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GLOSSARY

Space Vehicle	The total flight unit delivered by Boeing including propulsion but not shuttle adapter.
Spacecraft (or Satellite)	Space vehicle less propulsion, on orbit vehicle.
Shuttle Adapter (Flight Support Equipment)	Cradle to support space vehicle in shuttle.
Payload	Separate box containing experiment attached to front of spacecraft.
Baseline	Vehicle with 3 axis low cost attitude control system.
Option 1	Baseline with high accuracy star tracker and gyro unit added.
Option 2	Spin stabilized version.
Option 3	Carrier II TT&C module.
Option 4	Reduced Redundancy version.
Option 5	Spin stabilized version as a new baseline for costing only.
AEM	Atmospheric Explorer Mission
HCMM	Heat Capacity Mapping Mission
SAGE	Stratospheric Aerosol Gas Experiment
Base Module	The basic unit of the vehicle being developed for HCMM/SAGE missions; readily adaptable to many other missions.

Shuttle/Orbiter/IUS terminology is as defined in JSC 07700, Volume XIV.

1.0 INTRODUCTION

This document is the final report of a study conducted under NASA (Goddard) contract number NAS5-23537. It includes a technical description of a large diameter shuttle launched-AEM (LDSL-AEM), an AEM Base module adapted to carry 5' diameter payloads in the shuttle with propulsion for carrying payloads to higher altitude orbits from a 150 NM shuttle orbit.

The data is to be provided to the Rand Corporation for use in a parallel study comparing the characteristics and costs of four vehicles to do SAMSO Space Test Program (STP) type missions with the shuttle as a launch vehicle. The four vehicles being studied by Rand include:

- o AEM Base Module
- o Large diameter AEM (this study)
- o New STP standard satellite
- o NASA Multimission Satellite (MMS)

The AEM Base Module is being developed by The Boeing Aerospace Company under contract number NAS 5-22870. Two vehicles are involved in the original contract; AEM-A for the Heat Capacity Mapping Mission (HCMM), and AEM-B for the Statospheric Aerosol Gas Experiment (SAGE). The second, or SAGE vehicle is used to adapt to the large diameter payload, shuttle launched vehicle for this study. An artist's sketch of the SAGE is shown in Fig. 1.0-1.

The AEM is designed for launch on the Scout launch vehicle. On board equipment provides capability to despin, acquire the earth, and control the vehicle in an earth pointing mode using reaction wheels for torque with magnets for all attitude acquisition, wheel desaturation and nutation damping. Earth sensors in the wheels provide pitch and roll attitude. This system provides autonomous control capability to 1 degree in pitch and roll and 2 degrees in yaw. The attitude can be determined to .5 degrees in pitch and roll and 2 degrees in yaw.

ARTISTS SKETCH OF SAGE VEHICLE

FIGURE 1.0-1

An S band Space Tracking and Data Network (STDN) compatible telemetry tracking and command system in the AEM provides for 256 stored, addressable and timed commands, 8 KBPS real time data rate and 1 MBPS dump rate from the 10^8 bit tape recorder. The clock provides a resolution of 1 second with a clock time capability of 194 days. Tracking information is computed at STDN ground stations using minitrack interferometer equipment.

A 23-square foot solar array provides power at 28 ± 4 volts for spacecraft and payload equipment. The solar array is clocked in one axis by ground command to track the sun for the wide variation in beta angles.

The temperature of components within the AEM is controlled by thermal blankets, radiators and louvers to allow for variation in internal power and sun position.

The AEM Program provides for full qualification with qualification testing conducted on the first flight vehicle only. The program is to be accomplished in 20 months from ATP (considered PDR) to delivery of the first unit. The payload integration program is to be accomplished by NASA Goddard after delivery by Boeing.

The LDSL-AEM as described in this document is a modification to the SAGE design.

The Statement of Work for the LDSL-AEM Study included the following requirements:

- o The payload interface shall be hexagonal 60 inches in maximum diameter.
- o The spacecraft shall be 3-axis stabilized with control capability to .5 degrees in pitch and roll and 1 degree in yaw, with capability to be modified to control to 6 arc minutes or spin-stabilized with control capability to 1 degree.

- o Solid propulsion shall be provided to inject the spacecraft into a circular orbit at altitudes up to geosynchronous (orbiter altitude 150 NM). Provisions shall be made to use one or two motors - both to be installed on the same end (tandem).
- o A SGLS-compatible TT&C subsystem shall be provided using Carrier I with capability to also incorporate Carrier II for transmitting payload data at high data rates.
- o Provision shall be made for payload weights up to 1000 pounds.
- o The power system array shall be 1 axis articulated with settable 2nd axis with 100 square feet of array area. Power shall be stored in two 20 ampere hour batteries.
- o The thermal system shall use louvers, heaters and thermal blankets with a maximum power input from a payload of 10 watts.
- o There shall be no single-string failure modes.
- o The Shuttle interface shall be defined including an adapter to support one or more spacecraft with payloads in the Shuttle over the short and long spacelab tunnel, clear space or over Orbital Maneuvering System (OMS) kit.
- o The vehicle shall be capable of being attached to the IUS payload interface.
- o The vehicle shall be capable of being installed in the Shuttle as a secondary payload with a variety of USAF primary payloads including IUS.

2.0 SUMMARY

The LDSL-AEM provides control during separation from the shuttle, propulsion for transfer from shuttle orbit to payload orbit and on board services to the payload while on orbit for a wide variety of payloads including multiple experiment payloads. Standardization for many different payloads and the use of the "off the shelf" AEM base module with added redundancy

LDSL-AEM Requirements/Capability

Item	Requirement	Capability
Launch vehicle	Shuttle	Shuttle or delta
Maximum diameter	60-inches	Main structure 60 inches
Attitude control	3-axis or spin	Baseline 3 axis, option 2 spin
Attitude control accuracy	$.5^{\circ}$ P&R 1°Y Option 6 arc minutes Spin 1°	Complies
Data handling	Carrier I with option or Carrier II	Carrier I Carrier II (option 3)
Data rate		128 and 256 kbps 1 mbps (option 3)
Propulsion	Low earth to synchronous orbiter altitude 150 nmi payload weight 1,000 lb	Complies Low earth payload capability exceeds 1000 lb
Redundancy	No single point failure	Full redundancy No single point failures

TABLE 2.0-1

provides high probability of success with low cost and low risk. The LDSL-AEM is able to satisfy the mission requirements established for the study as shown in Table 2.0-1.

The LDSL-AEM is a 60 inch diameter hexagonal structure with the off-the-shelf AEM as the central core as shown in Figure 2.0-1 and 2.0-2. The AEM single axis articulated solar array is mounted symetrically on each side of the propulsion adapter. The solar array area has been increased to two 50 square feet arrays. Each array is divided into six 30" x 40" panels folded around the hexagonal structure during launch. The articulation system clocks the solar array by ground command to accommodate sun beta angle variation. Two S-Band conical log spiral antennas provide omni coverage during on orbit operation.

Three variations of the attitude control system (Baseline and Options 1 and 2) provide three levels of capability for stabilization. The baseline is a direct derivative of the AEM system using spin stabilization for injection and momentum bias for on-orbit operations. Option 1 provides high accuracy, all attitude capability by adding star trackers and a gyro reference unit. The gyro reference unit also provides attitude information for 3 axis stabilized injection. Option 2 provides spin stabilized control for injection and for on orbit operations using earth and sun sensor for attitude information. Spacecraft attitude is determined on the ground with commands then transmitted to precess the spacecraft to the desired attitude using N_2H_4 thrusters, for the spin stabilized concept. Active damping is provided by the N_2H_4 thrusters for torque with rate gyros as sensors.

Each version has a blowdown N_2H_4 reaction control system. The thrusters are used in the baseline to provide spin stability during firing of the orbit transfer motors and to precess the vehicle to the burn attitude. Redundant thrusters with series redundant valves assure operation with any one failure.

The temperature of the subsystem components is maintained well within allowable limits by thermal insulation blankets, louvers and heaters.

LDSL–AEM Baseline Orbital Configuration

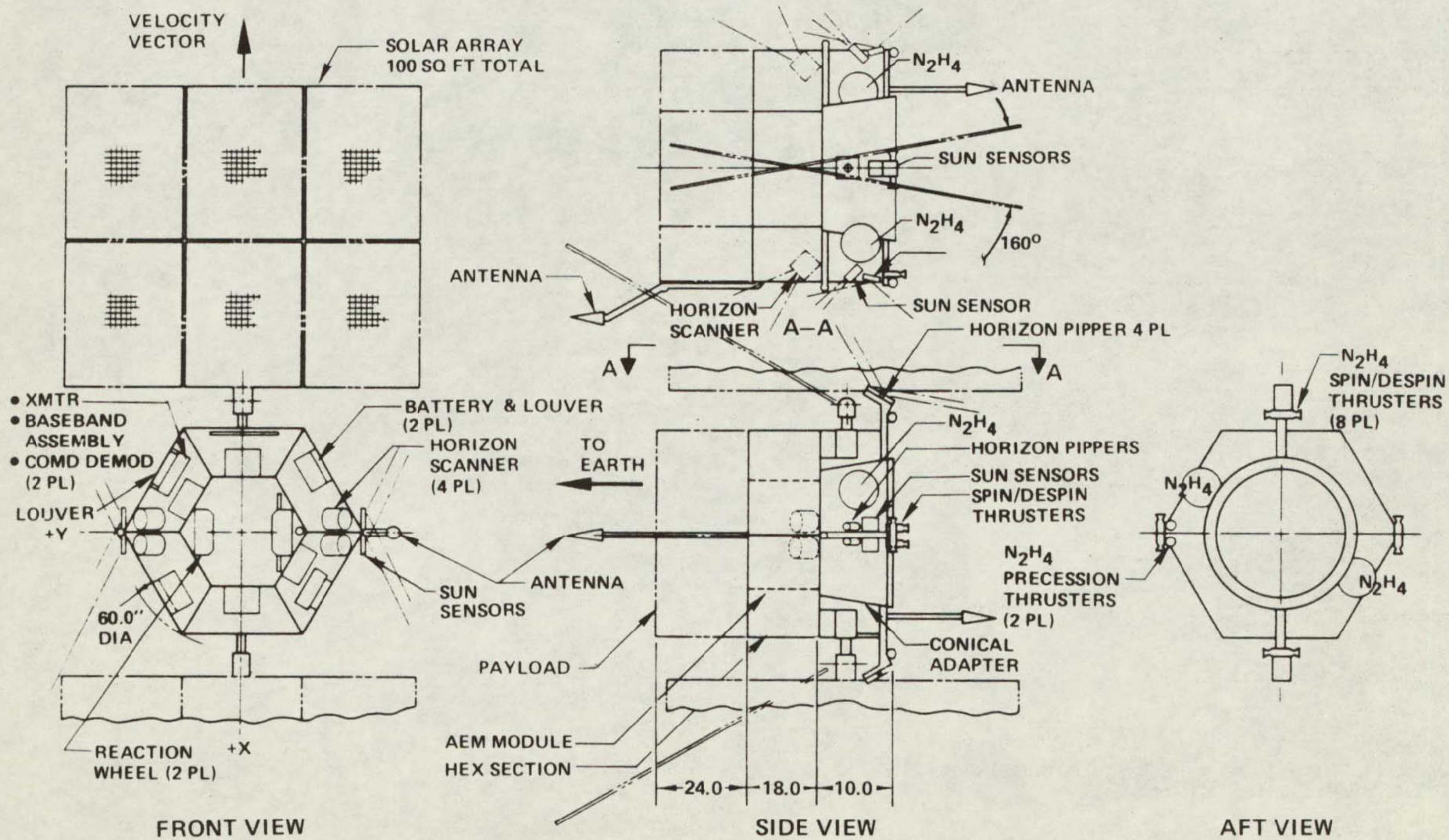


FIGURE 2.0-1

AEM Modified Launch Configuration

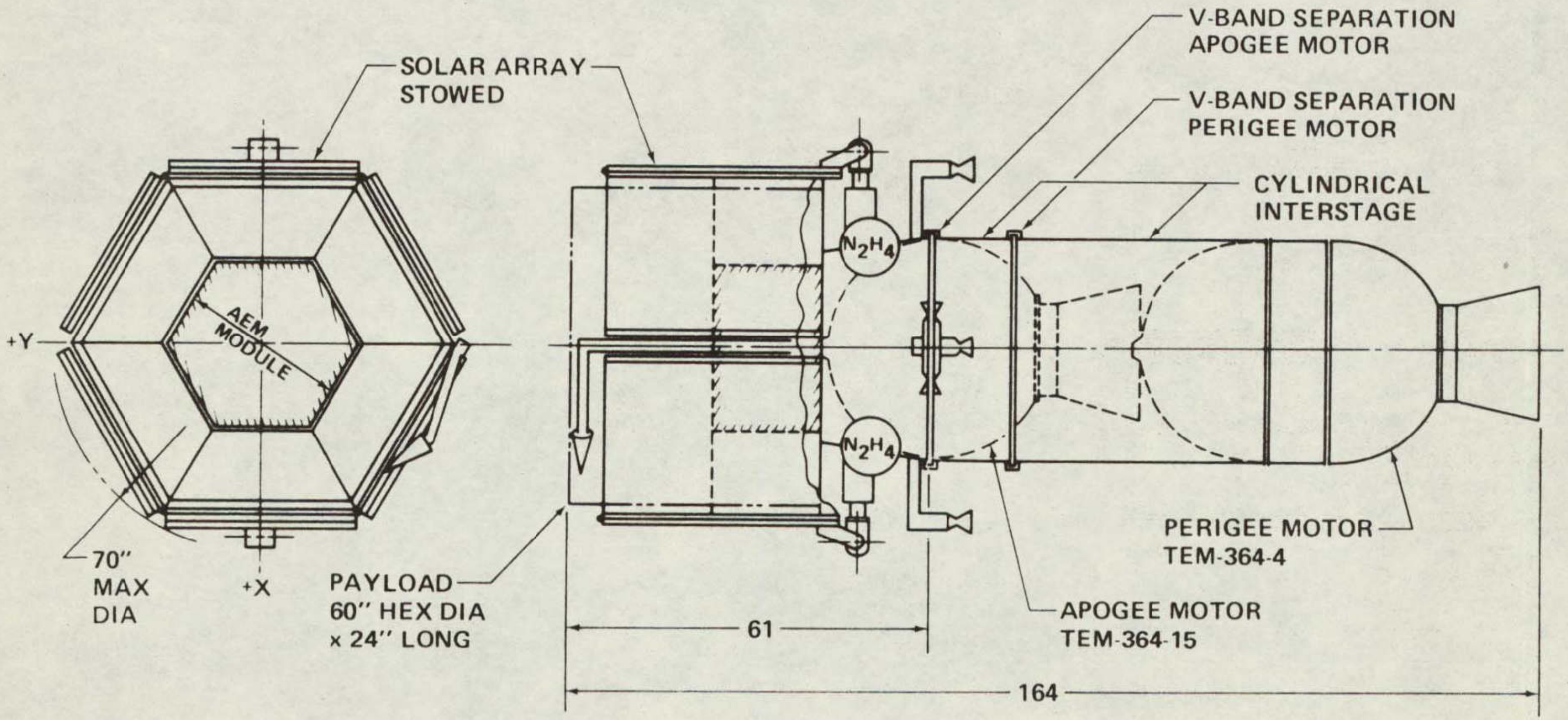


FIGURE 2.0-2

The transmitters and batteries are mounted near the outside of the hexagonal structure with louvers to maintain temperature control for wide variations in operating conditions. Flexible optical solar reflectors (FOSR's)are used to maintain radiating characteristics of louvers for varying sun positions.

Two redundant power conditioning and distribution systems assure flow of power to critical loads even after a component failure. A direct energy charging system provides efficient charging of batteries. Voltage is controlled to 28 $\pm$ 4 by floating the battery on the bus. An ampere hour meter controls charging conditions with charging rates selected by ground command. The power is generated by 100 square feet of solar array in two panels. The angle of the panels is controlled by ground command periodically to account for beta angle changes. The panels are fixed around the HEX spacecraft for the spin stabilized version (Option 2).

The baseline configuration uses the TE 364-4 and TE 364-15 solid propellant motors for perigee and apogee injection. These are considered to be the largest motors applicable to this vehicle. Many other motors are available for missions to different altitudes or for smaller payloads. An adapter from baseline structure to motor mounting flange provides for attachment of other motors.

The LDSL-AEM is mounted in the shuttle with a flight support adapter as shown in Fig. 2.0-3. The vertical and longitudinal loads are carried into the shuttle longerons. Lateral loads are taken out in the keel. All structure is designed to satisfy the worst case crash conditions.

Ground support equipment is provided to support the program through development and test and production.

The development assumes a complete development and qualification test program on the first flight vehicle.

The master phasing schedule provided an efficient time for design, procurement and production while maintaining technical personnel to successfully complete the test and production program. Production rate is based on use of one set of test equipment and one systems test crew.

LDSL-AEM Baseline Shuttle Short Tunnel Installation

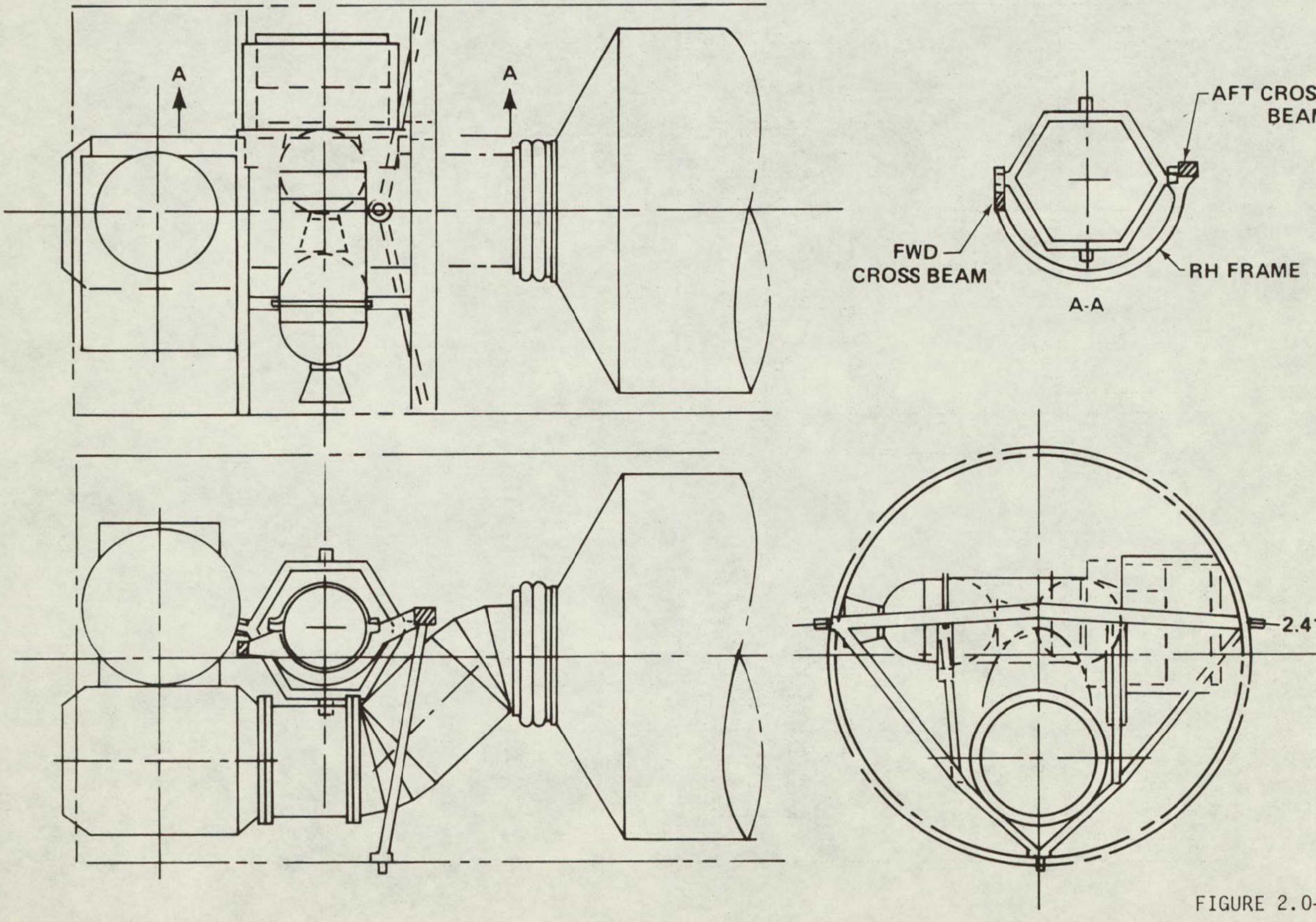


FIGURE 2.0-3

Cost and technical data were developed for several options as summarized in Table 2.0-2.

The LDSL-AEM as described in the following sections of this document can accomplish the mission requirements as given for the study, can interface with shuttle or IUS.

Option Summary

Option number	Title	Effect on subsystems				
		AC&D	TT&C	Power	Propulsion	Thermal
Baseline	3-axis with spin injection. Full redundancy	Redundant AEM	Carrier I Full redundancy. SGLS compatible	AEM, full redundancy	TE 364-4, TE 364-15, N2H4 RCS, full RCS redundancy	AEM
Option 1	High accuracy	Add redundant star tracker and GRU	N/A	N/A	Add RCS for injection phase	N/A
Option 2	Spin stabilized	Remove 3-axis Add spin sensors	Revise antennas	Fixed solar array	Same as baseline	Analyze for spin on orbit
Option 3	Add Carrier II	N/A	Add Carrier II	N/A	N/A	Add cooling for transmitter
Option 4	Single string	Use AEM	SGLS compatible AEM	Use AEM	Use AEM	Use AEM
Option 5	Spin stabilized program by itself (cost only)	Same as Option 2	Same as Option 2	Same as Option 2	Same as Option 2	Same as Option 2

TABLE 2.0-2

3.0 VEHICLE DESCRIPTION

3.1 Configuration and Structure

The baseline LDSL-AEM spacecraft is a hexagonal configuration 60" across the points with built up aluminum structure around the periphery and from the base module out to the outer shell at each corner as shown in Figure 3.1-1. The AEM Base Module forms the center core. Equipment is mounted in the central AEM as in the present design with added equipment mounted in the outer volume attached to the corner structure. The batteries and transmitters are mounted near the outer surface panel with a louver to control the temperature.

The configuration of Option 1 is similar to the baseline as shown in Figure 3.1-2. The star trackers are installed with 45° between the two units to obtain good star coverage. The gyro reference unit is installed in the outside structure.

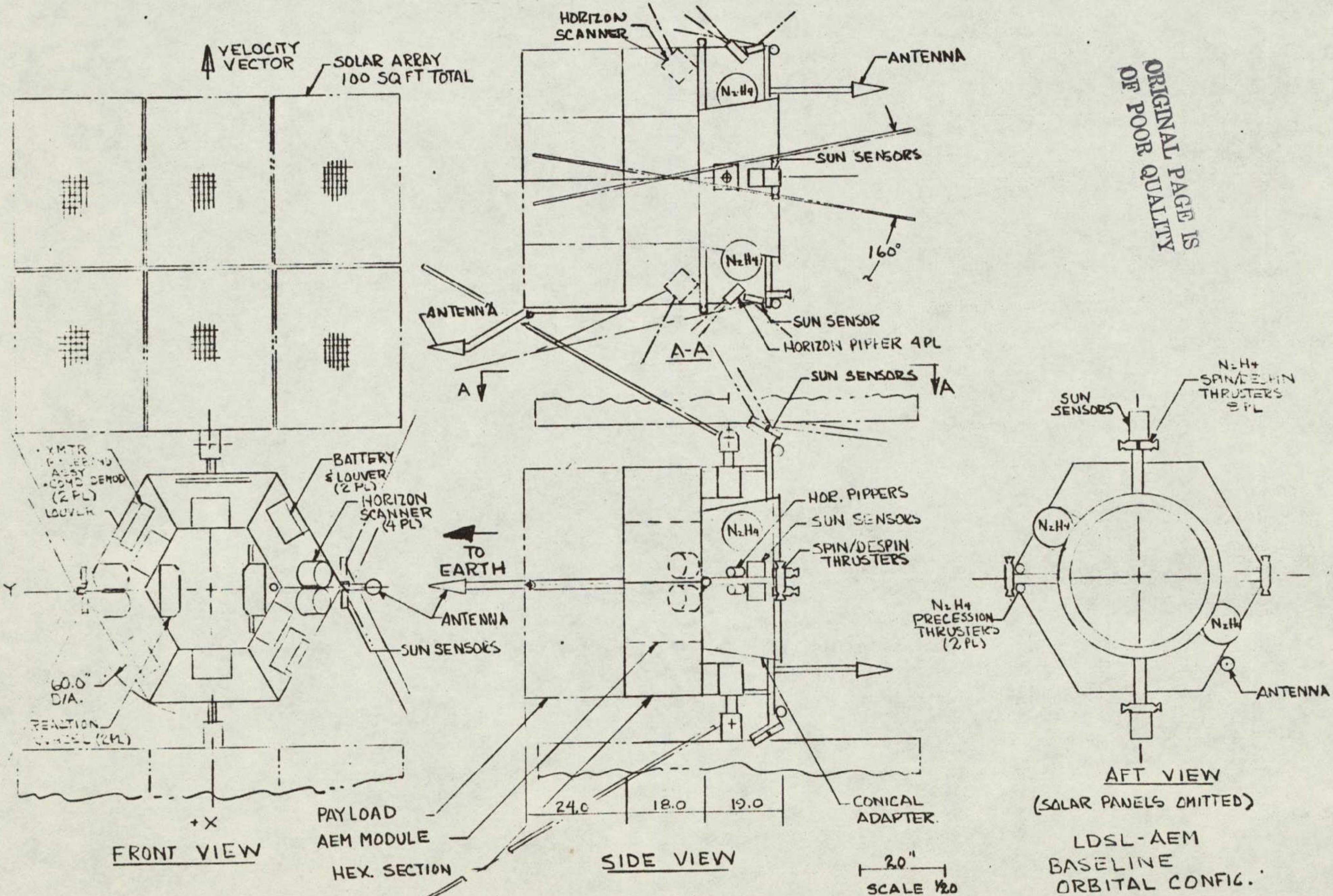
The spin stabilized configuration Option 2 is shown in Figure 3.1-3. The solar array is mounted fixed around the hexagonal structure. The sun and earth sensors are installed to provide attitude information as the spacecraft spins.

The baseline space vehicle with tandem TE-364-4 and -15 solid rocket motors installed in the launch configuration is shown in Figure 3.1-4. V-Band rings allow each motor to be separated after firing.

The LDSL-AEM can be installed in the shuttle over the short tunnel as shown in Figure 3.1-5, over the long tunnel (Figure 3.1-6) in clear payload bay, Figure 3.1-7, or over the OMS kit (Figure 3.1-8). A shuttle adapter structure and cradle allows the vehicle to be lifted out with the RMS. The same adapter configuration can be used for several locations.

3.1.1 Load Considerations

The LDSL-AEM, Figure 3.1-4 is a two stage vehicle launched from an orbiting space shuttle. It has the capability of being returned to earth in case of a failure prior to perigee motor burn. The LDSL-AEM is designed for the full set of shuttle induced environments and AEM free flight. The shuttle



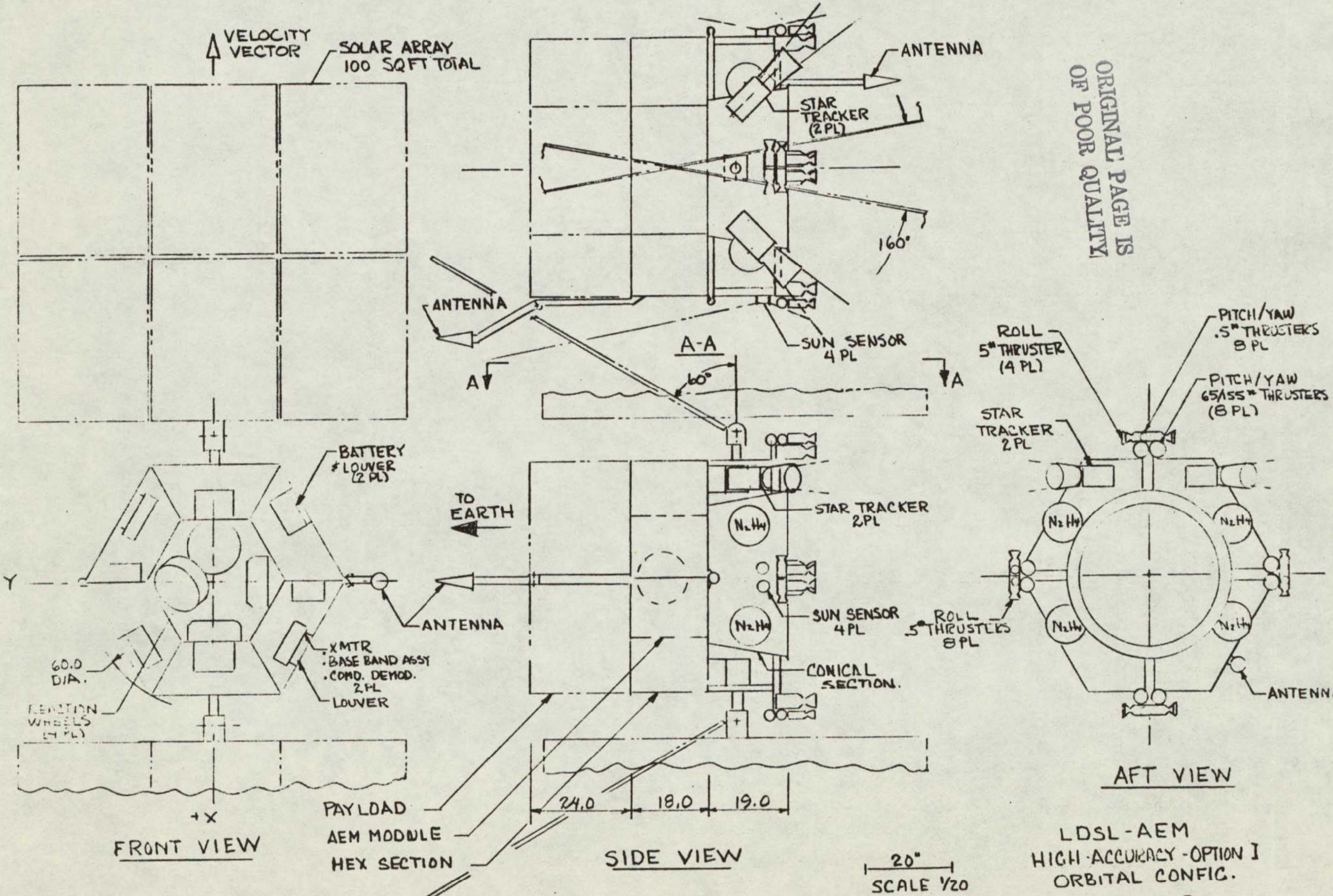
FRONT VIEW

SIDE VIEW

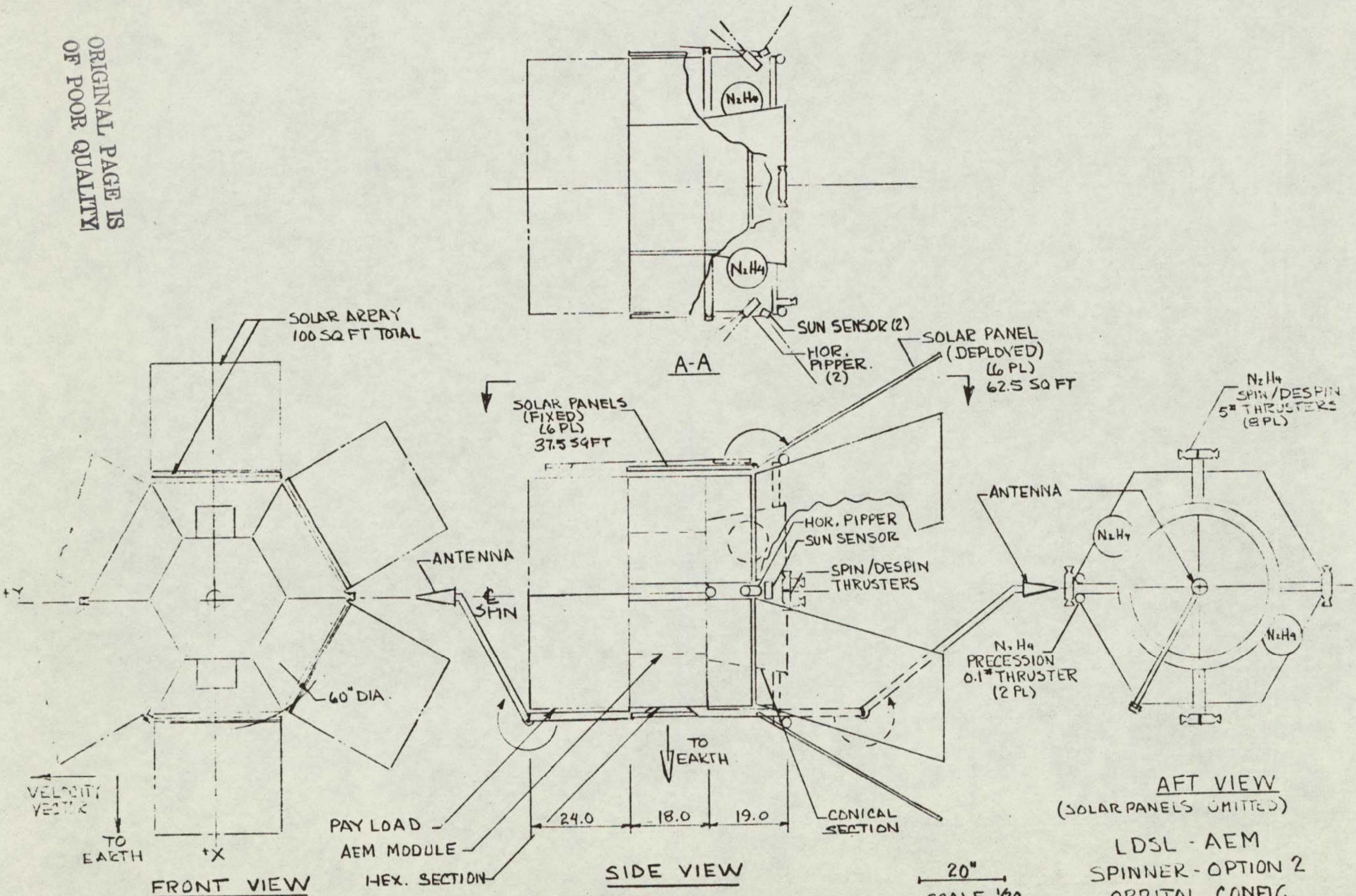
AFT VIEW

(SOLAR PANELS OMITTED)

