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APPLICATIONS EXPLORER MISSIONS (AEM)

MISSION PLANNERS HANDBOOK

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GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

APPLICATIONS EXPLORER MISSIONS (AEM)
MISSION PLANNERS HANDBOOK

Contributing Editor
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May 1974

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CAUTION TO HANDBOOK USERS

The AEM Spacecraft description presented in this document is based on a GSFC "in-house" design. However, the base module will be built by a spacecraft contractor. This contractor will have the option of using this design or to substitute one of his own as long as he meets the GSFC overall performance specifications. The performance specifications will incorporate the base and instrument module concepts.

Therefore, the user should be aware that the form factor and certain characteristics of the final spacecraft design could vary from those described, but functionally, the system will perform essentially the same tasks and have similar capabilities.

FOREWORD

This report describes the latest design of the AEM spacecraft as conceived by Goddard Space Flight Center (GSFC). The Heat Capacity Mapping Mission (HCMM) is the first Applications Explorer Mission and is discussed in Section 5.

The AEM spacecraft design is the result of a team effort of GSFC. Principal GSFC contributors and their areas of responsibility were:

P. G. Marcotte	Project Manager
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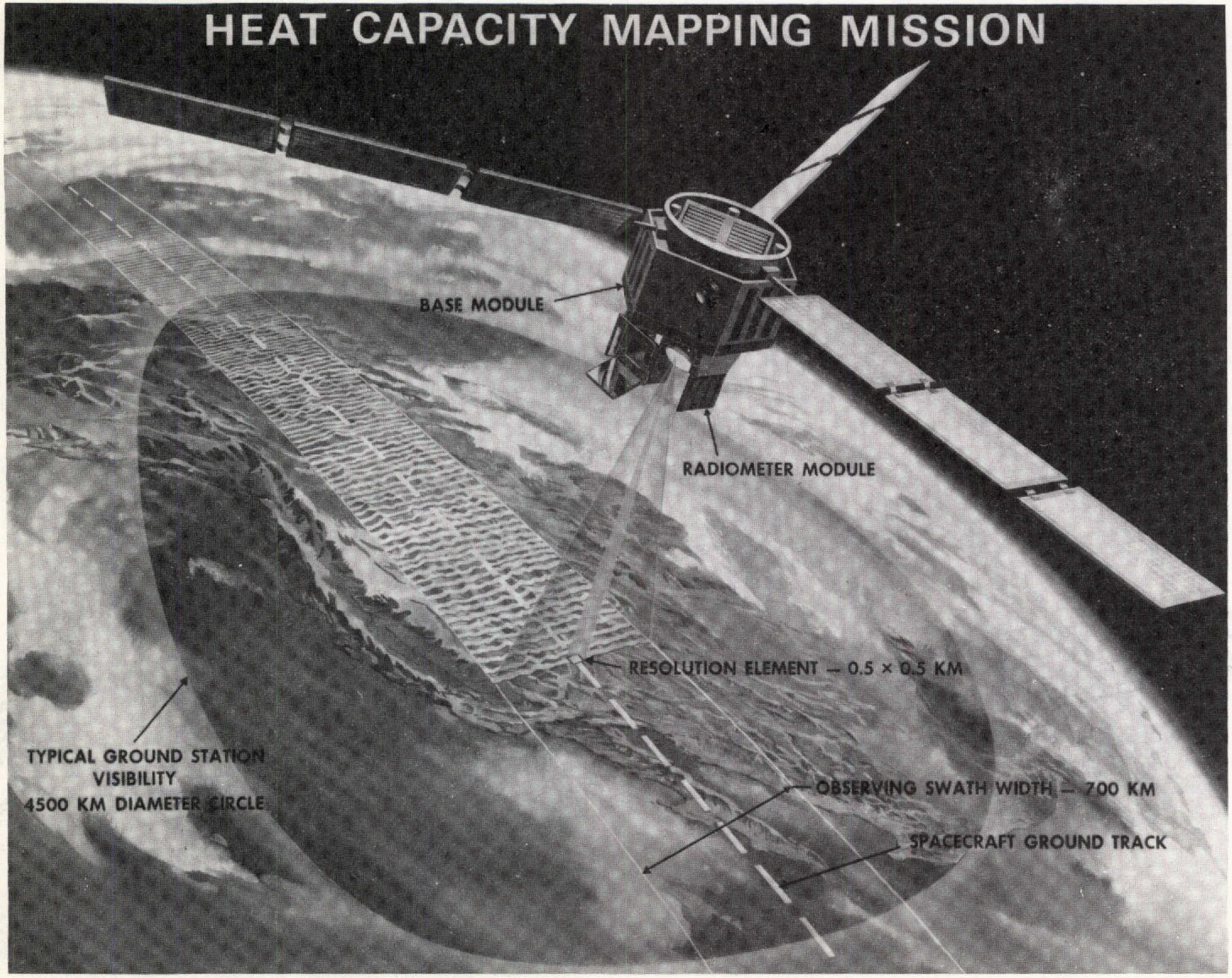
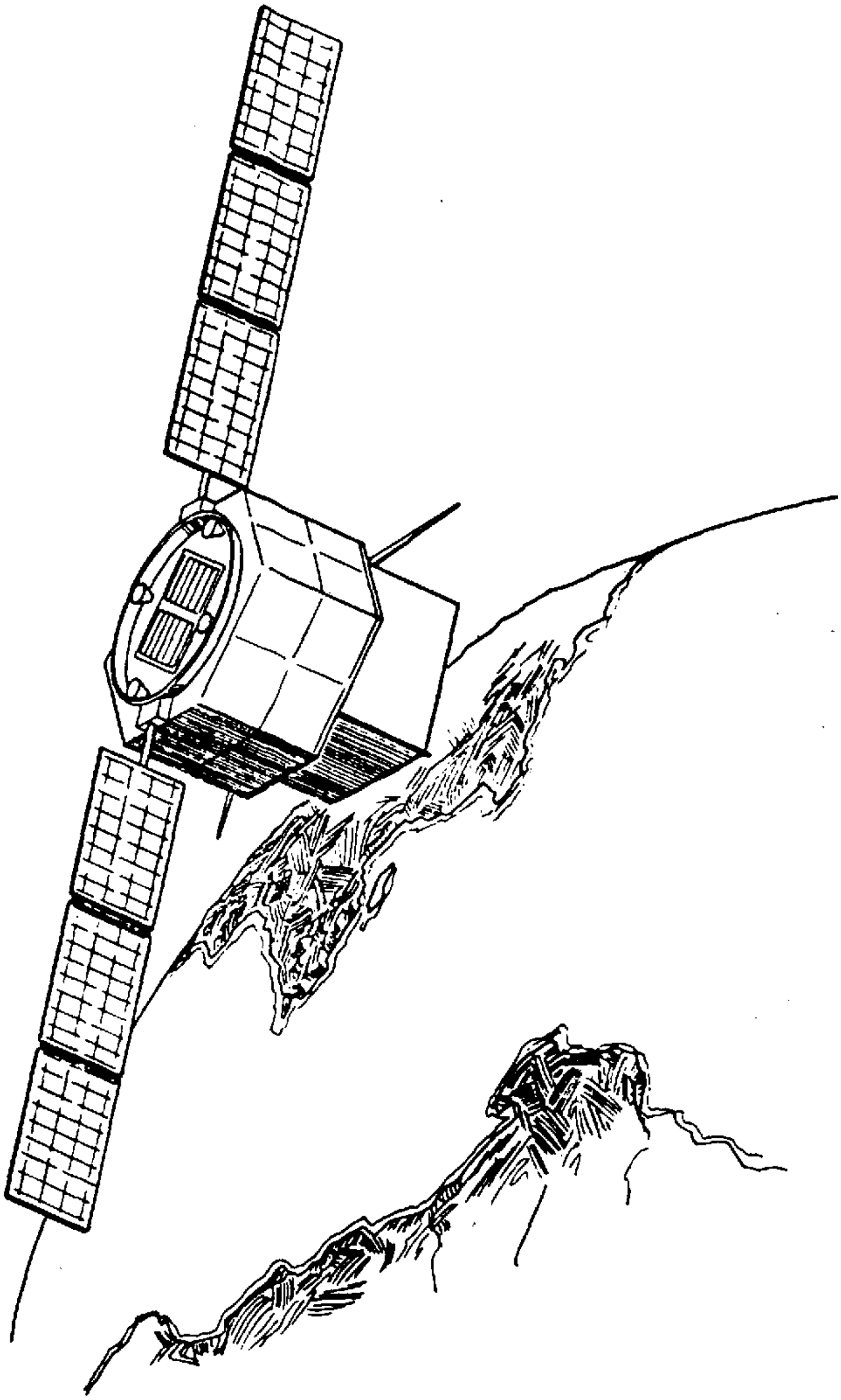
J. A. Dezio	Mechanical Systems
H. T. Wannemacher	Power Systems
E. M. Gaddy	Solar Array
L. C. Rogers	R.F. Systems
C. S. Vanek	VHF System
A. F. Block	S-Band System
R. B. Jackson	Antennas
D. C. Lokerson	Telemetry Encoder
B. K. Gabbert	Command System
T. H. Ratliff	Tape Recorder
J. M. Lidston	Attitude Control System
W. T. Tumulty	Thermal
D. I. Asato	Orbit Adjust System
H. F. Shaw	HCMR Instrument
J. B. Zegalia	Mission Operations
W. T. Davis	Network Support
L. R. Tant (La.R.C.)	Launch Vehicle



