminals. Attenuators are provided on all RF inputs and outputs for adjustment of signal levels. The operating characteristics will be as follows:

- RF Input Frequencies:

8025, 8150, 8250 MHz	X-band
2247 - 2253 MHz	S-band
1650 MHz	L-band
150 MHz	VHF band
- RF Output Frequencies:

7350, 7575, 7650 MHz	X-band
1797-1803 MHz	S-band
1550 MHz	L-band
850 MHz	UHF band
- Output Power:

1 milliwatt minimum -	all bands
- Operating Modes:

Frequency translation wideband data mode.
- Modulation Inputs, Video: 30 Hz to 4.5 MHz
Exp: 0.1 Hz to 450 kHz

17.10.2.2 Detail Description

The Transponder Simulators will consist of refurbished Engineering Model components and simulator peculiar parts packaged in RFI shielded containers for field use. The simulator will be powered from internal power supplies operating from 115 volts, 60 cycle power. The simulator will contain any redundant components or TWT amplifiers. The output will be taken directly from the up converter with an output power of one milliwatt minimum. See Figure 17.10-3 for a block diagram of the Transponder Simulator.

The input signals are fed to input attenuators, one for each band. Attenuation is adjustable over a range of at least 50 db. The attenuator output is fed to the down converters, 3 for X-band and one for each S-band and L-band. The VHF signals require only filtering and amplification. The converter and amplifier outputs are fed to the IF amplifiers via the IF distribution network which is manually controlled by switches on the front panel. Switches also select one of the three X-band input frequencies.

Each of the two IF amplifiers performs the following functions:

- Selects one of two initial bandwidths, 40 MHz or 12 MHz.
- Selects one of two post detection bandwidths, 1 MHz or 100 kHz.
- Selects one of two modes, linear or single sideband.
- Provides automatic gain control and indicates signal level on a panel meter.
- Provides a signal to the frequency synthesizer to enable coherent operation with any input signal.
- Amplifies beacon input signals.

The outputs of the IF amplifiers are fed to a second IF distribution network which selects any one of the six up-converters or the monopulse detector. After up-conversion, the signal is fed to the output attenuators, one for each band.

The X-band up-converter can manually select either of the two IF amplifiers or the VCO as its signal source. A second manual switch selects either of three output frequencies or turns it off.

The S-band, L-band and UHF band up-converter can manually select either of the two IF amplifiers as its signal source or turn off.

The discriminator can select either of the two IF amplifiers or turn off.

In the monopulse mode, the IF signal is switched to the monopulse detector which extracts the Apollo voice link.

In the output wideband data mode, a VCO is modulated with the wideband data and then switched to the selected X-band up-converter.

The beacon is manually controlled to either of the two IF amplifiers or turned off.

The synthesizer can select as its lock source either of two IF amplifiers or a crystal. An indicator light indicates when the synthesizer is in lock. A switch also provides for selection of either the high or low power mode of operation of the synthesizer.

A power supply operating from 115 volts AC 60 Hz supplies DC power to the DC-DC converter (a spacecraft component). The DC-DC converter supplies the various voltages required to the down converters, IF amplifiers, up converters, VCO, monopulse detector, synthesizer, etc. A control switch on the front panel of the Transponder Simulator controls the AC power on or off. An indicator light illuminates when the AC power is on. The on-off switch is a circuit breaker which limits AC input power, thus providing a dual function.

RFI filters will be used on the power input lines to prevent RFI via the power line. The shielded container which houses the Transponder Simulator will reduce stray emissions to a very low value. The case will be approximately 30 X connections will be provided for the X-band input and output.

Separate coax connectors will be provided for inputs and outputs at the other three bands. Coax connectors also provide for wide band and narrow band VCO inputs, video output from the discriminator and audio output from the monopulse detector.

Antennas will not be provided unless specifically requested. In that case, spiral type antennas will be used which provide broadband frequency coverage and circular polarization. Separate antennas will be provided for each band and for inputs and outputs; a total of eight antennas. This is a more economical approach than using diplexers to combine the receiver and transmit functions on one antenna.

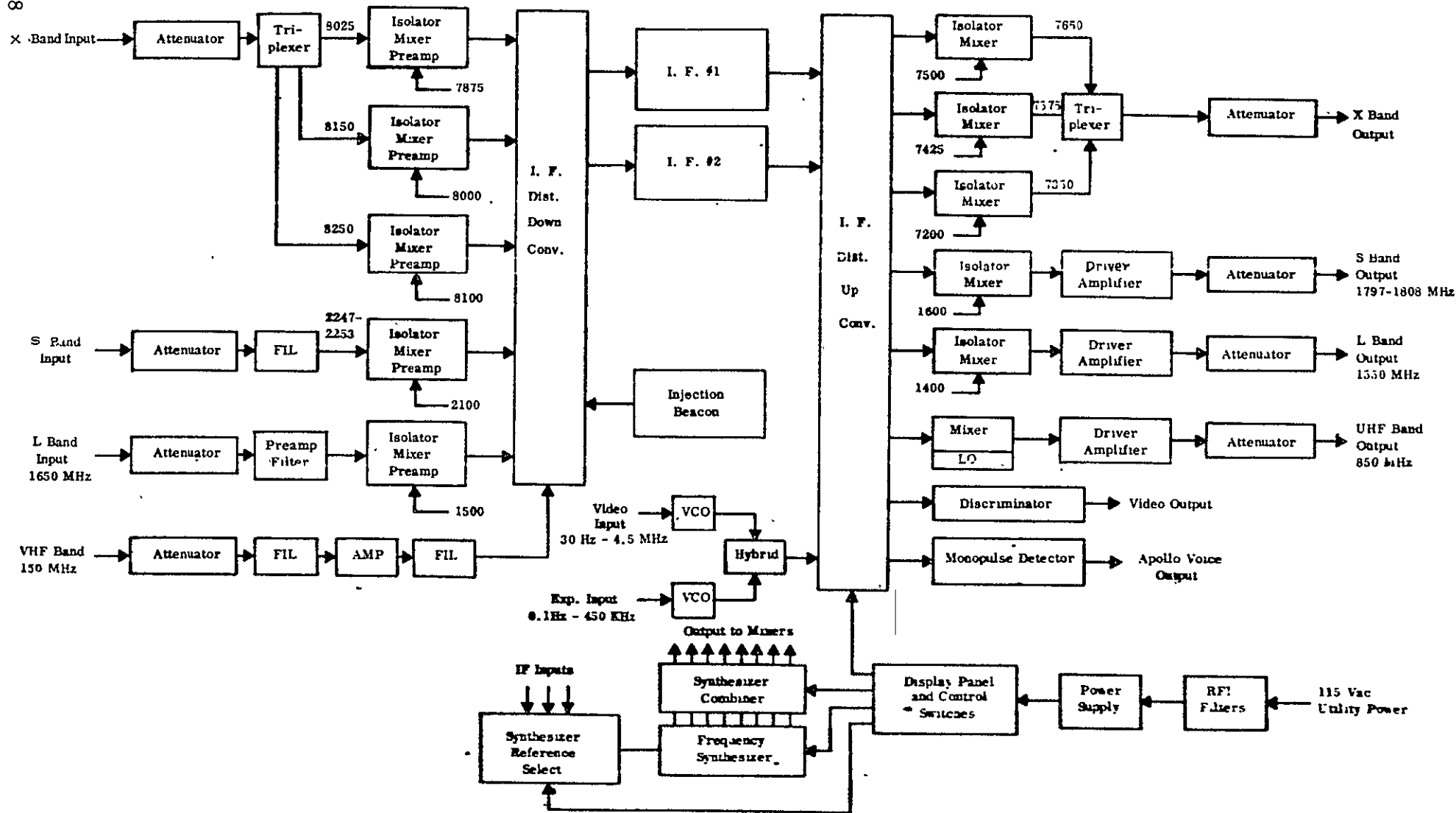


Figure 17.10-3. Transponder Simulator Block Diagram

SECTION XVIII

TEST PLANS

18.1 INTRODUCTION AND SUMMARY

This section presents Fairchild Hiller Corporation's (FHC) plans for the implementation of the ATS F&G Integrated Test Program. The Integrated Test Program is designed to encompass every test conducted during Phase D on all test specimens, parts, subassemblies, assemblies, components, subsystems and systems, on engineering, prototype, and flight hardware.

18.1.1 TEST PROGRAM OBJECTIVES

The primary objectives of the Integrated Test Program are as follows

- Obtain engineering data as required to verify or establish design concepts/details early in the program.
- Quality the ATS F&G spacecraft for flight.
- Demonstrate that the flight spacecraft is free from workmanship defects and ready for launch.

A fourth primary objective of the test program, which is implicit in the first three, is the establishment of a high degree of confidence that the spacecraft will achieve all mission objectives including a non-degraded life of at least two years in orbit with a five year design goal.

18.1.2 TEST PROGRAM APPROACH

Fairchild Hiller Corporation's approach to the ATS Test Program is illustrated and summarized in the functional flow block diagrams of Figure 18.1.1 (Thermal Structural Model and Prototype Spacecraft) and Figure 18.1-2 (Flight Spacecraft). The most significant characteristic of these plans is the degree to which FHC is able to take advantage of the modularized design concept of the ATS F&G spacecraft, both at the spacecraft and EVM levels. As illustrated by these flow diagrams, the overall spacecraft can be considered to be composed of, and tested as two lesser spacecraft assemblies: a relatively conventionally-structured spacecraft, housing integrated electronic subsystems and an active Thermal Control System (EVM); and a large open truss-boom spacecraft assembly which houses a 30-foot deployable

reflector, supports two deployable 208 square foot solar array assemblies, and contains relatively few electronic components (Tower).

The Tower, in turn, also can be considered to be composed of two major spacecraft elements, a Structural Subsystem and a Reflector Subsystem.

Treating these two widely different spacecraft assemblies separately for the major portion of the testing offers the following obvious advantages:

- Enables concurrent test flows, thereby reducing test program schedules with corresponding cost savings
- Enables the environmental testing of the integrated electronics subsystems (EVM) in conventional, smaller, and less costly test facilities
- Minimizes handling problems which would otherwise be prevalent during the entire program.

18.1.3 TEST APPROACH

It should be noted that the EVM, Tower, and Reflector assemblies are mated during the test flow to verify interfaces and alignment, and qualify the fully assembled ATS spacecraft for the launch vibration environment.

In a similar manner, the modular design of the EVM enables parallel subassembly test and integration efforts for the major subsystems. In addition to reducing total flow time, this approach enables subsystem integration to take place in the subsystem contractor's home plant, where equipment and trained personnel are concentrated and readily available.

Other test concepts employed by Fairchild to most effectively meet test program objectives include:

- Utilization of thermal-structural model (TSM) EVM to obtain early verification of EVM Structural Subsystem and Thermal Control Subsystem design adequacy
- Use of TSM/EVM to verify design adequacy and quality the prototype tower spacecraft structural assembly
- Use of TSM/EVM in conjunction with prototype tower assembly to verify Titan interface, demonstrate separation subsystem design adequacy, and establish reflector-feed alignment and antenna patterns



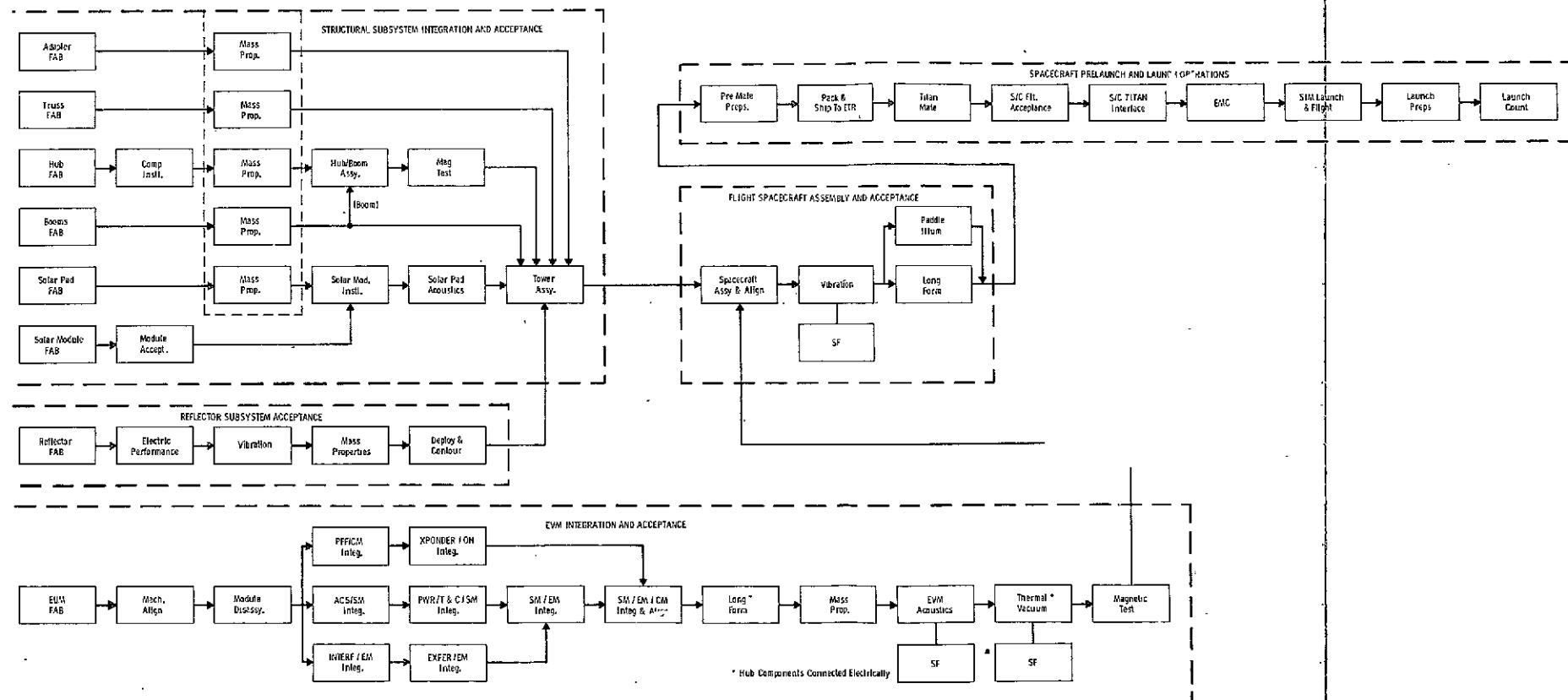


Figure 18.1-2. Functional Flow Block Diagram Flight Spacecraft Integrated Test Program

18.1.4 TEST PROGRAM DESCRIPTION

The Integrated Test Program is divided into the following major categories:

- Engineering Development Tests
- Qualification Tests
- Flight Acceptance Tests

These test program categories are briefly discussed in the following paragraphs.