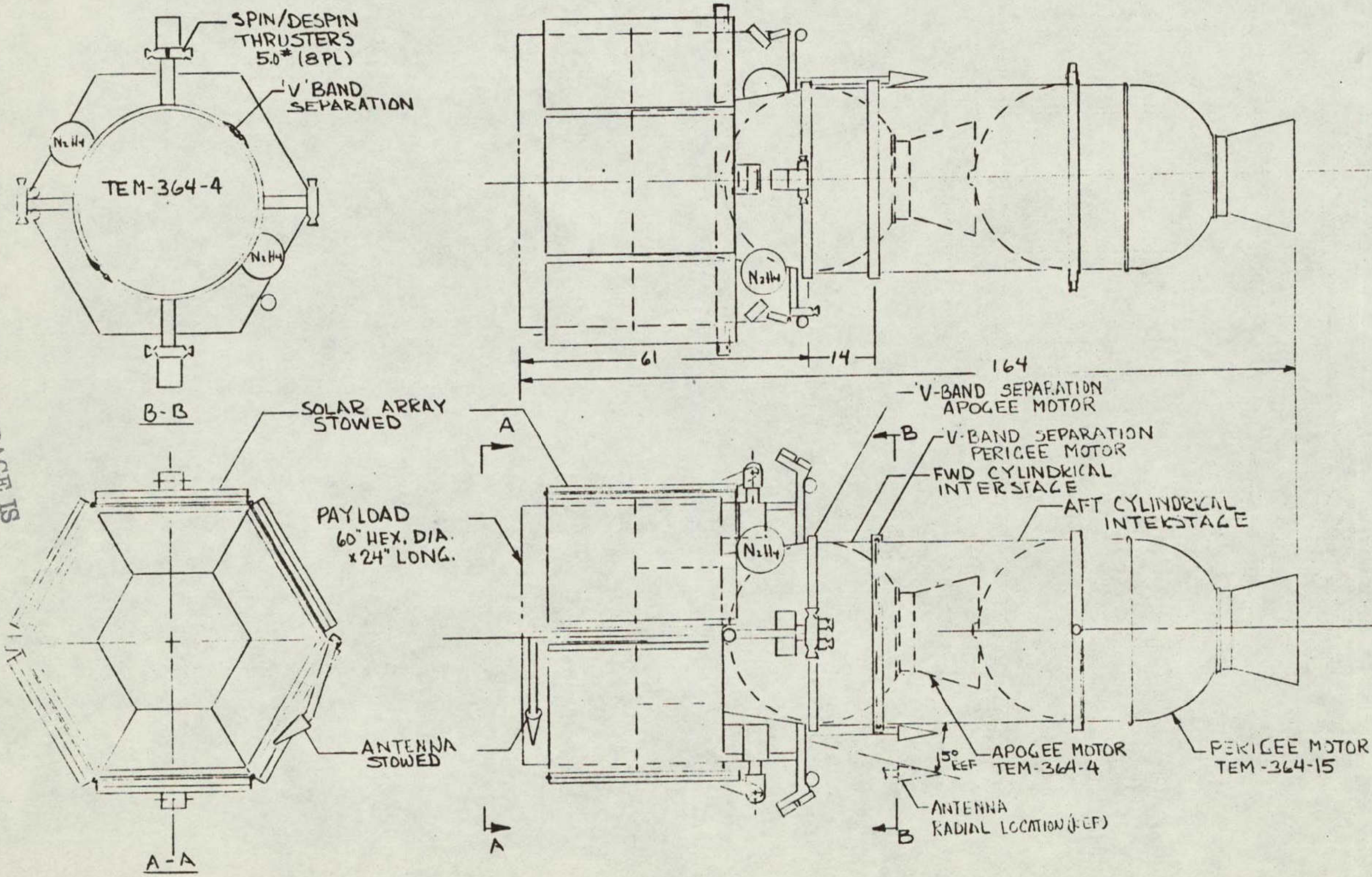
LDSL-AEM
BASELINE
ORBITAL CONFIG.
FIG 3.1-1



LDSL-AEM
HIGH ACCURACY - OPTION I
ORBITAL CONFIG.
FIG 3.1-2



LDSL - AEM
SPINNER - OPTION 2
ORBITAL CONFIG.
FIG 3.1-3



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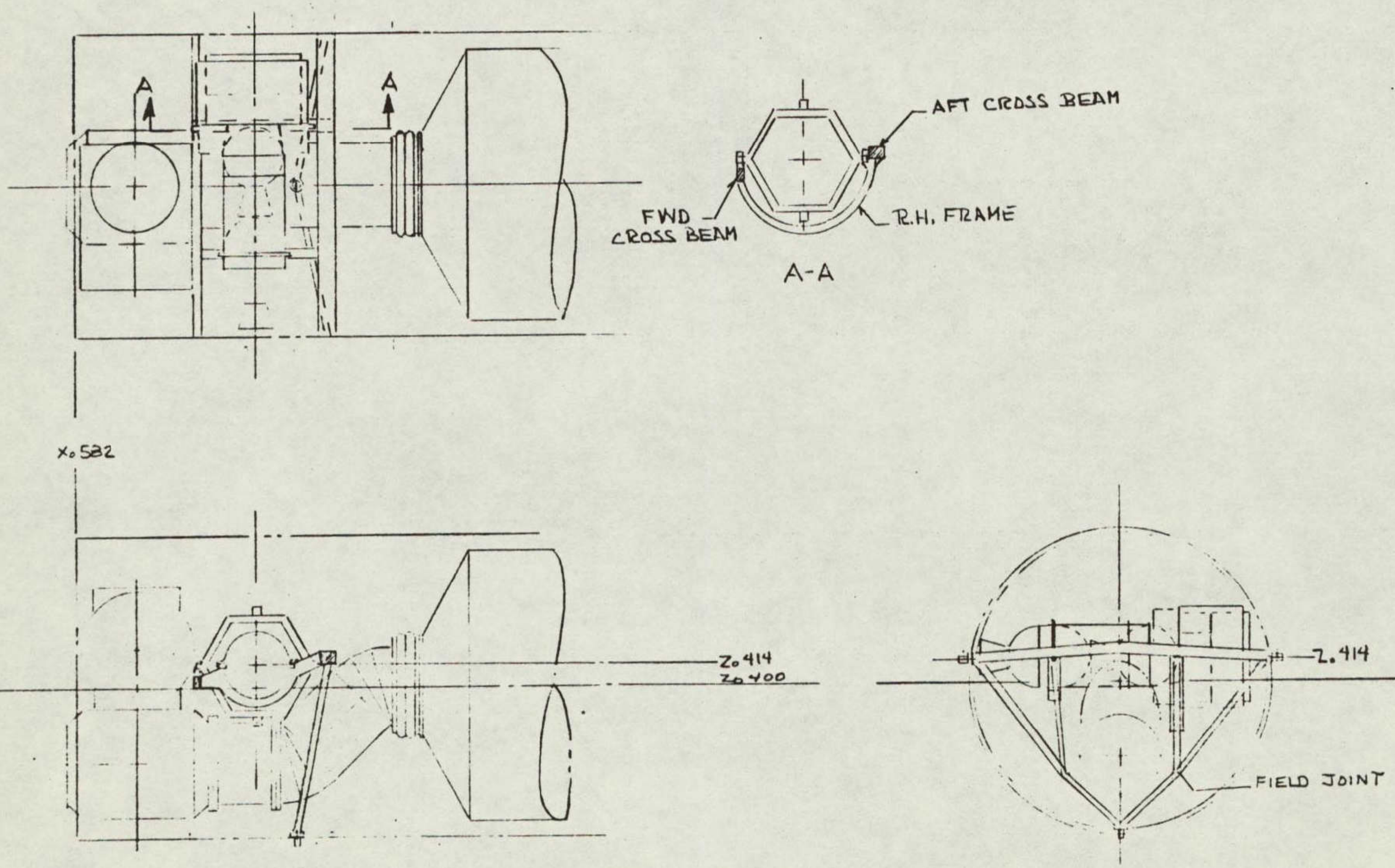
LDSL-AEM
BASELINE
LAUNCH CONFIG.
FIG 3.1-4



62"
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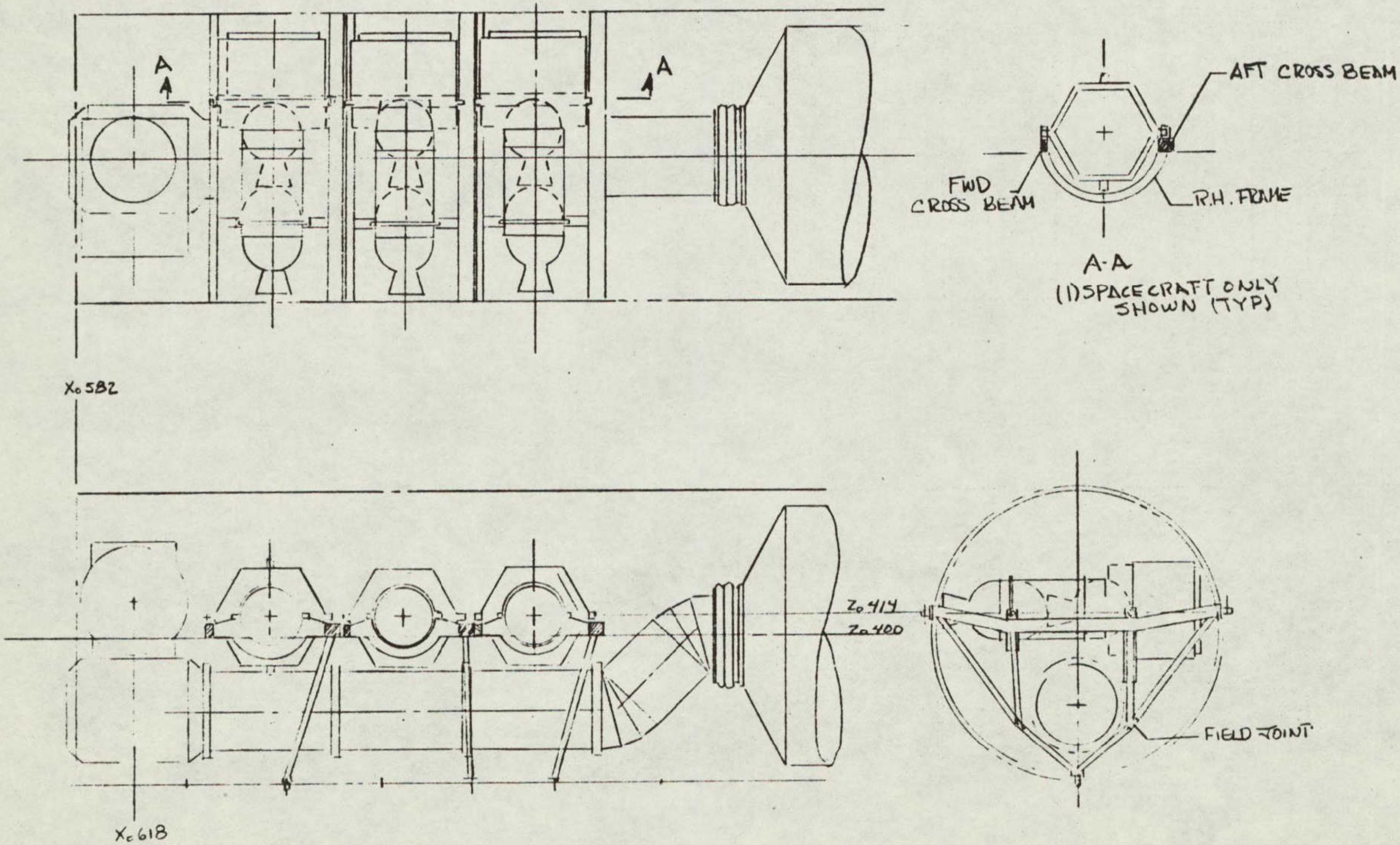
LDSL-AEM
BASELINE
SHUTTLE SHORT TUNNEL INST'L.
FIG 3.1-5



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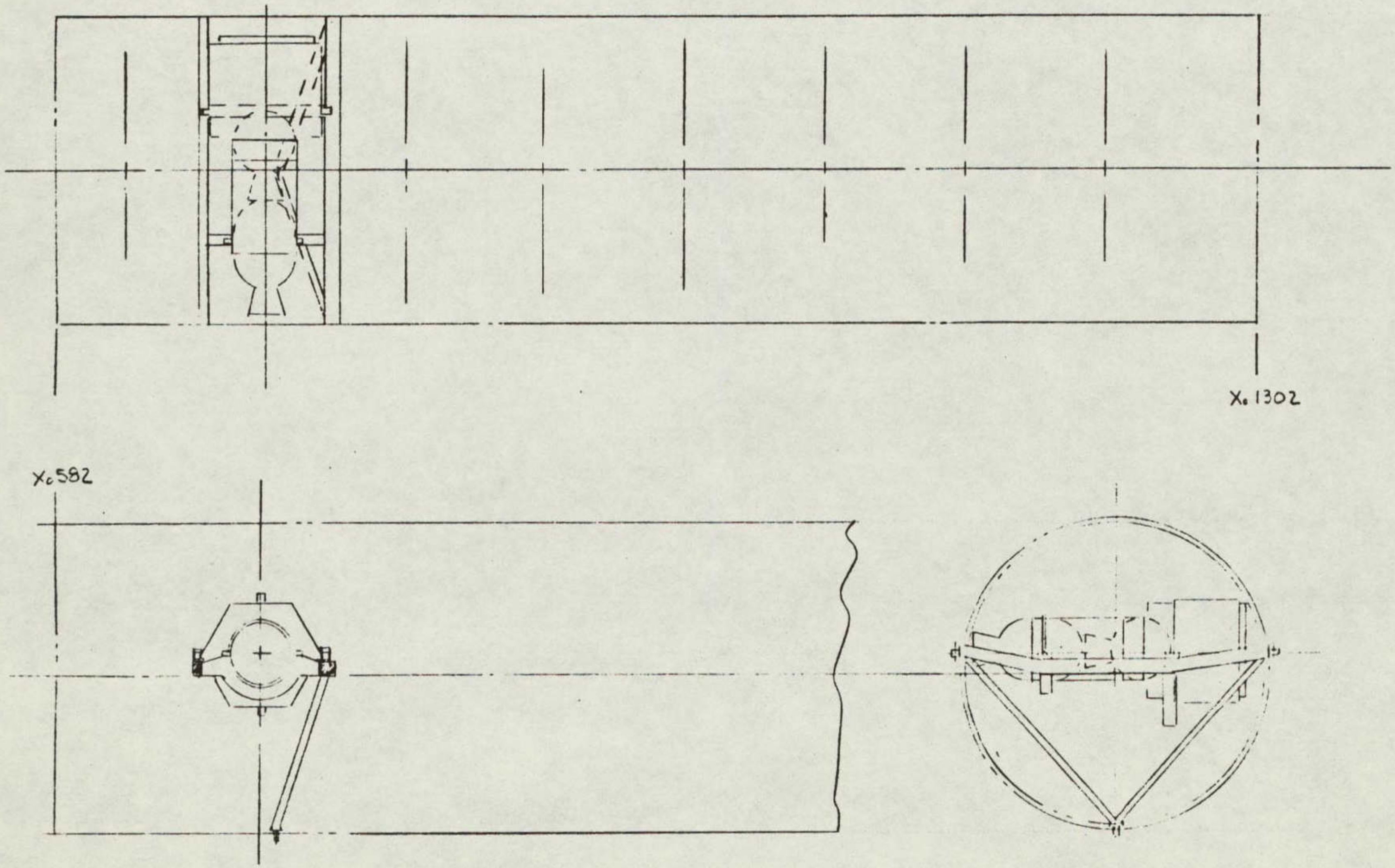
LDSL-AEM
BASELINE
SHUTTLE LONG TUNNEL INST.
FIG 3.1-6



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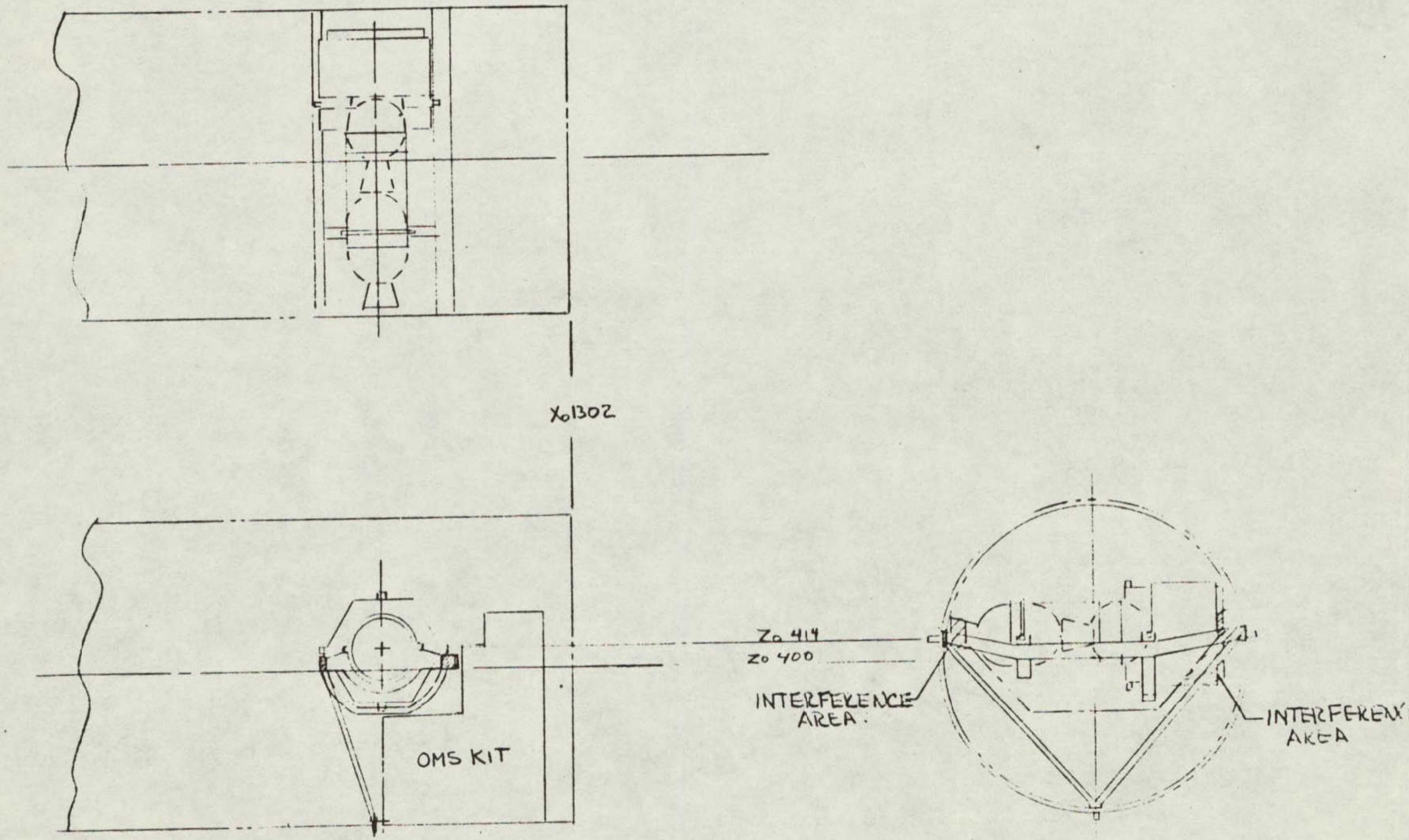
SCALE 1/62

LDSL-AEM
BASE LINE
SHUTTLE CLEAR SPACE INST.
FIG 3.1-7



62
SCALE

LDSL-AEM
BASELINE
OVER OMS KIT INSTL.

FIG 3.1-8

stowage area for the AEM requires the space vehicle be oriented across the shuttle equipment bay (Figure 3.1-5) and exposing the vehicle to accelerations in directions most difficult to accommodate. The shuttle adapter configuration minimizes weight penalties to the space vehicle from shuttle environemnts. The space vehicle plus payload is supported in the shuttle at the forward trunnions on the spacecraft and at the aft attachment ring of the aft rocket motor. Much of the structure of the spacecraft module is still designed by shuttle crash environment conditions. Crash acceleration requirements are taken from JSC 07700, Vol XIV, "Shuttle Payload Accommodations" as follows:

$$\left.\begin{aligned} N_x &= 9.0 \\ N_y &= 1.5 \\ N_z &= 4.5 \end{aligned}\right\} \quad \text{Ultimate}$$

3.1.2 Space Vehicle

The vehicle is made up of two primary sections; the spacecraft and the propulsion section (Figure 3.1-4). The spacecraft contains the experiments and supporting subsystems. The propulsion section contains two solid propellant motors, their connecting interstages and separation V-Band rings.

Figure 3.1-9 shows the vehicle shear and bending moment curve. The propulsion section interstage section and separation joints are designed by the first stage thrust at first stage burnout. Both interstages are cyclinders made of .080 in. 2024-T3 aluminum. The forward interstage contains the deployment attachment pick up fitting.

The spacecraft is a hexagonal structure, with provisions at the forward face for attaching a 1000 pound payload of compatible configuration. The spacecraft is built up aluminum structure with capability to accommodate equipment packages of the experiments and supporting subsystems. The primary structure of the spacecraft is a strong back containing the trunnions forming the forward attachment of the space vehicle to the shuttle adapter. The trunnion design is similar to the quick disconnects used to attach the shuttle adapter to the longerons. The vehicle's ultimate factor of safety is 2.0.

Space Vehicle Shear and Moment Installed in Shuttle Adapter

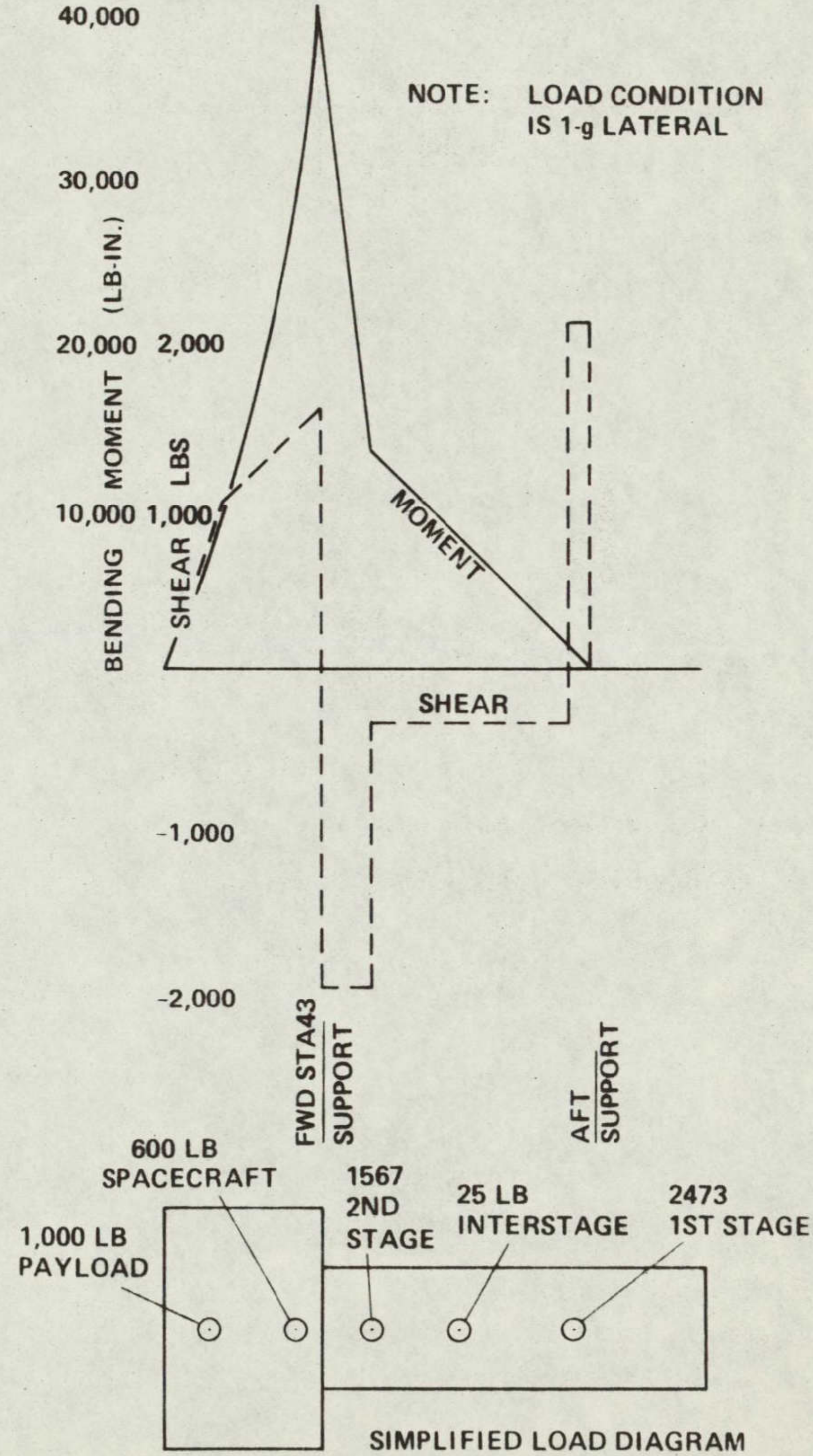


FIGURE 3.1-9

A cylinderical booster adapter section, attached to the aft face of the strongback is reinforced with stiffeners to distribute the high bending moment loads from the strongback into the propulsion section.

The strongback structure is not symmetrical about the vertical centerline. The N_x and N_y crash loads are reacted by the aft support beam of the shuttle adapter, with the left side of the spacecraft stronger than the right side.

The structure materials are principally 2024-T3 aluminum webs, and 7075-T73 chords.

Structural qualification will be by test and analysis. Proof tests to be conducted on the first vehicle will duplicate the critical loads experienced during shuttle carry and LDSL-AEM free flight. Qualification for crash conditions will be done by analysis. See Section 4.0.

3.1.3 Mass Properties

Table 3.1-2 shows weight for the baseline and the two optional configurations. Also shown is weight of the structural adapter required to support the space vehicle within the Orbiter Center of gravity and mass moments of inertia are controlled to meet requirements for thrust alignment, principle axis alignment and relationship of mass moments of inertia between axes. Final adjustments of mass properties is achieved by adding ballast to the space vehicle during the spin balance and MOI measurement tests as outlined in Section 4.0.

Weight Statement

	Baseline	Option 1	Option 2
Payload 1	(1,000.0)	(1,000.0)	(1,000.0)
Space vehicle structure	(244.0)	(244.0)	(244.0)
AEM primary and secondary structure	51.0	51.0	51.0
Added spacecraft primary structure	45.0	45.0	45.0
Added spacecraft secondary structure	50.0	50.0	50.0
Spacecraft adapter and separation	31.0	31.0	31.0
Propulsion interstage and separation 1	67.0	67.0	67.0
Attitude control and determination	(118.)	(141.5)	(27.4)
Reaction wheel	39.0	78.0	–
Horizon scanner	42.4	–	–
Magnetometer	4.0	4.0	–
Horizon pipper	2.6	–	2.6
Sun sensor–digital–acquisition	4.8	0.5	4.8
Rate gyro	3.0	–	3.0
Electro magnets	6.0	–	–
Computer	5.0	5.0	5.0
Interface electronics	12.0	12.0	12.0
Inertial reference unit	–	20.0	–
Star tracker	–	22.0	–
Telemetry tracking and command	(71.0)	(71.0)	(71.0)
Transponder baseband cmnd demod	15.0	15.0	15.0
Command and telemetry system	25.0	25.0	25.0
Tape recorder	30.0	30.0	30.0
S-band antenna	1.0	1.0	1.0
Electrical power	(286.2)	(286.2)	(279.2)
Solar array	110.0	110.0	110.0
Solar array drives	7.0	7.0	–
Batteries	74.0	74.0	74.0
Voltage limiter	5.4	5.4	5.4
DC converter	7.0	7.0	7.0
Ampere hour meter	7.0	7.0	7.0
Relays	24.0	24.0	24.0
Resistor panel	1.8	1.8	1.8
Wire and connectors	50.0	50.0	50.0
Thermal control	(13.8)	(13.8)	(13.8)
Louvers and louver radiators (6)	5.4	5.4	5.4
Multilayer insulation	4.6	4.6	4.6
Thermal coatings, FOSR, heaters	3.8	3.8	3.8
Reaction control	(44.4)	(113.8)	(44.4)
Spin/despin thrusters	16.0	–	16.0
Precession thrusters	1.0	–	1.0
Pitch-yaw thrusters	–	40.0	–
Roll thrusters	–	8.0	–
Coast control thrusters	–	8.0	–
Hydrazine tanks	8.0	16.0	8.0
Hydrazine and pressurant	13.4	31.2	13.4
Plumbing valves, filters, etc.	6.0	10.6	6.0
Propulsion 1	(4,040.3)	(4,040.3)	(4,040.3)
Apogee motor	1,567.0	1,567.0	1,567.0
Perigee motor	2,473.3	2,473.3	2,473.3
Ballast	(25.0)	(25.0)	(25.0)
Contingency	(80.0)	(80.0)	(80.0)
Subtotals–space vehicle and payload	5,923.5	6,015.6	5,825.1
Subtotals–spacecraft (does not include 1)	816.2	908.3	717.8
Shuttle adapter	(430)	(430)	(430)
Lateral support beam	108	108	108
Forward support beam	60	60	60
Aft support beam	210	210	210
Aft support ring	42	42	42
Forward fitting	10	10	10
Total (payload space vehicle shuttle adapter)	6,353.5	6,445.6	6,255.

TABLE 3.1-2

3.2 Attitude Control and Determination (AC&D)

The baseline attitude control and determination system is a direct derivative of the AEM System using spin stabilization for control during burn transfer and injection and momentum bias for on-orbit control. The transition from injection to on-orbit control including on-orbit attitude acquisition is accomplished autonomously. The expected performance is improved over the AEM capability by a factor of two yielding control in pitch and yaw to ± 0.5 degree and control in yaw to ± 1.0 degree. Expected attitude determination accuracy is ± 0.3 degree, ± 0.3 deg. and ± 0.5 degree for pitch, roll and yaw respectively. This performance is limited to orbit altitudes below 1000 nautical mile and for orbit inclinations greater than 30 degree.

The optional high accuracy AC&D system (option 1) uses a nominal zero momentum control with four skewed reaction control wheels for redundancy. Attitude reference is provided by 3 axis gyros updated by star trackers. Momentum dumping is by reaction control providing all altitude capability. Injection burn attitude control is also by reaction control with attitude reference from the gyro package. The gyros are run-up and uncaged while the space vehicle is still attached to the shuttle to establish an initial attitude reference. This system provides control to ± 0.05 deg and determination to ± 0.03 deg. about all axes. The system is not limited by orbit altitude or inclination.

The spinning spacecraft AC&D system operates with a ground control interface to close the basic loop. Spacecraft attitude is determined on the ground using data from the on-board horizon pippers and sun sensors.

Commands are then generated by the ground software to precess the spacecraft through a desired maneuver. Precession is achieved on board by pulsing the precession control ject using the sun pulse as reference. Active jet nutation damping is provided using measurements of the cross axis rates from the rate gyro. The expected performance for this system is ± 1 deg. for spin axis attitude control and 0.1 rpm for spin speed. Determination accuracies are 0.5 deg. and 0.01 rpm respectively.

3.2.1 Requirements

The performance requirements for the three AC&D systems, baseline, high accuracy and spinning are shown in Table 3.2-1. The major non-performance requirements imposed by the Shuttle are:

- o Space vehicle attitude relative to the shuttle is ill defined when it separates from the manipulator arm.
- o Attitude control propulsion systems can not be activated until the shuttle has maneuvered away and the space vehicle is in view of the astronauts.

These requirements imply that the space vehicle must have a post separation attitude reference and maneuver capability prior to the first injection motor burn. However, it is assumed permissible to have a gyro reference package active during separation.

3.2.2 Mission Profile

A preliminary mission profile for the baseline and both options (Table 3.2-2) satisfies their requirements.

3.2.3 System Description - Baseline

The baseline AC&D system is a 3 axis, stabilized, direct derivative of the AEM approach in that spin stabilization is used for injection burn control followed by autonomous transfer to an on-orbit momentum bias attitude control system using magnetic desaturation and nutation damping. A block diagram of this system is shown in Figure 3.2-1.