Frontispiece I—AEM 3-Paddle Configuration (Heat Capacity Mapping Mission)



Frontispiece II—AEM Two Paddle Configuration

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1.0 INTRODUCTION

The Applications Explorer Missions (AEM) Program is a planned series of space applications missions whose purpose is to perform various tasks that require a low cost, quick reaction, small spacecraft in a dedicated orbit. The Heat Capacity Mapping Mission (HCMM) is the first mission of this series.

The spacecraft described in this document was conceived to support a variety of applications instruments and the HCMM instrument in particular. The maximum use of commonality has been achieved. That is, all of the subsystems employed are taken directly or modified from other programs such as IUE, IMP, RAE, and NIMBUS. The result is a small versatile spacecraft.

The purpose of this document, the AEM Mission Planners Handbook (AEM/MPH) is to describe the spacecraft and its capabilities in general and the HCMM in particular.

This document will also serve as a guide for potential users as to the capabilities of the AEM spacecraft and its achievable orbits. It should enable each potential user to determine the suitability of the AEM concept to his mission.

1.1 ORGANIZATION OF THE MPH

The MPH is organized in the following manner.

- Summary of the AEM spacecraft capabilities. This includes a basic description with weight, power, data handling, and orbit parameters.

- Detailed systems description. Each spacecraft subsystem is presented. The launch vehicle is described. Tracking and data acquisition, data handling and processing methods and systems are discussed.

- Users guide. Aids are included to enable the user to determine the applicability of the AEM spacecraft to a proposed mission.

- HCMM description. The HCMM is described, including the details of how the basic AEM has been adapted to the peculiarities of this mission.

2.0 AEM SPACECRAFT SUMMARY

The AEM spacecraft, Frontispieces I and II has been designed to fly in a wide range of scout launched orbits from equatorial through polar inclinations. Frontispiece I depicts the HCMM and is typical of the three paddle configuration. The system flexibility is such that a variety of earth observation instrument requirements can be accommodated without significant subsystem modification. A modularized structure has been designed which separates the instrument and subsystem sections. This provides the ability to establish a well specified instrument interface, and to build and stock individual modules for rapid call-up to support an urgently needed mission. The three axis earth oriented attitude control system provides maximum capability for instruments in the applications disciplines. Table 2-1 lists the basic spacecraft parameters, including some optional equipment as described in Section 3.

The three paddle configuration, Frontispiece I, is used for polar, near noon sun-synchronous, and low inclination orbits. The two paddle configuration, Frontispiece II, is used for near twilight sun-synchronous orbits.

In order to make the most efficient use of available resources and yet derive an adequate and reliable spacecraft, maximum use of existing subsystem designs and hardware has been utilized. Table 2-2 lists the subsystems and the original sources of their design.