18.1.5 ENGINEERING DEVELOPMENT TEST PROGRAM

The purpose of conducting engineering development tests is to fulfill the objectives of paragraph 18.1.1 and obtain engineering data to verify or establish design concepts/details early in the program.

18.1.5.1 Engineering Components Test

Engineering model components are fabricated and subjected to test where it is deemed necessary to establish and/or verify design adequacy prior to the manufacture of prototype and flight components. Components which are of a new design and have not previously been flight tested or qualified fall into this category. In general, engineering model components will not be suitably fabricated for environmental exposure in order to reduce fabrication time and costs and proceed more quickly into performance evaluation tests. Selected environmental tests are conducted on components only where a specific need for early testing is established.

Since only selected engineering components are fabricated, no engineering model subsystems, per se, exists. In order to verify satisfactory subsystem performance prior to assembly of complete prototype subsystems, the engineering components are integrated with prototype components at the subsystem level. Functional integration and performance is conducted on the hybrid subsystems.

Fairchild Hiller Corporation's modularized design concept for the EVM, wherein each of the three major modules are integrated separately, eliminates the need for the engineering model systems level equipment demonstrator (SLED) previously considered by Fairchild in order to functionally integrate and evaluate the overall electronics system. This is due to the fact that major portions of systems integration are accomplished prior to final assembly and integration at FHC.

18.1.5.2 Thermal-Structural Model Tests

The primary objective of the thermal-structural model test program is to demonstrate the adequacy of the EVM Thermal Control Subsystem early in the

program, and conduct design verification and pre-qualification tests on the overall spacecraft structural subsystem.

To meet these objectives effectively, a thermal-structural model (TSM) of the EVM is fabricated and used along, and in conjunction with the tower assembly (adapter, truss, hub, booms, and paddles), to conduct the thermal-vacuum and structural tests shown in the functional block diagram of Figure 18.1-1.

The TSM/EVM is a near second prototype EVM which includes heat pipes and operational thermal louvers. EVM components are simulated by thermal-mass models. The TSM/EVM is subjected to a thermal balance test to evaluate the performance of the Thermal Subsystem and provide data for comparison with thermal math model study results.

The structural elements of the spacecraft are fabricated and tested individually and at specific points of assembly for design verification. For example, a solar paddle (with dummy solar-cell modules) is attached to a boom and subjected to "zero-G" boom deployment tests in order to verify design adequacy and demonstrate reliability of the boom deployment subsystem. The GFRP truss, in conjunction with the hub, adapter, and TSM/EVM is subjected to a static loads test to determine influence coefficients and measure local loads. Mass properties are determined by obtaining measurements on individual elements and in combination, and calculating the final results.

Upon completion of these tests, the tower assembly and the TSM/EVM are mated and subjected to launch vibration testing to verify design integrity, measure resonant frequencies, and determine critical loads.

Subsequent to these tests, the TSM spacecraft is used as a test tool in order to accomplish the following tests (Figure 18.1-1):

- Titan Interface (with transtage simulator)
- Separation (with transtage mass dummy)
- Reflector - PFF Integration (with proto reflector and PFF)
- Deployed Mode Shape Vibration (with prototype reflector)

Upon completion of this test sequence, the tower assembly could be refurbished as required to serve as the prototype tower during the qualification test program.

18.1.6 INTEGRATION, DESIGN VERIFICATION, AND QUALIFICATION OF THE EVM

After fabrication, assembly, and alignment of the prototype EVM structure, it is disassembled for concurrent installation and integration of the individual modules. The Communication Module (CM) is integrated and tested at Philco-Ford, the Service Module (SM) is integrated and tested by Honeywell (less the T&C and Power Components), and IBM install, aligns, and integrates the interferometer

in the Experiment Bay (EB). FHC then installs and integrates the T&C and Power Subsystems in the SM and installs and integrates the GFE experiments and other outstanding components in the EB. The SM and EB are then integrated. The CM is integrated with the SM/EB to complete the integration of the prototype EVM. The EVM is then subjected to the long form, or functional acceptance test, to verify satisfactory systems performance (the hub electronic components are electrically connected to the EVM for this test).

Upon completion of the long form test, the EVM's Mass Properties are determined and the unit is exposed to acoustic environment.

At this point in the test flow, the EVM could be mated to the tower structure for all-up spacecraft testing, or could proceed directly into thermal-vacuum and magnetic testing. The first approach provides for environmental exposure in accordance with the launch sequence; the second would be less costly since it would avoid one extra mate-demate sequence. Another consideration which favors the first approach is that the vibration test is a shorter and less expensive test than the vacuum test. Accordingly, it would be less costly to repeat the vibration test if a subsequent failure occurred during vacuum testing than it would be to repeat the vacuum test if a failure occurred in a subsequent vibration test. It is also desirable to conduct the magnetic test after vibration. The test flow shows vibration prior to thermal-vacuum. The spacecraft vibration test is discussed in the following paragraph. The remainder of the EVM-alone testing is covered in this paragraph.

The thermal-vacuum test is performed on the EVM alone for the following reasons:

- There is no technical requirement to subject the fully-assembled spacecraft to thermal-vacuum environment
- The facility support and operational requirements to conduct a thermal-vacuum test on the fully-assembled spacecraft are prohibitively expensive
- The ATS spacecraft can be demonstrably qualified for thermal-vacuum by selective testing of spacecraft subassemblies and components

Upon completion of the thermal vacuum test the EVM is transported to GSFC in order to measure its magnetic field as required by the presence of the magnetometer experiment.

18.1.7 QUALIFICATION OF ATS SPACECRAFT ASSEMBLY

Upon completion of the concurrent qualification testing of the major spacecraft elements discussed above, the all-up ATS F&G prototype spacecraft is assembled, integrated, aligned, and qualified for the launch vibration environment.

Only the T&C and other specific components which are powered for launch are powered during this test. A short form (functional) test is performed after exposure to verify no degradation in performance.

A separation shock test is conducted after vibration, utilizing much of the vibration test instrumentation. This test is conducted by suspending the spacecraft and firing separation squibs to drop the adapter and simulated transfer.

The reflector is deployed after the shock test to verify mechanical integrity (deployment reaction loads will have been checked during the initial deployment for the deployed mode shape vibration survey).^{*} The spacecraft is then subjected to a self-induced RFI/EMI test whereby various combinations of RF links are radiated through the feed system to determine if the EVM performance is adversely affected by the near field of the reflector.

Upon completion of the EVM thermal vacuum and magnetic tests, the spacecraft is reassembled and the final long form test conducted. Test results will be compared with the results of the initial long form tests to verify no degradation of performance.

The final operation on the prototype spacecraft is the Titan mate-demate demonstration. The purpose of this test is to verify mate-demate procedures prior to mating of the flight spacecraft on the launch pad. The prototype spacecraft is shipped in the spacecraft shipping container to Cape Kennedy for this purpose, thereby concurrently validating the shipping container and shipping route prior to delivery of the flight spacecraft. This test will be conducted 30 to 60 days prior to launch, dependent upon Titan pad schedule requirements (the ATS Titan need not be erected for this test; any Titan with a standard transtage interface can be utilized).

18.1.8 FLIGHT ACCEPTANCE TESTS

The purpose of conducting the flight acceptance test program is to verify that the flight components, subsystems, and spacecraft are free from workmanship defects and ready for launch. The tests performed at the spacecraft level are presented in the functional flow diagram of Figure 18.1-2.

18.2 ENGINEERING DEVELOPMENT TEST PROGRAM

18.2.1 GENERAL

The development test program provides a means for early verification of component and subsystem performance. Confidence in the design adequacy is obtained as soon as practical in the program to add confidence to the prototype qualification program which follows.

^{*} It is noted that the prototype reflector is deployed twice at FHC. The flight reflector is not deployed at all at FHC.

Considerable judgement goes into deciding if the complexity of design warrants the fabrication and test of an engineering model. In addition, judgement must influence the state of the design which is released for engineering model fabrication. The later the engineering model is fabricated, the more likely the design is similar to prototype design. However, the later the engineering model is tested, the less useful the test results can be in providing timely design changes for incorporation in the prototype design. The converse of this rationale is equally true; if the engineering model is fabricated too early in the design phase, it is less likely to represent the final design and provides less useful test data.

The development test program described in this section was established in consideration of those factors as well as program cost and schedule considerations. This program is summarized in Table 18.2-1. As exhibited by this table, major testing in this category is conducted on the engineering model components and the structural-thermal model EVM. These tests are discussed below.

18.2.2 SUBSYSTEM DEVELOPMENT TESTS

In general, engineering model components will not be fabricated suitable for environmental exposure to reduce fabrication time and costs and proceed more quickly into performance evaluation tests. Selected environmental tests are conducted on components only where a specific need for early testing is established.

Since only selected engineering components are fabricated, no engineering model subsystems, per se, exists. In order to verify satisfactory subsystem performance prior to assembly of complete prototype subsystems, the engineering components are integrated with prototype components at the subsystem level and functional integration and performance testing is conducted on the hybrid subsystems.

A brief description of the highlights of each subsystem development test plan is provided in the following paragraphs.

18.2.2.1 ACS

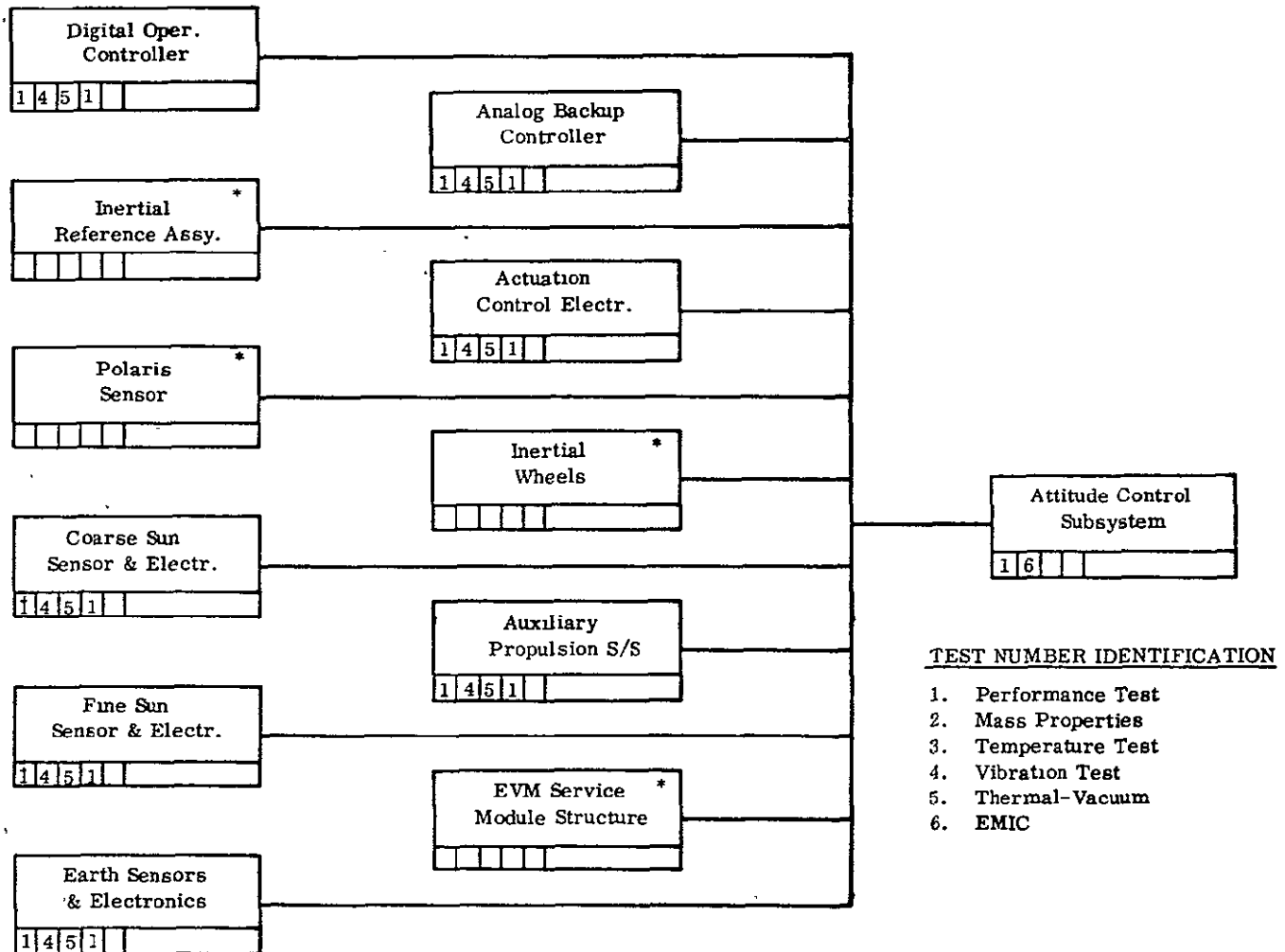
The ACS consists of many sensors functionally interfacing with the DOC or ABC which, in turn, provides actuation commands to the ACE for jet or wheel control. Development tests for these areas are provided in Figure 18.2-1.