The mission profile is as outlined in Table 3.2-2. Subsequent to separation from the shuttle the attitude and reaction control systems are activated and the vehicle is spun up to approximately 60 rpm. Attitude determination is performed by ground software using data from the horizon pippers, magnetometer and digital sun sensor. The vehicle is maneuvered to the desired transfer attitude by time phasing the precession control jet pulses relative to the sun sensor observation time within the spin cycle. Rate gyros provide cross axis rate measurement for active jet nutation damping to corrct for the expected unfavorable inertia ratio

TABLE 3.2-1

AC&D SYSTEM PERFORMANCE REQUIREMENTS

PARAMETER		BASELINE	HIGH ACCURACY	SPINNING
Orbit Altitude	n.m	$<$1000	all	all
Orbit Inclination	deg.	$>$ 30	all	all
Orbit Injection Control	deg	± 3.0	±1 deg First burn ±3 deg Second burn	±3.0
Altitude Control Accuracy				
Pitch	deg	± 0.5	±0.1	±1.0
Roll	deg	± 0.5	±0.1	±1.0
Yaw	deg	± 1.0	±0.1	N.A.
Attitude Determination Accuracy				
Pitch	deg	± 0.2	± .03	±0.5
Roll	deg	± 0.2	± .03	±0.5
Yaw	deg	± 0.5	± .03	±0.5 (spin angle)
Attitude Rate Control	deg/sec	$<$ .01	.01	$<$ .01/$<$.01 rpm spin speed
Payload Disturbance				
Torque	ft lb	0.1	0.1	N.A.
Momentum	ft lb sec	0.05	0.5	N.A.

TABLE 3.2-2
AC&D MISSION PROFILE

BASELINE	HIGH ACCURACY	SPINNER
	Run up and align gyros	
	Separate from Shuttle	
	Shuttle Retro	
Spin Up (60 rpm)		Spin Up 60 rpm
Determine attitude (Ground Software)		Determine attitude (Ground Software)
Precess to First Burn Attitude (Ground Command)	Maneuver to First Burn Attitude (On Board Control)	Precess to First Burn Attitude (Ground Command)
	FIRST BURN	
Determine attitude (Ground Software)		Determine attitude (Ground Software)
Precess to Second Burn Attitude (Ground Command Approx 180 deg)	Maneuver to Second Burn Attitude (On Board Command)	Precess to Second Burn Attitude (Ground Command) Approx 180 deg)
	SECOND BURN	
Despin to nominally 0 rpm	Maneuver to On Orbit Nominal attitude	Despin to on orbit spin rate
Spin up momentum Wheels	Initiate Wheel Control	Determine attitude
Precess to on orbit attitude (on board magnetometer control)	Initiate IRU Acqusi- tion and Update using star tracker	Precess to on orbit attitude (Ground Command)
Activate horizon sensors		
Acquire earth		
Hold attitude	Hold altitude	Hold attitude (Ground Command)

Baseline Momentum Bias AC&D System

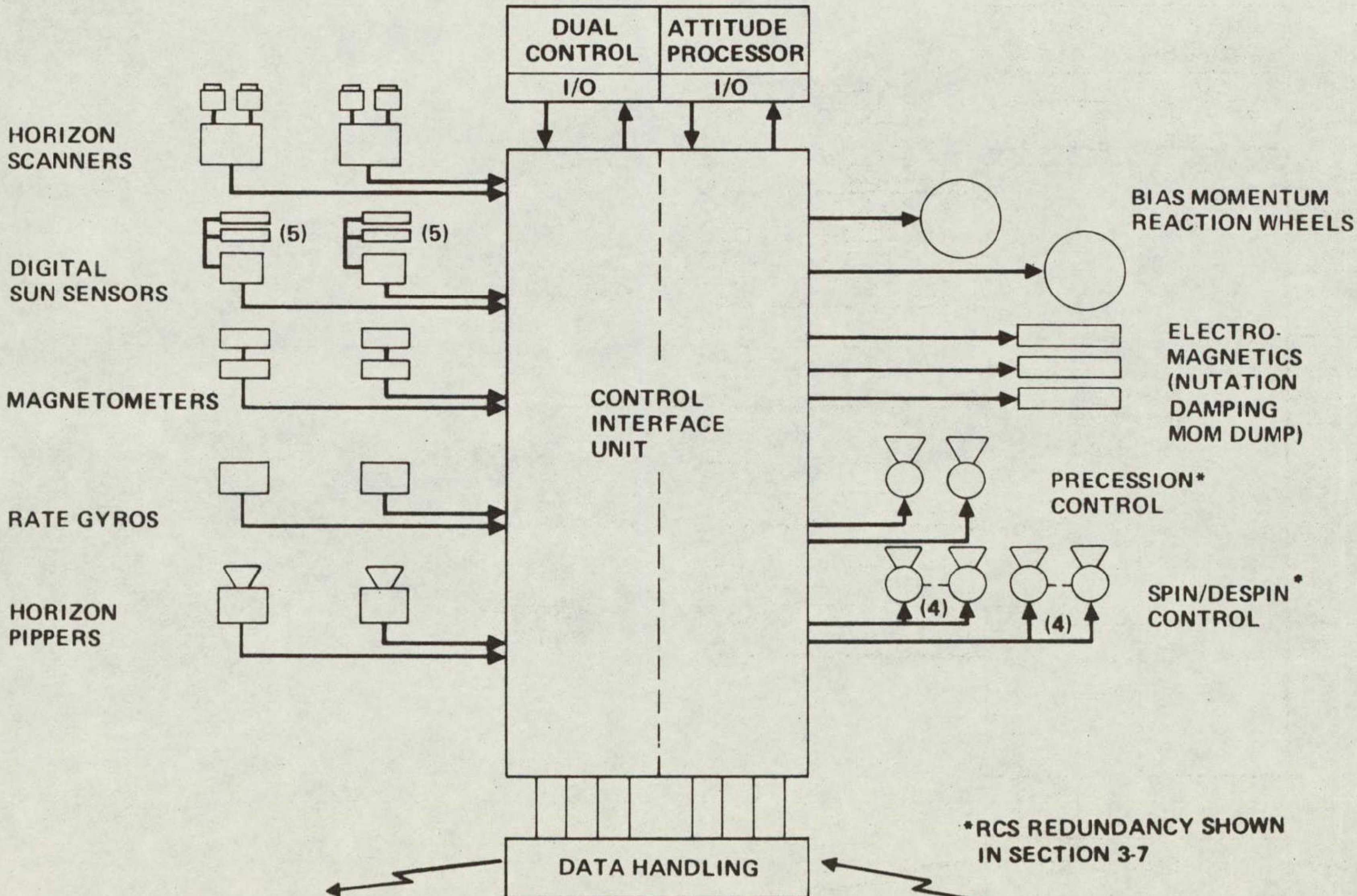


FIGURE 3.2-1

and space vehicle energy dissipation characteristics.

Subsequent to the first burn the mission proceeds using the injection spin control system until the termination of the second burn as shown in Table 3.2-2. After the second burn the vehicle is despun to nominally zero rpm using the jets. The momentum wheel is run up to partial speed and the horizon scanners activated. The vehicle at this point is tumbling arbitrarily. On-orbit attitude is achieved from this condition using the AEM magnetic controls. The horizon pippers and rate gyros are no longer required and could be disconnected. The spacecraft having achieved its local vertical attitude is controlled by the horizon sensor in a normal momentum bias system implementation with momentum dumping, precession, and nutation damping provided by active control of the electromagnets based upon the magnetometer measurements and the appropriate scaling of the AEM control laws.

The no single failure criterion is met by provision of a completely redundant set of equipment.

3.2.4 System Description - Option 1 High Accuracy

The high accuracy AC&D system, shown in Figure 3.2-2 uses 3 axis control for transfer and injection burns and for on-orbit control. The basic attitude reference is a 3-axis gyro package. The gyro reference is originally aligned in the shuttle and relies on the quality performance (low drift) of the gyros to provide sufficiently accurate attitude information during the phase from separation through the injection burns. Control torques are provided by high level reaction control thrusters for the motor burn phases low level thrusters for coast and maneuver. Rate information from the gyros is used for damping.

The gyro reference is updated by the star tracker information. This requires the storing of a star catalog in the on-board computer and providing the capability for updating the ephemeris state vector sufficiently often that accurate attitude estimates can be made. Autonomous navigation is not a planned capability of the system but could be incorporated. Control torques are derived from a zero net momentum reaction wheel system.

A 4π steradian coarse sun sensor is provided for re-acquisition in case of loss of the primary system prior to switching to the redundant system.

Momentum dumping is achieved using the low level reaction control system.

Option 1—High Accuracy AC&D System

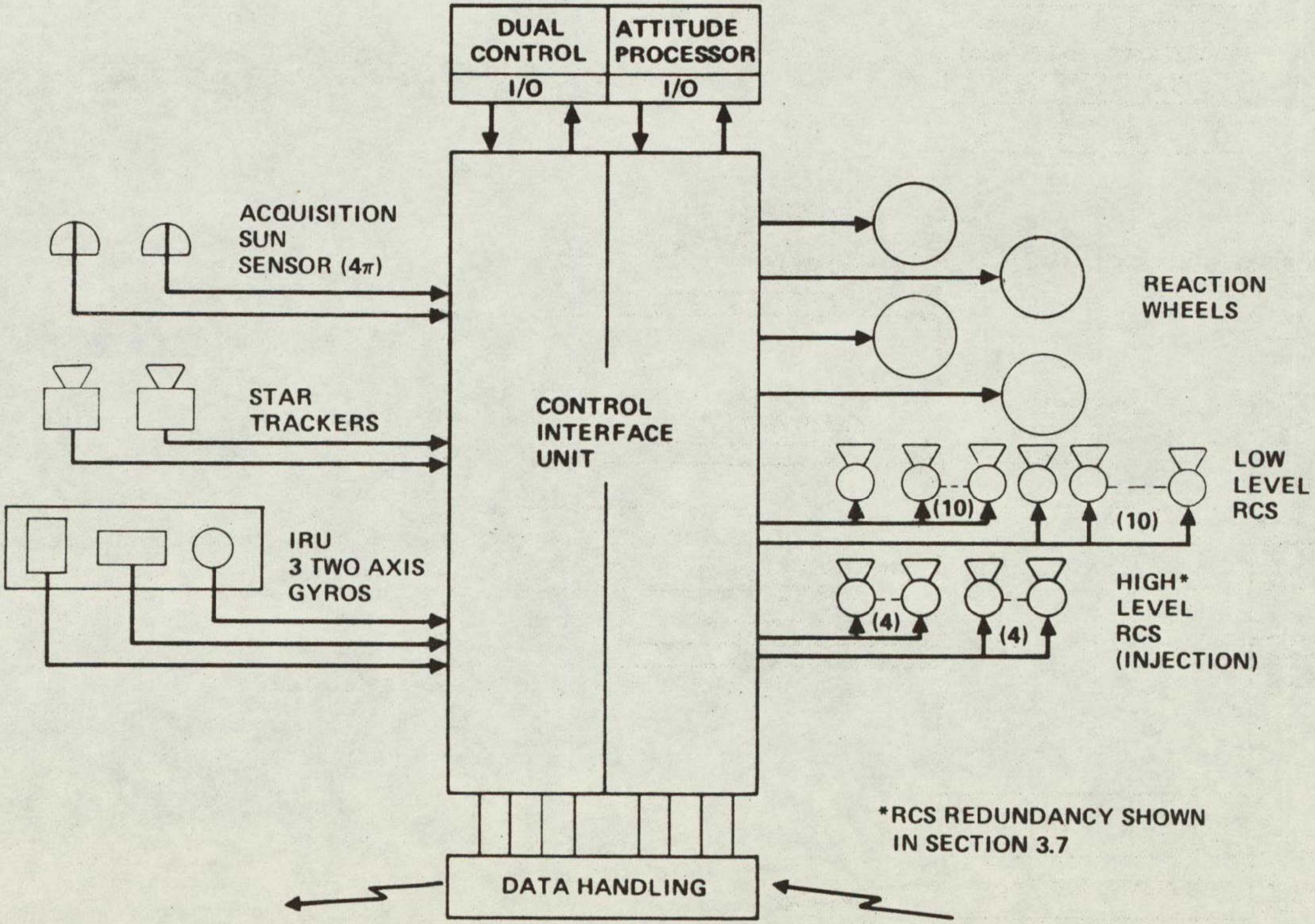


FIGURE 3.2-2

The implementation of this system does not use complete duplication of equipment to meet the "no single failure" criterion. In the gyro package there are two different approaches possible depending on the type of gyro used. If single axis gyros are used then functional redundancy can be achieved by adding a skewed fourth gyro allowing any three out of four to provide 3 axis rate measurement as in the Block VD System. If two axis gyros are used as currently planned and shown in the equipment list, then redundancy is achieved by orthogonally mounting three gyros. Failure of one gyro then leaves complete orthogonal 3-axis measurement capability. Reaction wheel redundancy is achieved by skewing so that three axis torquing can be achieved using any three of the four wheels.

3.2.5 System Description - Option 2 Spin Stabilization

The spinning AC&D system provides spin stabilization during both the motor burn phases and on orbit. A block diagram of this system is shown in Figure 3.2-3. The system operation through the injection burn phase is identical to the baseline system. Spin down is discontinued at the operational spin rate. The vehicle is then precessed to the on orbit attitude, usually the spin axis normal to the orbit plane. This desired attitude and spin speed is then maintained by the ground interface determining attitude and selecting the required precession and spin commands. Active nutation damping is provided by the precession jet and the rate gyro.

3.2.6 Single String AC&D System Option 4

A single string AC&D system is the same as the baseline except as shown in Figure 3.2-4.

3.2.7 Hardware Characteristics

The hardware characteristics of candidate equipment to implement the three AC&D systems is shown in Table 3.2-3. All candidate equipment, except the interface unit, is flight proven and, where possible, long production or NASA standard items have been selected. Some minor modifications would be required to the horizon pipper and digital sun sensor as described below. Since these changes have been implemented in similar types of equipment by these manufactur -s, their qualification status should not be compromised.

3.2.7.1 Equipment Modifications

Horizon Pipper - The Lockheed Type 8 horizon pipper was originally designed and operated for a 60 rpm spin speed. This is acceptable for the spinning injection

Option 2–Spinning AC&D System

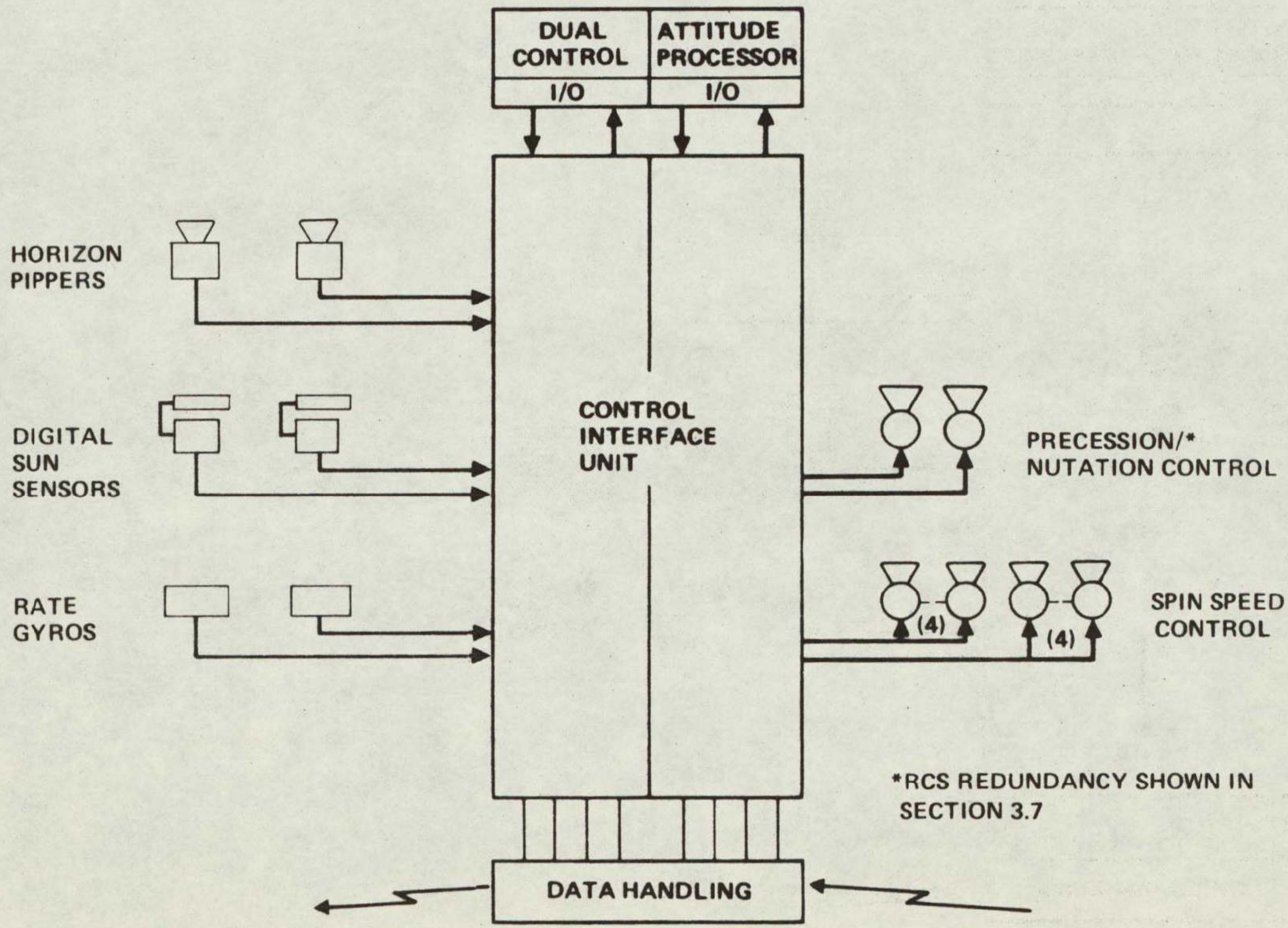


FIGURE 3.2-3

Single String Momentum Bias AC&D System

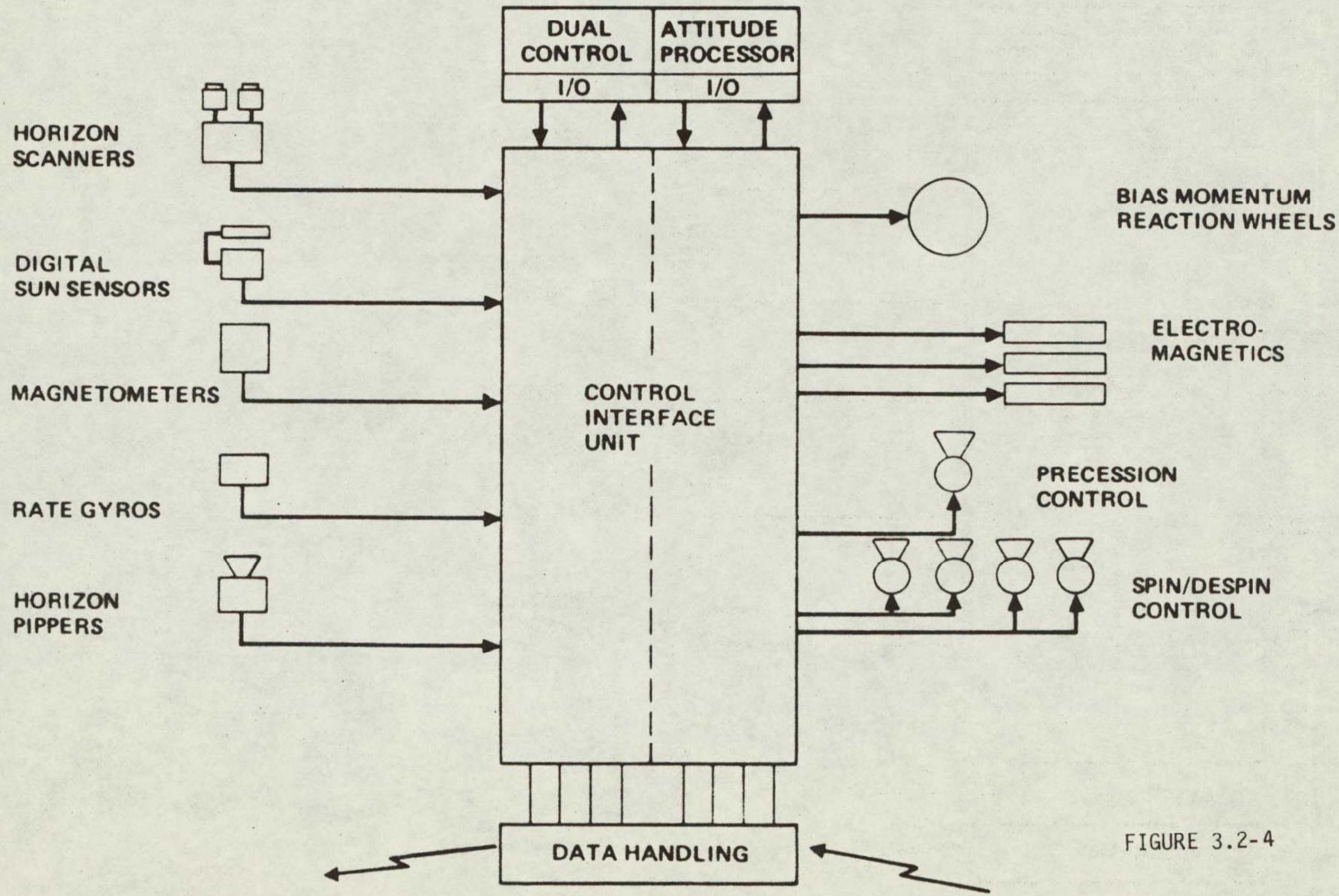


FIGURE 3.2-4

TABLE 3.2-3 AC&D EQUIPMENT CHARACTERISTICS

COMPONENT	CHARACTERISTICS	WEIGHT lb	POWER watts	USED ON	QUANTITY B/L	QUANTITY HA	QUANTITY SPIN
Reaction Wheel	Bendix Model 1964327* 8.5 ft lb sec @ 1250 rpm 30 oz in Torque Motor	19.5		ATS, LES, VELA	2	4	-
Horizon Scanner	Barnes 13-166 Conical Scanner ±0.05 deg accy	21.2	20	P50, P95 High Rel	2	-	-
Magnetometer	Schonstedt SAM 63C-1 3 axis fluxgate 2 Level	2.0	1.1	S3	2	-	-
Horizon Pipper	Lockheed Type 8	1.3	1.2	P72-1	2	-	2
Digital Sun Sensor	Adcole Model 16764 with added spin slit	2.4	.25	AEM	2	-	2
Rate Gyro	Northrup GR/G5	1.1	4		2	-	2
Inertial Ref. Unit	Honeywell IMU (4 gyro) or Kearfott DRIRU*	21 20	30 21	Block VD MSS	- -	1 or 1	 -
Star Tracker	Ball Bros CT401*	11	3	SAS-C	-	2	-
Electromagnets	Boeing 10-70440 Dual Wdg	2	1	S3	3	-	-
Attitude Control Processor	Typical-Bendix BDX910	2.5	17	B1 ESRO Spacelab	2[1]	2	2
Interface Control Unit	New	12	8	-	2[1]	2	2
Acquisition Sun Sensor	Ball Bros. Model C105	.25	0		-	2	-

NOTES * NASA Standard

1 Two units in one package.

application but performance could be degraded if the on orbit spin rate for the spinning AC&D is out of the range of acceptable performance for the internal filter networks. Additional switchable networks may be required for the on-orbit operation. These networks have been previously implemented in this sensor for the STP P72-1 with an operational spin rate of 10-14 rpm.

Digital Sun Sensor - The digital sun sensor axis that provides measurement of the sun angle relative to the spin axis in both the baseline and spinning AC&D systems will require a timing slit in one sensor head that will both strobe the angle measurement in the spinning mode of operation and provide a timing reference of the sun observation. A dual operating sensor has been implemented by Adcole in their Model 18273.

3.2.7.2 Computer Selection

The selection of the computer implementation approach of central versus dedicated computer is a trade that is not completely resolved at this time. A feasible approach that should be available within the time frame of this application is the use of a central executive machine with subsidiary dedicated microprocessors for special purposes, the most likely of which is attitude control. Since such equipment development is still in the early development stage, but progressing rapidly, the Bendix BBS910 was selected as being typical of the class of computing equipment required. Two BB5910 computers, integrated into one package meet the no single point failure criteria.

3.2.7.3 Control Interface Unit

The control interface unit is the spacecraft peculiar electronics required for the signal conditioning necessary to tie the attitude sensors and torquers, data handling system and control computer together. The unit is a standard piece of equipment capable of handling all of the systems, i.e., baseline, high accuracy and spinning. The redundánct unit is capable of performing the following basic functions.

- o Sensor output conversion from analog to digital form compatible for both the control computer input and output to the data handling system.
- o On Board transmission of sensor time tagging signals to the data handling system.

- On Board transmission of commands and data updates from the data handling system to the control computer. A limited re-programming capability is also incorporated.

- Conditioning of the computer control outputs (DIA) into the form required for reaction wheel, electromagnet and reaction jet operation.

- Onward transmission of selected computer outputs to the data handling system.

3.2.7.4 Software

The software for the on board computer for the baseline system is implemented in two parts, covering the spinning injection phase and the on orbit phase. The spinning system requires only a timing function response to the variable time delay of the precession jet pulsing relative to the sun sensor observation time and the thresholding limit of the rate gyro output for nutation damping. The basic computation of the precession jet delay time is a ground computation function. The on orbit control software will be a digital implementation of the AEM control laws with suitable scaling.

The software for the high accuracy system is a significantly larger program than for baseline system. Some of the basic tasks are:

- Use the outputs of the strapdown gyro package as inputs to a quaternion attitude integration algorithm
- Using this output determine the vehicle altitude relative to the desired reference frames.
- Periodically update the attitude state using star sensor measurements in a Kalman Filter estimater and obtain estimates of gyro rate bias errors.
- Perform task queing based on priority including interrupts of lower priority tasks such as telemetry.
- Based on the control law algorithm generate reaction wheel torque commands and reaction jet on-off commands for both powered and unpowered flight.
- Monitor wheel speeds and generate momentum transfer and dumping requirements.
- Process data for telemetry output
- Recall from the stored catalog the required star coordinates

- Accept updates of ephemeris state vectors and propagate to the times of star sensor observation
- Monitor commanded wheel failure and select appropriate transformation matrix
- Perform rudimentary reasonableness checks on sensor and wheel outputs to detect gross anomalies.

The spinning system software implementation will be the same as for the injection system of the baseline.

3.3 THERMAL CONTROL SUBSYSTEM

The thermal control subsystem for the Large Diameter Shuttle Launched AEM Spacecraft is similar to that of the AEM spacecraft. Thermal control is provided by a combination of multilayer insulation blankets, low conductance payload attachment fittings, heaters, thermal coatings, and louvers. Thermal insulation blankets isolate the spacecraft from the external environment and low conductance attachment fittings isolate the spacecraft from the payload. The hydrazine reaction control system temperatures are maintained by heaters. Six internal thermal-control louver mounted on the sides of the hexagonal structure provide temperature control over the range of internal heat dissipation. Flexible-optical-solar-reflectors (FOSR) cover the louver assemblies to minimize the effects of varying solar flux. The spacecraft overall heat balance is adjusted by means of fixed area radiators covered with FOSR. The interior spacecraft surfaces and components are painted with a high emittance paint to enhance internal heat transfer. Exposed exterior surfaces are painted with white paint to minimize solar heating effects.

3.3.1 Design Requirements

The thermal control system is required to maintain the spacecraft component temperatures within specified limits for all orbit conditions. Table 3.3-1 shows the component qualification temperature limits and the design requirement temperature limits. Since the spacecraft is to fly a wide variety of missions, the thermal control system must function at any spacecraft attitude in low-earth to geosynchronous orbits.

Table 3.3-1 COMPONENT TEMPERATURE LIMITS

Item	Qualification -Limits °C	Design Requirement °C
Batteries	-5 to +35	+10 to +30
Electronics	-10 to +50	0 to +40
Scan Wheels	-15 to +55	0 to +35
Hydrazine System	---	+10 to +65
Sun Sensors	-20 to +60	-10 to +50
Solar Panels	-100 to +120	-85 to +100

3.3.2 Thermal Control System Description

The LDSL AEM incorporates a combination of passive thermal-control components plus reaction-control-system heaters to provide adequate temperature control. The selection of thermal-control components and design philosophy is based on the AEM spacecraft design.

Figure 3.3-1 depicts the thermal control system. The majority of spacecraft components are mounted on the inside surfaces of the hexagonal structure. This structure is covered with multilayer insulation blankets (except for controlled radiating surfaces) to thermally insulate the interior region from the external environment. Exposed surfaces are covered with flexible optical solar reflector minimize solar heating effects. The thermal mass within the spacecraft is used to damp transient equipment heat loads. Internal radiation and conduction coupling reduce internal temperature gradients. High emittance painted surfaces enhance the radiation coupling. Thermal control louvers with FOSR surfaced covers are mounted on the hexagonal surfaces to provide temperature control. The overall spacecraft thermal balance is adjusted by using exposed radiator surfaces covered with FOSR.

Multilayer Insulation Blankets

The multilayer insulation blankets consist of (1) an outer layer of aluminized 1-mil Kapton (Kapton facing outward), (2) 10 layers of doubly aluminized 1/8-mil perforated Mylar separated by silk net spacers, (3) a single layer of Dacron scrim cloth to act as a filter, and (4) an inner layer of aluminized 1-mil Teflon (Teflon facing outward). The edges of the blanket are taped or sewn. The design value of effective emittance for the blankets is 0.02.

Thermal Control System

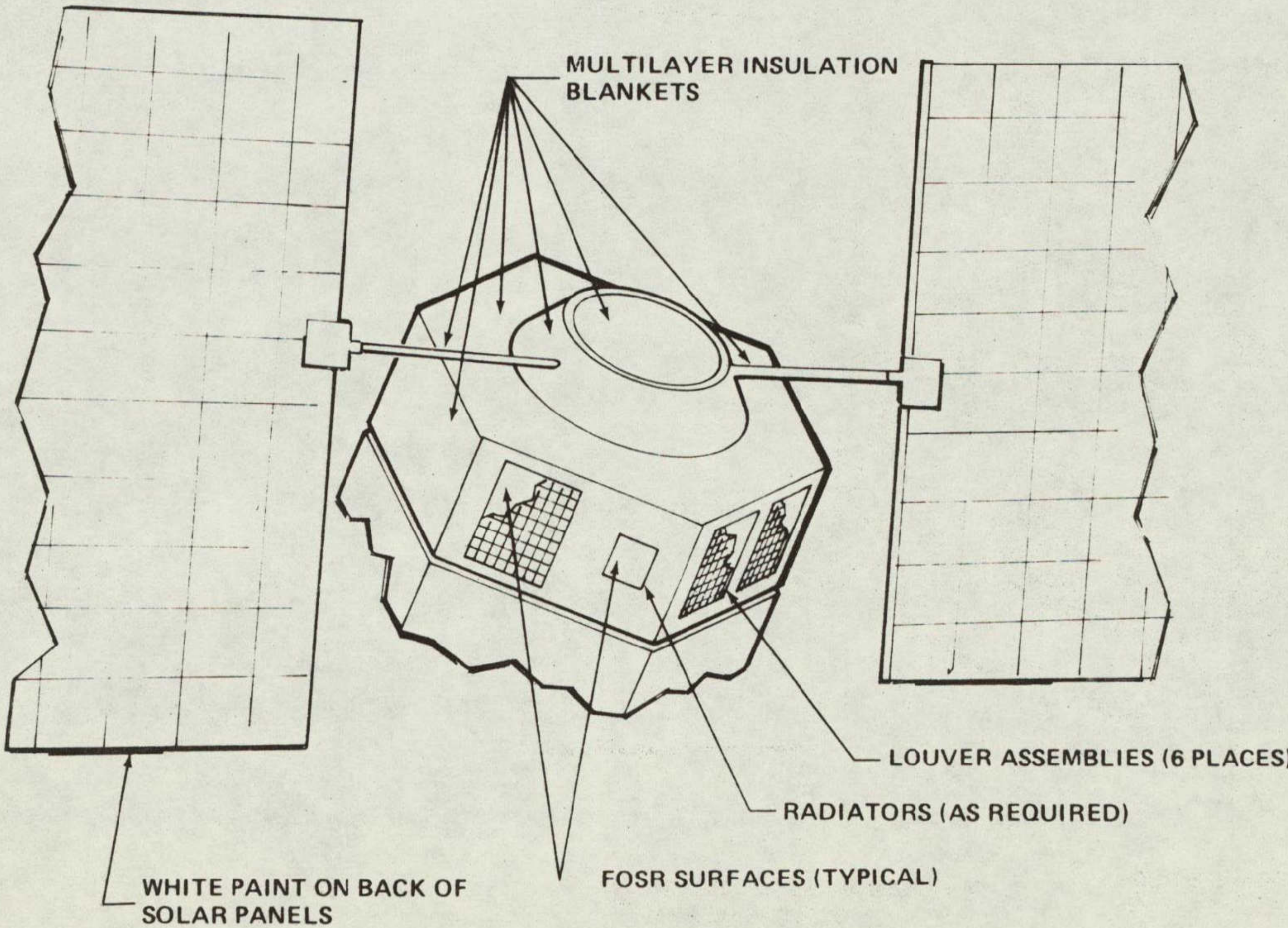


FIGURE 3.3-1

Thermal Control Coatings

The white paint (MS 74), used on the back of the solar panels and various other exposed surfaces, has a solar absorptance to infrared emittance ratio of 0.2/0.85. The FOSR have a solar absorptance to infrared emittance ratio of 0.08/0.78.