Table 2-1

Spacecraft Parameters

Power (orbital average)	28 volts ±2% regulation
Base Module	30 watts
Instrument	30 watts (minimum)
Data Handling	
1 to 40 Kilobits/sec.	
Two switchable bit rates	
Four Program Formats (2 fixed, 2 programmable)	
Data Storage	
GSFC standard 10^8 bit tape recorder	

Table 2-1 (Continued)

Command

Real Time - 1200 bits/sec.

63 Individual 37 bit serial digital commands

64 Impulse commands

Delayed execution of any or all commands

Thermal

Isolation between instrument and base modules

Utilizes active and passive techniques

Attitude Control

Three axis earth oriented

Stability ±1° pitch and roll; ±2° yaw;

Maximum body control rate 0.01°/sec

Determination ±0.5° pitch and roll; ±2° yaw

Communication

VHF Command receivers

VHF for low data rate instrument and housekeeping data

S-band for wideband instrument data and tape recorder dump

Weight (depending on required orbit)

Spacecraft at Launch:	up to 165kg (364 lbs)
Available for Instrument:	up to 63kg (150 lbs)

Volume Available for Instrument — approx. 196 liters (7 ft^3)

Orbit

Scout Launch Vehicle, low altitude, typical inclinations
of 0°, 38°, 50° or sun-synchronous at various local times

Table 2-2

AEM Commonality Matrix

		AEM	Deriv:
Att. Control	Reac. Wheel Scanners	S	NIMI
	Sun Sensor (Medium)	S	IU:
	Magnetometer	S	GE(
	Electronics	M&S	NIMI
	Electro-Magnets	M	NIMI
Power	Array	M	IM
	Electronics	S	IU]
	Battery	M	ITC
Data Handling	Dataplexer	S	IU]
	Converter	S	IU]
RF System	S-Band Transmitter	S	IU]
	VHF XMTR	M	RA
	VHF Recvr. & Ant.	S	IUI
	S-Band Ant.	S	IUI
CMD System	Decoder	S	IUI
	Relay Unit	S	IUI
Thermal System	Thermal Louvers	M	OA
Structure	Base Mod.	M	IM]
	Instrument Mod.	M	IM]

NOTES: S = SAME SUBSYS. FOR EACH CASE – SAME PHYSICAL BLACK BOX
M = MODIFIED DESIGN DERIVED FROM LISTED PROGRAMS

3.0 AEM SYSTEMS DESCRIPTION

Within the modularized structure of the AEM spacecraft, provisions have been made for optional subsystems to satisfy the peculiarities of future missions. Section three describes all of the standard base module subsystems as well as the optional ones. In some cases the standard as well as optional configurations are included in a single section, as in the case of the command system where one receiver is standard and a redundant one is optional. Table 3-1 lists the weight and average power requirements for the standard subsystems. Table 3-2 lists the same information for the optional ones. It should be noted that due to weight, space, and power constraints all options could probably not be employed on a single mission.

Table 3-1

Weight and Power Summary for Standard AEM Subsystems

Subsystem/Component	Quantity	Weight (Kg.)	Power (Watts)
Attitude Control			
Sun Sensor (Medium)	5	1.20	0.0
Sun Sensor Electronics	1	1.13	0.5
Horizon Scanner & Reaction Wheel (Right)	1	4.07	3.3
Horizon Scanner & Reaction Wheel (Left)	1	4.07	3.3
Control Magnets	3	1.36	2.0
ACS Electronics	1	3.72	7.4
3-Axis Magnetometer	1	0.23	0.0
Total		15.78	16.5

Table 3-1 (Continued)

Subsystem/Component	Quantity	Weight (Kg.)	Power (Watts)
Power			Power Requirements Not Specified. Losses Included as System Efficiency in Power Calculations.
Power Module	1	3.54	
Mission Adaptor	1	1.00	
Battery	1	5.90	
Dump Resistors	36	0.50	
Subtotal		10.94	
Solar Paddles			
3-Paddle System			
Double Sided Panel	6	9.47	
Single Sided Panel	3	3.16	
Main Hinges	3	2.04	
Segment Hinges	12	2.18	
Restraint System	1	0.91	
Subtotal		17.76	
2-Paddle System			
Single Sided Module	6	6.32	
Main Hinges	2	1.36	
Segment Hinges	8	1.45	
Restraint System	1	0.91	
Subtotal		10.04	
Power System Totals			
3-Paddle		28.70	
2-Paddle		20.98	

Table 3-1 (Continued)

Subsystem/Component	Quantity	Weight (Kg.)	Power (Watts)
<u>Communications</u>			
VHF Transmitter	1	1.00	1.0
VHF Receiver	1	0.55	0.4
S-Band Transmitter	1	1.00	8.0
Diplexer & Hybrid	1	0.49	0.0
VHF Antennas	4	0.40	0.0
S-Band Antenna	1	0.25	0.0
R-F Cables	Set	0.45	0.0
Subtotal		4.14	9.4
<u>Command</u>			
Decoder	1	1.13	1.0
Relay Unit	1	3.20	0.1
Operations Electronics	(See Table 3-2)		
Subtotal		4.33	1.1
<u>Telemetry</u>			
Encoder			
Subplexer	1	0.70	0.4
Dataplexer	1	1.36	1.0
Converter	1	0.68	2.0
Subtotal		2.74	3.4

Table 3-1 (Continued)

Subsystem/Component	Quantity	Weight (Kg.)	Power (Watts)
Structure			
Center Tube	1	2.45	
Platform	1	2.54	
Posts	6	0.82	
Skins	6	2.27	
Strong-Ring	1	1.22	
Angle-Frame	1	0.32	
Handling Brackets	3	0.23	
Misc. Bolts & Brackets	N/A	1.81	
Yo-Yo	1	0.68	
Paddle Brackets	3	0.23	
Subtotal		12.57	
Thermal			
Blankets		0.14	
Insulators		0.59	
Paint		0.91	
Louvers		1.13	
Subtotal		2.77	
Miscellaneous			
Harness	1	4.54	
Pyrotechnics	Set	0.10	
Balance Weights	Set	2.75	
Subtotal		7.39	
Basic Spacecraft Totals			
Two Paddle Conf.		70.70	22.4*
Three Paddle Conf.		78.42	22.4*

*Does not include S-Band transmitter, which is not operated continuously.