18.2.2.2 APS

The APS is an ammonia expellant subsystem which is based largely on techniques developed during ATS D&E by AVCO Corporation. The general areas of concern are associated with the verification expellant flow at low temperatures and verification of the mechanical design of the APS. The development test plan summary is provided in Figure 18.2-2.

Table 18.2-1. Subsystem/Component Development

TEST SUMMARY					
	Performance Test	Vibration Survey	Temperature	EMIC	Thermal-Vacuum
Attitude Control Subsystem	X				
DOC	X	X			X
ABC	X	X			X
ACE	X	X			X
Inertia Wheel (Drive Electronics)	X				
Earth Sensor	X	X			X
Fine Sun Sensor	X	X			X
Coarse Sun Sensor	X	X			X
C/F Sun Sensor Elect.	X	X			X
Auxiliary Propulsion Subsystem	X	1			X
Interferometer	X				
Communication Subsystem	X			X	
Transponder	X	X	X	X	
Prime Feed Assembly	X				
Telemetry & Command Subsystem	*				
DACU	X				
DSU	X				
Spacecraft Clock	X				
CD/D	X				
* Less RF Components					
Radio Beacon Experiment	X	X		X	
Gravity Gradient Experiment	X	X		X	X
Power Subsystem	X				
LIC	X				
SMR	X				
Battery	X				
Battery Charger	X				
Solar Array	X				



* Engineering model not fabricated

Figure 18.2-1. Attitude Control Subsystem Development Test Summary

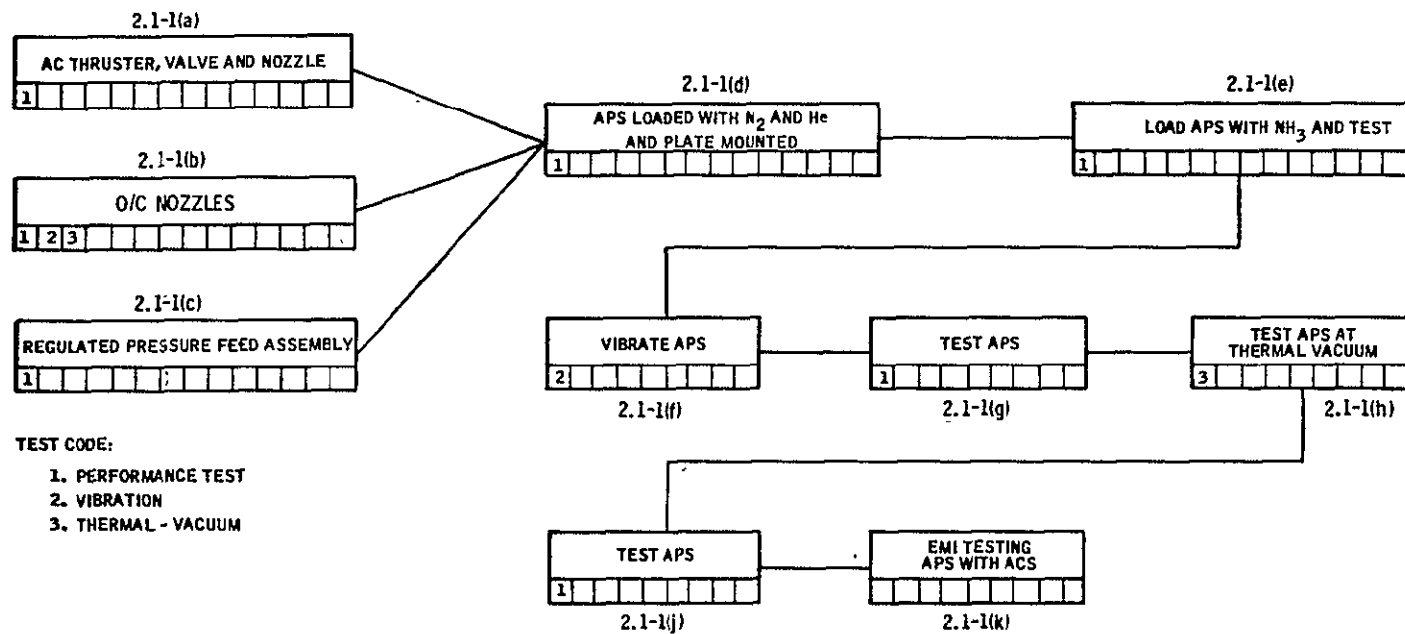


Figure 18.2-2. APS Development Test Summary

18.2.2.3 Interferometer

The Interferometer consists of two components; the array assembly and the receiver/converter. The development test program for the Interferometer includes engineering model components for both the array and the receiver/converter. Development tests include performance verification at both the subsystem and component level. Alignment and calibration procedures and accuracy are critical areas for the Interferometer. Development tests are proposed to develop the techniques for aligning the array front end for zero error on axis and for performing calibration tests accurate to better than 0.02 degree.

18.2.2.4 T&C Subsystem

The T&C Subsystem is made up of two groups of components; a digital group and an RF group. The RF group consists of hardware previously developed and qualified for other space programs. Thus, for the RF group engineering model, components are not fabricated or development tested.

The digital group is made up of the DCAU, DSU, CDD and Spacecraft Clock. These units require development tests on engineering model hardware to isolate potential noise coupling and logic race conditions. The development test plan summary is shown in Figure 18.2-3.

18.2.2.5 Communication Subsystem

The Communication Subsystem has two primary functional elements, the Transponder and Prime Feed. The Transponder is made up of six RF up-links and six RF down-links including combinations of simultaneous operation. The development of tuning alignment procedures and the verification of design performance is required early in the program to provide confidence in the prototype design. An engineering model of the Transponder is fabricated without redundant components. (Interface circuits with the redundant components are provided with appropriate loads). An engineering model feed assembly is required to verify mechanical alignment procedures and RF patterns. The Communication Subsystem is tested during the engineering development phase of the program to verify the design with respect to EMIC and to isolate potential vibration problems. The EMIC test is expected to produce limited results due to the non-redundant nature of the Transponder but gross EMIC problems are isolated during this test. The Communication Subsystem development test program is summarized in Figures 18.2-4 and 18.2-5.

18.2.2.6 Power Subsystem

The components of the Power Subsystem are interconnected to almost every spacecraft component and as such, represent a critical system function from a system performance point of view. The noise levels of the switching mode regulators and the transient interrelationships between Power Subsystem components require the fabrication of a non-redundant Power Subsystem in order to provide early verification. These components provide an operating subsystem to the extent that

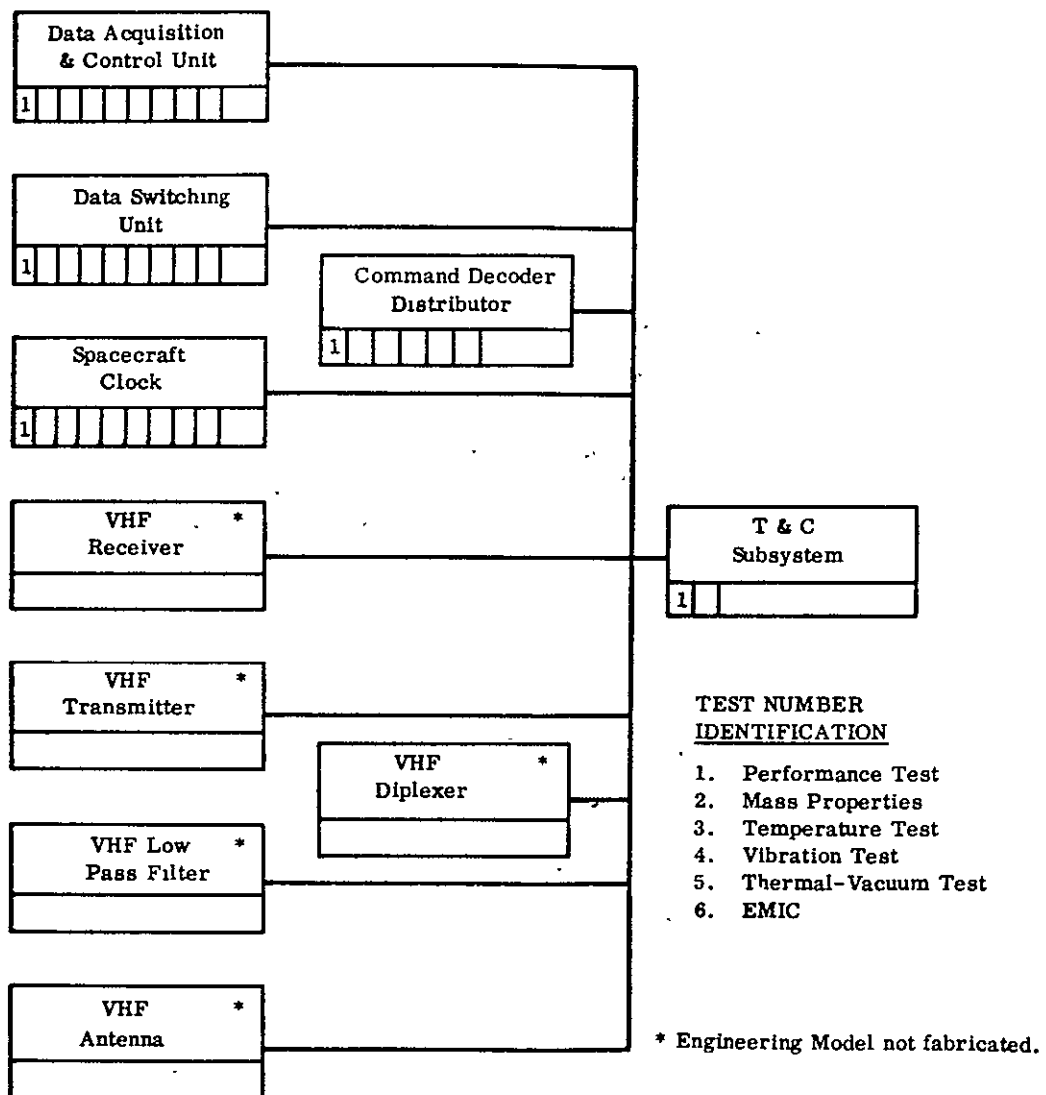
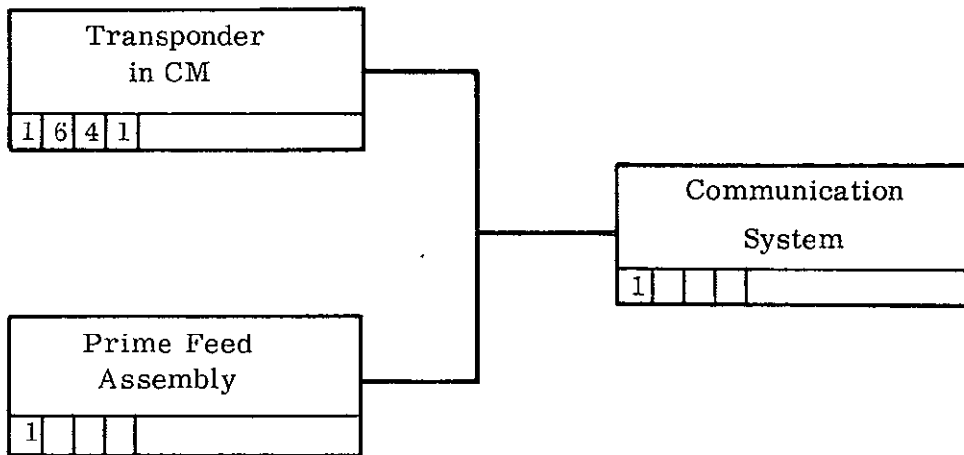


Figure 18.2-3. T & C Subsystem Development Test Summary



**TEST NUMBER
IDENTIFICATION**

1. Performance Test
2. Mass Properties
3. Temperature Test
4. Vibration Test
5. Thermal-Vacuum Test
6. EMIC

Figure 18.2-4. Communication Subsystem Development
Test Summary

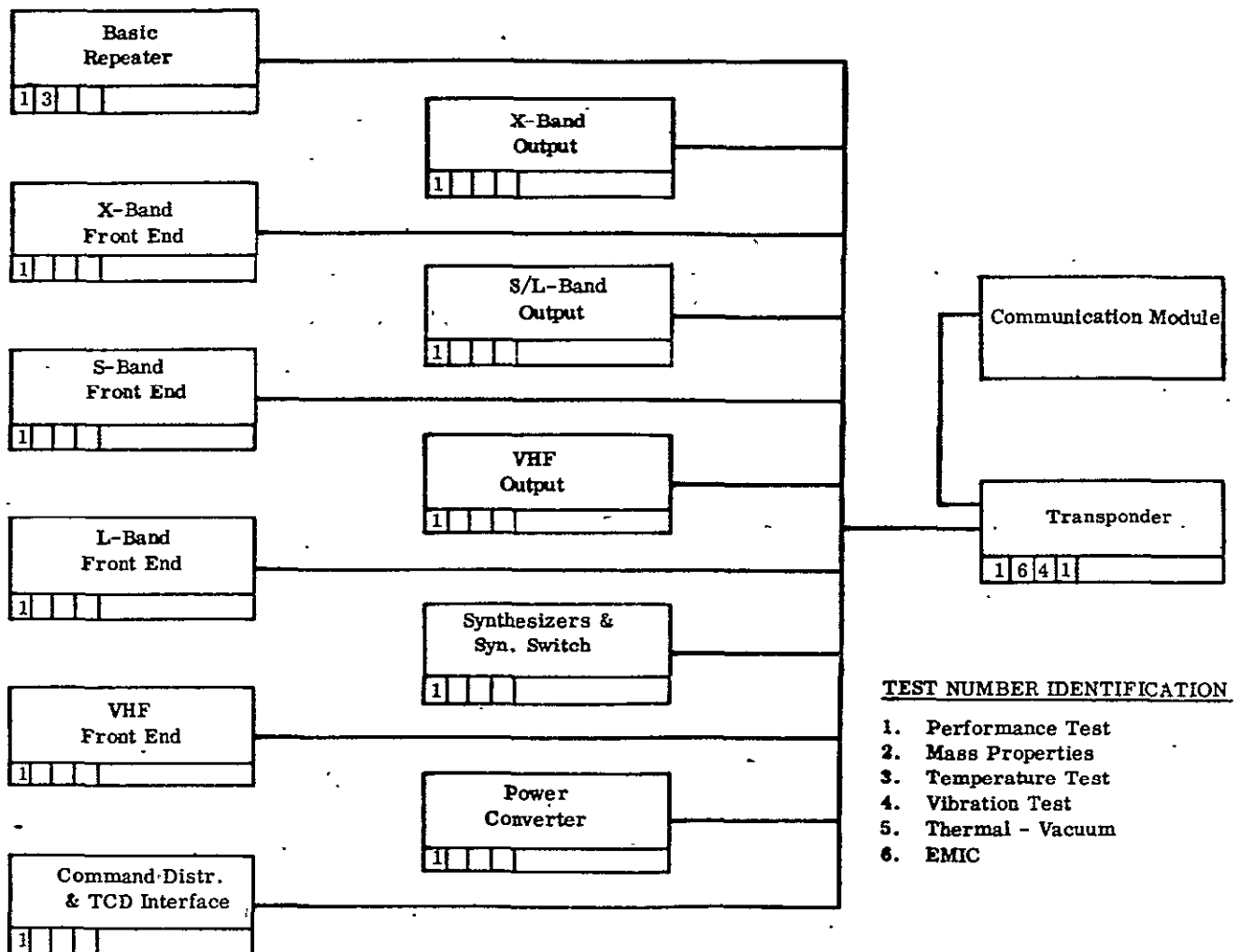


Figure 18.2-5. Transponder Development Test Summary

performance verification and operating modes are accomplished. The Power Subsystem development test summary is shown in Figure 18.2-6.