Thermal Control Louvers

The thermal control louvers are Northrup 8 x 16 inch louvers with a FOSR coated cover to minimize the effects of solar illumination. Figure 3.3-2 shows the effective emittance of the louver assembly. The variable emittance, over a selected temperature range, provides thermal control of the spacecraft.

Thermal Performance

Figure 3.3-3 shows the thermal control system performance. This performance was determined for a system with no exposed radiator surfaces to adjust the overall thermal balance. The extreme cold case assumes no environmental heating.

The extreme hot case assumes direct solar heating of the mutilayer insulation covering on top of the spacecraft. For these extremes, Figure 3.3-3 shows thermal control (10 - 35° C) for heat loads between 40 and 90 watts. This performance may be improved by refining the thermal control system design. This refinement requires detailed thermal analyses.

3.3.3 Thermal Design Verification Tests

The thermal design is verified at the system level during the thermal-vacuum testing. The test is designed to provide two basic verifications.

1. Verify the satellite system performance and the ability of the thermal control system to maintain temperature control under orbital operating conditions (in a simulated environment).
2. Verify the computer math model used to predict actual orbital temperatures, and to analyze temperature sensitivities due to expected system and environmental variations.

Verification of item 1 is made by: a) adjusting the test chamber environment as close as practicable to the range of real orbital conditions (solar simulation, earth IR and albedo heating and spacecraft orientation), b) operating the satellite over its planned timelines corresponding to worst hot and cold conditions sufficient to meet the system test objectives.

Effective Emittance of Northrup Louver With Radiator Cover Installed

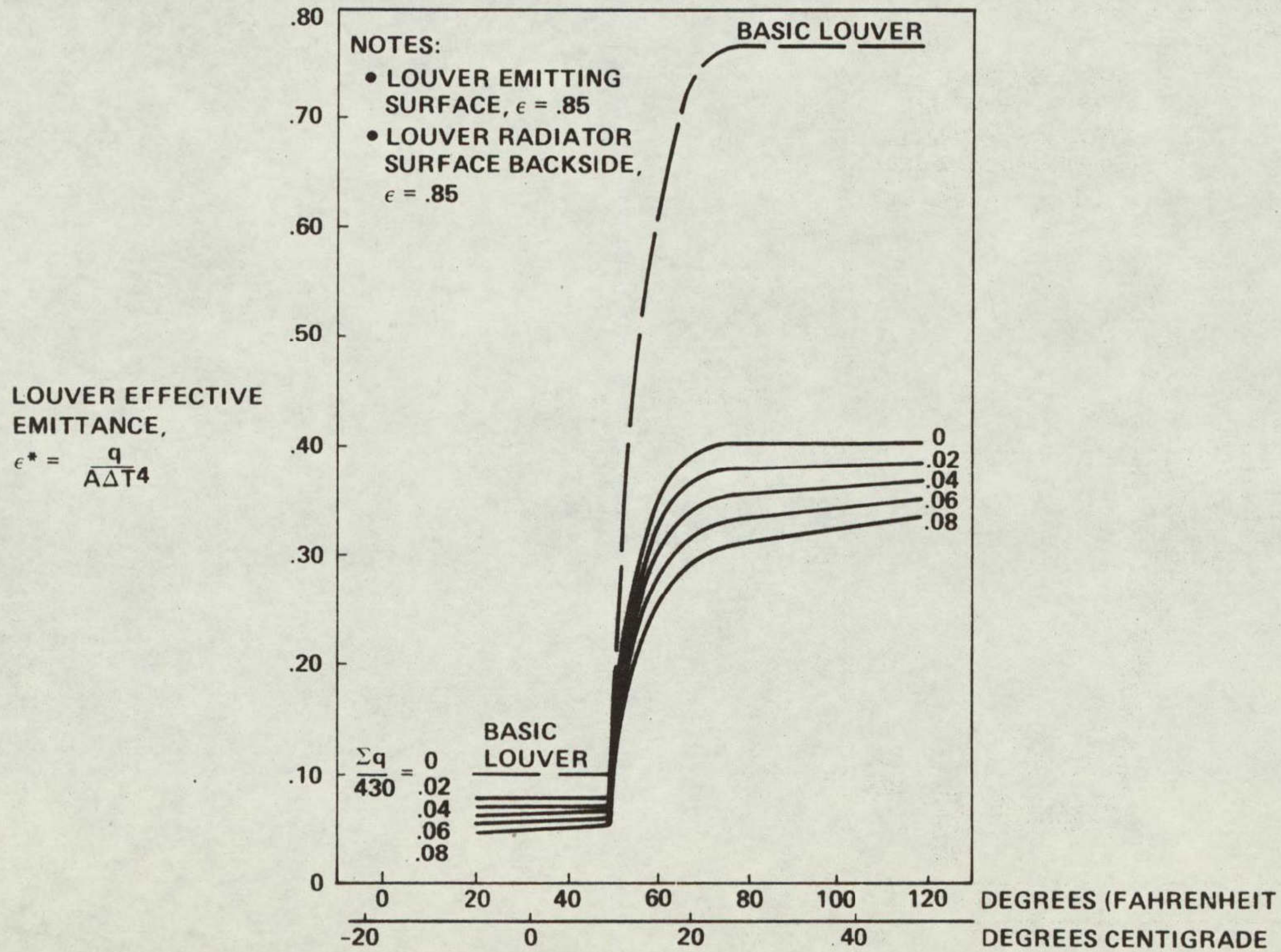


FIGURE 3.3-2

Thermal Control Performance

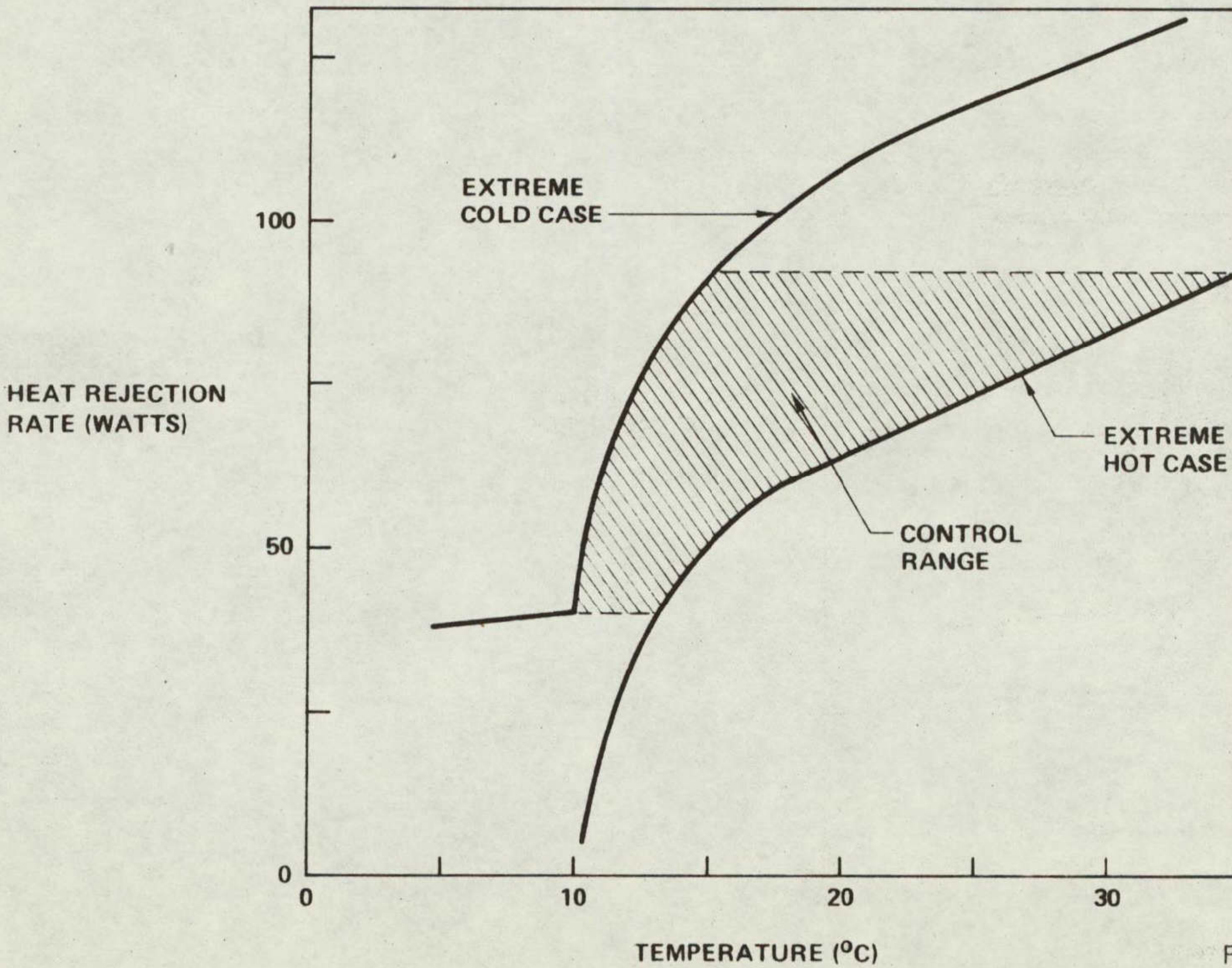


FIGURE 3.3-3

Verification of item 2 is accomplished by inputting the actual chamber environments (thermal fluxes) impinging on the external satellite surfaces, during the cold and hot steady state phase of the thermal vacuum tests as boundary conditions to the computer thermal model. This model is then used to predict thermal vacuum temperature levels and distributions. This data is compared to the actual test data and the math model is upgraded as required until satisfactory correlation is attained.

3.4 Electrical Power Subsystem

3.4.1 Subsystem Operation

The electrical power subsystem Figure 3.4-1 generates, stores, controls, and distributes the electrical energy required to operate the spacecraft systems and experiments. Power generated in two solar panel arrays is either stored in two 20 AH nickel cadimum batteries or used directly by electrical loads. Each array section, when normally illuminated provides 500 watts. Voltage limiters control the excess current shunted thru a resistor load bank to maintain the spacecraft bus at 28 $\pm$4 VDC. Battery charging is controlled by an ampere-hour meter. The battery is allowed to charge at rates up to C/3, this rate is determined by the array output capability. The ampere-hour meter measured battery state-of-charge (SOC) by integrating battery current with time. When full charge is reached, the voltage limiter switches to a trickle charge mode.

Loads are connected to one of two independently controlled power busses. Approximately 1/2 of the available solar array output is distributed to spacecraft loads attached to load bus #1 and the remainder of available power is distributed to loads attached to load Bus #2. Control of each bus is independent, each charging a separate battery. The batteries are allowed to charge and discharge at rates and duty cycles dependent on ground configuring of loads and the experiment timeline operations required.

The solar array for the baseline and option #1 is comprised of two 50 ft^2 panels. Both are capable of being rotated about the spacecraft roll axis in increments to take advantage of optmum solar inputs as the spacecraft orbit precesses with respect to the sun vector. The solar array performance in a circular orbit is shown in Figure 3.4-2 as a function of the angle measured in the orbit plane referenced to the point of intersection with the ecliptic plane. The output profile as it varies with the β angle (angle of solar vector measured from the orbit plane) is shown

Power Subsystem
(LDSL–AEM)

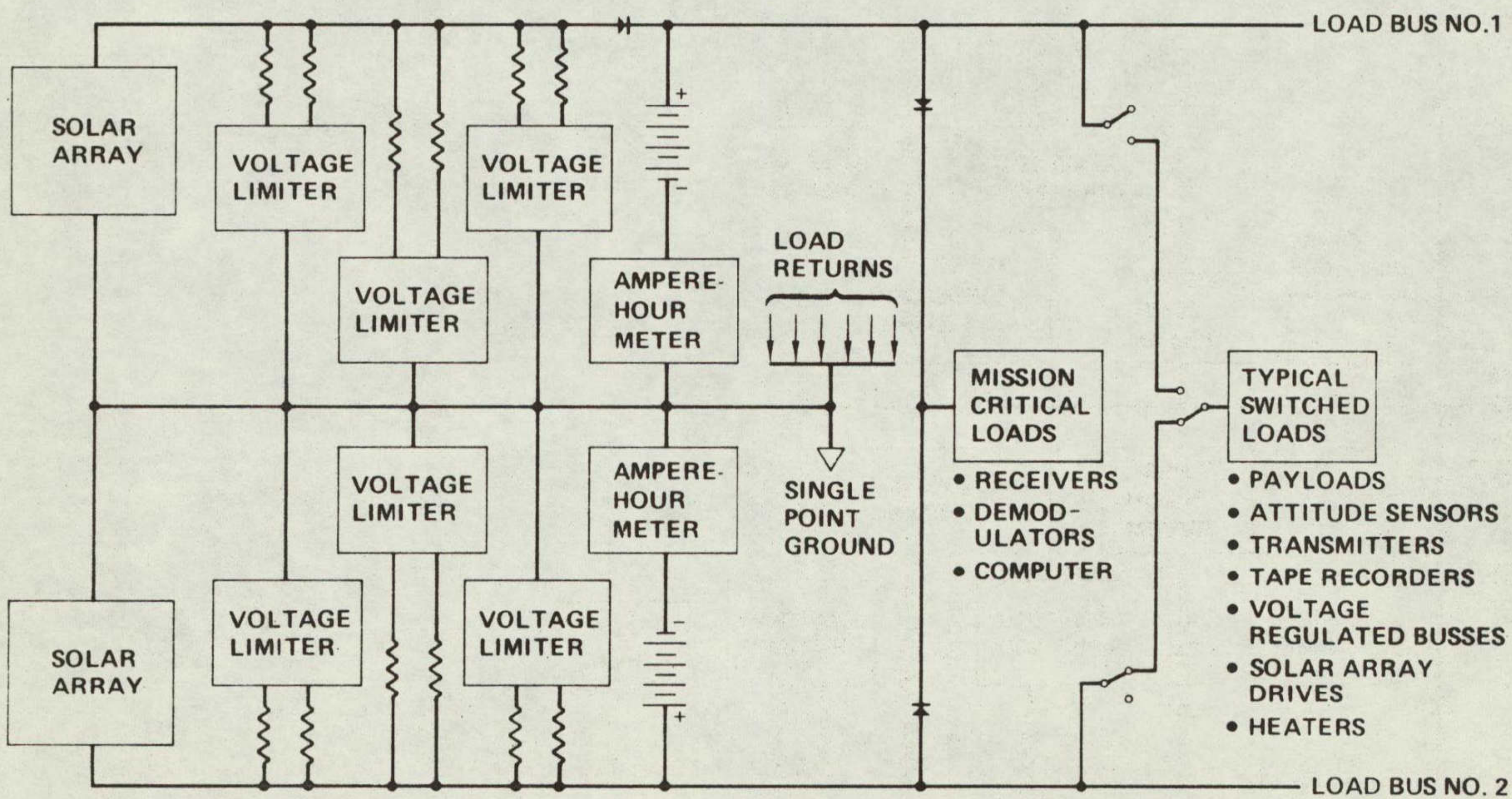
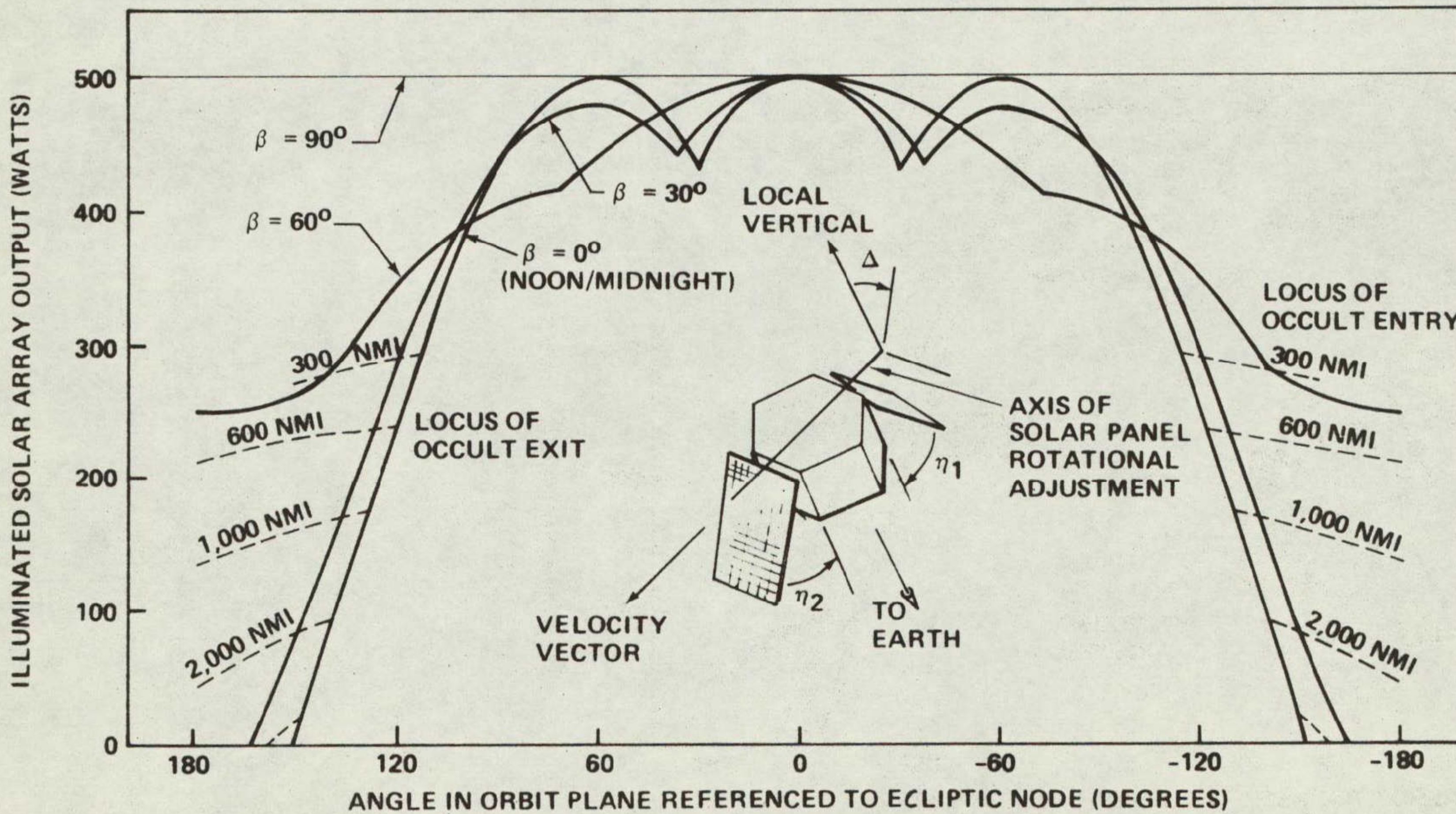


FIGURE 3.4-1

Typical Solar Array Performance of AEM Adaptation for Large Payloads in Low Earth Orbits (Application for All Sun Angles)



ASSUMPTIONS:

$\eta_1 = \eta_2 = 30^{o}$ FIXED BEFORE LAUNCH

$\Delta = 90^{o} - \beta$ ADJUSTED ON ORBIT BY COMMAND

ARRAY OUTPUT – 10 WATTS/FT^2

FIGURE 3.4-2

as a third variable. Occult entry and exit are shown as a function of altitude. The angles η are variable but fixed before launch. The 30^0 angle was chosen as representative of desireable uniform array output during the illuminated portion of an orbit. The η angle provides flexibility to adjust the array output to meet the mission profile requirements.

The 100 ft^2 of solar panels are body mounted on the cylindrical surface for the spin stabilized configuration, option #2, to provide uniform output throughout the spin cycle.

The solar array performance in terms of average illuminated solar array output is presented in Figure 3.4-3 for the baseline and options #1, #2, as a function of the β angle. The performance is based on 100 ft^2 of solar array subject to the notes listed on the figure. Both the fixed and spin stabilized configurations will benefit from earth albedo especially at low altitudes and low β angles. It is estimated that earth albedo will increase the output performance $\sim$ 10 to 15% at β = 0 and 300 N.Mi altitude.

The subsystem operates in the following modes as determined by the orbital parameters and operating routines.

- o When the solar array is illuminated, electrical energy generated by each array section is supplied directly to the spacecraft and payload systems attached to load bus #1 and #2 respective bus charges the battery attached to that bus. Note that the load bus voltage is as determined by the battery state-of-charge, charge rate, and temperature but not in excess of 32.0 VDC. Each load bus voltage, therefore, 'floats' at the battery, terminal level and is limited only when the 100% state-of-charge signal is present or when a maximum of 32.0 VDC is reached.

- o When the spacecraft load demand on either bus exceeds the solar array capability, the battery provides the additional load current. The load bus ampere hour meter monitors battery state-of-charge and shuts off all but essential loads at 29% SOC. Such loads are retained in a disable state until the ampere hour meter is reset or disabled, the 29% SOC signal is disabled, or

Dependence of Solar Array Output on Altitude, Configuration Option and β Angle

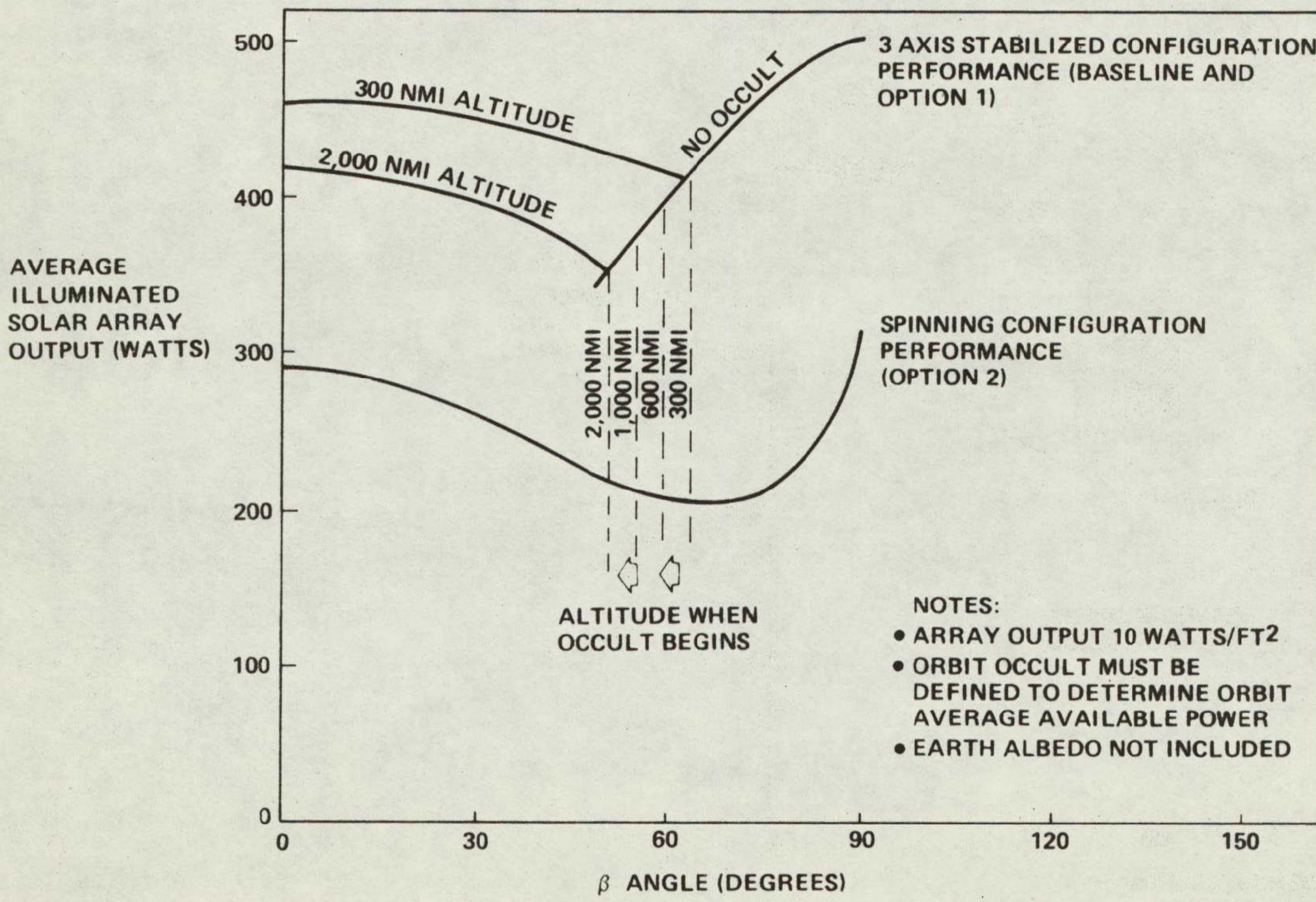


FIGURE 3.4-3

the battery SOC signal rises to 50%. Loads attached to the other load bus are controlled in a similar manner.

- All but essential loads may be configured to operate from either but #1 or bus #2. Essential loads are connected to both load busses thru isolation diodes and in such a way that a power interuption due to ampere hour low state of charge signal will not disable them.

- When one of the spacecraft batteries reaches a state-of-charge of 100% the voltage limiters limit that load bus voltage at one of 8 levels selectable by ground command. Excess power generated in the solar array section is then shunted through power dissipating resistors. Voltage limiting will continue until the battery state-of-charge drops to 99.1%.

- Excess power generated by the solar array while batteries are fully charged is available power unused. The operations requiring the most power should be programmed to permit maximum utilization of energy produced by the solar panels, thus minimizing the amount of energy drawn from the battery.

- Each solar array (Baseline & Option #1) panel is oriented about the spacecraft roll axis using stepping motor array drive. The angle of rotation about the roll axis is controlled to maximize array output as the solar vector moves in and out of the orbit plane. Note that articulation is not continuous during an orbital period, therefore no slip rings are required.

No single point failure will render the spacecraft without useable power. Listed below are a few of the design features and alternate modes of operation.

- Redundant cross strapping of components is unnecessary to achieve high reliability. High reliability is inherent in design of each load bus. Each battery is designed to operate without mission degradation with single shorted cell failure. A third voltage limiter attached to each load bus compensates automatically for malfunction of either of the other two limiters.

- o Spacecraft system operations are programmed to limit battery depth of discharge and allow battery to recover charge more frequently. In the event of an ampere-hour meter failure.

- o Subsystem testing is performed as if the system was single thread design, eliminating costly redundant path functional checkouts.

- o High degree of operational flexibility and performance assessment is incorporated.

- o Minimal loss of energy from array to spacecraft loads. Operational efficiencies exceed 95%.

- o The power subsystem is identical for the baseline and options except for the spin stabilized option #2 where the solar array is body mounted on the cylindrical surface and therefore does not require the stepping motor for orientation.

3.4.2 Electrical Load Summary

A loads summary is presented in Table 3.4-1 indicating the typical load requirements for the satellite less payloads. Derived from this requirement and the array output, a net power allocation for the payload module is defined. Note that the array output is typical and that the basic design allows for additional growth as required.

3.4.3 Single String, Electrical Power Subsystem

The single string configuration, Option 4, power subsystem is the same as the baseline except as shown in Figure 3.4-4.

Electrical Load Demands

					Option 2	
Subsystem/unit	Standby	Operate	Standby	Operate	Standby	Operate
Command and telemetry						
Baseband unit	3.0	3.0	3.0	3.0	3.0	3.0
Command demodulator	2.0	2.0	2.0	2.0	2.0	2.0
CD/TM system	1.1	9.0	1.1	9.0	1.1	9.0
Tape recorder No. 1	–	–	–	–	–	–
Tape recorder No. 2	3.0	16.0	3.0	16.0	3.0	16.0
Transponder transmitter	–	16.0	–	16.0	–	16.0
Transponder receiver	4.0	4.0	4.0	4.0	4.0	4.0
ACS						
Reaction wheel	8.0	8.0	12.0	12.0	N/A	N/A
Horizon scanner	20.0	20.0	N/A	N/A	N/A	N/A
Magnetometer	1.1	1.1	1.1	1.1	N/A	N/A
Horizon pipper	1.2*	1.2*	N/A	N/A	1.2	1.2
Digital sun sensor	0.3	0.3	N/A	N/A	0.3	0.3
Rate gyro	8.0*	8.0*	N/A	N/A	8.0	8.0
Electro magnets	1.0	1.0	N/A	N/A	N/A	N/A
Computer/ICU	25.5	25.5	25.5	25.5	25.5	25.5
IRU	N/A	N/A	21.0	21.0	N/A	N/A
Star tracker	N/A	N/A	3.0	3.0	N/A	N/A
Acq sun sensor	N/A	N/A	–	–	N/A	N/A
Power						
Voltage limiter	2.7	2.7	2.7	2.7	2.7	2.7
Voltage regulator	2.9	2.9	2.9	2.9	2.9	2.9
Ampere Hour meter	3.6	3.6	3.6	3.6	3.6	3.6
Solar array drive	–	–	–	–	N/A	N/A
Relay box	8.5	8.5	8.5	8.5	6.0	6.0
Propulsion						
Tank patch heater (10% DC)	2.0	2.0	1.0	1.0	2.0	2.0
Line heaters (10% DC)	2.0	2.0	1.0	1.0	2.0	2.0
Subtotal (watts)	90.7	127.6	95.4	132.3	67.3	104.2

*Orbit insertion only

Standby—Spacecraft not recording data

Operate—Spacecraft recording data and in contact with ground

TABLE 3.4-1

Power Subsystem (LDSL-AEM) Single String

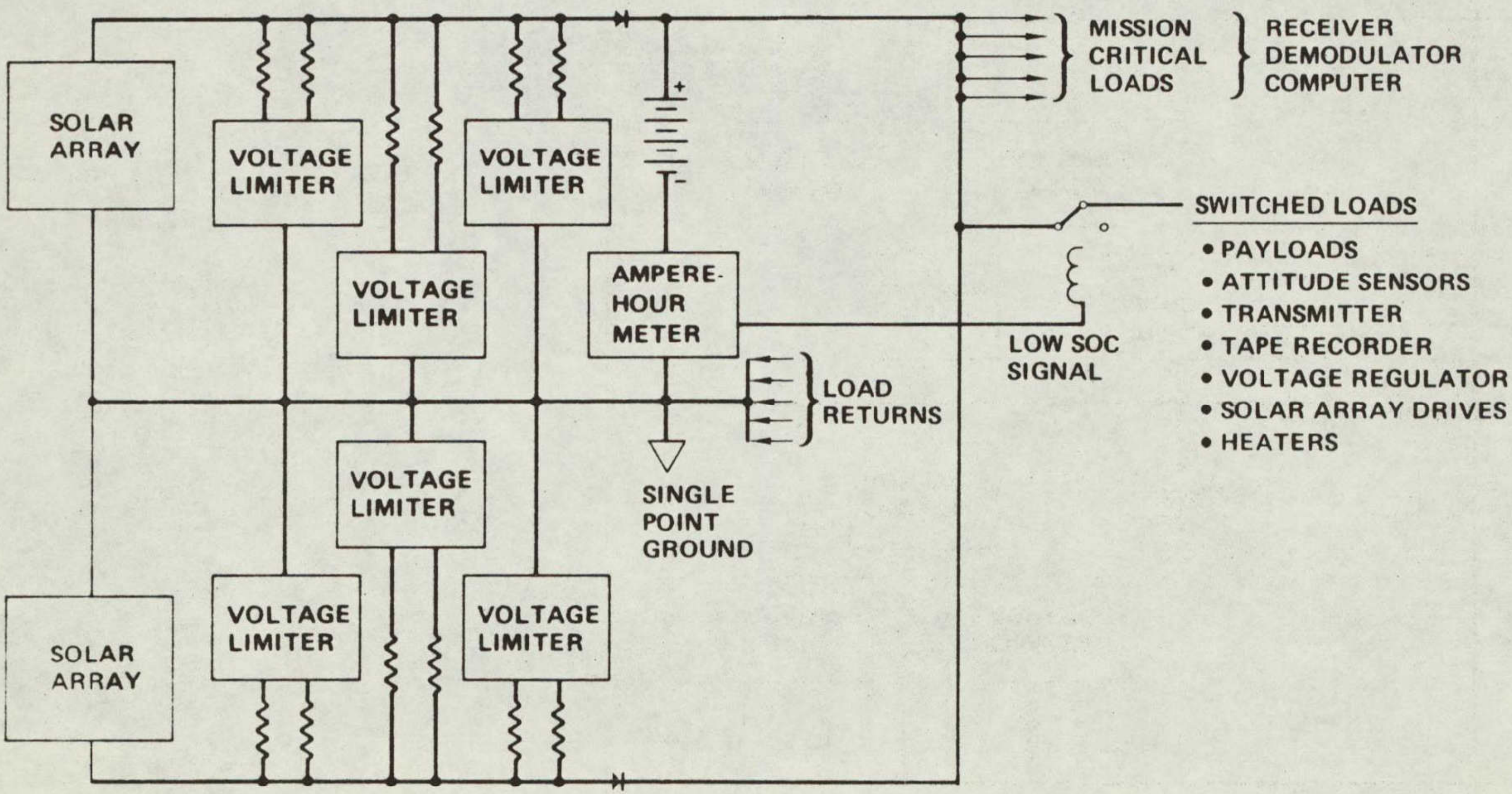


FIGURE 3.4-4

3.5 Telemetry, Tracking and Command (TT&C)

The TT&C subsystem is a completely redundant system made up of components space-qualified on other programs. A redundant integrated command and telemetry system features microprocessor control for flexibility of telemetry formats and storage and processing of delayed commands. Redundant remote units are used for distribution of commands and acquisition of telemetry data. These units are modular in numbers and types of interfacing command and telemetry channels so a system can be configured for a wide variety of space-craft requirements without requalification of the hardware. The redundant SGLS compatible transponder is a Motorola M-Series unit, and the dual integrated command and telemetry system is Spacetac's "Mod-CATS" system soon to be flown on P78-1. The redundant tape recorders are NASA's standard 4.5×10^8 bit units built by RCA. The antennas are space-qualified conical log-spiral units built by Boeing and previously flown on several spacecraft including S-3, HCMM, and SAGE.