Table 3-2

Weight and Power Summary For Optional AEM Subsystems

Subsystem/Component	Quantity	Weight (Kg.)	Power (Watts)
<u>Attitude Control</u>			
Third Reaction Wheel	1	2.27	1.0
ACS Electronics	1	0.45	1.2
Total		2.72	2.2
<u>Power</u>			
3-Paddle (solar cells on both sides of all panels)			
Delete Single Sided Panel	3	-3.16	
Add Double Sided Panel	3	4.74	
Net Total		1.58	
<u>Communications</u>			
VHF Receiver (redundant)	1	0.55	0.4
S-Band Transponder			
Delete Transmitter	1	-1.00	-8.0
Add Transponder	1	3.50	11.0
Net Addition		2.50	3.0
<u>Command</u>			
Decoder (redundant)	1	1.13	1.0
Operations Electronics (mission unique, HCMM unit specified here)	1	1.00	1.0

Table 3-2 (Continued)

Subsystem/Component	Quantity	Weight (Kg.)	Power (Watts)
Telemetry			
Encoder			
Subplexer		0.70	0.4
Tape Recorder		4.55	2.5 to 10.0 (speed dependent)
Orbit Adjust			
Thruster & Valve	1	0.41	
Fill Valve	2	0.27	
Filter	1	0.05	
Pressure Transducer	1	0.11	
Temperature Transducer	2	0.18	
Connector & Wiring	1	0.45	
Lines & Fitting	1	0.23	
Tube Support & Brackets	1	0.11	
Misc. Hardware	1	0.45	
Tank		1.36	
N_2H_4		4.77	
GN_2		0.05	
Total		8.44	

3.1 MECHANICAL SYSTEMS

3.1.1 Structure

The base module (Figures 3-1a and 3-1b) is a structure that houses the subsystem packages. This module is hexagonal in shape, 75 cm (30 in.) across the flats and 56 cm (22 in.) in height. Most of the subsystem components will be mounted to the base module platform in a nonsymmetric arrangement (Figure 3-2) so that they can be fitted onto this thermal control surface. Attached to the top of the base module through a simple interface is the instrument deck. This honeycomb deck is the primary instrument module structure and can support electronic packages, antennas, sensors, thermal controls etc. Separation of the base and instrument modules will take place often and will not affect the integrity of either structure.

Structural design is based on the new Scout 107 cm (42 inch) heat shield and new 61 cm (24 inch) adaptor. Arranged within the fairing (Figure 3-3) is the FW-4S motor, 61 cm (24 inch) adaptor, the base module with solar paddles folded against its sides, and the instrument module. As Figure 3-4 shows, the structure can accommodate either a two or three paddle solar array.

3.1.1.1 Structural Characteristics—Spacecraft design goals have resulted in the following structural traits:

a. Subsystem packages are mounted to the shelf in a nonsymmetric manner except for earth and space pointed sensors.

b. The subsystem mounting shelf is the thermal control surface of the base module.

c. Nonmagnetic materials are used throughout the structure to minimize the residual magnetic moment.

d. A strong lightweight honeycomb deck is the primary instrument module structure.

e. Base to instrument module attachment is through a simple bolt circle interface.

3.1.1.2 Base Module Description—A base module design having six equal sides has been chosen because it maximizes both the package mounting area and solar cell area. Parametric studies to determine the optimum shape were based on the fairing dynamic envelope and a number of spacecraft requirements such as average power, number and size of packages, thermal dissipation, testing, weight, reliability, etc. A typical parametric summary is shown in Table 3-3.