18.2.2.7 Structural Subsystem

Selected areas of design concern in the Structural Subsystem have been selected for development testing. These areas are GFRP and end fittings, solar boom hinge joint, solar panel release mechanism, and spacecraft truss tests. The GFRP is tested for verification of design margins in static and tension loads. Thermal distortion of GFRP is verified to be small and within design tolerances. The solar boom hinge joint is tested for friction and damping and the solar panel release mechanism is evaluated for program operation.

18.2.2.8 Reflector Subsystem

An engineering model reflector is required to demonstrate manufacturing and alignment techniques and to obtain early verification of expected thermal performance. In addition, the RF performance is verified on the engineering model reflector.

18.2.2.9 Thermal Control Subsystem

Engineering model units of Thermal Control Subsystem components are fabricated to verify design analyses in the following areas:

- Thermal Louvers - A solar simulation test is designed to demonstrate the louver radiation properties under solar impingement.
- Heat Pipe - The efficiency of the heat pipe system configuration will be verified by fabricating and testing a model of the "H" configuration of heat pipes at a single spacecraft elevation.

Engineering models are not proposed for the Radio Beacon and Gravity Gradient experiments. This philosophy is in concert with the general program requirements for experiments.

The development test program is discussed in greater depth in the following paragraphs. The implementation of the development test program in the fabrication and test of engineering model components/subsystems greatly enhances the confidence in the design details at a point in the program where design deficiencies are easily rectified.

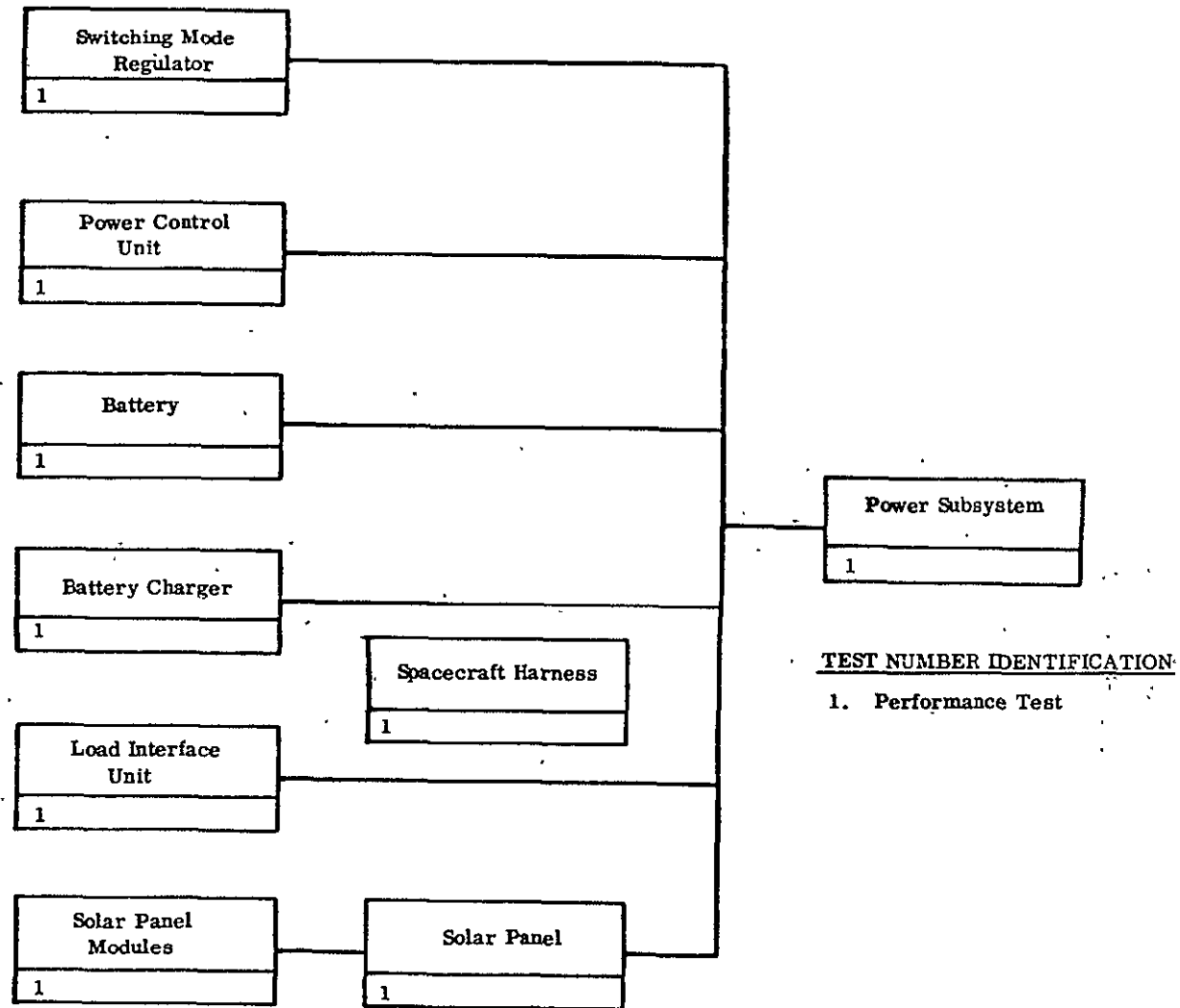


Figure 18.2-6. Power Subsystem Development Test Summary

18.2.3 THERMAL/STRUCTURAL MODEL TESTS

18.2.3.1 Summary

The Thermal/Structural Model (TSM) is subjected to a partial qualification test sequence as shown in Figure 18.1-1. These tests are conducted prior to the prototype qualification of the integrated spacecraft unit. The TSM is intended to demonstrate proof of design as well as to provide a greater degree of confidence in the success of the prototype qualification test program. The completion of the TSM series of tests precede the start of the spacecraft level qualification tests with sufficient schedule time to permit correction of any design deficiencies that may be uncovered during the TSM sequence. This series of tests also serves to establish and/or verify the environmental levels for component qualification and acceptance.

18.2.3.2 TSM Test Sequence

The thermal/structural model test sequence consists of the following tests:

- Mass Properties
- EVM Thermal Balance Test
- Zero G Boom Deployment Test
- Launch vibration test
- Static Loads Test
- Titan interface-mate
- Separation test
- Deployed mode shape vibration
- Solar Paddle Acoustic Test

18.2.3.4 TSM Description

The TSM configuration includes the EVM, of prototype quality, with Thermal/Structural mock-ups of each component and experiment. The Thermal Control Subsystem heat pipes, louver assemblies, and thermal insulation are included in the EVM. A dummy harness is fabricated and installed in the EVM to provide a simulated mass for dynamic testing. The TSM includes the prototype GFRP feed truss and hub structure. Since the prototype LSM reflector will not be available until January 1971, the major portion of the test sequence is performed with a dummy reflector mass. Upon receipt of the prototype reflector and prior to installation on the prototype spacecraft, the reflector is installed on the TSM for a deployed mode shape vibration survey which includes the deployed prototype solar boom and paddle.

QUALIFICATION TESTS

The following items are noted from Fairchild Hiller Corporation's qualification test plans:

- One set of qualification test components is fabricated and used for component/subsystem qualification testing and installed in the prototype spacecraft for spacecraft qualification. Redundant components are acceptance tested prior to prototype spacecraft installation. This is justified on the basis of cost and the fact that no destructive testing is planned or anticipated.
- Components are subjected to environmental exposure at either the component or multiple component level - duplicate exposure is not performed.
- The prototype spacecraft is identical to the flight spacecraft with the exception of the solar panel which will contain only approximately 10% solar cell coverage since this is adequate to provide proof of mechanical design integrity.
- The spacecraft is qualified for the thermal-vacuum space environment on the basis of the following tests:
 1. Thermal performance and selected thermal balance tests on the EVM alone, electrically interconnected to the hub separately located in the chamber.
 2. Thermal performance tests on the solar paddles, reflector, and other remotely located components as part of the individual component qualification testing. Misalignment between the feed and the reflector due to thermal-gradient-induced distortions will be determined analytically (thermal coefficients for the GFRP truss material will be verified on laboratory test samples).
- The reflector-feed system is verified as follows:

The prototype reflector is contour-checked cup-up and cup-down and at several isothermal conditions in order to verify the LMSC contour analysis. RF antenna pattern tests are conducted with the reflector in a "forced" zero G configuration and a point source feed in order to prove the RF analysis. Finally, the reflector is installed on the TSM/EVM-truss-hub assembly with the prototype

feed subsystem for electrical alignment and RF antenna pattern tests.

- The spacecraft's magnetic field is determined by obtaining field measurements on the hub and EVM separately in the GSFC Magnetic Test Facility since the facility is not large enough to accommodate the fully-assembled spacecraft readily and safely.
- Two reflector deployments at the spacecraft level are planned: one for the deployed mode shape vibration test on the prototype TSM/EVM spacecraft; the second is performed after the shock test on the prototype spacecraft is performed to verify deployment and check contour after vibration and shock exposure, and to conduct a self-induced systems RFI/EMI test.

18.3.1 SUBSYSTEM QUALIFICATION

Qualification of the components is demonstrated two ways:

- Qualification testing of prototype hardware to ATS qualification environmental levels
- Providing proof of qualification of design on other programs where qualification environmental levels were equal to or more severe than ATS requirements

Qualification environmental testing is performed at levels more severe than those expected during the flight spacecraft mission to demonstrate margin in the design implementation. Qualification environmental levels are described in detail in GSFC Specification S-320-ATS-1B.

Components will be acceptance tested prior to subsystem/spacecraft integration. All other components will require fabrication of hardware for qualification testing. Although more than one component of a given design is required in the spacecraft, only one prototype component is required to be qualification tested. Redundant components are acceptance tested to verify the absence of workmanship defects. This method of prototype spacecraft qualification has the feature of preserving the flight worthiness of many components and allowing them to be used as flight spares or ATS-G flight components. This rationale holds true because the prototype spacecraft qualification test levels are lower than flight acceptance component levels.

Highlights of the component/subsystem testing for each subsystem is contained in the following paragraphs.

18.3.1.1

ACS Qualification Testing

ACS qualification testing is performed at both the component and subsystem level. The vibration and thermal-vacuum tests are conducted at the component level and EMIC at the subsystem assembly level. Performance tests are conducted at both assembly levels. The ACS is integrated into the service module of the EVM for subsystem level tests. A summary of ACS qualification tests is shown in Figure 18.3-1. Two ACS components have prior qualification status, the polaris sensor and initial reference assembly. The IRA prior qualification status exceeds ATS requirements and thus no further qualification tests will be required. The polaris sensor requires vibration environment requalification because previous vibration qualification spectrum tests were lower than ATS requirements.

18.3.1.2

APS Qualification Testing

The APS qualification testing is accomplished primarily at the subsystem level. The Regulated Pressure Feed Assembly and the Altitude Control Thrusters are vibrated as a complete assembly. The Orbit Control lines and Thrusters are not assembled during vibration testing because of the difficulty in supporting the long feed lines. The Orbit Control Thrusters are vibration-tested separately. Thermal-vacuum tests are performed on the complete APS assembly. An adaptor is used during these tests to allow the orbit control lines to "lie down" in the plane of the attitude control lines. This allows the test to be conveniently performed in a considerably smaller vacuum chamber.

Particularly desirable features of the APS qualification testing include the absence of planned NH_3 filling, emptying and cleaning after the APS has been filled with NH_3 prior to vibration and thermal vacuum testing. The APS tanks are filled with NH_3 at the RPFA assembly level and barring failure, no further NH_3 operations are required.

The APS is integrated into the service module with the ACS and performance tested prior to shipment. (Refer to Figure 18.3-2 for a summary of the APS qualification tests.)

18.3.1.3

Interferometer

The Interferometer consists of two components, the array and front end assembly and the receiver/converter. These components are vibration and thermal-vacuum tested individually. However, calibration is only performed at the Interferometer assembly level.

Calibration and alignment of the Interferometer is a critical performance factor and an area of particular concern during testing. The calibration of the Interferometer in the anechoic chamber before and after environmental testing is included to demonstrate that the calibration data does not change as a result of environmental exposure.