A block diagram of the TT&C subsystem is shown in Figure 3.5-1. A table showing equipment characteristics is shown in Table 3.5-1.

3.5.1 Antennas

Communication link performance between the spacecraft and ground is assured by the use of flight proven S-Band conical log-spiral antennas located on booms at opposite ends of the spacecraft, to provide complete spherical coverage. The antenna element is shown in Figure 3.5-2. It is a self-complement, two arm conical log-spiral of cone angle 15 degrees and spiral wrap angle of 53 degrees, and operates over the SGLS frequency range from 1750 to 2300 MHz. Figure 3.5-3 shows the pattern from a single element at the telemetry frequency. The antenna peak gain is 1.4 db above a circularly polarized isotropic antenna. Ellipticity is less than 2 db over $+80^o$ off boresight. Its VSWR is less than 1.5 to 1 from 1750 to 2300 MHz. The antenna operates successfully in a vacuum with 25 watts of RF power applied.

The TT&C System Is Completely Redundant With Remote Units for Command Distribution and Telemetry Data Acquisition

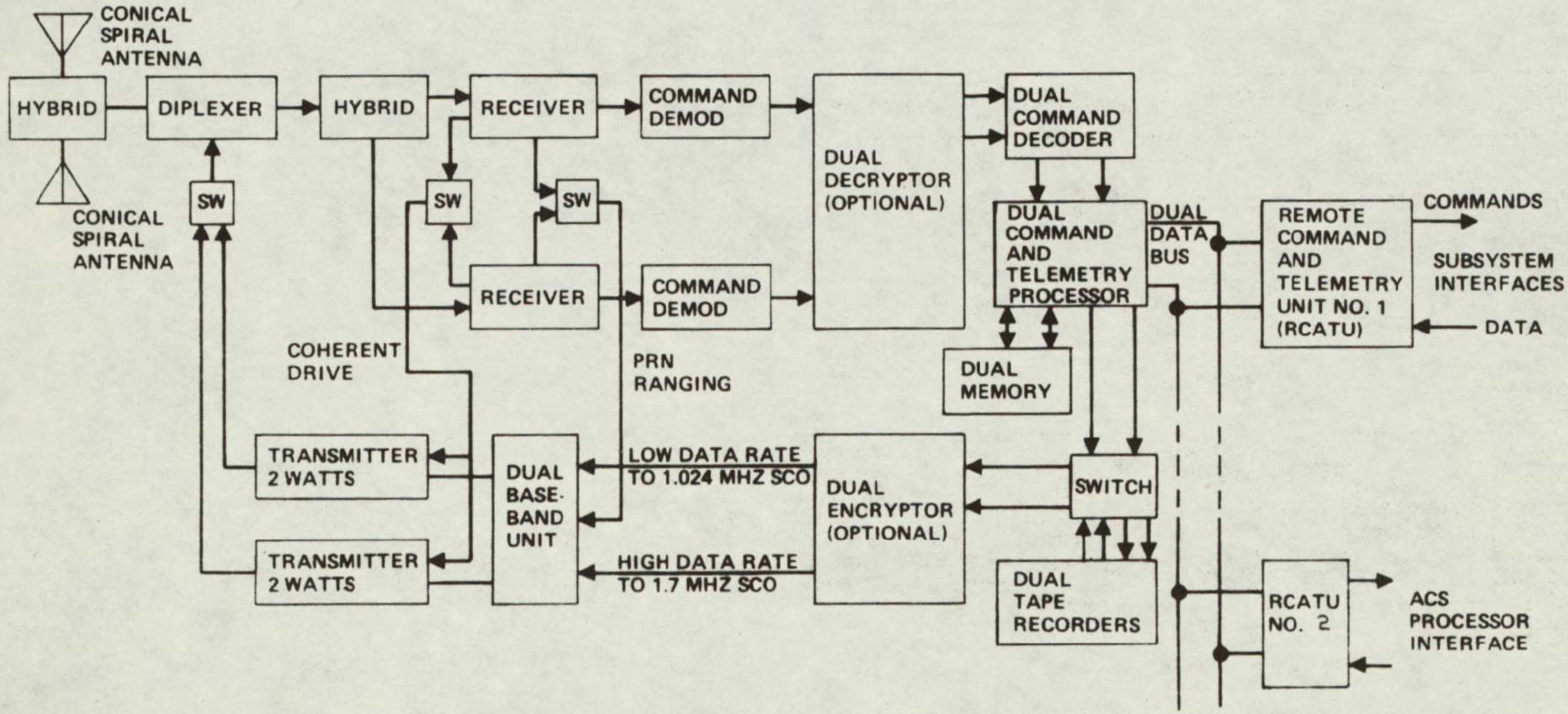


FIGURE 3.5-1

TABLE 3.5-1 TT&C EQUIPMENT LIST

COMPONENT	VENDOR MODEL	WT. LBS.	SIZE INCHES	POWER WATTS	PREVIOUS USE
Redundant SGLS Transponder	Motorola M-Series				Similar to NASA STD. XPDR, and Several SGLS Units.
Baseband Assembly Unit	Motorola M-Series	15	12 x 11 x 3	25;(5,RCVR ONLY)	
Command Demodulator	Motorola M-Series				
Command Decoder and Processor		6	6 x 6 x 6	5 (Central unit)	Will fly on P-78-1.
PCM Telemetry Encoder	SPACETAC "MOD-CATS"	6	6 x 6 x 6	1 (remote unit)	
Tape Recorders	RCA	30	8 x 16 x 7	16	Will fly on SAGE
Timer/Sequencer	Cyclomatic	3	3.5 x 6 x 2.4	1	S-3, HCMM, SAGE
S-Band Antenna	Boeing; Conical Log-Spiral	1	6 x 3 DIAM.	—	S-3, HCMM, SAGE

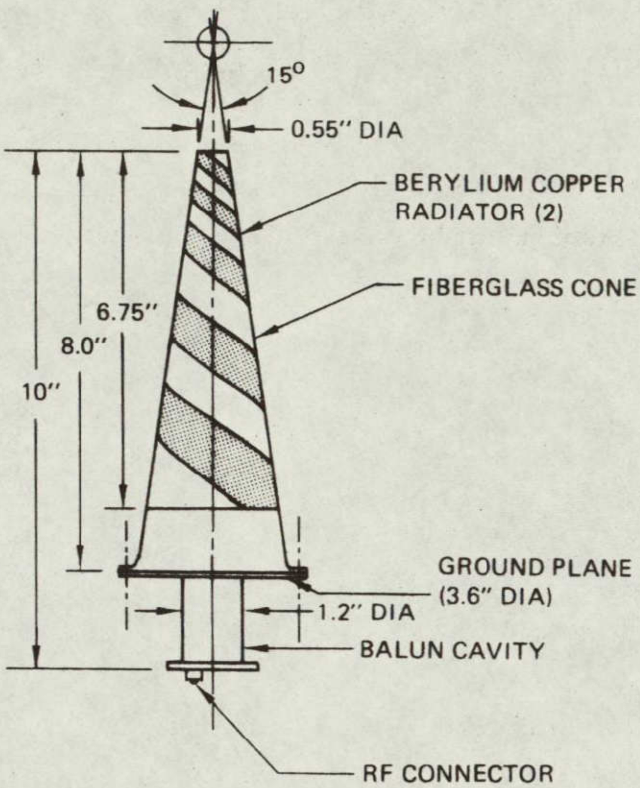


Figure 3.5-2. Flight Proven Conical Log Spiral Design

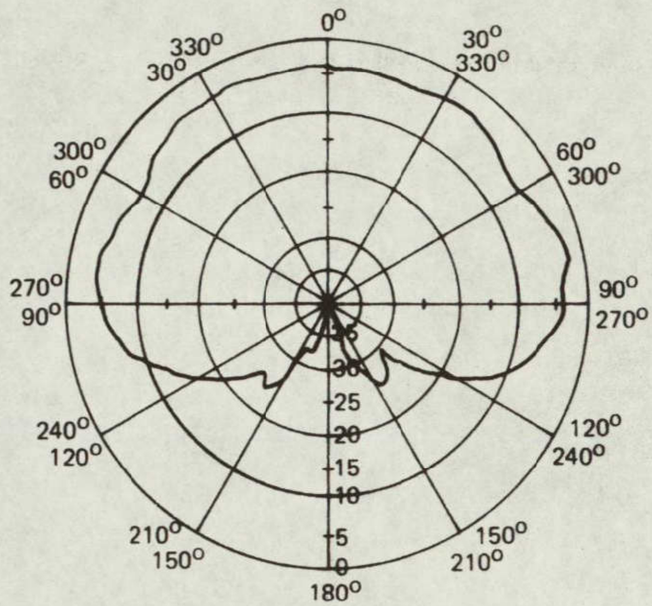


Figure 3.5-3. Measured S-Band Antenna Patterns Provides Hemispherical Coverage

The antenna RF feed configuration is a broadband balun and tapered transmission line connected to the spiral arms at the truncated apex. The balun printed circuit network is enclosed in a cavity attached to the base of the cone and the output is connected to the balanced twin-lead transmission line, tapered along its length, to provide the proper impedance transformation to the conical spiral element. This results in equal amplitude, out-of-phase signals at each of the spiral arms. The resultant radiation pattern is symmetrical about the cone axis. A small ground plane (3.6 inch diameter) is located at the base of the conical element to enhance the radiation pattern beamwidth characteristics at large angles off the antenna axis.

The spiral arms are bonded to the antenna cone form constructed from .030 inch epoxy fiberglass. The feed network balun cavity is rigidly attached to the base of the fiberglass cone. The pattern radiated from two antennas on opposite ends of the spacecraft is shown in Figure 3.5-4.

3.5.2 Transponder

The redundant SGLS compatible transponder is the space qualified Motorola M-Series unit. This newly developed series of transponders features high performance in a miniaturized size and the flexibility to tailor the transponder configuration and performance to mission requirements. A block diagram of this unit is shown in Figure 3.5-5. The unit's redundancy is not shown. The command demodulator and the baseband assembly unit, both redundant, will be contained as separate modules within the transponder package. After filtering for separation of the PRN range code, the command demodulator module filters and detects the S, 0, and 1 tones at 65, 76, and 95 KHz for input, as digital signals, with clock and enable signals, to the command decoder.

The baseband assembly unit accepts the ranging code and two channels of PCM data, modulates subcarriers at 1.024 and 1.7 MHz, and sums the three signals for application of the composite baseband signal to the modulator of the transponder transmitter. A block diagram of this unit is shown in Figure 3.5-6. The unit permits adjustment of the

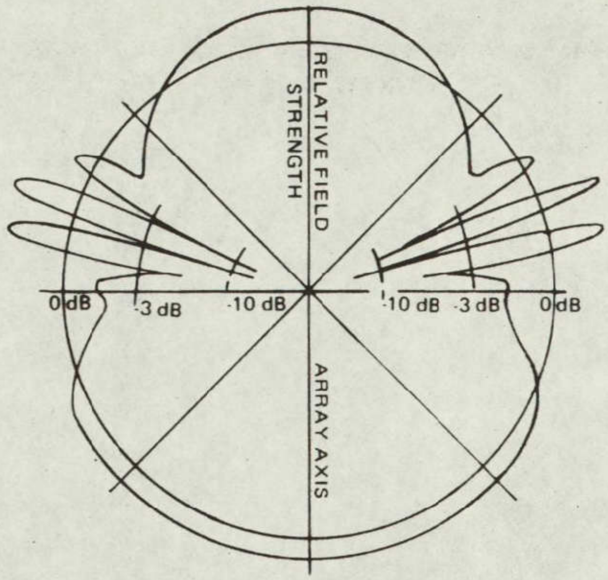


Figure 3.5-4. Back-to-Back Conical Log Spiral Antenna Pattern Meets Spherical Coverage Requirements

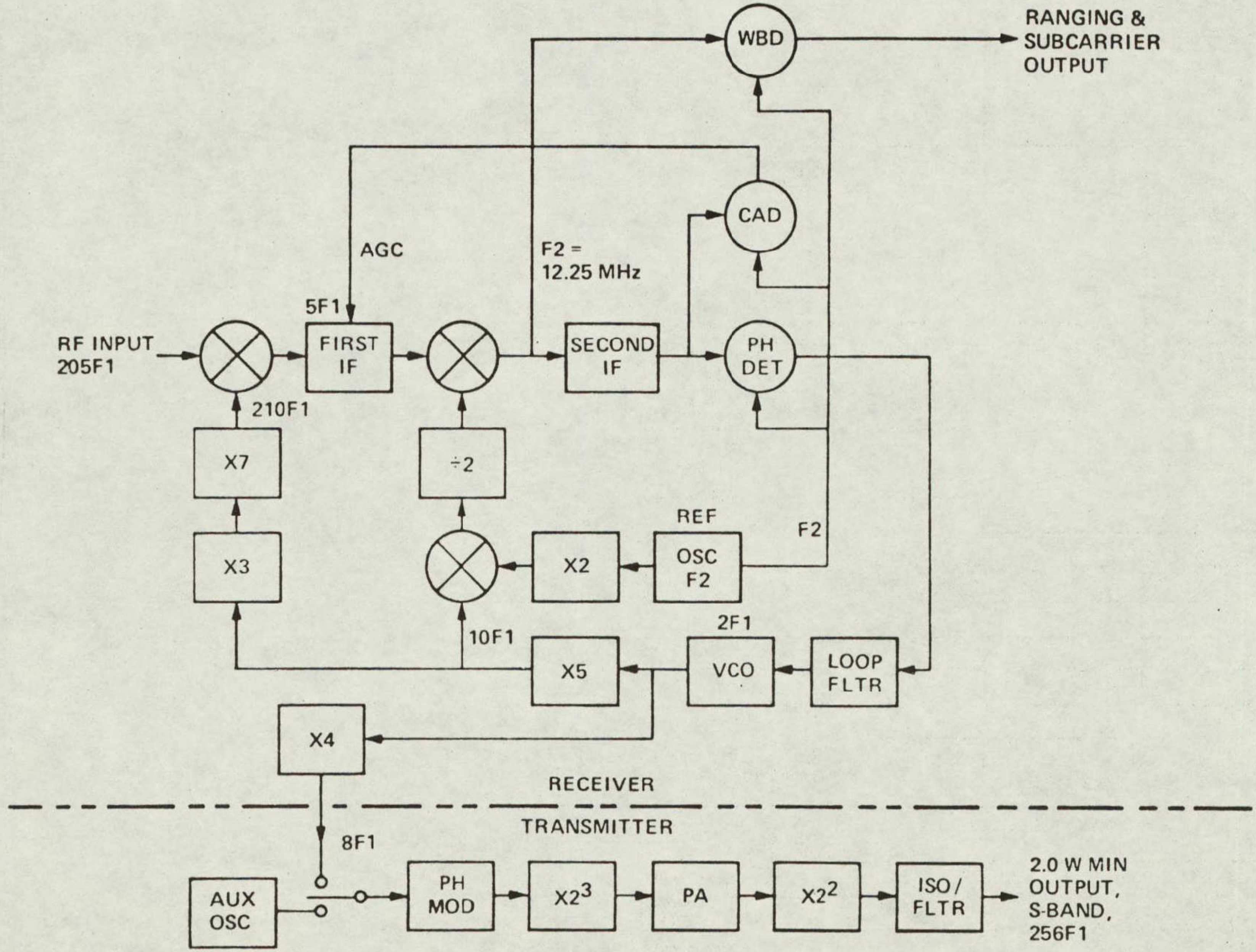


Figure 3.5-5. SGLS, Transponder Block Diagram (Redundancy Not Shown)

Baseband Assembly Unit (Redundancy Not Shown)

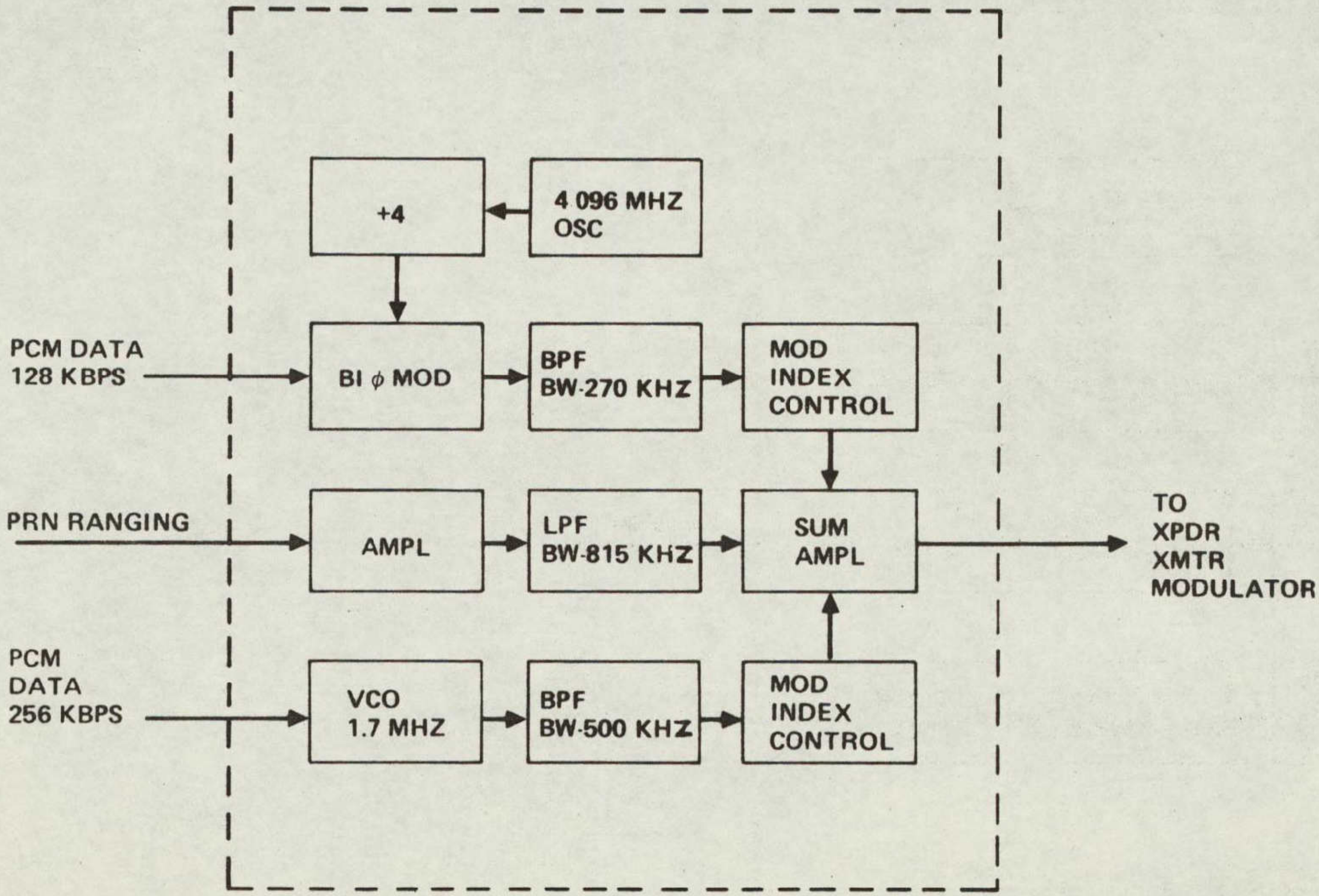


FIGURE 3.5-6

relative modulation index among the three channels to easily accommodate different data rates on different missions.

The total transponder design permits command rates of 1 or 2 kbps; range and coherent range-rate tracking with the SGLS standard up/down frequency ratio of 256/205; 1 Mbps PRN ranging; and two data channels (low rate up to 128 kbps and medium rate up to 256 kbps) on two subcarriers at 1.024 and 1.7 MHz. One data channel can be used for housekeeping data and the other for tape recorder dump or other mission data.

3.5.3 Integrated Command and Telemetry

Command decoding and distribution and telemetry formatting and data acquisition are all controlled by a redundant, microprocessor controlled, integrated system called "Mod CATS" and built by Spacetac, Inc. This system will soon be space qualified and will fly on P78-1.

A block diagram of the system is shown in Figure 3.5-7. Redundant primary, or central, units are connected to redundant remote units by redundant control and data buses. The bus operation permits both data acquisition for telemetry and command distribution on the same data bus. The primary unit controls the entire operation using the SMS-300 microprocessor built by Signetics. This is a Schottky - TTL device that features a cycle time of 300 nanoseconds. This unit, operating with modular RAMs and ROMs and a PROM for telemetry formatting, controls all command and telemetry operations. The primary unit also has a number of modular telemetry multiplexers and command distribution units. More details of the primary unit's architecture are shown in Figure 3.5-8. Modular units are shown for bilevels, analog and serial digital telemetry data channels. The interface with the attitude control computer will be made with one of these interface units. Modular Command distribution units accommodate groups of discrete and serial digital commands. The remote units have similar multiplexing modules and control modules.

The command format used is the SGLS compatible "Madcom-0" format, similar to "Module-0" formats. It is a 33 bit format, with 5 bits for

Integrated Command and Telemetry System Block Diagram

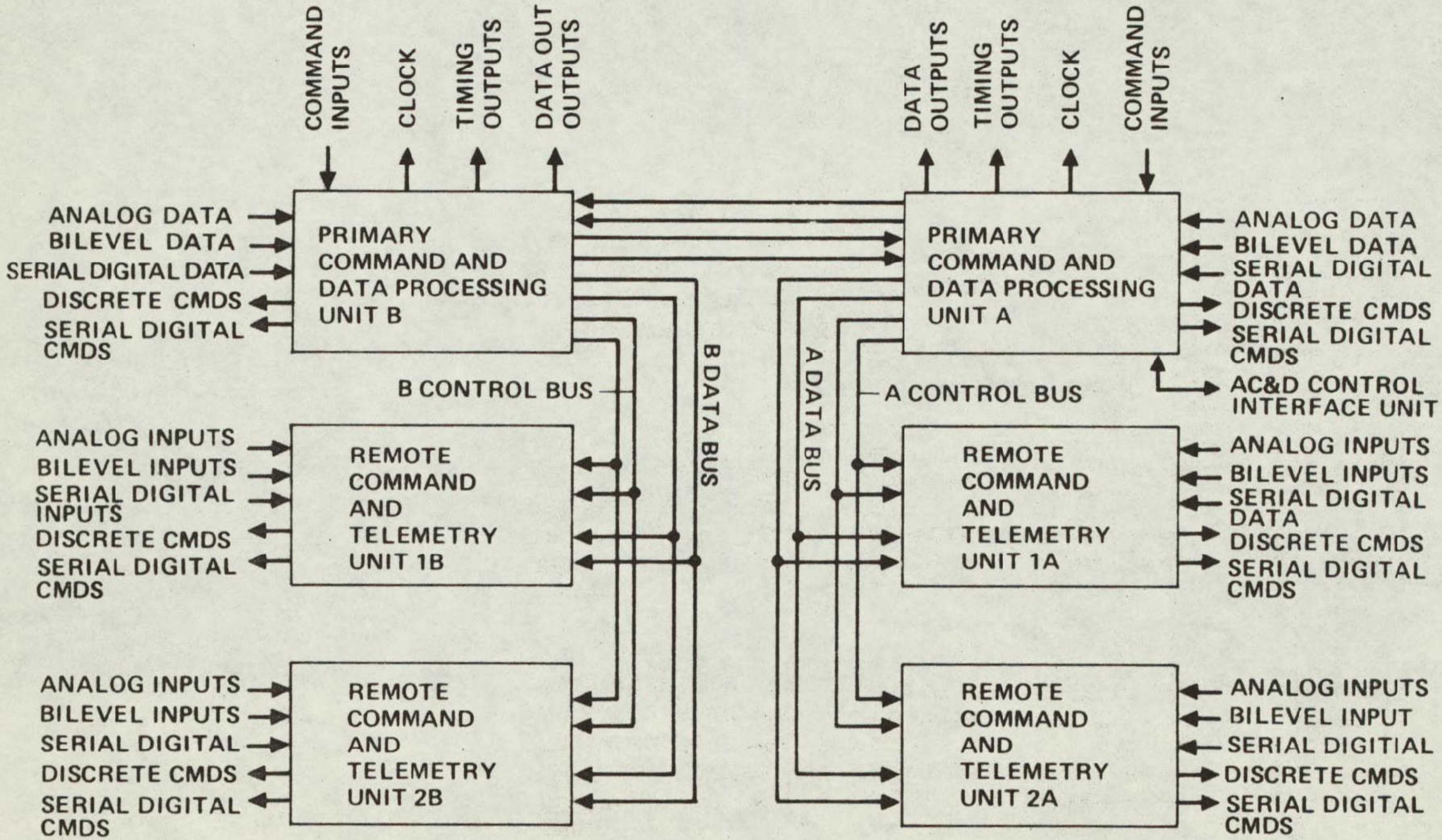


FIGURE 3.5-7

Architecture of Primary Unit of Command and Telemetry System, Showing Command Distribution and Telemetry Multiplexing Units

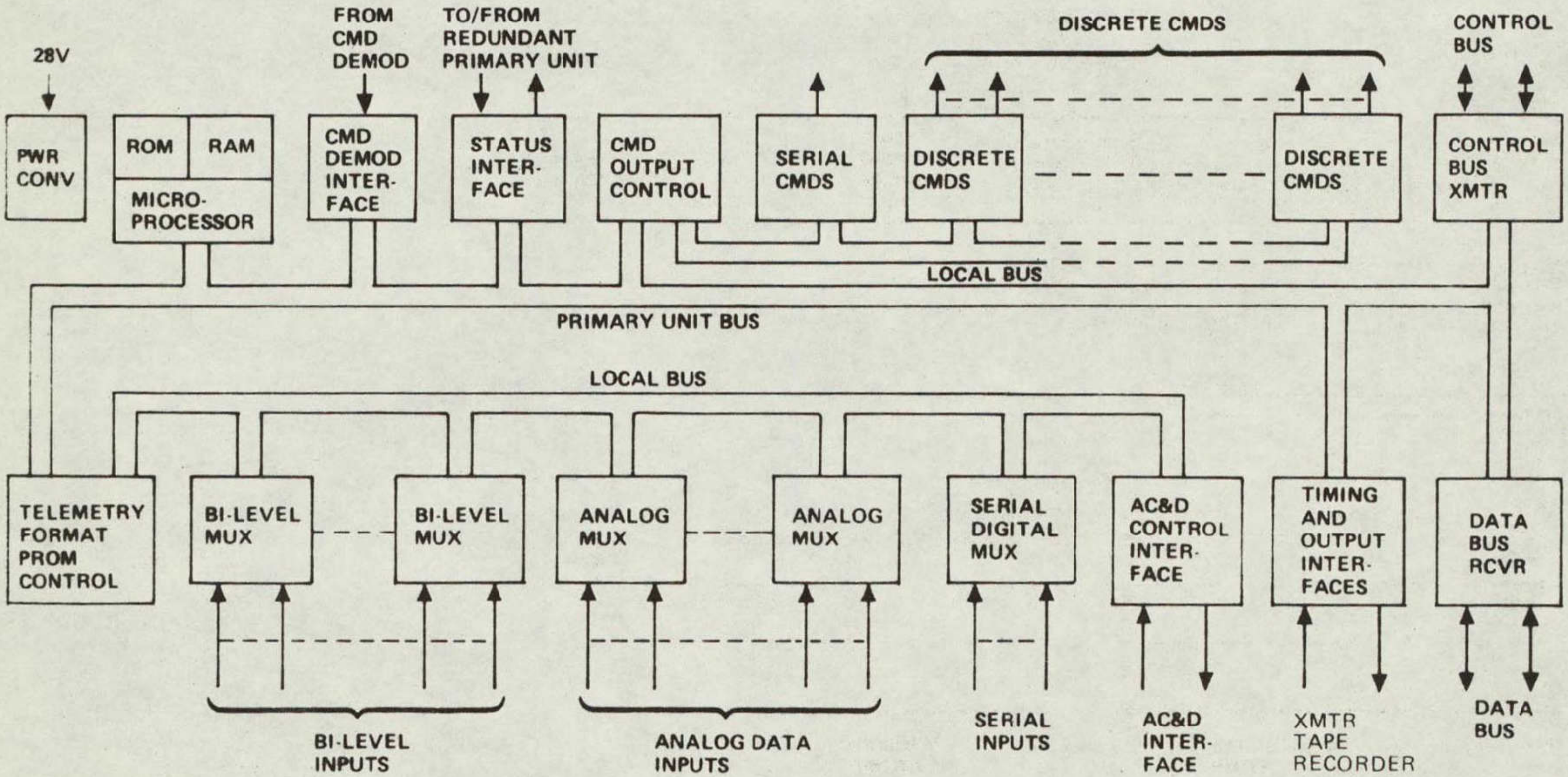


FIGURE 3.5-8

vehicle address, 3 bits for function or operation codes, 24 bits for the command field, and a parity bit. The 24 bit command field includes up to 21 serial digital data bits and 21 bit time tags. Two words are required for stored commands; one word contains the data and the other word contains the time tag. The system is capable of storing up to 512 commands in the modular RAM associated with the microprocessor. The microprocessor scans the memory contents and compares the time tags with a real time clock. When the commands are "timed-out", they are executed on a time shared basis with real time commands using the same decoding and distribution circuits.

The PROM controlled telemetry formatter permits great flexibility in sub commutation and super commutation without requalification of hardware. Four formats, selectable by ground command, can be used for different modes of operation such as launch, and orbital formats for use with different groups of sensor or experiments. The remote multiplexers will minimize the spacecraft wiring required. The timing and control module includes a stable oscillator and a count down chain, so that operation over a wide range of selectable bit rates, up to 256 kbps is possible. With the PROM format control, two telemetry formats can be generated simultaneously - one for housekeeping data, and one for experiment or mission data.

3.5.4 Timer-Sequencer

This unit controls the timing of all deployment events and other events needed to activate or initialize the system during injection and early orbits, before ground stations are in view for real time commanding. It is Cyclomatic's Model 4051, previously used on several spacecraft including S-3, HCMM and SAGE. It is a magnetic logic device, non-volatile, and programmable for any timing desired. It provides 23 outputs, with the interval between each output separately programmable. Two units are used for redundancy. They can also be used to back-up certain control functions normally accomplished with the command system.

3.5.5 Tape Recorders

The standard NASA 4.5 x 10^8 bit tape recorder provides for on-board storage of mission data for later dump via the S-Band transponder, or bia the Carrier II transmitter if that option is selected. A redundant pair of tape recorders is used for reliability. The recorder is capable of recording and playback at 23 rates in each of 3 modes: 1 track, 4 track or 8 track. Data rates from 2 kbps to 2.56 Mbps are possible at tape speeds from 0.209 to 33.465 ips. The recorder mode, speed, and bit rates are controlled by a single serial digital command. Electronic units can be cross-strapped to redundant transport units. These units will be qualified and flown on SAGE. The redundant set is 8 x 16 x 7 inches, 30 lbs, and require 16 watts of power in the record or playback mode.

3.5.6 Single String TT&C Option 4

The optional, single-string TT&C subsystem is made up of components space qualified and used on AEM-HCMM and SAGE, except for the transponder, which is a single-string version of the Motorola M-Series dual redundant transponder used on the baseline design. A block diagram of the TT&C subsystem is shown in figure 3.5-9.

The telemetry system includes a variable format, PROM controlled, PCM encoder which can operate at any rate up to 256 kbps. These data are transmitted via the 1.024 or 1.7 MHz subcarriers of the Carrier I SGLS transponder transmitter. The transponder also provides range and range rate tracking using the PRN ranging codes and the 205/256 coherent turn-around ratio for up-link and down-link frequencies.

The antennas are dual conical log-spirals mounted on booms at opposite ends of the spacecraft to provide complete spherical coverage. These antennas have been used by Boeing on S-3, HCMM, SAGE, and other spacecraft.