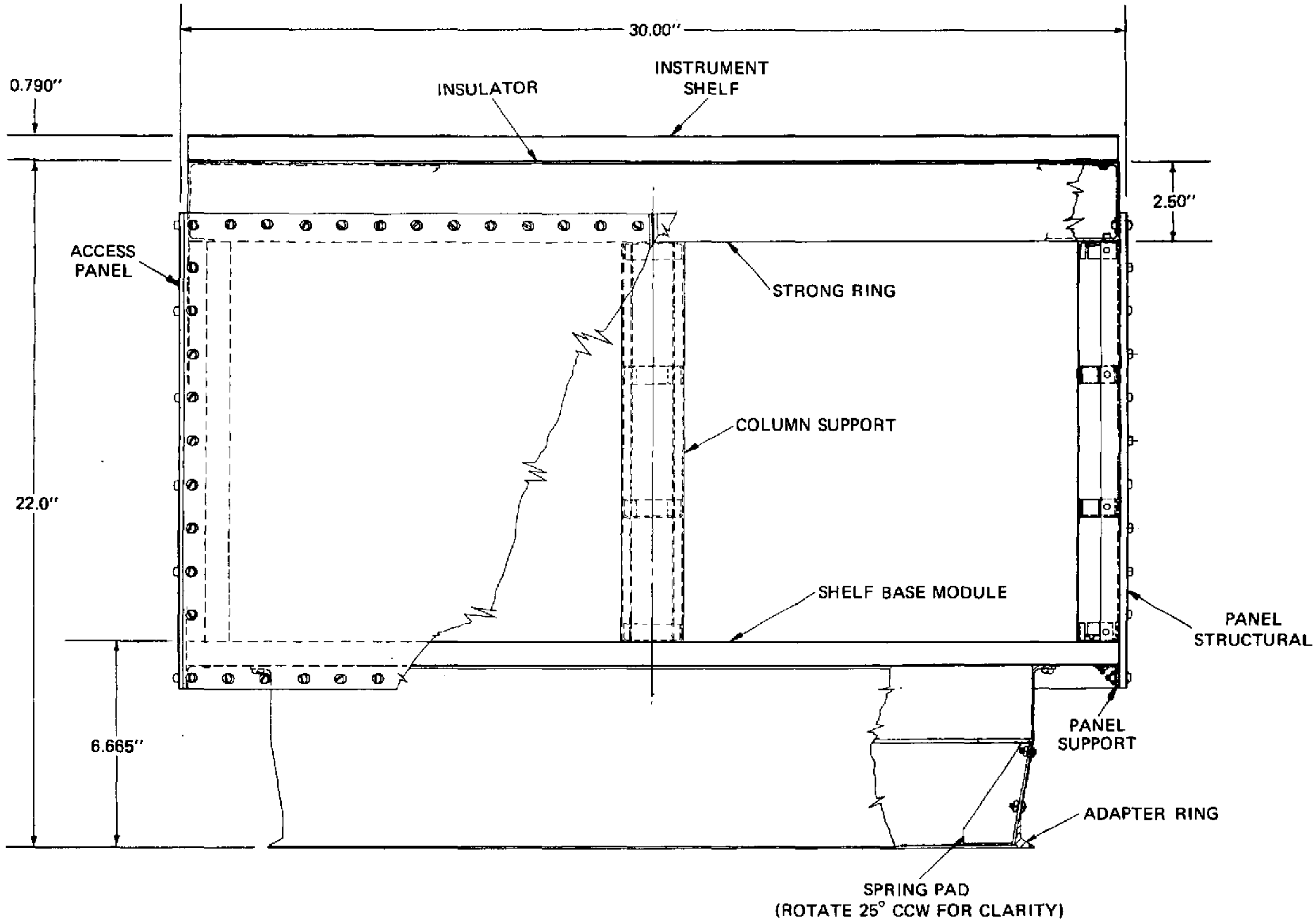


Figure 3-1a. Base Module Structure