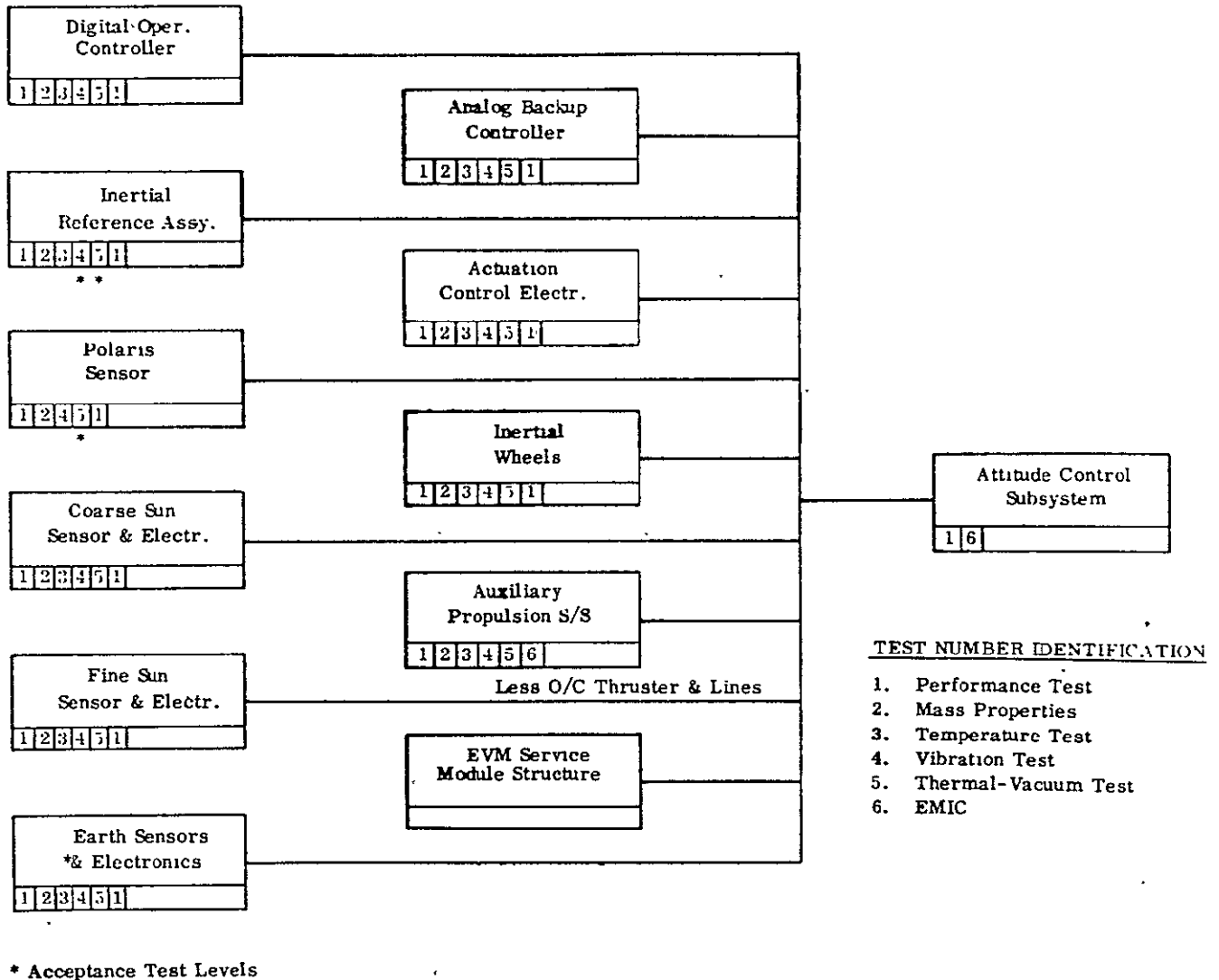


Figure 18.3-1. Attitude Control Subsystem Qualification Test Summary

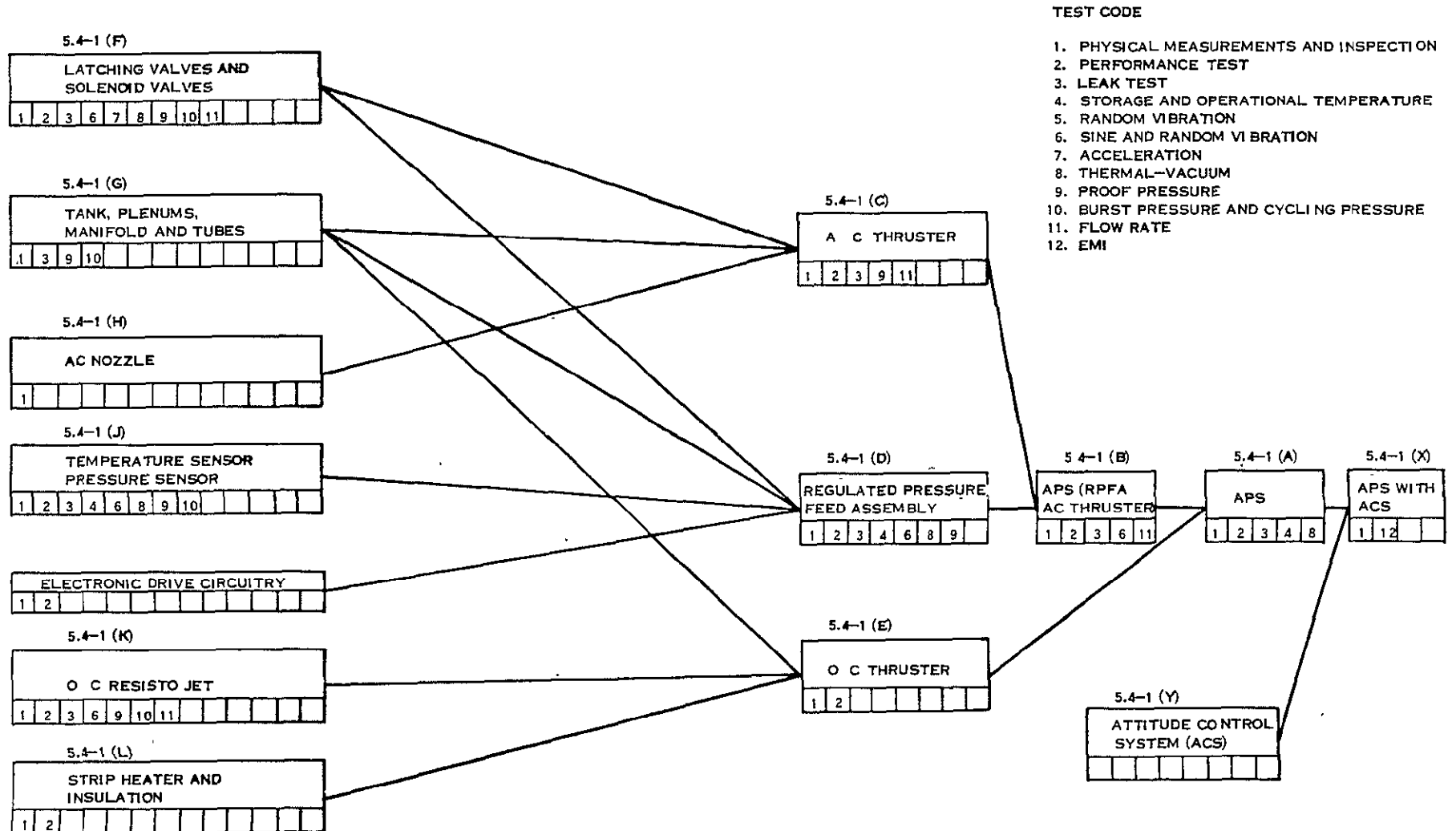


Figure 18.3-2. APS Qualification Test Flow Diagram

18.3.1.4 T&C Subsystem

The T&C subsystem is made up of two groups of components, the RF group and the digital group. All of the RF components have been previously qualified and do not require additional qualification tests. Acceptance vibration tests are performed on the RF components prior to integration into the T&C Subsystem. The digital components are vibration tested to qualification levels prior to T&C subsystem integration and thermal vacuum testing to qualification levels. The T&C qualification test summary is shown in Figure 18.3-3.

18.3.1.5 Communications Subsystem

The Communications Subsystem consists of two major components, the Transponder and the Prime Feed assembly. Qualification tests are conducted on the components and subsystem as indicated in Figures 18.3-4 and 18.3-5. A thermal vacuum test is not performed on the Prime Feed assembly. Thermal vacuum exposures at the prototype spacecraft level demonstrate complete Prime Feed qualification. Analysis indicates that there is no aspect of the Prime Feed assembly which appears to be subjected to thermal-vacuum degradation. Temperature performance tests will be conducted to verify that there is no change in the feed VSWR's. This approach is considered to contain a minimum risk since there are no electronic circuits or exposed mechanically moving joints.

18.3.1.6 Power Subsystem

A prototype of each type of component in the Power Subsystem is fabricated and qualified at the component level. A subsystem performance test is conducted after component qualification. It should be noted that the spacecraft harnesses are included as a part of the Power Subsystem. The harnesses are functionally tested as a component, but qualification testing occurs at the spacecraft assembly level. Figure 18.3-6 shows the qualification tests performed for the Power Subsystem.

18.3.1.7 Reflector Subsystem

The prototype Reflector will be completely qualified prior to spacecraft assembly. Vibration tests in the stowed configuration and thermal-vacuum tests in the deployed configuration are performed. Contour measurements before and after environmental exposures demonstrate no degradation. The prototype Reflector is integrated with the Prime Feed assembly and electrical tests are conducted to demonstrate the accuracy and tolerances on the combined Feed/Reflector focal lengths.

18.3.1.8 Thermal Control Subsystem

Qualification testing for the Thermal Control Subsystem is accomplished at the spacecraft level. Component level tests are conducted to demonstrate performance of the heat pipe and louvers.