The command decoder and processor is the same as used on HCMM and SAGE except for minor modifications to acommodate the SGLS ternary format and word length changes. The "MADCOM-0" or Module 0 format of 33 bits per word will be used. Time tags of 16 bits provide 2 second resolution over 24 hours. Any intermix of discrete and serial digital commands

Single String TT&C

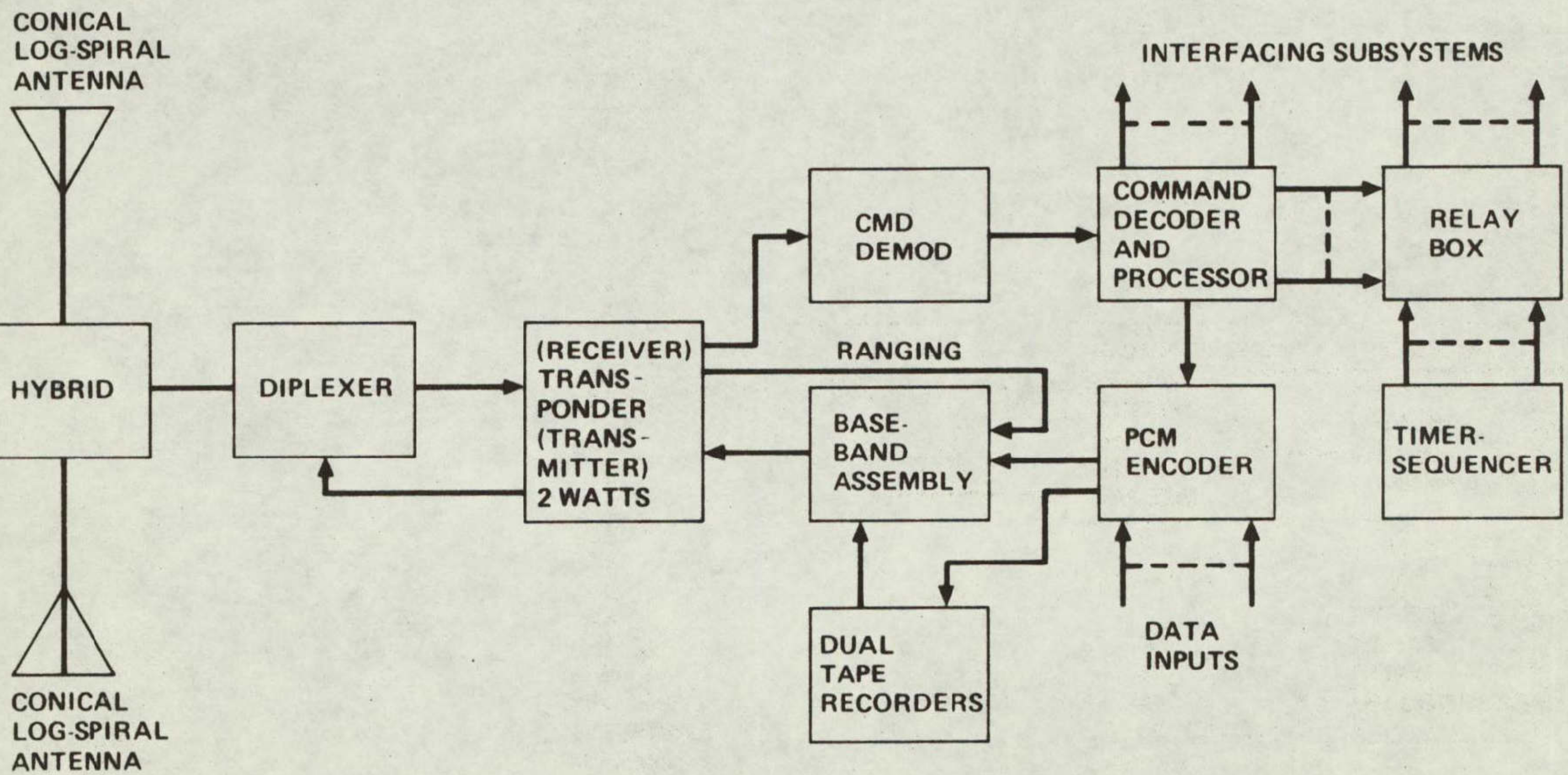


FIGURE 3.5-9

3.5.6 Single String TT&C Option 4 (Cont'd)

up to a total of 256 commands can be stored in memory for later execution. The PCM encoder provides accurate timing for the stored commands. The command decoder provides 128 +28 volt discrete commands, 32 +5 volt impulse commands, and 16 serial digital commands. It is modular in construction, so additional command outputs can be provided without requalification.

The command demodulator unit and the baseband assembly unit are separate modules packaged within the M-Series transponder. The baseband assembly unit combines the turn-around PRN ranging code with two channels of telemetry on the 1.024 and 1.7 MHz subcarriers. The subcarriers are biphase modulated with PCM data within the baseband unit.

The system also includes a Timer/Sequencer which is used for all deployment and initialization events. This is a Cyclomatics model 4051, and it is programmable and non-volatile. Twenty-Three separate outputs are provided with a timing precision of 0.5%.

Dual redundant NASA standard 4.5 x 10^8 bit tape recorders are used in this "single-string" option to provide high reliability data sotrage with a high degree of flexibility of operation with 23 record and playback speeds with either 1 track, 4 track, or 8 track operation. Data rates from 2 kbps to 2.56 M bps are possible at speeds from 0.209 to 33.465 ips.

3.6 Reaction Control System (RCS)

Reaction control is provided by monopropellant hydrazine thrusters in all three configurations. The baseline and option 2 require spin up and either despin or spin rate reduction capability. Precession torque is also required. Option 1 requires three axis control both in coast and powered flight.

3.6.1 RCS Requirements

The requirements imposed upon the RCS by the configurations and the system capabilities are summarized in Table 3.6-1. The requirement to avoid single point failures results in doubling the quantity of many of the components. All functional and performance requirements are met.

3.6.2 RCS Description

The reaction control systems are schematically illustrated in Figures 3.6-1 and 3.6-2. The systems are identical for the baseline and option 2 (Figure 3.6-1). The single string Configuration Option 4 RCS system is the same as the baseline except as shown in Figure 3.6-3. The system is based entirely on existing technology and hardware. The tanks are positive expulsion with an elastomeric bladder. Pressurization is by blowdown of the prepressurized ullage space on the gas side of the bladder.

TABLE 3.6-1 RCS REQUIREMENTS AND CAPABILITIES

OPTION	REQUIREMENT	CAPABILITY
All	No single point failures	Redundant valves, thrusters, heaters and sensors where applicable
Baseline and Option 2 Spinner	• Spin up to 60 RPM Despin to zero or low rate. • Resolution ± 0.1 RPM • Precession Capability	90 RPM with full tanks ± 0.03 RPM Yes
Option 1 - High Accuracy 3 Axis Stabilized	• Powered Flight Attitude Control • Coast attitude control and on-orbit wheel desaturation • Torque x time $\leq$ 0.02 ft.lb. sec.	• Two level 65 lb and 155 lb pitch/yaw thrusters and 5 lb roll thrusters handle handle lateral and roll moments of 364-15 & 364-4. • Sixteen 0.5 lb thrusters • 0.0188 with blow down to ½ thrust & 15 ms pulse

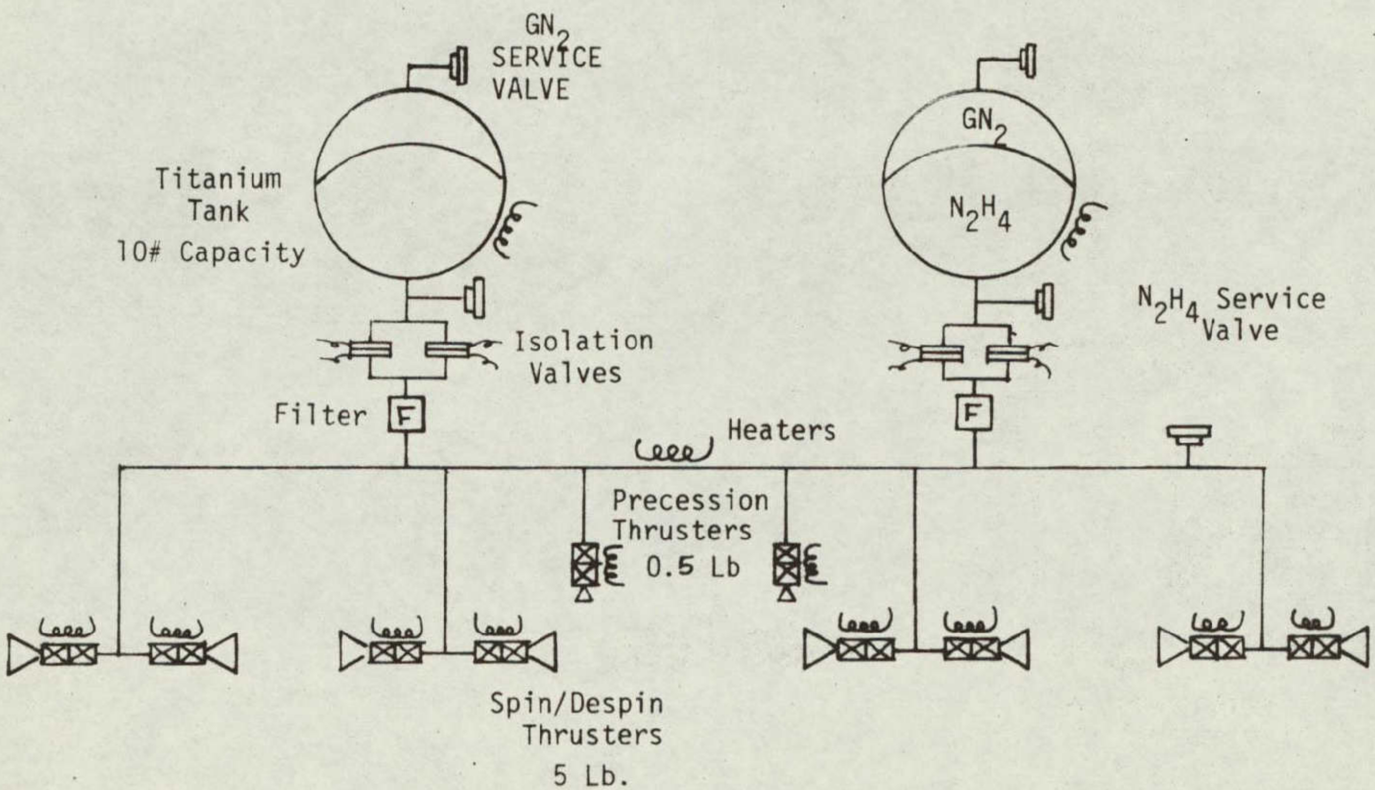


FIGURE 3.6-1 BASELINE AND OPTION 2 SPINNER
REACTION CONTROL SYSTEM

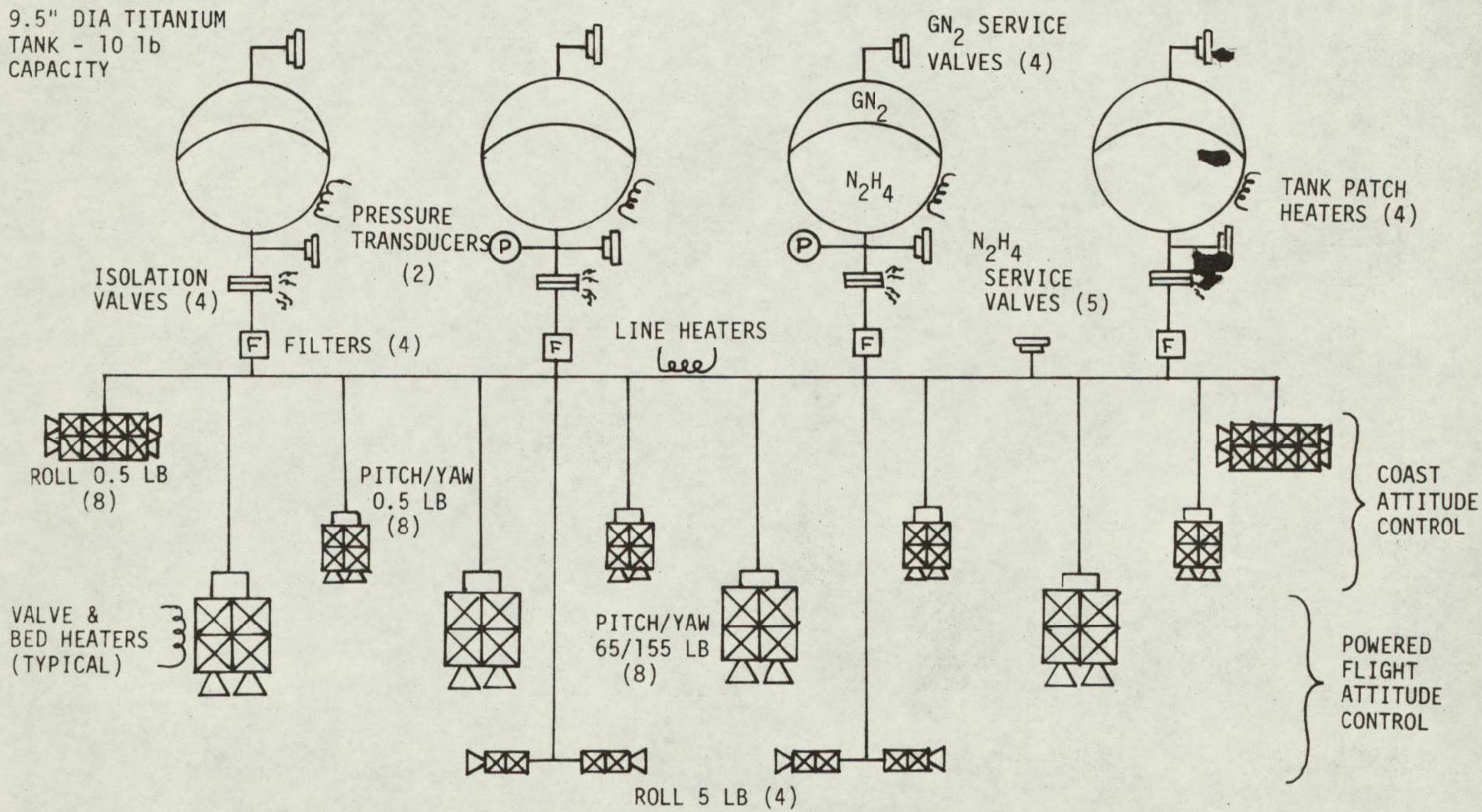


FIGURE 3.6-2 HIGH ACCURACY 3 AXIS STABILIZED REACTION CONTROL SYSTEM

Option 4—Single String RCS

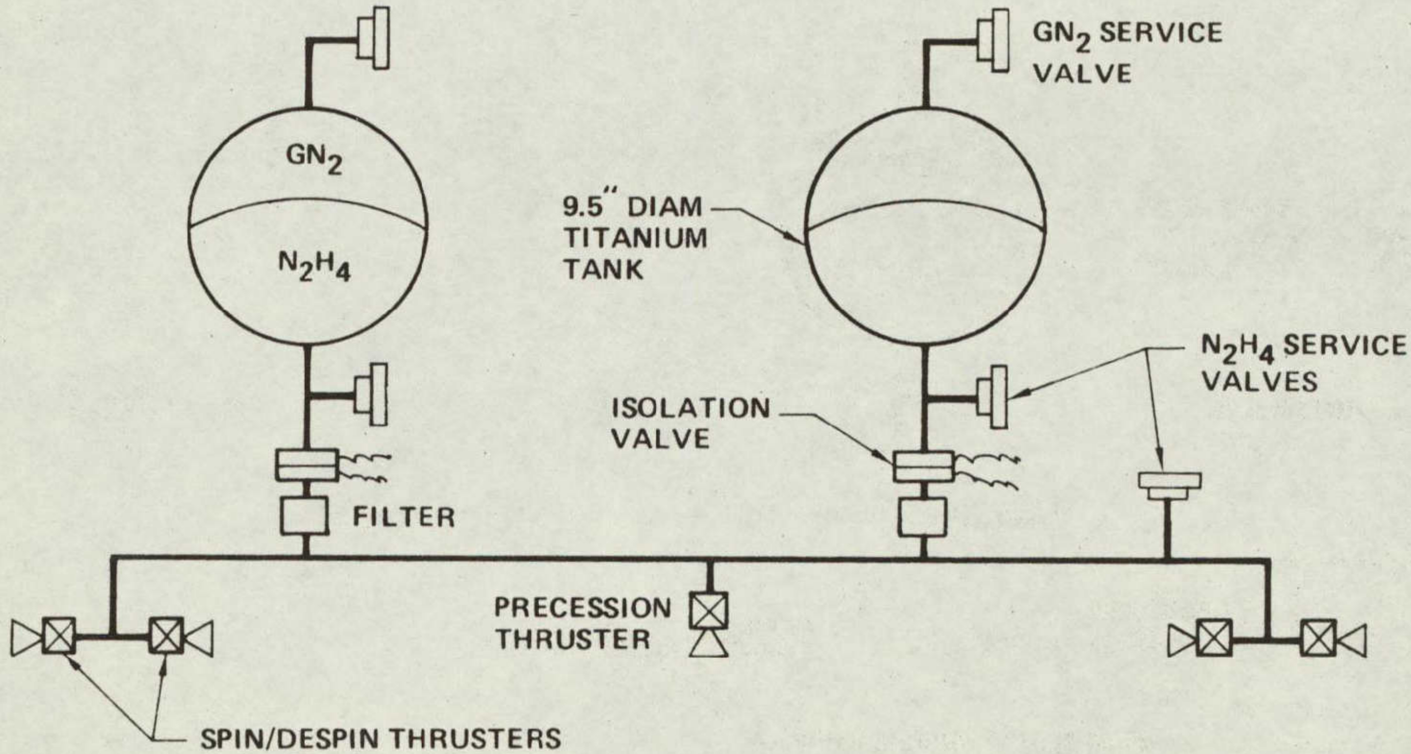


FIGURE3.6-3

required 5 lb. thrust for spin/despin and 0.5 lb. for precession. Spin up to 60 RPM takes 111 seconds. Typical available hardware for this application is shown in Table 3.6-2. Valving, heaters and filters all exist in the sizes required.

Figure 3.6-2 illustrates the more complex RCS required for the three axis stabilized high accuracy, all altitude system. The tanks are the same size as in the baseline but four are used instead of two. Twenty eight thrusters are required including three sizes. Coast control thrusters are small (0.5 lb) with a small impulse to provide wheel desaturation during spacecraft on orbit operation. They are large enough, however, to provide a 180^{0} maneuver prior to first solid motor burn in 134 seconds.

Control during powered flight is provided by 8 large thrusters for pitch/yaw control and 4 smaller thrusters for roll. The pitch/yaw control thrusters are the same bi-level (65 pound and 155 pound) thrusters used on Block 5D. The roll thrusters are 5 lb. thrust.

Service valves for this system are the same as for the baseline, however, the isolation valves are larger and are dual squib to avoid the requirement for redundant isolation valves. Typical hardware for this system is also shown in Table 3.6-2. All tubing joints are brazed to eliminate leakage.

3.6.3 RCS Performance

Since spin and despin are the primary functions of the RCS in the baseline and Option 2, the system performance has been calculated in those terms. Figure 3.6-4 shows the system capability as a function of payload weight. The two tanks each have a capacity of 10 pounds of propellant. Payload weights above 470 pounds require more than one tank. Two tanks easily provide more than 60 RPM for all payload weights. 92 RPM is possible with two full tanks and a payload of 1000 pounds.

TABLE 3.6-2 RCS HARDWARE CHARACTERISTICS

OPTION	FUNCTION	SIZE/TYPE	TYPICAL HARDWARE
BASELINE AND OPTION 2	Spin/Despin Thrusters	5 lb. thrust	Ham. Std. RE 16-4, Rocket Res. MR-50A
	Precession Thrusters	0.1 lb. thrust	Ham. Std. REA 10-13, Rocket Res. MR-74 etc
	Tanks	9.5 in diam.	Pressure Systems Inc. P/N 80156-1
	Service Valves	1/4 in.	Pyronetics 1831 series
	Isolation Valves	1/4 in.	Pyronetics 1047
	Filters	25 u	
	Heaters	Patch & Tape	
OPTION 1 3 Axis	o Powered flt pitch/yaw thrusters	Bi level 65/155	Marquardt R-30
	o Powered flt roll	5 lb.	Ham. Std. RE 16-4, Rocket Res. MR-50A
	o Coast thrusters	0.5 lb.	Ham. Std. REA 17-7, Rocket Res. MR-6@
	Tanks	9.5 in. diam.	Pressure Systems P/N 80156-1
	Service Valves	1/4 in.	Pyronetics 1831 series
	Isolation Valves	3/4 in.	Pyronetics 1420 dual squib
	Filters	25 u	
	Heaters	Patch & Tape	

RCS Propellant Requirements for Spin Injection—Baseline and Option 2

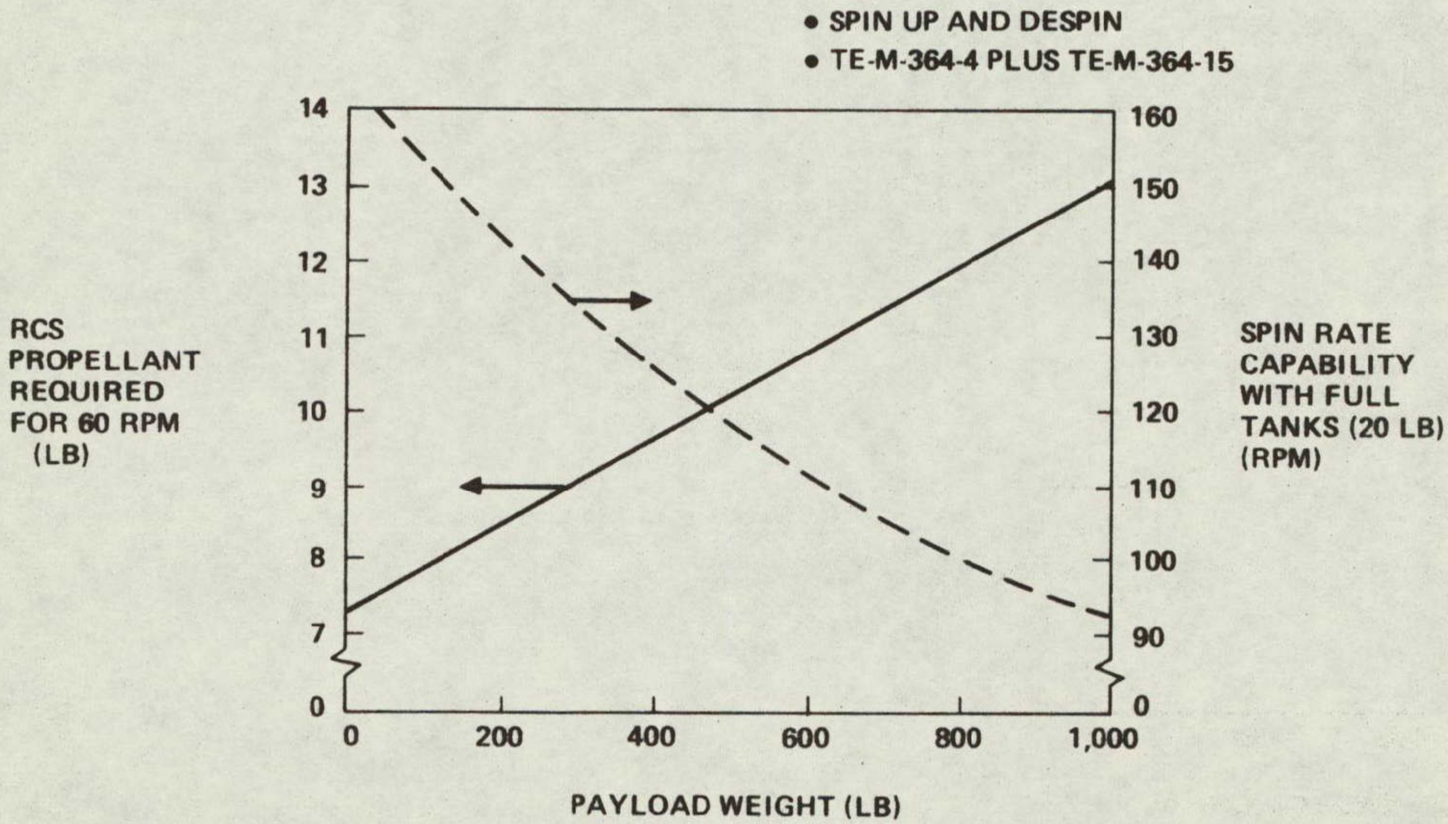


FIGURE 3.6-4

The Option 1 (high accuracy three axis stab ilized vehicle) requires four tanks to overcome the 3 sigma worst case lateral and roll forces from TE-M-364-4 and TE-M-364-15 solid motors (top of Figure 3.6-5). Total propellant required for the 3 sigma case is 30.4 pounds primarily for the powered flight control. The average propellant requirement is 13.5 pounds. The frequency occurring excess propellant will be used for orbit adjustment. The bottom of Figure 3.6-4 shows the orbit adjust ΔV available as a function of excess hydrazine and payload weight. At least 9.6 pounds will be available with four tanks providing 40 to 85 feet per second ΔV capability. This could range up to 120 to 230 feet per second on an average flight. The addition of more tanks could provide more capability.

3.6.4 Safety

Hydrazine systems are relatively safe if leakage is eliminated. The RCS systems defined here isolate the hydrazine in the tanks prior to deployment from the orbiter payload bay. The resulting dry propellant manifold thereby minimizes the leakage potential during prelaunch and launch operations. The hydrazine will also be preloaded in the tanks and service valves capped prior to installation in the payload bay eliminating all fluid interfaces with the orbiter.

3.7 Propulsion

3.7.1 Baseline Motors

A variety of payloads and missions could lead to a variety of solid motors for orbit transfer, however this study has concentrated on the largest motors likely to be used namely the TE-M-364-4 for the first stage and the TE-M-364-15 for the second stage. These two motors will not provide geosynchronous or sun synchronous capability from shuttle orbits launched from ETR, but do provide large payload capability to low earth orbits as discussed in section 3.8. The characteristics of these two motors are summarized in Figures 3.7-1 and 3.7-2.

Potential Orbit Adjust ΔV−Option 1 3-Axis Stabilized

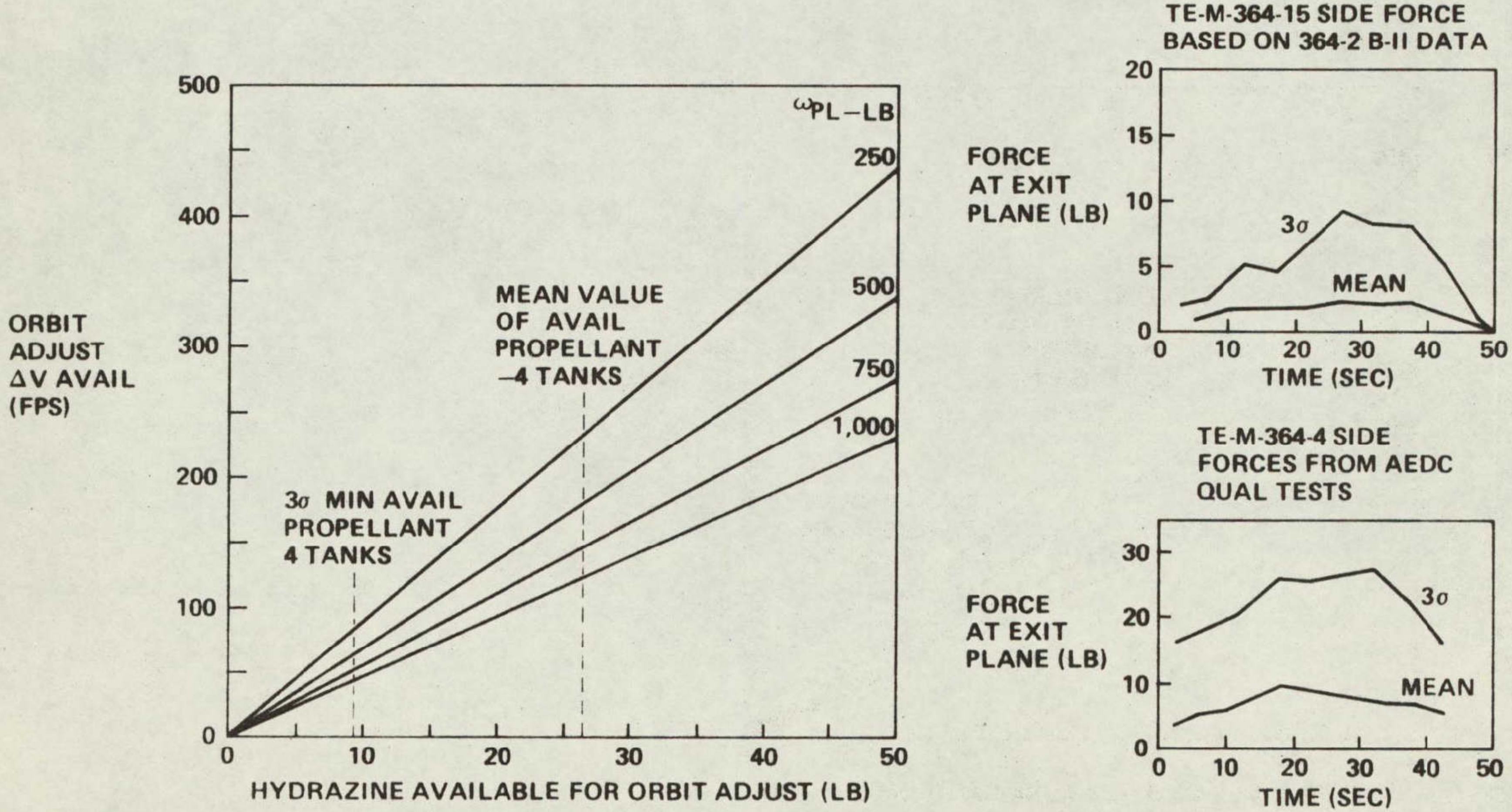


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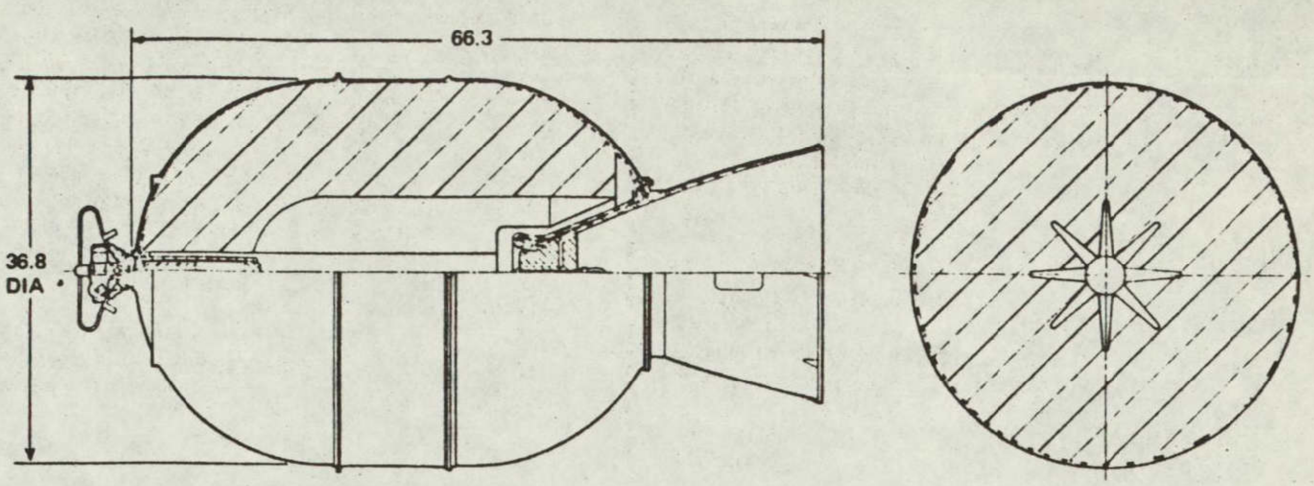
FIGURE 3.6-5



MOTOR PERFORMANCE*

Burn Time/Action Time (t_b/t_a), sec	41.96/43.65
Ignition Delay Time (t_d), sec	0.220
Burn Time Avg. Cham. Press. ($\bar{P}_b$), psia	561
Action Time Avg. Cham. Press. ($\bar{P}_a$), psia	548
Maximum Chamber Pressure (P_{max}), psia	600
Total Impulse (I_T), lbf-sec	654,400
Burn Time Impulse (I_b), lbf-sec	649,200
Motor Specific Impulse (I_{mo}), lbf-sec/lbm	264.5
Propellant Specific Impulse (I_{sp}), lbf-sec/lbm	285.5
Burn Time Average Thrust ($\bar{F}_b$), lbf	15,472
Action Time Average Thrust ($\bar{F}_a$), lbf	15,060
Maximum Thrust (F_{max}), lbf	16,870
Measured Thrust Coefficient (C_f)	1,809
Theoretical Thrust Coefficient (C_f°)	1,873
Discharge Coefficient (C_d)	0.966

*75°F, Vacuum.