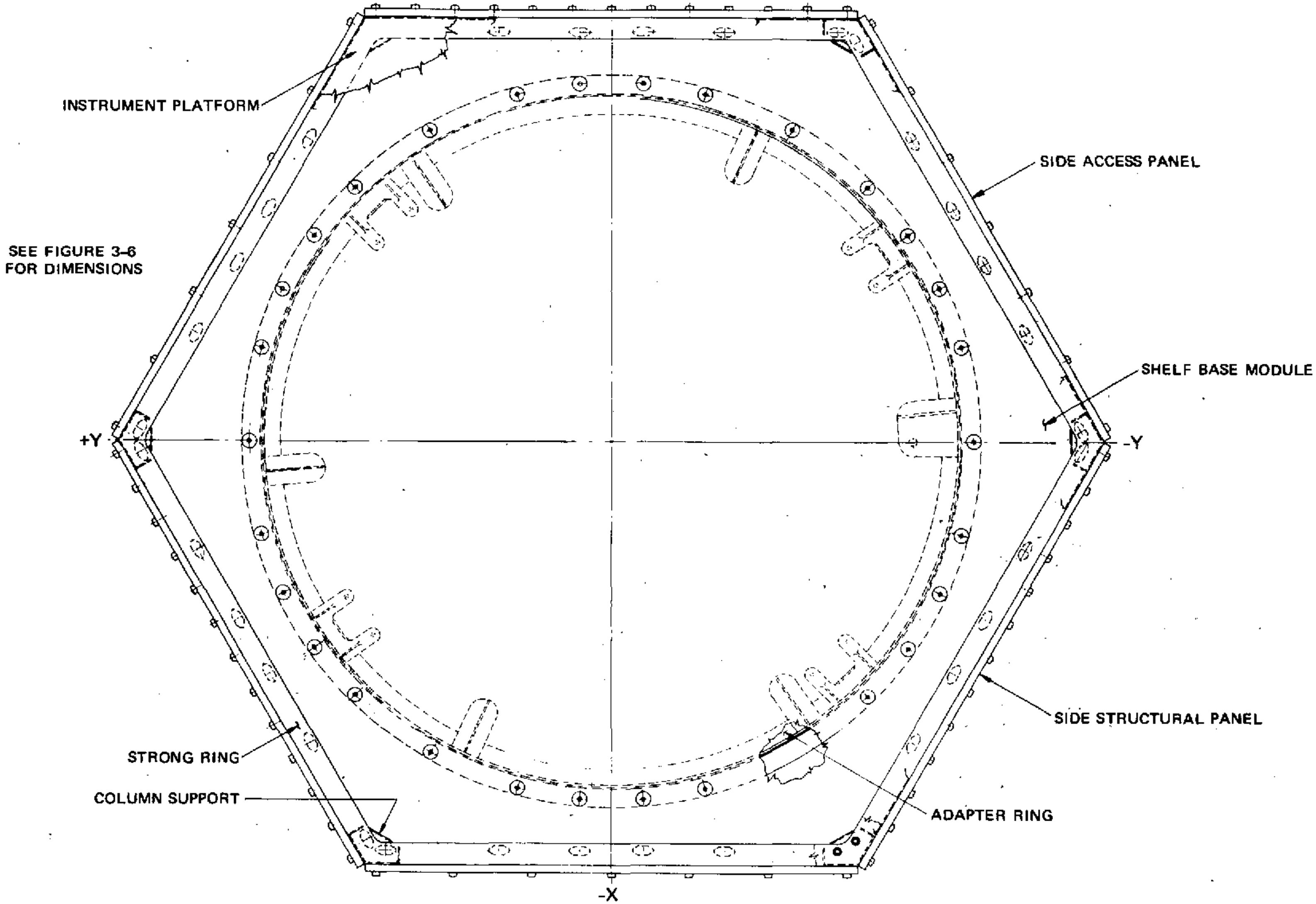
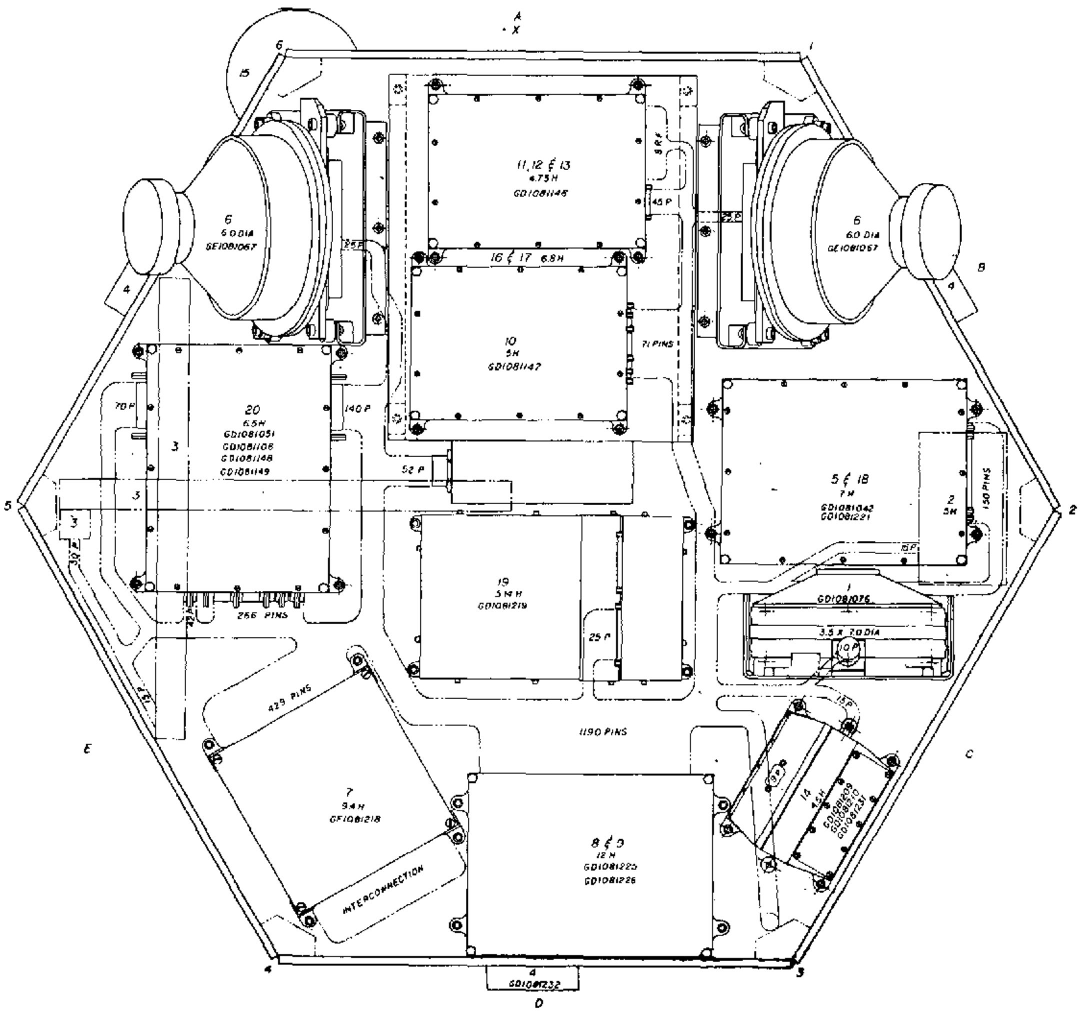