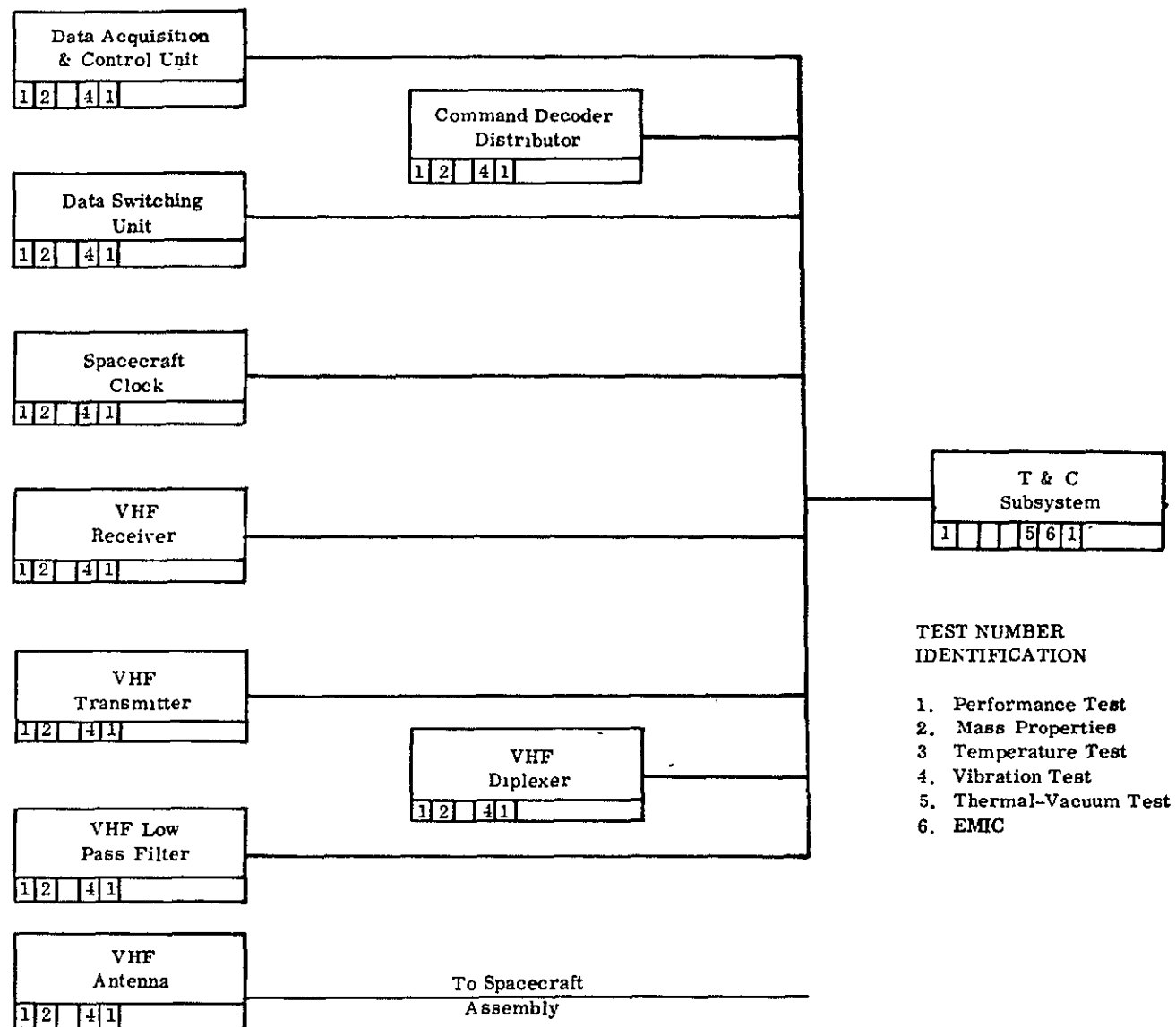


Figure 18.3-3. T&C Subsystem Qualification Test Summary

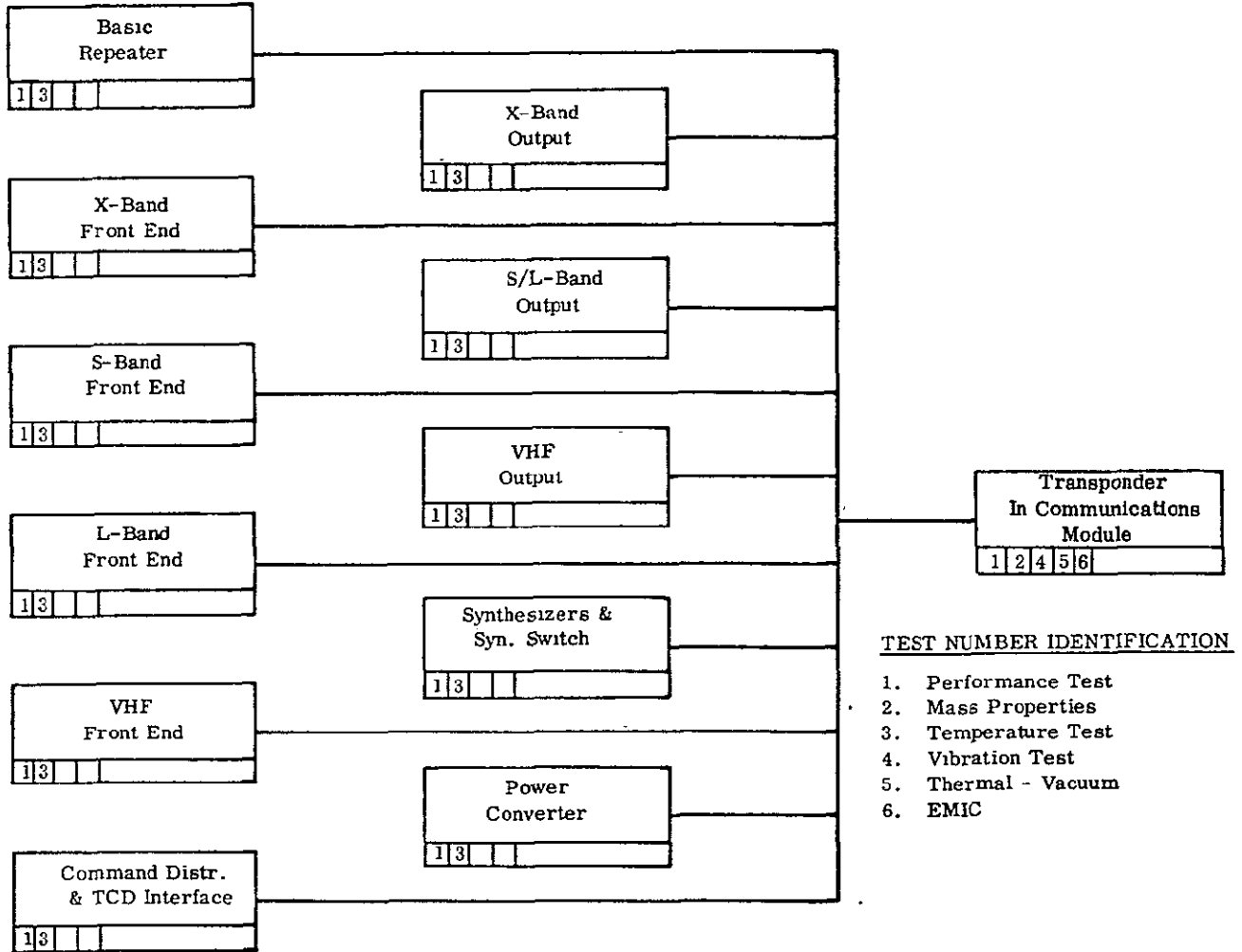


Figure 18.3-4. Transponder Qualification Test Summary

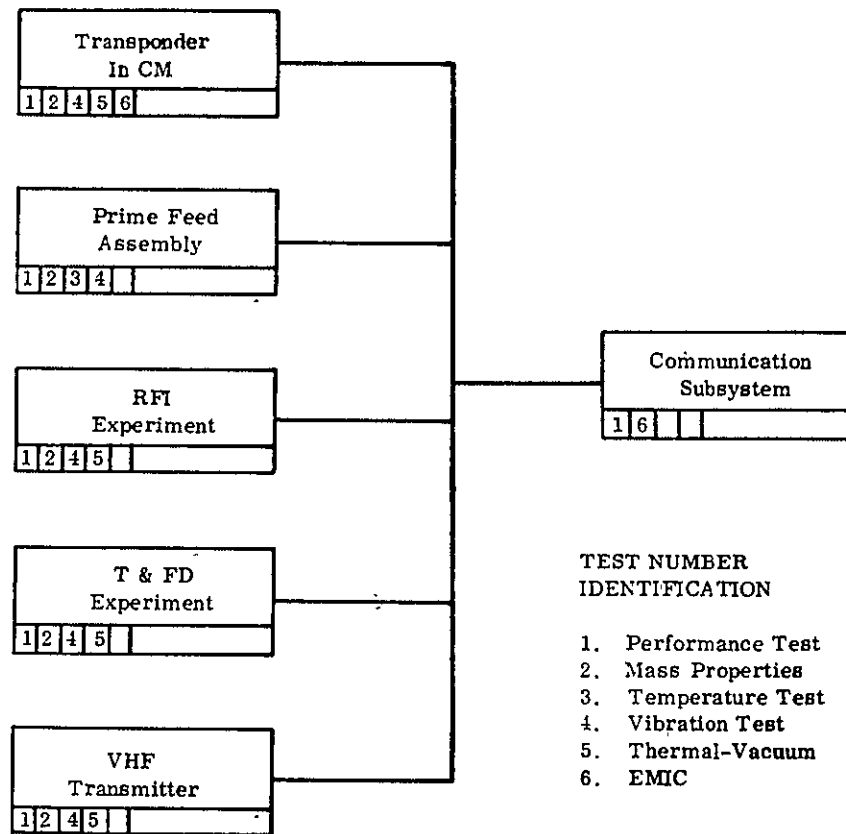


Figure 18.3-5. Communication Subsystem Qualification Test Summary

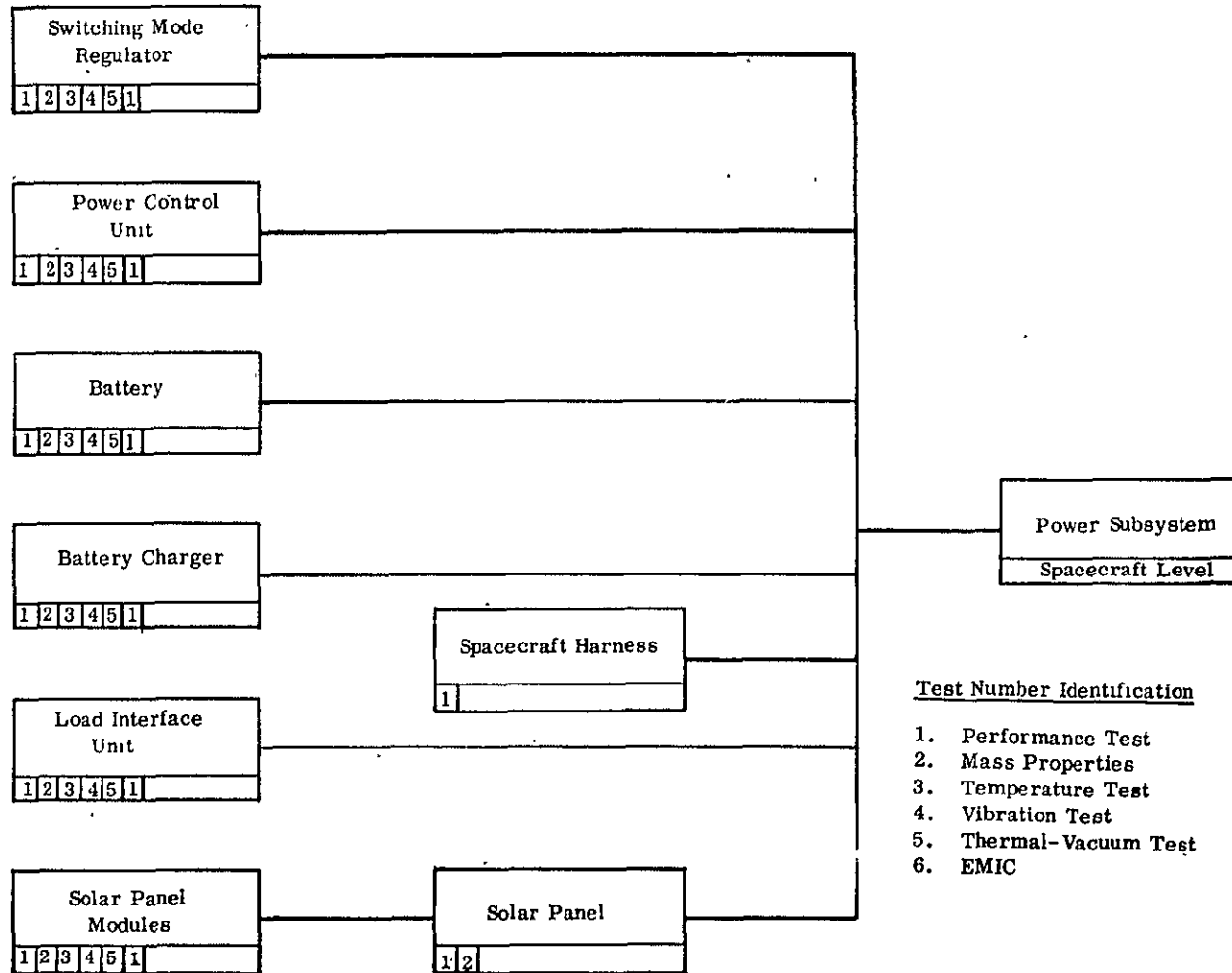


Figure 18.3-6. Power Subsystem Qualification Test Summary

18.3.1.9 Experiments

Contractor-furnished experiments, Radio Beacon and Gravity Gradient, are qualification tested as components. The Radio Beacon experiment is integrated into the CM during Communication Subsystem integration. The Gravity Gradient experiment is integrated into the system during prototype spacecraft integration.

18.3.1.10 Structural Subsystem

The Structural Subsystem is essentially qualified during the TSM test program which includes static loads, thermal vacuum, separation shock, and vibration environment testing. The prototype spacecraft structure (which may be obtained in part by refurbishment of the TSM tower structure) is not tested alone, but in conduction with all ATS systems after installation and check out.

18.3.2 PROTOTYPE SPACECRAFT QUALIFICATION

A formal qualification test program shall be conducted on the prototype spacecraft to verify the adequacy of the integrated system design for specified electrical and mechanical function under overstressed environmental conditions. All subsystems and components within the subsystem are mounted to the prototype spacecraft and electrically connected in a configuration identical to the launch and/or flight configuration as applicable to the tests being performed.

The design verification and qualification of the Structural Subsystem is performed during the testing of the TSM spacecraft which precedes the formal prototype qualification. It is to be noted that the design of the feed truss, solar booms and paddles, hub and adapter used during the TSM period of tests are considered to be near prototype design and therefore qualification of the Structural Subsystem design is demonstrated during that Phase of testing.

An extensive amount of systems tests is conducted prior to the start of the qualification sequence. This data is used as the base line for subsequent system performance prior to and following each environmental test.

18.3.2.1 Prototype Spacecraft Description

The prototype spacecraft system will consist of a fully integrated EVM, feed truss, hub structure and components, solar array boom and structure, and a deployable reflector, fabricated to flight hardware drawings and specifications.

Prior to installation and integration into the spacecraft, all components and/or subsystems will have been subjected to qualification testing.

The specific spacecraft configuration is included in each applicable test description. The major portions of the qualification program are conducted with the integrated EVM as a subsystem and the upper structure hub as modular entities. Vibration, deployment, and systems functional testing are performed on the all-up spacecraft to demonstrate compliance for the fully assembled spacecraft.

18.3.2.2 Systems Integration

There are three categories of systems integration at the spacecraft level; EVM integration, reflector - prime focus feed integration, and EVM - Tower integration. EVM integration and reflector - feed integration are discussed below.

18.3.2.2.1 EVM Integration

EVM integration includes the mechanical mounting of all EVM structural and subsystem components and the electrical verification of system functional operation. The sequence of events performed during EVM integration is shown in Figure 18.3-7.

The communications Module is received from Philco-Ford with the Communication Subsystem completely integrated. The earth viewing horn and waveguide is removed for convenience in handling the CM. After installation of the experiments, the communications module is ready for complete EVM integration.

The Service Module is received from Honeywell with the ACS components installed and aligned. The completely assembled APS is included and shipping support provided for the orbit control assembly and feed lines. After the power components and T&C components are installed, the SM is ready for complete EVM integration.

The Experiment Module structure is assembled to the Picture Frame Holding Fixture. The Power Subsystem components and harness are installed and a functional check performed to verify battery charging and the presence of power and grounds at the proper connector interfaces for the experiments and ACS components.

The Earth Sensors, Earth Sensor Electronics, both Polaris Sensors and Interferometer are installed and aligned in the EM after completion of the initial power checks. With the installation and alignment of the experiments, the EM is ready for complete EVM integration.