WEIGHTS, lbm

Total Loaded	2473.3
Propellant	2290.0
Case Assembly	75.3
Nozzle Assembly	63.0
Igniter Assembly (with Model 2129 S&A)	7.4
Internal Insulation	34.9
External Insulation	0.0
Liner	1.3
Miscellaneous	1.1
Total Inert (Igniter Propellant not Included),	183.3
Burnout	168.8
Propellant Mass Fraction	0.926

TEMPERATURE LIMITS

Operation	40°F to 100°F
Storage	40°F to 100°F

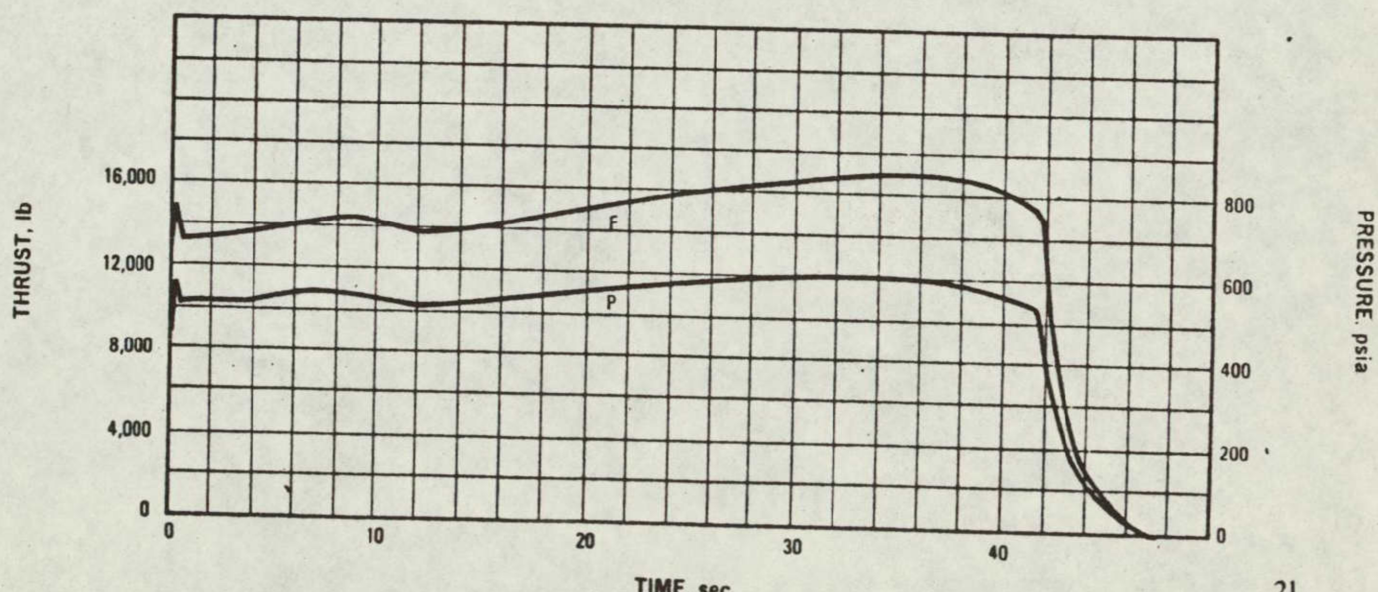
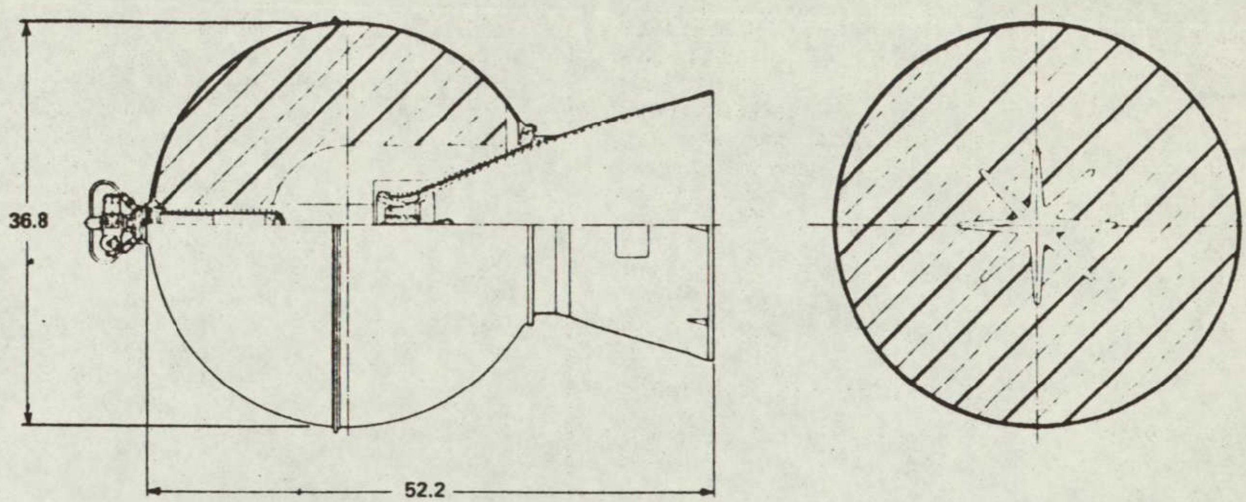


Figure 3.7-1 TE-M-364-4 Characteristics



MOTOR PERFORMANCE*

Burn Time/Action Time (t_b/t_a), sec	42.2/43.9
Ignition Delay Time (t_d), sec	0.161
Burn Time Avg. Cham. Press. ($\bar{P}_b$), psia	597
Action Time Avg. Cham. Press. ($\bar{P}_a$), psia	587
Maximum Chamber Pressure (P_{max}), psia	637
Total Impulse (I_T), lbf-sec	420,430
Burn Time Impulse (I_b), lbf-sec	409,430
Motor Specific Impulse (I_{mo}), lbf-sec/lbm	265
Propellant Specific Impulse (I_{sp}), lbf-sec/lbm	289.9
Burn Time Average Thrust ($\bar{F}_b$), lbf	9,790
Action Time Average Thrust ($\bar{F}_a$), lbf	959
Maximum Thrust (F_{max}), lbf	10,470
Measured Thrust Coefficient (C_f)	1.89
Theoretical Thrust Coefficient (C_f^*)	1.95
Discharge Coefficient (C_d)	0.97

*75°F, Vacuum

WEIGHTS, lbm

Total Loaded	1,567
Propellant	1,450
Case Assembly	42.4
Nozzle Assembly	45.4
Igniter Assembly (with Model 2129 S&A)	7.6
Internal Insulation	19.3
External Insulation	0
Liner	0.9
Miscellaneous	1.4
Total Inert (Igniter Propellant not Included)	117
Burnout	105.4
Propellant Mass Fraction	0.925

TEMPERATURE LIMITS

Operation	30°F to 100°F
Storage	20°F to 110°F

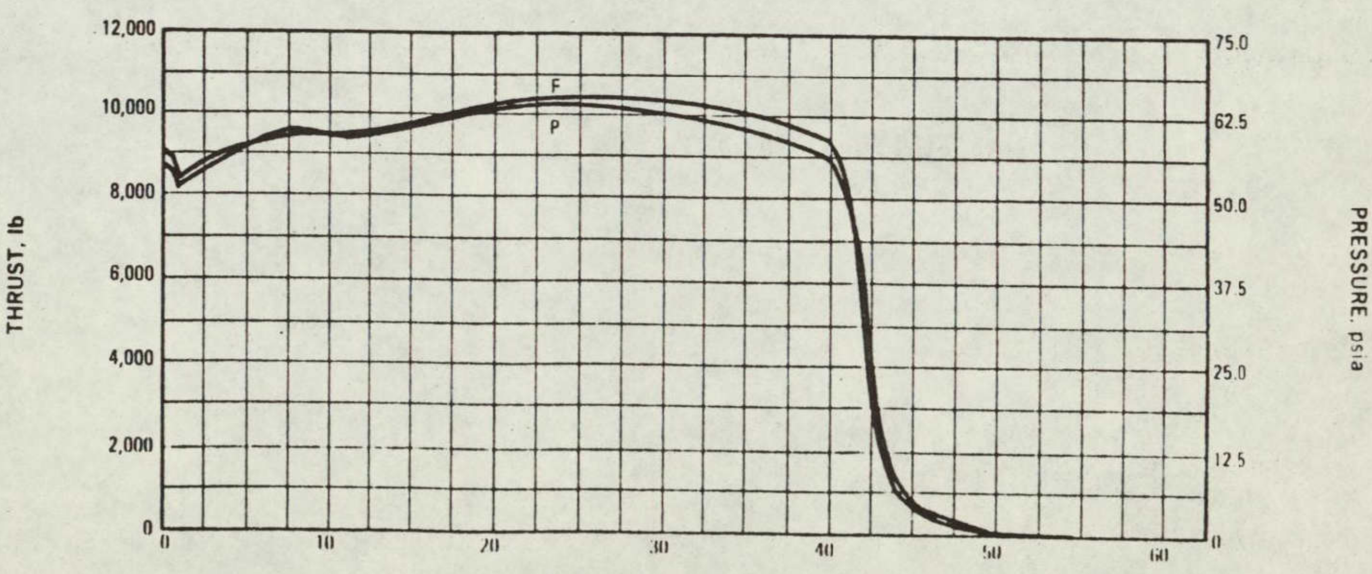


Figure 3.7-2 TE-M-364-15 Characteristics

Redundancy is provided where practical. Two remote safe and arm devices for each motor together with two sets of explosive transfer assemblies provide completely redundant ignition systems.

3.7.2 Other Motors

A variety of existing motors could be used for missions other than heavy payload low earth orbit. Replacement of the TE-M-364-15 second stage with a TE-M-616 for example permits high altitude orbits with moderate sized payloads.

The characteristics of the TE-M-616 is shown in Figure 3.7-3. Other applicable existing motors are summarized in Table 3.7-1.

3.7.3 Safety

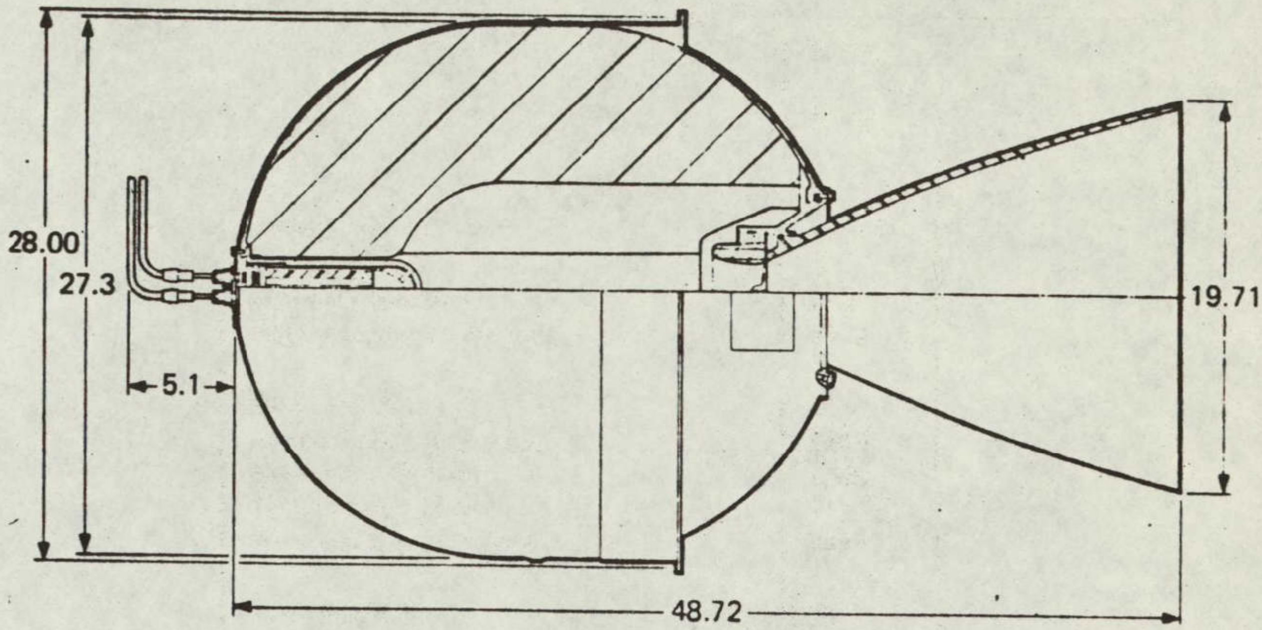
All motors considered above use Class II propellant which has been specified as an IUS requirement for use in the shuttle payload bay. This is a non-explosive propellant. The safe and arm devices are also the same as the IUS safe and arm device. Positive interlocks will prevent an arming signal prior to deployment from the payload bay.

3.8 Performance

Performance data are shown in the form of payload as a function of orbit altitude for specific and parametric configurations. The baseline spacecraft-motor combination provides payload capability in excess of the 1000 pound requirement at altitudes of 4000 n.miles and lower. Substituting the TE-M-616 motor in place of the TE-M-364-15 motor in the second stage increases the orbit altitude capability substantially at the lower payload weights, providing a 12-hour orbit capability with a 250 pound payload.

3.8.1 Requirements

The performance requirements are to provide the capability to transfer payloads from the Shuttle parking orbit (150 n.mile circular) to higher circular orbits. Altitudes up to geosynchronous are to be considered



Model Number	TE-M-616
Principal Diameter, in.	27.3
Motor Length (without initiator), in.	48.72
Burn Time (t_b), sec*	34.9
Average Thrust ($\bar{F}_b$), lbf*	6,000
Maximum Thrust, lbf*	6,480
Total Impulse, lbf-sec*	215,200
Propellant I_{sp}, lbf-sec/lbm*	293.1
Total Weight, lbm	799
Propellant Weight, lbm	734.2
Burnout Weight, lbm	57.7
Propellant Mass Fraction	0.919
Operating Temperature Range, ^{0}F	20 to 100
Operating Spin Rate, rpm	100
Vehicle Environmental Qualification	Delta

Figure 3.7-3 TE-M-616 Characteristics

TABLE 3.7-1 EXISTING SPACE MOTORS

MOTOR DESIGNATION	SUPPLIER	PROPELLANT WT LB.
TE-M-516	Thiokol	73
SVM-3	Aerojet	137
SVM-2	Aerojet	306
TE-M-604-1	Thiokol	437
TE-M-442-1	Thiokol	524
FW-5	United Technologies	577
SVM-5	Aerojet	630
TE-M-616	Thiokol	730
SVM-7	Aerojet	895
TE-M-364-15	Thiokol	1450
TE-M-364-19	Thiokol	1863
TE-M-364-4	Thiokol	2290

and provision shall be made for payload weights up to 1000 pounds. Specific mission requirements (payload, altitude, and inclination) are not defined.

3.8.2 Capability

A two-stage solid propellant vehicle has the capability to deliver one specific payload weight to one specific circular orbit altitude (coplanar) using a Hohmann transfer and expending the total energy at each burn. The payload-altitude capability is a function of the total propellant loading and the propellant ratio (ratio of first stage propellent to second stage propellant, W_{P1}/W_{P2}) as shown in Figure 3.8-1. It is shown that the optimum propellant ratio varies from about 1.2 for a 1000 n.mile orbit to about 3.0 for a 12-hour orbit (10,900 n.miles). The range of propellant loadings shown in Figure 3.8-1 generally cover the range available with motors which are within the dimensional constraints of the configuration. A propellant loading of 4000 pounds at a propellant ratio of 3.9 will deliver a 330 pound payload to geosynchronous altitude.

Figure 3.8-2 provides the payload-altitude capability for two specific configurations, the baseline spacecraft with the baseline motor combination and a configuration where a smaller second stage motor is used. To define the total region of payload-altitude capability for a specific configuration, non-Hohmann transfer trajectories are designed to use the total ΔV available at each burn. The baseline motor combination has a large payload capability for the lower orbit altitudes, exceeding 1000 pounds at altitudes below 4000 n.miles. When a 28.5 deg plane change is required, the payload is reduced to about 400 pounds. Using the TE-M-616 motor in the second stage in place of the TE-M-364-15 motor increases the altitude capability at the lower payload weights, providing the capability to reach a 12-hour orbit with a payload of 250 pounds.

3.9 Flight Support Equipment

The Baseline Installed Configuration (short tunnel) Figure 3.1-5 is the most critical from a space available standpoint.

Parametric Performance Capability

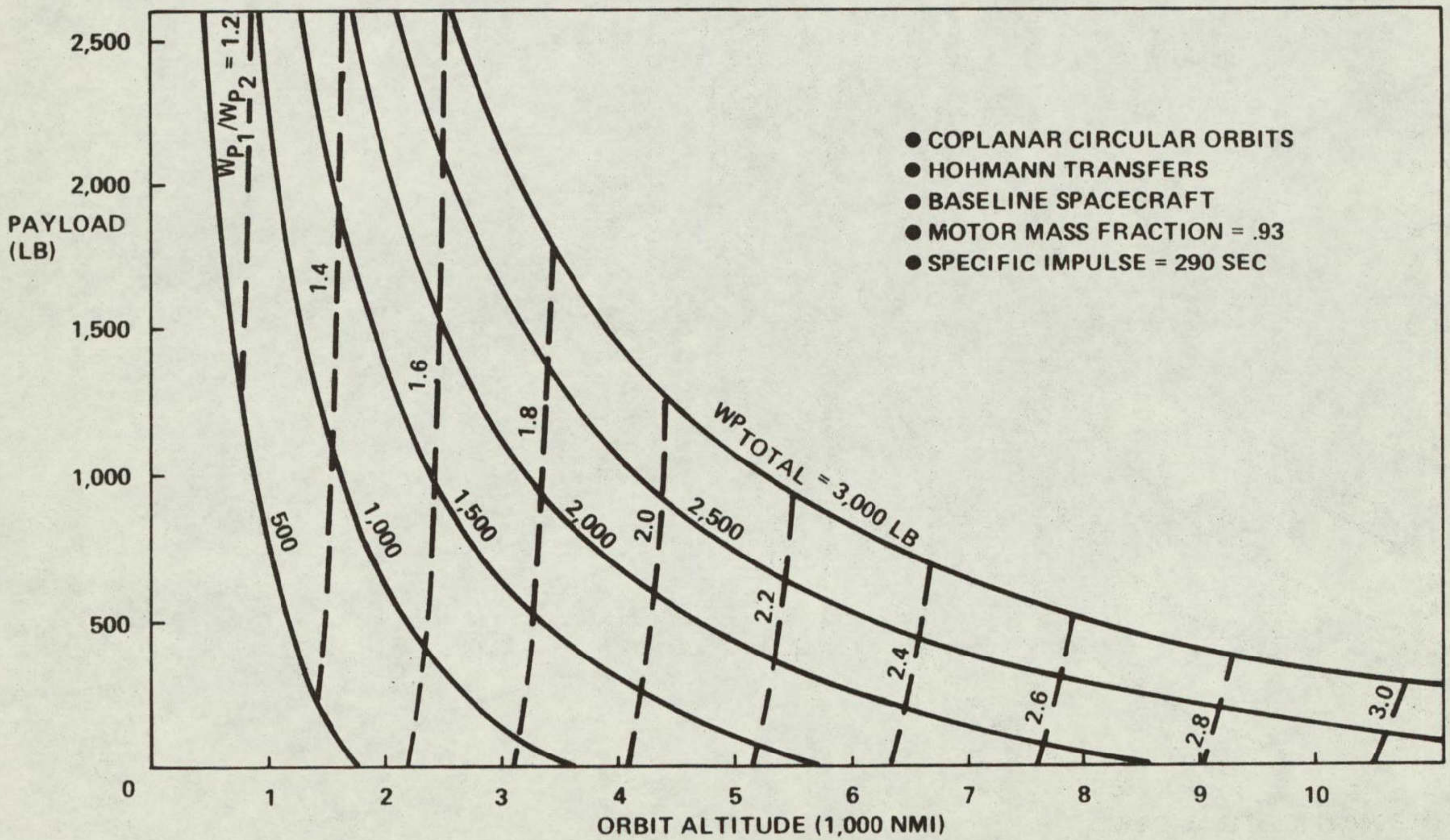


FIGURE 3.8-1

Circular Orbit Capability
Baseline Spacecraft

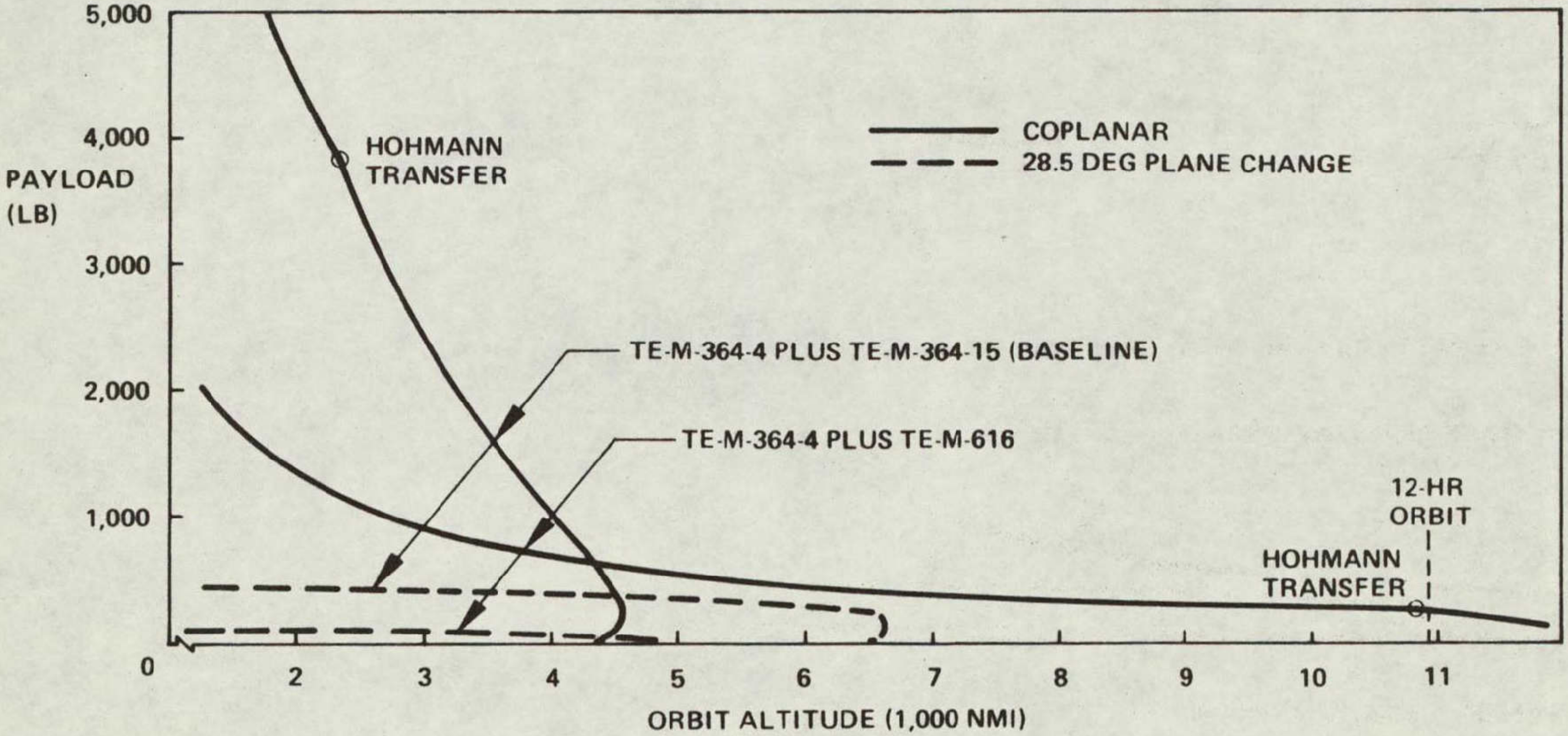


FIGURE 3.8-2

Payload attachment restrictions in the shuttle limit reaction loads to vertical and longitudinal only on the longerons and lateral only on the keel. The shuttle adapter configuration complies with these restrictions, and in addition, will allow the shuttle structure to "breath" around the adapter. This is accomplished by allowing adapter trunnion slippage in all directions except in the load reaction direction. The friction loads introduced at the longeron trunnions are carried by the adapter beams as a column.

The design conditions for the adapter are the singular crash conditions in three directions. The beam design presented here was selected for simplicity. It consists of two beams spanning the shuttle longerons with one beam connecting the aft beam to the keel. The adapter also contains a hinged split ring forming the aft space vehicle support. The forward vehicle trunnions attach directly to the two spanwise beams. This configuration will require only three unlock points for space vehicle deployment. The lock/unlock points will be similar to the longeron attachment fittings.

With the split ring opened and the space vehicle forward trunnions unlocked the manipulator can lift the vehicle out of the adapter for deployment. In case of a vehicle checkout no-go, the vehicle can be returned to the adapter and re-locked.

The N_x (forward) loads are reacted by the aft adapter beam only. The N_y (lateral) loads are reacted by the aft beam and the beam attaching to the keel. The N_z (vertical) loads are split between the fore and aft beams.

The reaction loads at the longerons and keel are within shuttle capabilities assuming that the aft adapter beam is in the vicinity of X_o 715. See Figure 3.9-1.

All adapter components are made of 7075-T73 aluminum. The forward beam is an I beam 6 in. deep with flanges 4 in. wide and 0.3 in.

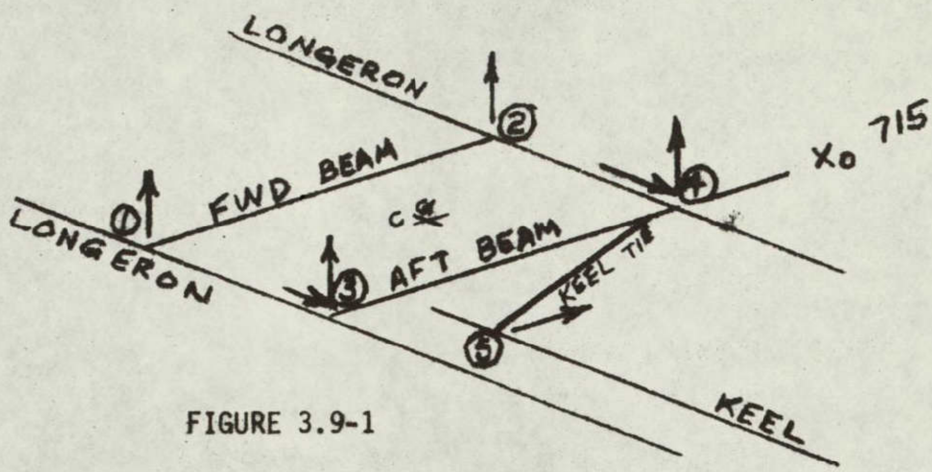


FIGURE 3.9-1

SHUTTLE REACTIONS TO ADAPTER CRASH LOADS

ALL REACTIONS ARE ULTIMATE

Load \ Location	1	2	3	4	5
N_x = 9.0			25,500H	25,500H	
N_y = 1.5			1,890H 4,390V	1,890H 4,390V	8,500
N_z = 4.5	6,370V	6,370V	6,370V	6,370V	
Shuttle* Capability	22,000V	22,000V	27,000H 22,000V	27,000H 22,000V	49,000

H = Horizontal

V = Vertical

* Assuming 1.5 Factor of Safety to Ultimate

thick. The web is 0.1 in. thick. The aft beam is a box 6 in. high and 10 in. wide. The sides are 0.5 in. thick, and the top and bottom are 0.25 in. thick.

The keel attachment beam is built up and tapers from 10 in. square at the aft beam attachment to the keel. The corner angles are 2 in. by 2 in. by 0.25 in. thick with 0.08 in. thick webs at the aft beam attachment.

3.10 Ground Support Equipment (GSE)

The GSE planned to support spacecraft integration and test includes maximum use of existing equipment or design from previous space programs and from Boeing inventory of capital equipment. This equipment is designed to also support payload integration and test and prelaunch operations support at the AFETR. Payload peculiar GSE is not included. Table 3.10-1 identifies existing and new GFE and Boeing-furnished GSE required to support the program.

3.10.1 Electrical GSE

The electrical GSE provides the capability to test, integrate and launch the space vehicle. It incorporates isolation/protection features to preclude damage to flight hardware. The major items are discussed below and show in a functional test setup in Figures 3.10-1, 3.10-2 & 3.10-3.

Mobile Satellite Test System (MSTS) - The MSTS is a complete command and telemetry ground station with the following capabilities:

- o Record test data
- o Transmit spacecraft commands
- o Monitor redline measurements
- o Identify out-of-tolerance measurements
- o Format, convert, and display selected test data
- o Service operator requests for data processing, command generation and transmission, display formatting, measurement conversions, and other software operations

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Ground Support Equipment

Item	Part number	Qty	Function/use	CFE	GFE	Source/remarks
Electrical GSE						
Mobile satellite test system		1	S/C checkout	x		S3-Boeing cap equipment
Contr and monitor test set	233-10064-1	1	S/C checkout		x	S3 initializing racks
Trickle charger and alarm box	233-10084-1 & Gulton EMAL 803	1	Charge S/C batteries		x	S3
Squib simulator	233-10065-2	1	S/C checkout		x	Mod for SEP squibs
Payload elect simulator	New	1	S/C checkout		x	
Shuttle I/F elect simulator	New	1	S/C checkout		x	
Earth sensor stimulator	SK 233-11004-1	1	S/C checkout	x		S3
Breakout boxes	SK 233-11002	1 set	Subsystem integ & test		x	S3 mod for new connectors
EMI breakout boxes	SK 233-11007	1 set	EMC test		x	S3 mod for new connectors
Test cables		1 set		x		Partial set avail (S3)
Standard test equip		as reqd	S/C checkout	x		Boeing inventory
Mechanical GSE						
Payload mass simulator	New	1	Dev test acoustic/vib	x		Make
Shuttle mech I/F sim	New	1	Physical integ	x		Make
Component mass sim's	New	1 ea	Dev test acoustic/vib	x		Make
Shock recorder	SK 233-11028	1	Record transp shock		x	S3
Thermal vac test fixt	SK 11-04xxx	1	Support S/C for T/V test		x	S3 modified
Modal survey test fixt	SK 11-040156C	1	Support S/C for mod surv	x		P72-1 modified
EMC test fixt	New	1	Support S/C for EMC test	x		Wood support stand
Mag survey test fixt	New	1	Support S/C for mag surv	x		Non-magnetic adapter
Spin balance test fixt	SK 276	1	Support S/C for spin bal	x		P72-1 modified
Moment of inertia test fixt	SK 272	1	Support S/C for MOI meas	x		P72-1 modified
S/C handling sling	New	1	Vertical lift S/C adapters	x		Make (4-leg)
Space vehicle sling (horiz)	New	1	Horiz lift S/V	x		Make (strong back 4 leg)
Space vehicle sling (vert)	New	1	Vert lift S/V	x		Make (4 leg)
Hoisting beam assy	E23014-01	1	Lift rocket motors	x		Buy-Thiokol
Handling ring, TE-M-364-4	E 22980-1	1	Handle & rotate R/M	x		Buy-Thiokol
Handling ring TE-M-364-15	TBD	1	Handle & rotate R/M	x		Buy-Thiokol
Support stand	TBD	1	Support & rotate R/M's	x		Buy-Thiokol
Space vehicle integ stand	New	1	Vertical stackup & rotate	x		Make
Space vehicle dolly	New	1	Support S/V horiz	x		Make
Spacecraft dolly	New	12	Support S/C test & ship	x		Make
S/C dolly overbox	New	12	Enclose S/C for shipping	x		Make
Solar panel ship cont	New	12	Shipping	x		Holds 12 panels
Solar panel protective cover	New	12	Protect installed panels	x		Make (2 piece plexiglass)
Cylindrical adapt ship cont	New	12	Shipping	x		Make
Conical adapt ship cont	New	12	Shipping	x		Make
TE-M-364-4 ship cont	TBD	12	R/M shipping	x		Buy-Thiokol
TE-M-364-15 ship cont	TBD	12	R/M shipping	x		Buy-Thiokol
Spacecraft prot cover	New	12	Shipping	x		Double bag plastic
Conical adapt cover	New	12	Shipping	x		Plastic
Cylindrical adapt cover	New	12	Shipping	x		Plastic
RCS hydrazine serv cart	New	1	Service RCS		x	
GN_2 service equipment	New	1	Service RCS		x	
Pressure monitor panel	New	1	Monitor RCS pressure	x		Make
Inert TE-M-364-4 R/M	TBD		Physical integ	x		Thiokol
Inert TE-M	TBD		Physical integ	x		Thiokol
Sep bolts and squibs	TBD	4 sets	Sep/depl shock test	x		Buy