Figure 3-1b. Base Module (Top View)



1 REACTION WHEEL -1
2 MAGNETOMETER -1
3 MAGNET -3
4 SUN SENSOR -5
5 SUN SENSOR ELECTRONICS -1
6 REACTION WHEEL W/HOR. SCAN -2
7 ACS ELECTRONICS -1
8 COMMAND DECODER -2
9 COMMAND RELAY -1
10 STORE COMMAND PROCESSOR -1
11 VHF DIPLEXER & HYBRID -1
12 VHF RECEIVER -2
13 VHF TRANSMITTER -1
14 S-BAND TRANSMITTER -1
15 S-BAND ANTENNA -2
16 TAPE RECORDER TRANS. -1
17 TAPE RECORDER ELECT. -1
18 POWER CONDITIONING ELECT. -1
19 BATTERY -1
20 SUBPLEXER -1
DATAPLEXER -1
CONVERTER -1

Figure 3-2. Nonsymmetric Component Placement

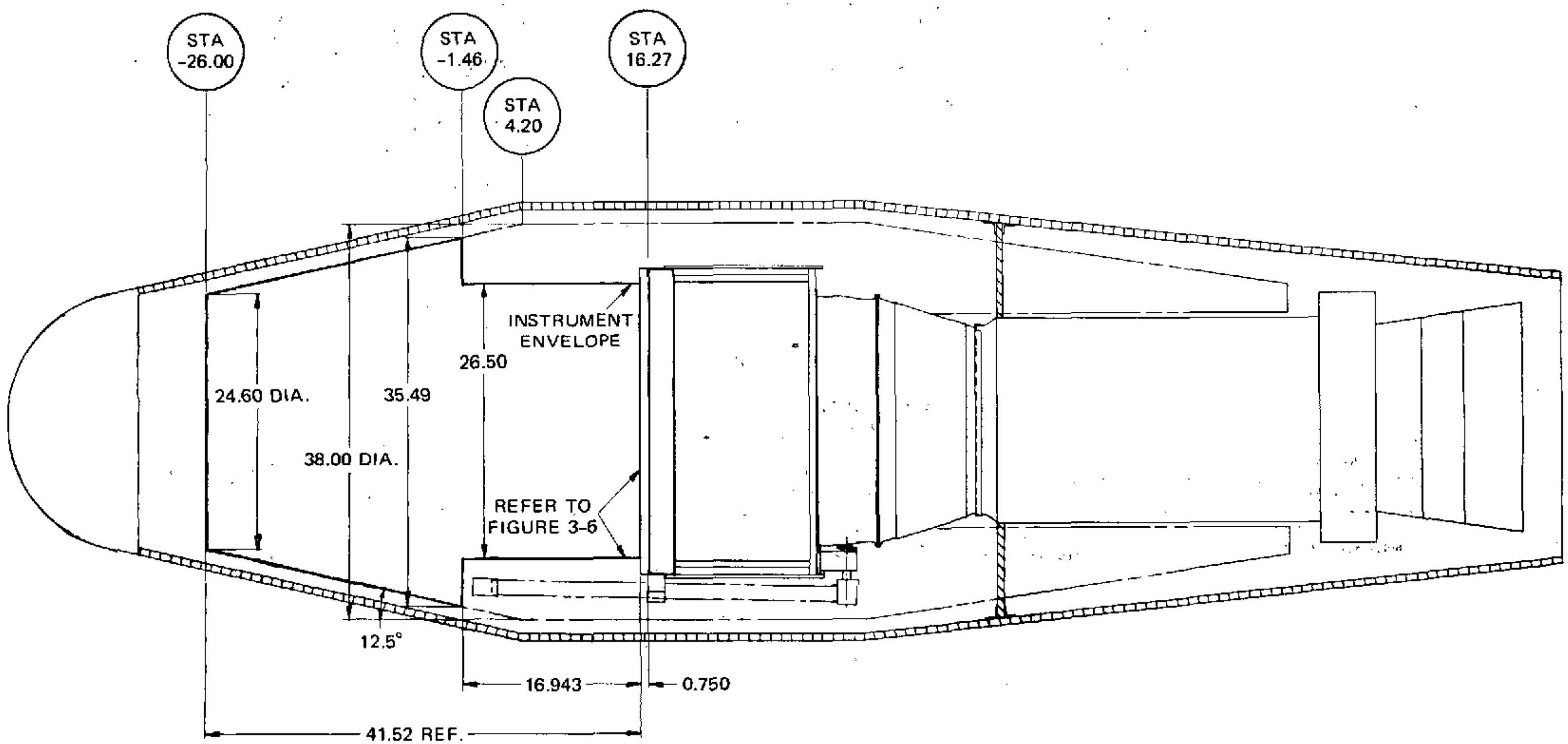


Figure 3-3. AEM in Scout Fairing

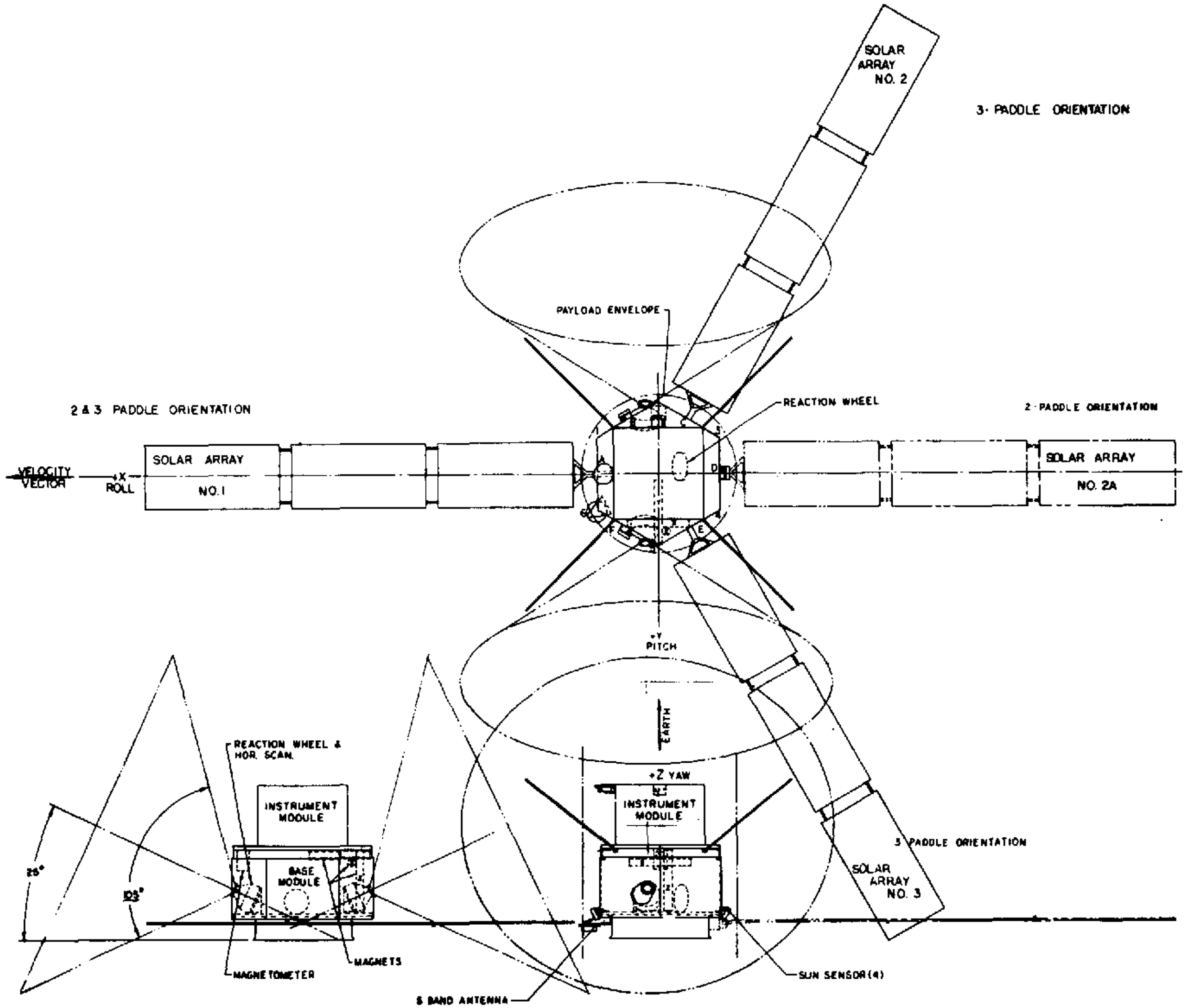


Figure 3-4. Spacecraft Layout

Table 3-3

Typical Shape Parameter Study

No. of Sides	Platform Mounting Area cm^2 ($in.^2$)	Maximum Solar Paddle Width (in.)	Maximum Paddle Folds	Paddle Area* One Side One Paddle cm^2 ($in.^2$)	Optimum Product ($x10^6$) (Mounting Area) x (Paddle Area)
4	3875 (600)	19	3	12135 (1881)	47.023 (1.129)
6	5000 (778)	16	3	10219 (1584)	51.095 (1.23) Max. Prod.
8	5450 (848)	11	3	7026 (1089)	38.292 (0.92)

*Typical Paddle Segment Length 84 cm (33 inches).

Briefly, the base module is a monocoque structure made from as few as six (6) different part designs (Figure 3-1). A monocoque (stressed skin) structure has been chosen because it is very efficient and leaves an uncluttered interior for subsystem placement. The primary building material is aluminum honeycomb as well as regular aluminum sheet and plate stock. Inserts are installed in the honeycomb where attachment is required.