The EVM electrical integration tests will generally proceed in the following sequence:

- Power
- T & C
- Communications
- ACS
- Experiment

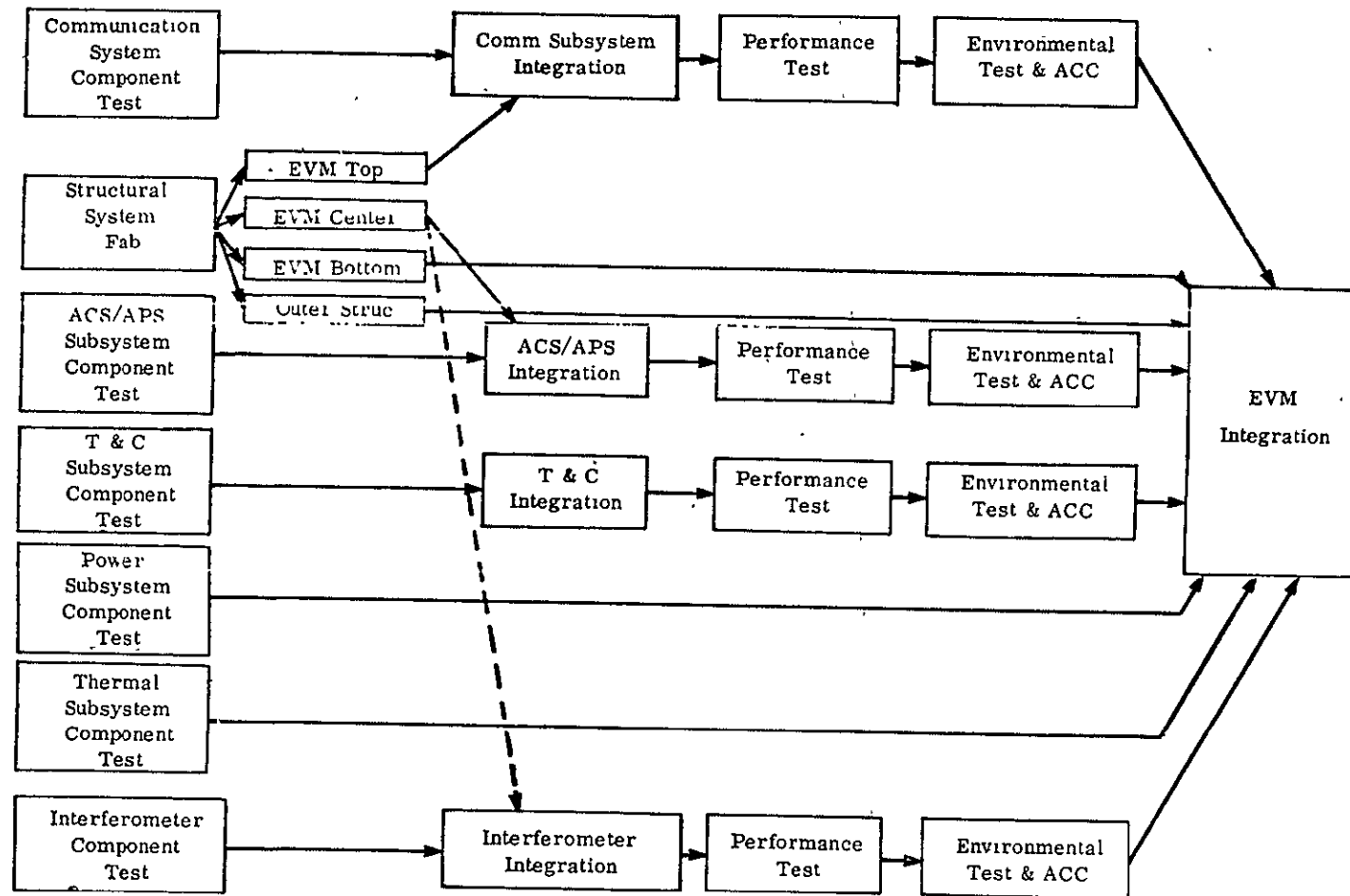


Figure 18.3-7. EVM Integration

18.3.2.2.2 Power Subsystem

The Power Subsystem functional checks will demonstrate the proper power interfaces with all power users. (Connections to users remain disconnected at this point.) Once the proper power supply voltages have been verified at the proper connector pins and ground returns have been checked, the Power Subsystem is considered ready to supply power to the remaining subsystems.

18.3.2.2.3 T & C Subsystem

The T&C Subsystem is connected after the Power Subsystem functional checks are complete. Power is applied to the spacecraft, commands are sent to the Power Subsystem and telemetry is monitored for correct response. The telemetry data format is checked from both DACUs through both high gain and omnidirectional RF links. Once the ability to receive commands and transmit telemetry has been verified, the T&C Subsystem is ready for additional subsystem integration.

18.3.2.2.4 Communications Subsystem

The Communications Subsystem connectors are mated and power is applied to the spacecraft. The hat coupler output lines are connected to the Communication STC. Commands are sent through the command links to operate the Communications Subsystem through each long form operating mode. Results are verified through the telemetry link and the Communications Subsystem STC. Upon completing the long form sequence of operating modes, the Communications Subsystem is ready for the "test of record" long form.

18.3.2.2.5 ACS

ACS connectors are mated and the hub sun sensors are connected from the Hub Component Test Fixture (mounting plate). The ACS targets are positioned for the zero error condition and the commands are sent through the Command Subsystem to exercise the ACS in the long form modes of operation. After verifying proper ACS operation in all primary and backup modes, the ACS integration is complete.

18.3.2.2.6 Experiment Integration

Experiment integration is accomplished by connecting the experiments into the system one at a time and verifying proper operation by sending commands and observing results transmitted through telemetry and prime feed links. For particularly complicated experiments, the experimenter will be required to participate in the experiment integration. The following experiments are considered to require experimenter participation during integration:

- EME
- LASER
- MM Wave
- Radiometer
- Camera

18.3.2.2.7 Prototype Reflector and Composite Feed RF Integration Test

The RF integration of the reflector with the composite feed is performed on the prototype models of these two subsystems, using a partial subassembly of the prototype EVM, reflector support truss and reflector supporting structural hub. The test is to be conducted at the LMSC Large Aperture Antenna Test Facility at Santa Cruz, California. The purpose of the test is to:

- Determine the required axial positioning (distance) of the reflector relative to the feed and EVM for maximum on-axis gain performance of the combined antenna system at the highest operating frequency (x-band).
- Verify the analytically determined maximum permissible tolerance on the above positioning distance, from consideration of RF performance (on-axis gain at x-band)

The reflector is adjusted to and maintained in the "design contour" by means of the RF test positioning fixture, and mounted to the antenna positioning pedestal. The properly aligned subassembly of composite feed, EVM, trusses and structural supporting hub is hoisted into position and attached to the reflector at the mounting interface. The reflector alignment tool is used, in conjunction with an auto-collimating telescope to align the reflector axis to point precisely at the center of the x-band feed horn, located on the spacecraft Z-axis. Differential shimming in the reflector to structural supporting hub interface is used to accomplish this alignment.

First, pattern and gain measurements are made in vertical and horizontal planes through the maximum gain direction of the antenna, at a number of various axial distances between the reflector and the feed. These distances span a range of approximately 0.6 to 0.9 inches on both sides of the design position, defined as that for which the nominal design focal point of the reflector divides with the nominal design Phase center of the x-band feed. The distance is changed from one extreme end of this range to the other in 0.15 increments, by adding to the reflector structural support hub interface, underneath the reflector mounting points, uniform precision shims of identical thickness.

Upon completion of this test series, the maximum gain position and the associated allowable tolerance are determined from a plot of the peak gain versus total shim thickness used in each pattern measurement. The antenna is set to this optimum value, and the corresponding distance from the reflector to the feed and the

EVM are determined through measurement using pre-established reference points on these components. This dimension is determined within an accuracy of ± 0.030 inches.

The next Phase of the test is to determine RF performance of all feeds in all operating modes and frequency bands. Measured are:

- Gain
- Axial Ratio
- Beam Position
- Pattern
- Beam Width
- Sidelobes
- Monopulse Null Depth
- Monopulse Difference Pattern Slopes

The above measurements are performed at the following frequency bands:

- X-band, Including Monopulse Mode

18.4 FLIGHT ACCEPTANCE TESTS

18.4.1 INTRODUCTION

Implementation of the flight acceptance test program will be accomplished by two environmental exposures, vibration and thermal-vacuum testing. With few exceptions, the flight components/subsystem will be acceptance tested prior to spacecraft tests. At the spacecraft level, a second series of acceptance tests will be performed to prove flight worthiness.

Spacecraft acceptance testing centers on the EVM which contains nearly all of the electrical components. Thermal-vacuum tests at the spacecraft level will be performed on the EVM with hub components (EME, Gravity Gradient and coarse Sun Sensors) included. Vibration testing will be performed on the completely assembled spacecraft.

18.4.2 SUBSYSTEM ACCEPTANCE

The concept of concurrent testing of the major spacecraft subassemblies will be retained for the flight acceptance test program. The tower, reflector, and EVM subassemblies will be fabricated individually, assembled, integrated, and then combined for launch vibration acceptance.

A large portion of the EVM integration will be performed in parallel, with the Communication Subsystem integrated into the CM, the ACS & APS integrated into the SM and the experiment integrated into the EM.

18.4.2.1

Component and Subsystem Tests

The component and subsystem acceptance tests will be performed in a manner very similar to the tests described for the prototype components and subsystems. Several unique features of the acceptance testing of components and subsystems warrant emphasis:

- Where practical, the subsystems will be integrated into the EVM modules prior to spacecraft integration. This is true of the Communication Subsystem, Attitude Control Subsystem (less several sensors located outside the service module) and Auxiliary Propulsion Subsystem.
- The Auxiliary Propulsion Subsystem tanks will be filled with ammonia during subassembly testing and will remain filled during subsystem and spacecraft testing. This approach minimizes the handling and contamination hazard.

Component/Subsystem acceptance tests are summarized in Figures 18.4-1 through 18.4-5.

18.4.3

FLIGHT SPACECRAFT ACCEPTANCE

18.4.3.1

Summary

The flight spacecraft system will be subjected to flight acceptance environmental tests to locate material and workmanship defects prior to launch. The environmental levels are equal to those expected during ground handling, launch and orbit. Figure 17.1-2 defines the test sequence for the flight spacecraft system and includes the following:

- Systems Integration
- Mechanical Alignment
- EVM Long Form Test
- Mass Properties
- Acoustics
- Thermal Vacuum
- Magnetic Test
- Spacecraft Assembly
- Vibration
- Final Long Form Test
- Shipment to ETR