TABLE3.10-1

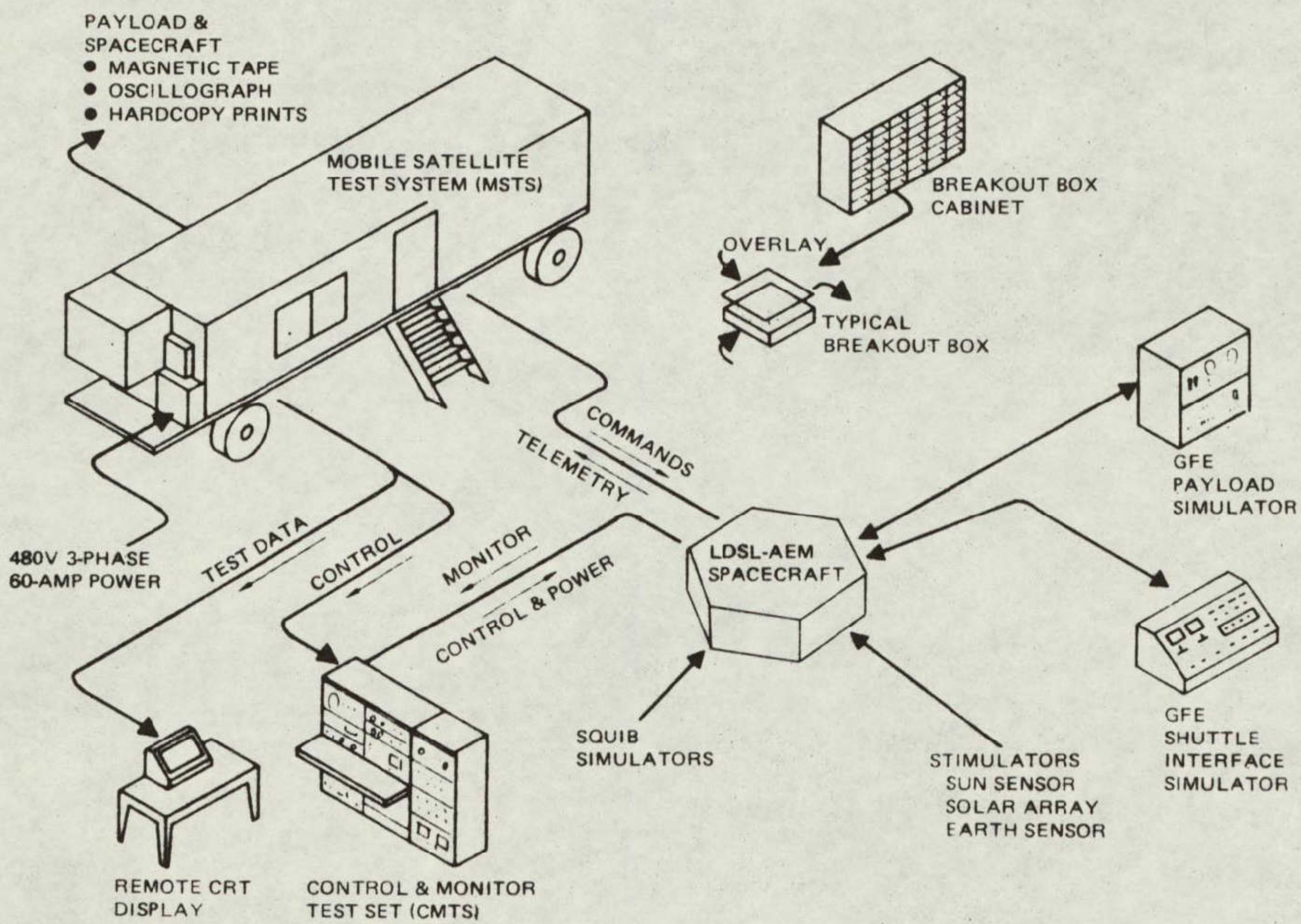


Figure 3.10-1. Electrical GSE

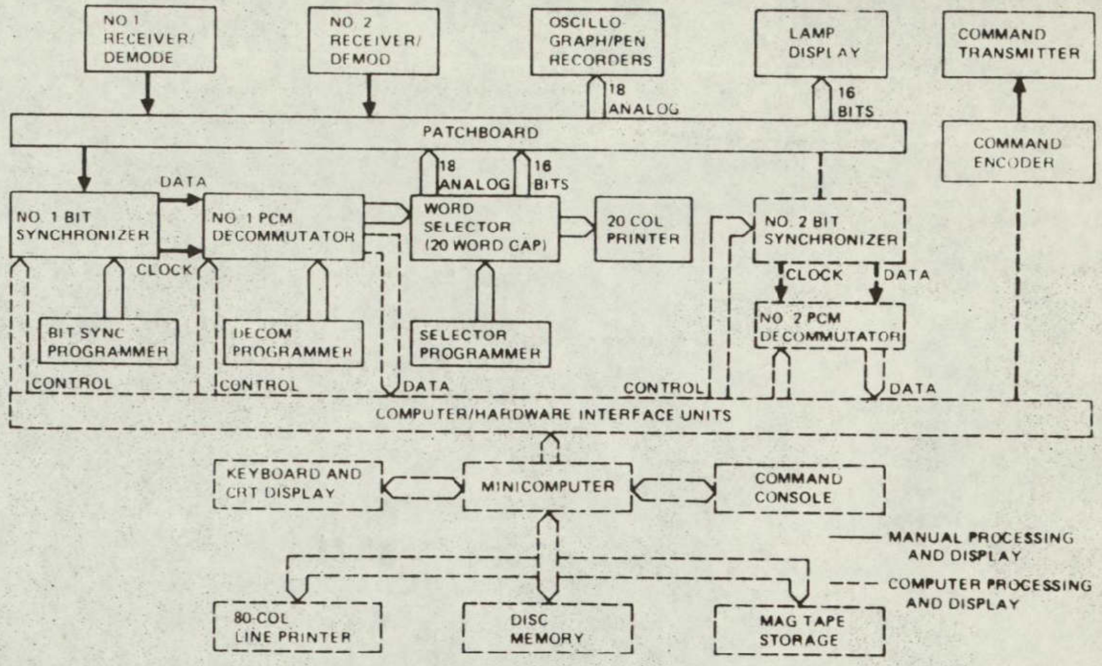
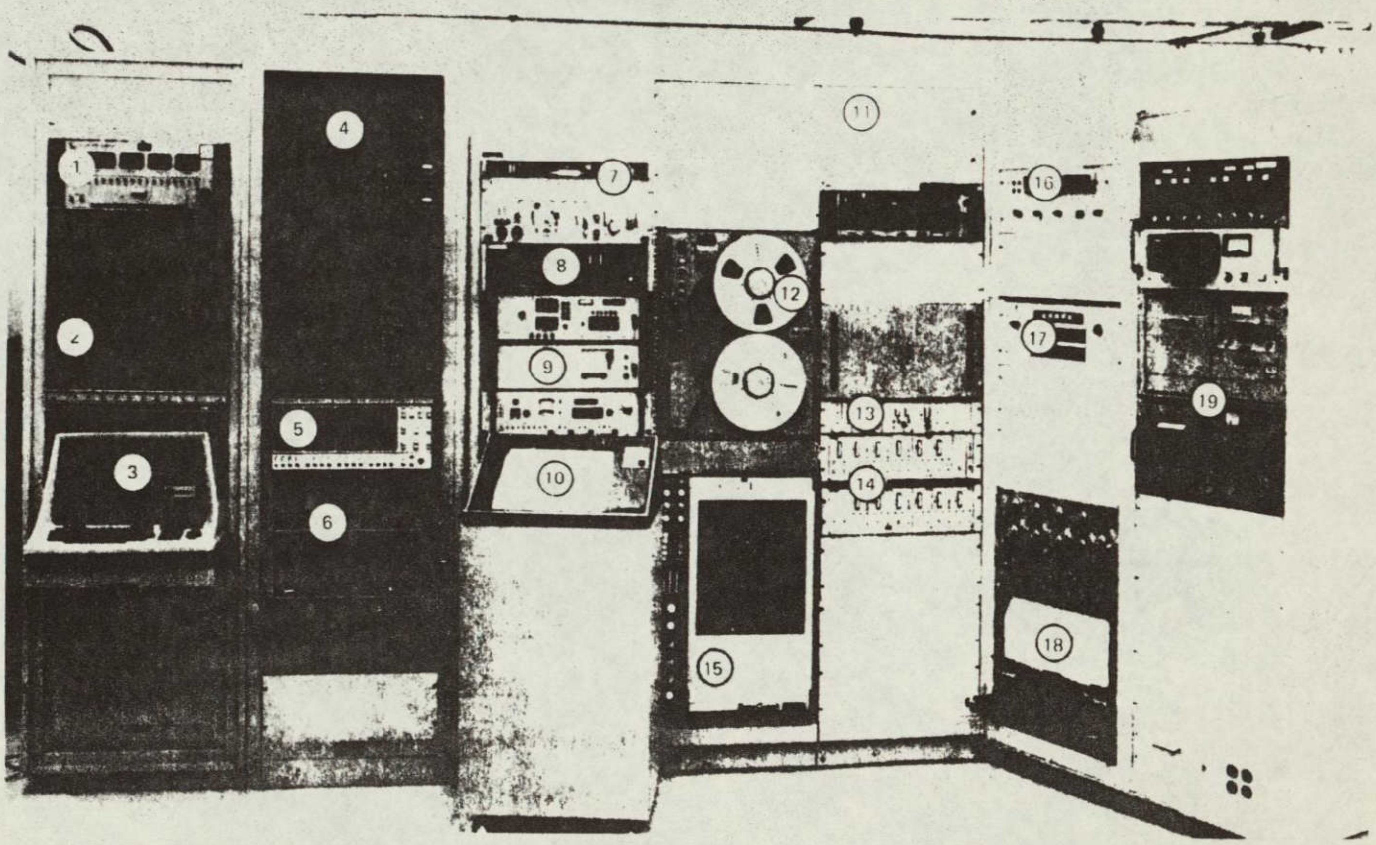


Figure 3.10-2. Mobile Satellite Test System Functional Diagram



1 TIME CODE TRANSLATOR
2 NOVA 820 MINICOMPUTER
3 VISITAR CRT DISPLAY
4 WANG DIGITAL RECORDER
5 COMMAND ENCODER
6 DISC UNITS (2)
7 RECEIVER DEMODULATOR
8 PCM BIT SYNC
9 PCM DECOMMUTATOR
10 VERSATEC PRINTER PLOTTER
11 STRIP CHART RECORDER WORD SELECTOR ELECTRONICS
12 FR-1300 RECORDER FR-1900 ANALOG RECORDER
13 PATCH PANEL
14 DATA DEMODULATORS IRIG FM & PSK
15 OSCILLOGRAPH
16 TIME CODE TRANSLATOR
17 TAPE SEARCH UNIT
18 PEN RECORDER
19 PRINTER/PLOTTER

Figure 3.10-3. Mobile Satellite Test System Equipment

- o Operate special test programs tailored to space vehicle testing such as tape-recorder bit error testing and AC&D simulation for orbital software verification test tapes.

Control and Monitor Test Set (CMTS)

The CMTS is contained in three standard equipment racks and was used during the S3 test program. It provides the following functions:

- o Electrical power to the spacecraft main bus and simulation of solar array output
- o The MSTS computer interface relay control of spacecraft functions
- o Visual monitoring and limit alarms for critical spacecraf functions (such as battery voltage and transmitter temperature)
- o Manual control of spacecraft power, sequencing, and monitoring functions during test operations
- o Troubleshooting with self-contained standard test equipment (oscilloscope, digital voltmeter, etc.)

Trickle Charger - A trickle charger is used to provide 250 $\pm$ 50 mA charge in the space vehicle battery at all times. It is in an explosion-proof case and has an automatic alarm system to indicate over or under voltage current.

Simulators - A squib simulator is used to simulate all ordnance bridgewires. It contains calibrated circuit breakers, lights, and test jacks to permit verification of ordnance circuit firing and timing.

A GFE payload simulator includes satellite power load interface circuitry to verify proper receipt of commands and simulated telemetry and pay load data inputs to the spacecraft.

A GFE shuttle interface simulator includes all control and monitor functions required in the shuttle during prelaunch, launch ascent and may incorporate a shuttle console control panel.

Stimulators - An Earth-sensor stimulator is provided to stimulate the Earth sensor during system tests. The stimulator is positioned in the field of view of the Earth sensor and provides periodic heat source to simulate horizon crossings to the Earth sensor. A Sun-sensor stimulator is provided to stimulate the Sun sensors during system tests. The stimulator provides a periodic stimulation with a high-intensity lamp within the field of view of the Sun sensor.

3.10.2 Mechanical GSE

The mechanical GSE provides for all assembling, handling, protecting, servicing, transporting and mechanically testing the spacecraft and propulsion sections of the space vehicle at the factory and the launch base. The major equipment categories are discussed below.

Spacecraft Handling Slings - Multipurpose slings are designed to lift the complete space vehicle including rocket motors, payload and shuttle attachment hardware or segments of the spacecraft as required for assembly, integration and test.

Rocket Motor Handling Equipment - Hoisting slings, support stands and handling rings are procured with the rocket motors to support motor rotations and installation tasks. Some handling equipment items are common to the TE-M-364-4 and -15 motors and do not require separate procurement.

Spacecraft Dolly - This dolly supports the spacecraft in the vertical position for transportation and test operations. It provides the capability to rotate the spacecraft to the horizontal position as required for component installation.

Space Vehicle Dolly - This dolly supports the complete space vehicle in the horizontal attitude for launch test and transportation to the shuttle integration facility.

Mechanical Simulators - Mass simulators for the payload and spacecraft components support initial mass properties, acoustic, modal survey, physical integration and static loads tests.

Inert rocket motors provide mechanical simulation of the TE-M-364-4 and TE-M-364-15 motors. A shuttle mechanical interface simulator simulates the mechanical interface between the spacecraft/shuttle adapter structure and the shuttle for static loads test and physical integration.

Shipping Containers and Covers - All major elements of the space vehicle are protected by special containers during storage and shipping. All containers except rocket motor containers, procured from Thiokol, are designed and built by Boeing.

Servicing and Monitoring Equipment - A GFE RCS hydrazine servicing equipment provide fill and monitor capability for the spacecraft RCS. A pressure monitor panel provides monitoring of the RCS during prelaunch operations.

Test Ordnance - Separation/deployment and demonstration shock tests require four sets of flight quality separation bolts, pin pullers and squibs that are expended during test.

4.0 INTEGRATION AND TEST

The LPSL-AEM spacecraft integration and test program is shown on the schedule Figure 4.0-1 and includes the development, qualification and acceptance tests briefly described in the following sections. The qualification program is for the baseline configurations. Qualification for option 1 and option 2 is on the basis of similarity.

Antenna Pattern Test - The optimum location, orientation, and configuration of the spacecraft antennas are determined by radiation-pattern measurements. The antennas are mounted on a full-size model simulating the significant spacecraft RF features and rotated about two axes to produce complete spherical coverage data. Radiation patterns are produced by transmitting a signal to the antenna under test and recording the signal amplitude received as the model is rotated. Radiation density plots are automatically produced. Actual isotropic gain levels are obtained by computer processing of the recorded data.

Preliminary Mass Properties Test - The spacecraft with mass simulators is weighed and balanced prior to the start of development testing to verify analytical predictions and to ensure that final weight and balance requirements are met without relocating components or payloads. The spacecraft is statically and dynamically balanced using Boeing's model E-50 Schenck dynamic balancing machine and then weighed. The reduction in balance weight that could be achieved by relocating components is evaluated. Spin-axis inertia is measured on the Schenck moment-of-inertia (MOI) machine to verify analytical predictions.

Modal Survey Test - A modal survey test is performed to verify mode shapes and frequencies and validate the dynamic model. The test configuration is rigidly mounted at the TE-M-364-15 rocket motor separation interface to a base fixture and excited with electromagnetic shakers. Accelerometers are installed along the structure and on designated equipment components to define resonances and measure mode shapes. The test includes low-level sine sweeps in three orthogonal axes followed by resonant dwells to identify all

Integration and Test Schedule

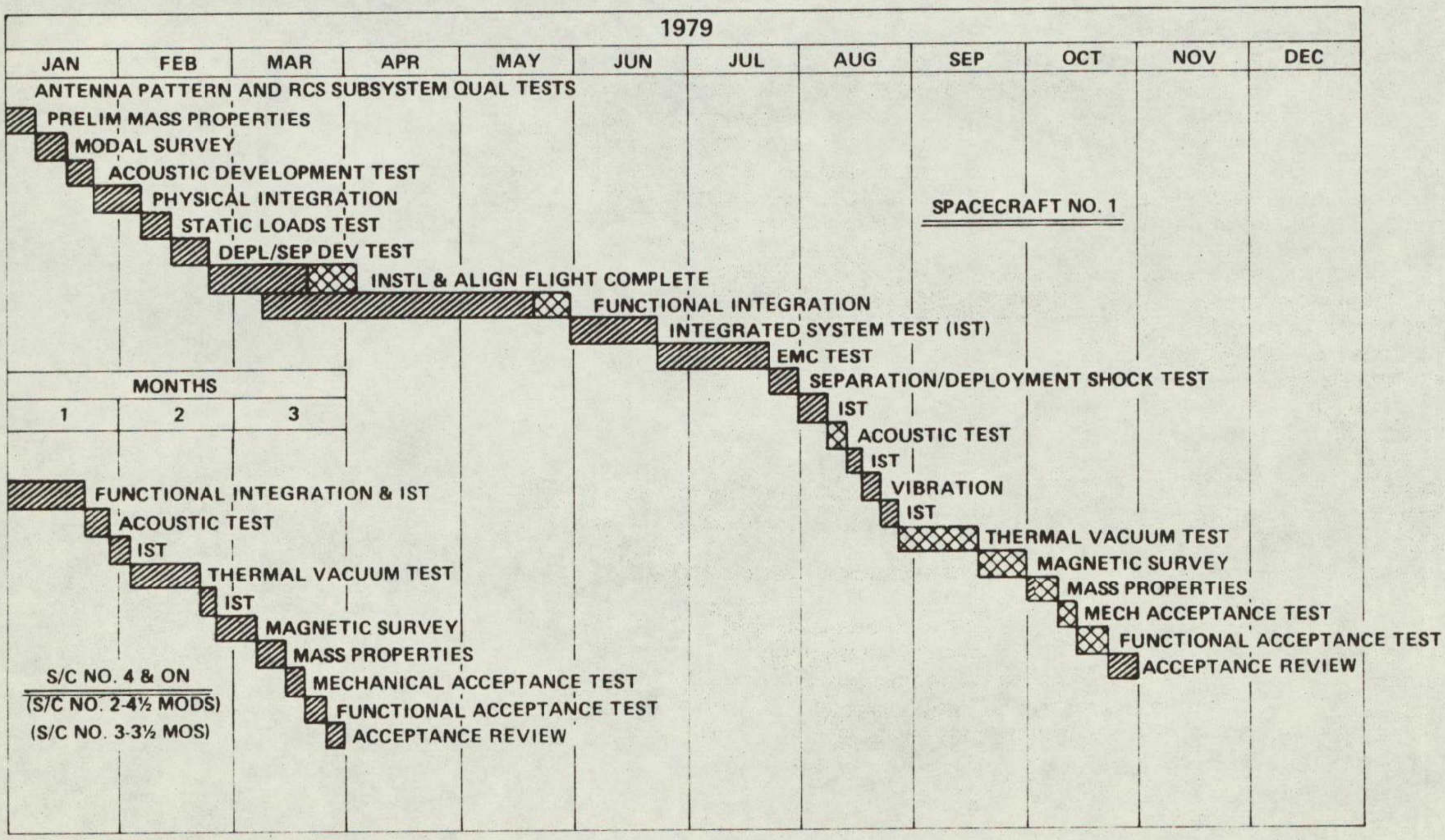


FIGURE 4.0-1

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modes below 50 Hz. Vibration levels are controlled to limit maximum responses to predetermined safe values. Response data verify that the dynamic model contains no modal mass coupling terms in the normalized generalized mass matrix outside the range of ± 0.1.

Acoustic Development Test - An acoustic test is performed to verify that the payload and equipment vibration responses to the acoustic environment do not exceed their allowable limits. The spacecraft is mounted vertically in the Boeing acoustic chamber and subjected to an acoustic spectrum 3 dB above the maximum predicted flight environment for 3 minutes. Response accelerometers are installed at selected component interfaces to record responses; microphones are used to record the actual test environment. An initial test run is made at 6 dB below full level to verify proper shaping of the acoustic spectrum.

Physical Integration - A physical integration test is performed to verify physical interfaces, clearances, access, RCS servicing, ordnance connections, mechanical support equipment, and procedures. The hardware is assembled and handled in the same sequence as at the launch base. RCS servicing operations are simulated to verify physical interfaces and procedures. Hydrazine is not loaded into the system. The interface and alignment between the spacecraft, adapters and rocket motors are verified by vertical assembly of the space vehicle on the space vehicle integration stand. The space vehicle is then rotated to the horizontal attitude and installed on a dolly. Procedures and handling equipment for horizontal installation in the shuttle interface structure are verified by mating the space vehicle to the structure.

Static Loads Test - The space vehicle including mass simulated components and payload, adapters and inert rocket motors is installed in the flight support structure and simulator and rigidly supported in the shuttle boost attitude. The static loads test is conducted using this test configuration to demonstrate the adequacy of the space vehicle structure and shuttle interface structure to meet strength requirements. Instrumentation is installed to measure stresses at critical locations, to define deflection characteristics and to preclude overloading conditions. Static proof loads are applied at specific locations and deflections measured and evaluated to verify the capability of the space vehicle, attaching mechanisms and structure to withstand shuttle boost and space vehicle free flight loads without sustaining permanent deformation or damage.

Separation/Deployment Test - Separation/deployment tests are conducted to verify functional characteristics and establish payload and equipment responses to the pyrotechnic release devices.

The spacecraft, payload mass simulator, adapters and inert rocket motors are assembled in a vertical attitude on a test fixture and the rocket motor separation ordnance fired sequentially. A counterbalance system is used to simulate zero-g on the section to be separated. Instrumentation includes accelerometers at selected payload and equipment locations to record shock response, high-speed movies to record separation, and linear transducers to record stroke vs. time.

Deployment tests of the solar array are conducted to verify deployment force margins, alignment repeatability, proper functioning of the stop-spring assembly, and boom stiffness in the deployed position. The test configuration is positioned to allow the deployment plane to be horizontal, and a spring counter-balance system is used to simulate zero-g. Spring forces and friction moment vs. stroke are measured and compared with design requirements. One deployment test is conducted by firing flight-equivalent ordnance. Instrumentation includes accelerometers at selected payload and equipment locations to measure shock response, high-speed movies to record deployment, and linear or torsional transducers to record stroke vs. time. Precision levels and transits are used to measure deployment repeatability.

Alignment Checks - The rocket-motor interface and payload and equipment alignments are verified by standard optical-alignment techniques; the alignments are recorded in the alignment test document. Critical alignments are adjusted to predetermined tolerances. Component alignment is accomplished by shimming and by reaming the attachment holes to full size. Attachment holes are initially piloted to permit final adjustment.

All alignments are made with respect to the spacecraft reference axes defined as two mutually perpendicular axes in the spacecraft/TE-M-364-15 rocket motor separation plane and a geometric centerline perpendicular to and passing through the theoretical centerline of this plane. All static and dynamic balancing is performed with respect to these axes to accurately relate CG and principal-axis alignment to the geometric axes.

Functional Integration - Wire harness continuity, insulation resistance, high-potential tests, and electrical-connector/pin-retention tests are performed as part of the manufacturing process after the wire harnesses are installed on the structure. Also, RCS proof and leak tests are performed during the manufacturing process.

Subsystem components are incrementally aligned, if required, installed, and connected. Each subsystem is tested separately where practical and then connected to interfacing subsystems to verify combined performance.

Functional integration testing verifies proper operation of all equipment and redundancy including:

- o The use of breakout boxes to verify proper power and command interfaces within the spacecraft.
- o Functional testing of the power subsystem using simulated solar-array power and test batteries.
- o Functional testing of the TT&C subsystem, including exercising redundant channels, processing of telemetry data, demodulation of RF commands, and transmitting RF telemetry signals by the respective transmitters and and antennas.
- o End-to-end telemetry calibration and noise measurements.
- o AC&D subsystem functional checkout to verify response to commands and stimulation.
- o Functional checkout of payload and shuttle interfaces using electrical simulators to verify compatibility of control, command and telemetry signals.
- o Verification of functional compatibility of the spacecraft and electrical GSE.

Integrated System Test (IST) - The IST provides a complete functional test of all the electrical/electronic systems in the space vehicle, including ordnance circuits and redundancy. The test is automated using the MSTS and test software. The test is performed in a flight sequence from liftoff through orbital operations to the maximum extent possible. All commands are sent to the spacecraft and proper response verified. Spacecraft equipment is exercised through injection and typical orbital sequences with sensors stimulated. Payload and shuttle interfaces are verified using electrical simulators. The first IST verifies operation of the complete spacecraft at high-, low-, and nominal-voltage levels. Subsequent IST's performed before and after environmental tests are run at nominal bus voltage.

Electromagnetic Compatibility (EMC) - Spacecraft EMC tests are performed to verify the EMI safety margin requirements of MIL-STD-1541. These safety margins are 6 dB for power and signal circuits and 20 dB for ordnance circuits. The test includes a magnetic and electric field survey and steady state and transient interference measurements on the operating spacecraft. Data from this test are compared with EMI susceptibility threshold data for EMI safety margin comparisons and verification. Testing also includes RF susceptibility to verify EMC with launch and ascent electromagnetic environments.

Deployment/Separation System Test - The spacecraft is subjected to three firings of all pyrotechnic devices to verify that the resulting shock response causes no performance degradation of spacecraft equipment. This test also serves to qualify and accept components for the shock environment.

The V-band clamp separators that separate the TE-M-364-4 and -15 rocket motors and ordnance devices that release the solar array are sequentially fired.

The test configuration is positioned verically and the spacecraft counter-balance to simulate a zero-g environment. The solar array is allowed to partially deploy to verify no hangups result from the pyrotechnic firing. During the test, electrical power is applied, all subsystems are operating in their planned mode, and critical parameters monitored. An abbreviated IST is performed after the test to verify functional integrity.

Acoustic System Test - An acoustic test of the flight-ready spacecraft is conducted to detect possible workmanship defects, such as faulty solder joints, loose connectors, etc., not found by physical inspection. The space vehicle is installed vertically in the acoustic test chamber and subjected to an acoustic environment 3 dB above maximum predicted flight level for 1 minute. The acoustic level is then lowered 6 dB to allow completion of functional test. Spacecraft operation is monitored for signs of failure, degradation, or intermittent operation. An abbreviated IST is conducted before, during, and after the test to verify no functional degradation resulted from the acoustic environment.

Vibration Test - a vibration test of the spacecraft is conducted to demonstrate its ability to withstand the vibratory excitations associated with launch operations. Sine and random tests in three orthogonal axes to the levels predicted for launch and powered flight are conducted using an electro-mechanical vibrator. The spacecraft is instrumented with accelerometers to monitor and record the response of the structure elements and mounted components to the imposed environment.

Thermal Vacuum/Balance Test - The spacecraft thermal vacuum test combines a thermal cycling test with a thermal balance test to achieve the following:

- o Verify thermal control subsystem performance.
- o Detect material, process, and workmanship defects that respond to thermal vacuum conditions.
- o Verify the spacecraft analytical thermal model.
- o Verify all spacecraft subsystem operation in the thermal vacuum environment.
- o Monitor areas where potential outgassing may be detrimental to any spacecraft function.

The spacecraft without the solar array is installed in the thermal vacuum chamber and subjected to 48 hours of typical orbital operation after four temperature cycles between the worst case hot and worst case cold conditions.

The data is obtained on thermal vacuum environments and thermal response of coatings to the simulated solar beam and, after satisfactory data correlation and thermal-model updates, the analytical thermal model is used to predict flight temperatures and verify thermal subsystem performance. If necessary, thermal trim will then be provided to fine-tune the thermal-control subsystem prior to delivery.

At the conclusion of the thermal vacuum test, the spacecraft, while still mounted on the spin fixture with connections through slip rings, will be operated and spun at the maximum predicted spin rate while being monitored to verify no out-of-tolerance performance degradation occurs due to the spinning environment.

Magnetic Survey Test - The spacecraft is compensated for magnetic dipole moment to reduce extraneous torques on the attitude control system. Measurements of magnetic fields at and near the payload are made to verify compliance with interface requirements. Measurements are made under the following space vehicle conditions:

- o As assembled
- o Permed
- o Depermed
- o Powered up

Measurements are compared to baseline values determined from magnetic survey of each package before installation.

Final Mass Properties Test - Weight, spin balance and MOI testing provide the necessary data to verify that weight, center of gravity, dynamic balance, and MOI requirements are met for boost, injection and on-orbit operation. The spacecraft is spin-balanced in the boost configuration without the rocket motors or payload (solar array stowed) while supported from the TE-M-364-15 rocket motor interface. The rocket motor is weighed and balanced by the motor subcontractor. Both spin axis and transverse axis MOI are measured.

Mechanical Acceptance Test - A functional test of the solar array deployment mechanism is conducted to verify performance margins and alignments prior to delivery. Deployment forces and friction moments are measured under simulated zero-g conditions and operation with no potential hangups verified. Alignment in the deployed position is also verified.

The RCS is leak checked by pressurizing with GN_2 and checking all joints fittings and thruster value seats using "snoop" solution bubbleometer or volumetric leak detector. Nitrogen pressure is then removed and the system pressurized with helium. A helium mass spectrometer with hand probe is used to determine individual joint leak rates. This test is a repeat of the same test performed on the RCS early in the program after assembly of the system.

Electrical Acceptance Test - Prior to delivery of the spacecraft, a complete IST is performed. End-to-end VSWR and RF-insertion loss measurements on all RF cables is also performed, and the modulation index on each RF transmission link for each subcarrier and mode of operation is verified.

Spacecraft Acceptance Tests (Nos. 2 through 12) - System tests for spacecraft Nos. 2 through 12 are shown in Figure 4.0-1. These tests are essentially the same as the corresponding tests described for the first spacecraft except reduced in schedule time as result of learning and procedure validation occurring during testing of the first spacecraft. These tests include:

Functional integration
Integrated System Test
Thermal Vacuum and Spin Test
Acoustic Test
Magnetic Survey
Mass Properties
Mechanical Acceptance Test
Functional Acceptance Test

Prior to the above listed tests, the spacecraft RCS is proof and leak tested and the wire harness is checked for continuity and insulation resistance.

5.0 PROGRAM SCHEDULE

The LDSL-AEM Master Phasing Schedule, Figure 5.0-1, uses similar flow times for similar tasks as on the AEM Project. Key aspects of the schedule are: 1) Delivery of the first spacecraft vehicle is twenty months from authorization to proceed (ATP), 2) Spacecraft buy-off, Material Inspection and Receiving Report - DD-250 will be accomplished at Contractor's facility, 3) No Instrument Module (payload) integration will be accomplished by the Contractor on this contract, 4) CDR will be accomplished the fifth month, 5) Drawing release will be complete the sixth month, 6) PDR will be accomplished prior to ATP. The Preliminary Design phase and PDR will be funded under another contract, 7) The twelve vehicle buy is in two parts of 6 each; ATP for the second buy is in the twenty-third month, 8) The test program for the first vehicle is 7-1/2 months (subsystem and system tests), the test time decreases to three (3) months for #4 spacecraft and on. The test time is predicated upon one set of test hardware and one test crew, 9) Eleven months flow time has been allowed for subcontractor procurement, this time period may change, dependent upon further hardware definition, 10) Long-lead procurement authorizations will be released within two weeks after ATP.

The LDSL-AEM Master Phasing Schedule flow times are consistent, not only with the current AEM Program, but also with other applicable programs that have been accomplished by The Boeing Company (i.e., S3, Burner II, etc.). The schedule represents a realistic plan for accomplishment of the LDSL-AEM Program.

LDSL–AEM Master Phasing Schedule

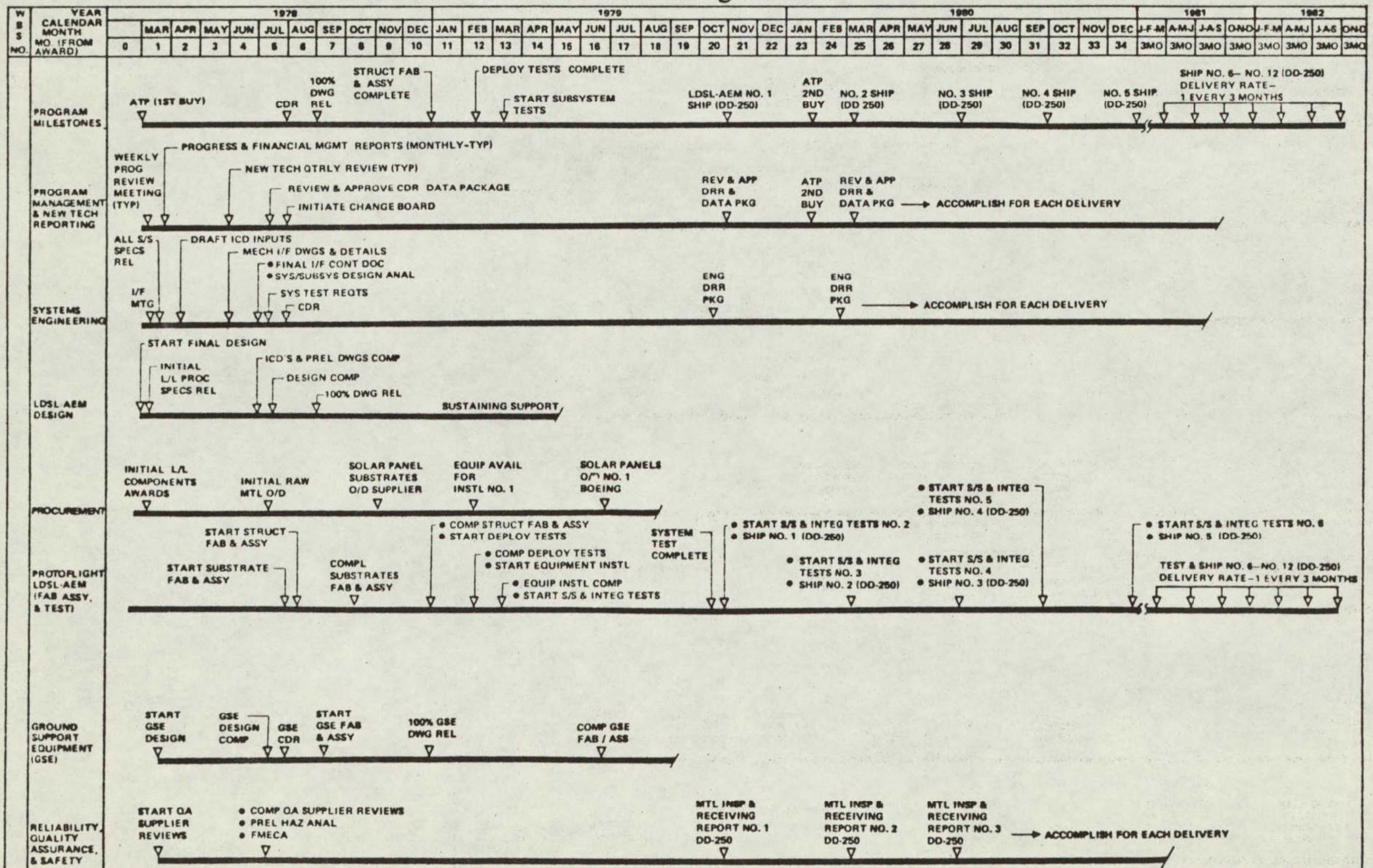

FIGURE 5.0-1