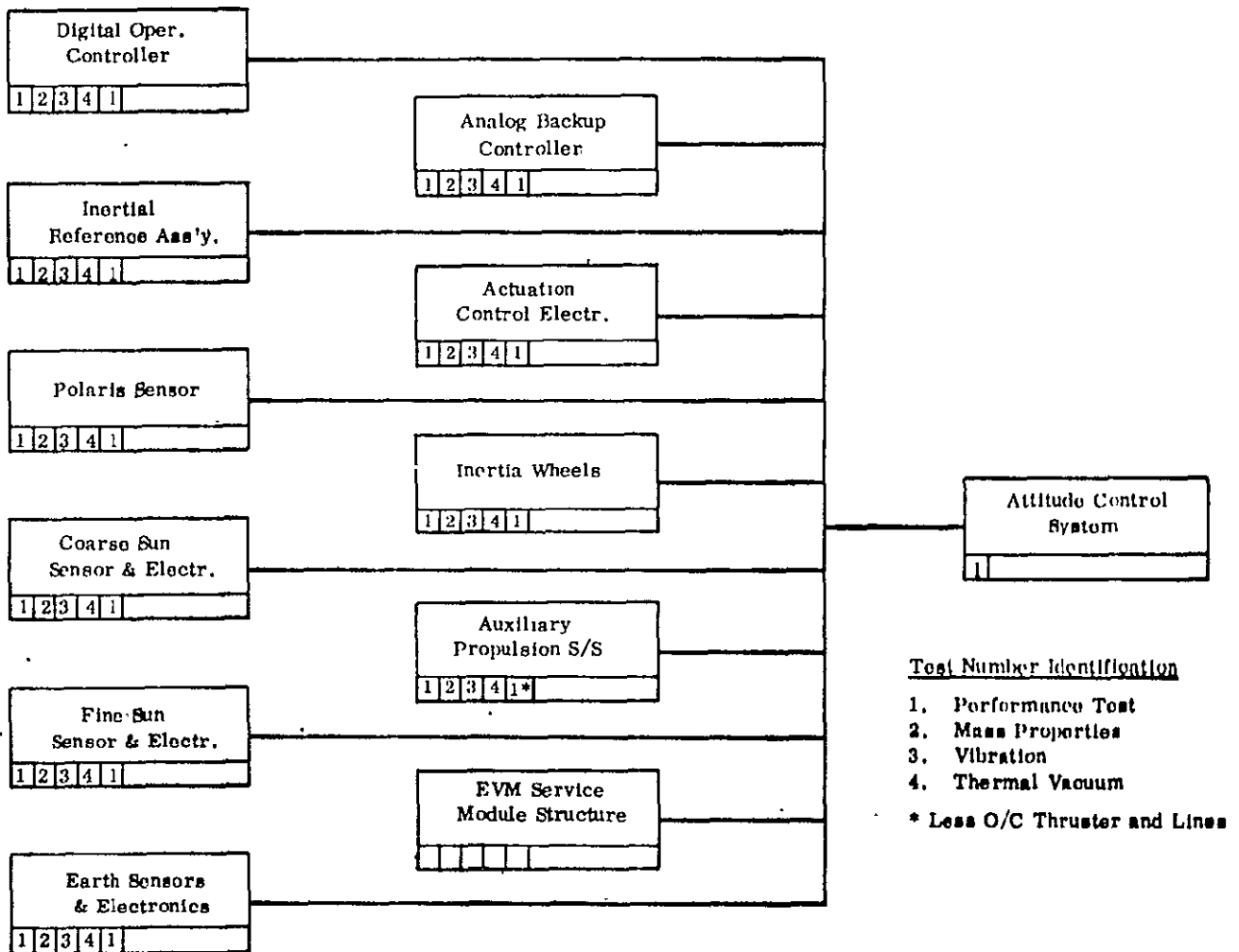


Figure 18.4-1. Acceptance Test Summary - Attitude Control Subsystem

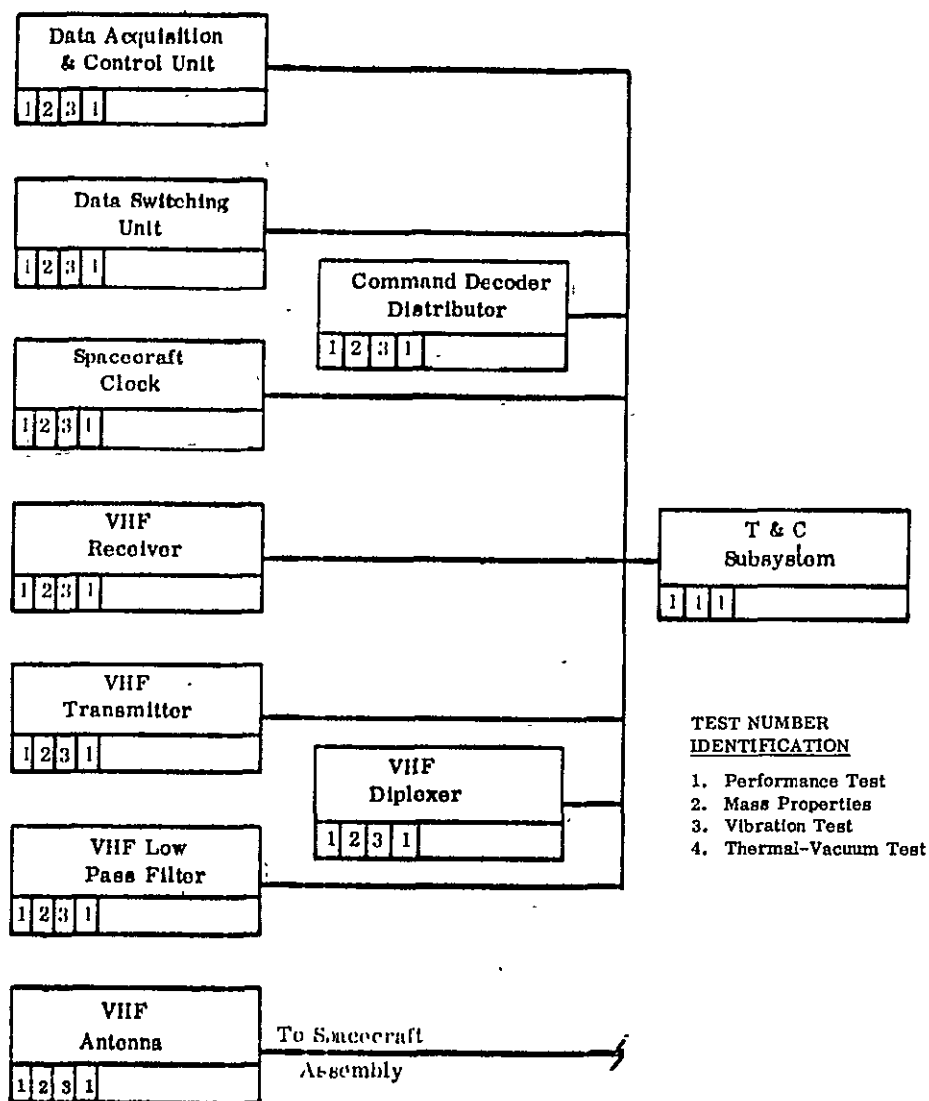


Figure 18.4-2. Acceptance Test Summary - T&C Subsystem

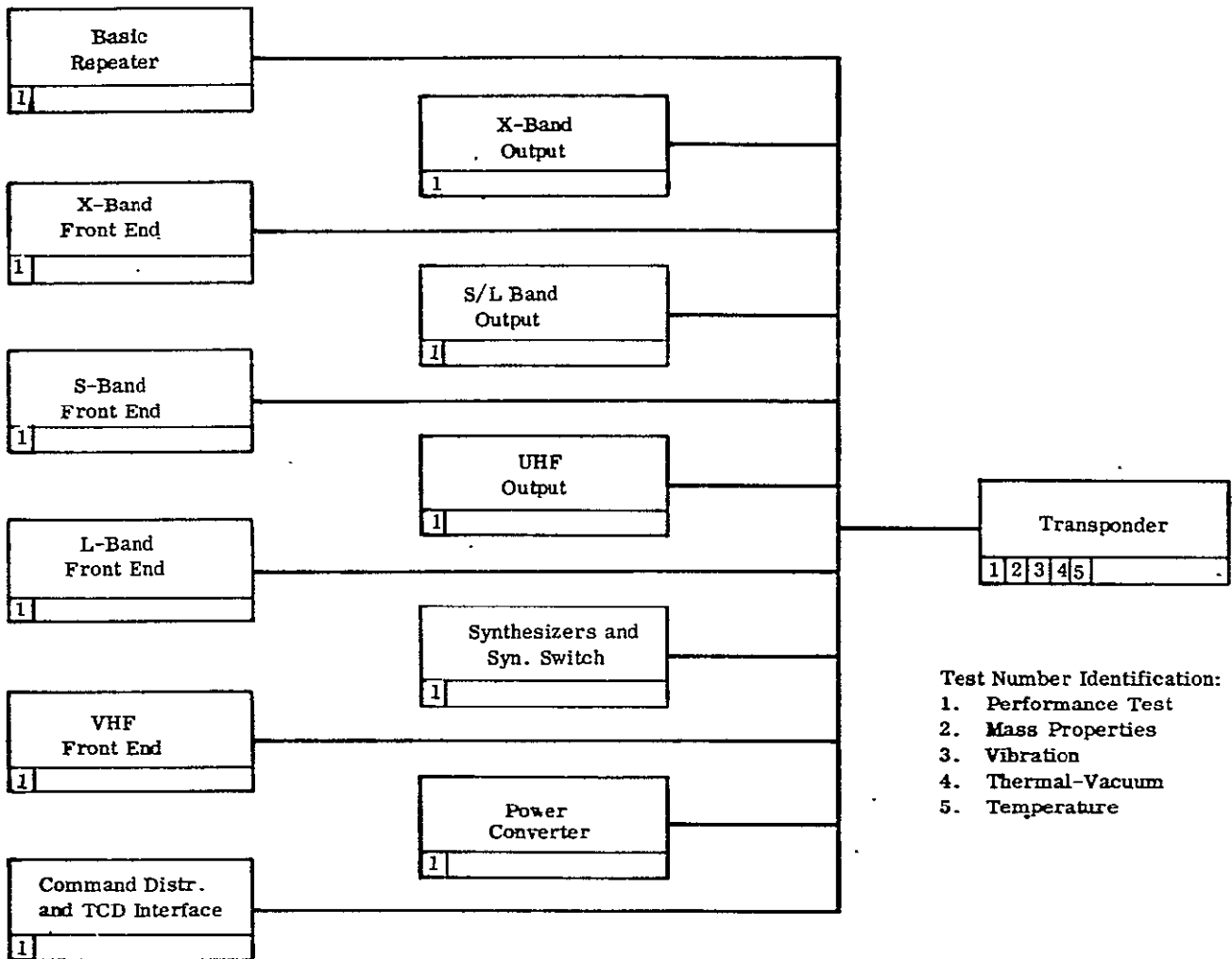
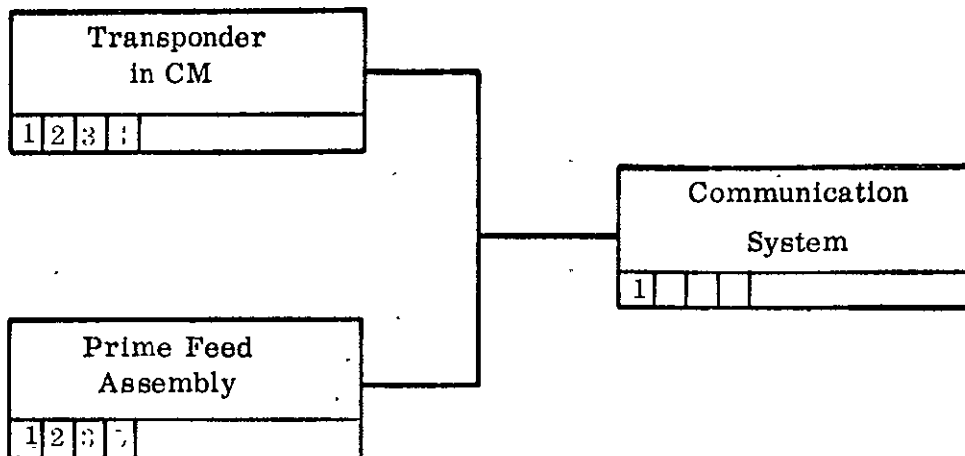


Figure 18.4-3. Acceptance Test Summary - Transponder



Test Number Identification

1. Performance Test
2. Mass Properties
3. Vibration Test
4. Thermal-Vacuum Test
5. Temperature

Figure 18.4-4. Acceptance Test Summary - Communication Subsystem

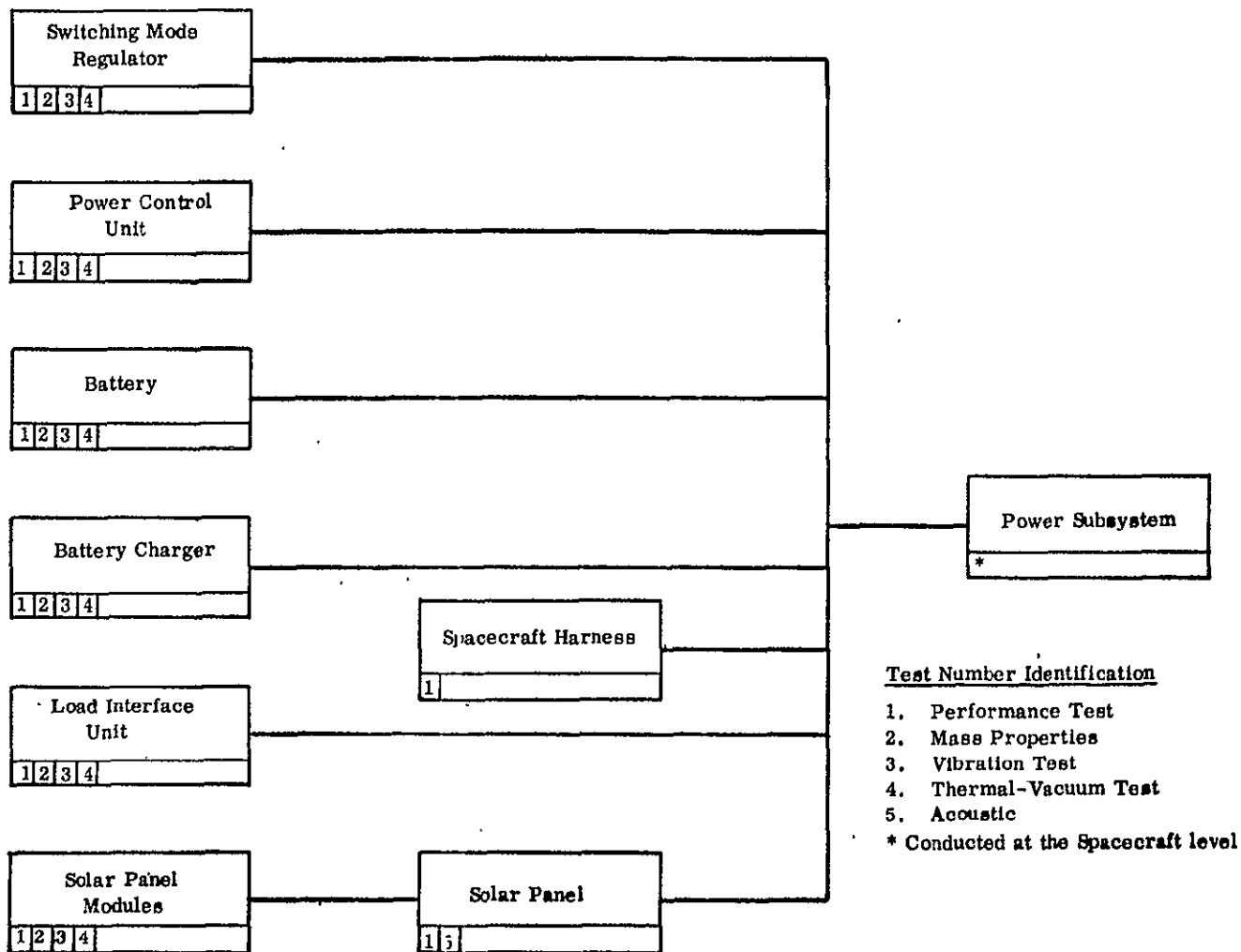


Figure 18.4-5. Acceptance Test Summary - Power Subsystem

Upon completion of the spacecraft acceptance, shipment to ETR will be accomplished in the environmentally controlled shipping container. The spacecraft is transported directly to the launch complex and mated to the Titan IIIC. A system performance and EMC test will be conducted to ensure final system functionality prior to launch.

18.4.3.2 Flight Spacecraft Test Sequence

The flight spacecraft system consists of the fully integrated EVM, feed truss, hub structure and experiments, solar array boom and structure, and a deployable reflector.

Prior to installation and integration into the spacecraft, all components and/or subsystems will be subjected to acceptance testing.

The flight acceptance will be performed on major spacecraft subsystems such as the EVM, Hub and Reflector to simplify handling procedures, reduce schedule time, and reduce costs which would be required to test the all-up spacecraft. The environmental tests to be performed on the spacecraft subsystems are:

- Long form performance
- Mechanical alignment
- Mass properties
- Acoustics

Short form performance

- Thermal vacuum
- Short form performance

Magnetic test

Following the magnetic test of the EVM and hub modules, the spacecraft will be assembled to the all-up launch configuration.

An optical alignment check of critical components will be accomplished and a boresight alignment of the reflector/hub is performed relative to the EVM.

A launch vibration test at acceptance levels will be conducted. Following vibration, the solar panels are removed and subjected to an illumination check. Concurrent with the panel check, a long form performance test will be conducted on the spacecraft system. The solar panels are then reinstalled and the spacecraft is prepared for shipment to ETR.