The major structural parts are:

1. Centertube (1)
2. Skin attachment angles (6)
3. Platform (1)
4. Posts (6)
5. Skin Panels (6)
6. Strong ring (1)

3.1.1.3 Attach Fitting—A monocoque structure such as the base module is most efficient when the skin loads can be transferred directly into the launch vehicle. Therefore the newly developed 61 cm (24 inch) Scout adaptor (Figure 3-5) with its large interface diameter is ideal for the AEM structure. This

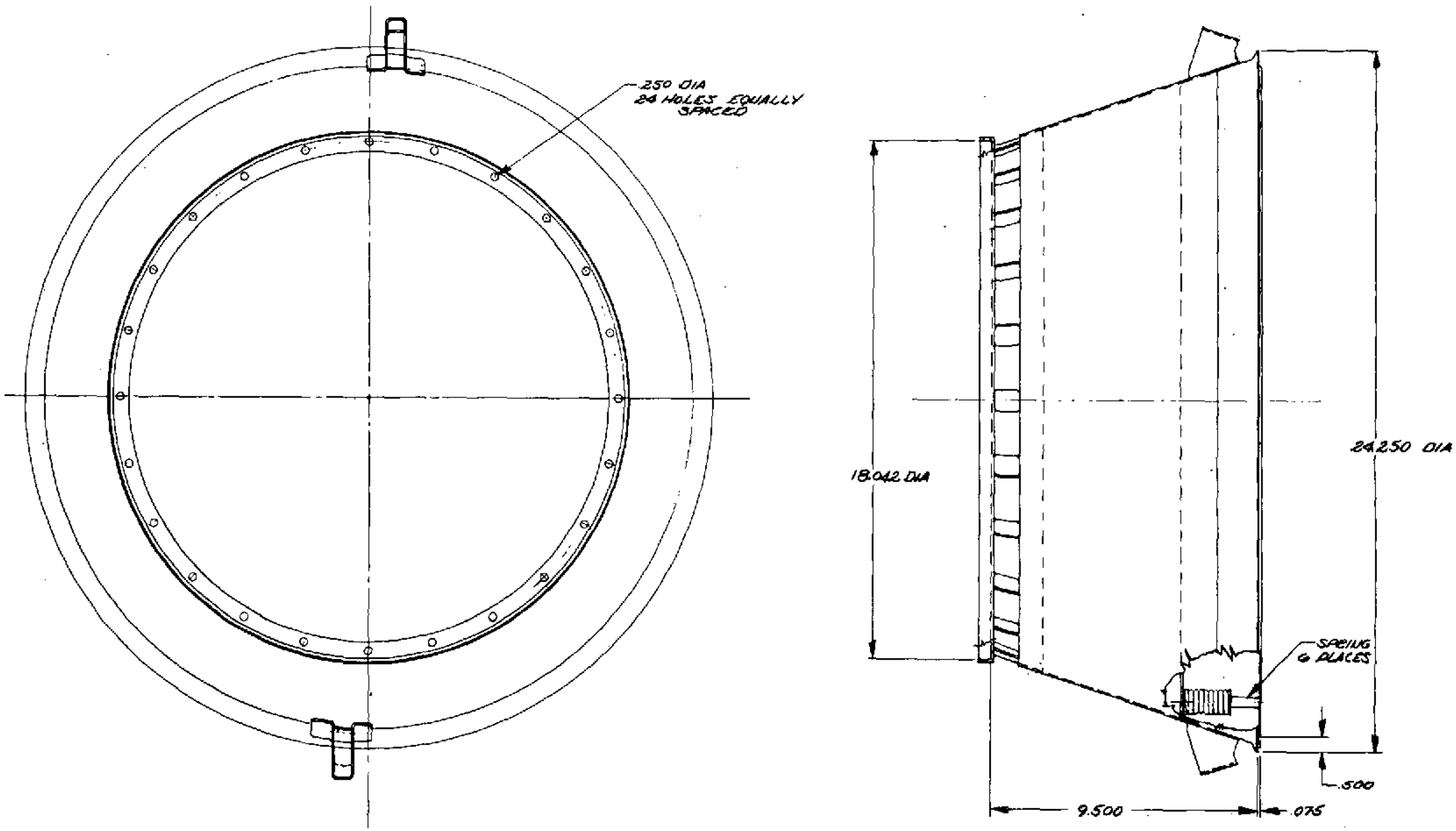


Figure 3-5. AEM/Scout Adaptor

adaptor was developed to meet the trend of larger spacecraft diameters that the 107 cm (42 inch) heatshield will accommodate.

Magnesium is used throughout the adaptor in order to keep its weight down to 4.8 kg (10.5 lbs) including the clampband and pyrotechnics.

3.1.1.4 Platform—All of the spacecraft electronic packages are mounted to this aluminum honeycomb surface because it is a primary structural element and also the thermal control surface of the base module. Platform face sheets are 0.020 inches thick and made from 2024-T3 aluminum. The core is 0.750 inches thick made of 5052-H39 aluminum with a cell width of 0.125 inch. Both the base and instrument module platforms are of the same construction.

3.1.1.5 Posts—Six aluminum sheetmetal posts perform two functions; to coupl the skin panels together in shear, and to support the instrument module in a ground handling environment while the skin panels are not in place. Removal o all six panels allows maximum accessibility to the spacecraft interior.

3.1.1.6 Strong Ring—The top strong ring can be machined from a flat plate of aluminum stock or fabricated from standard aluminum channel. This ring has four functions:

- Interface with the instrument module.
- Attachment for posts and skins.
- Impart dimensional stability and distribute loads to the skin panels.
- Provide a mechanical mounting surface for the Yo-Yo, thermal blanket, and solar panel tiedown system.

3.1.1.7 Instrument Module—The instrument module is a strong flat mounting surface that bolts to the base module through a simple interface. With this basic structure (Figure 3-6) as a starting place an instrument module can be tailored to meet a wide range of instrument requirements.

3.1.1.7.1 Design Criteria. Instrument module design criteria is influenced by the fact that many instruments are self contained units that only need a mounting surface for support. When an instrument does need a support structure, a strong flat surface is usually the ideal foundation to begin assembly.

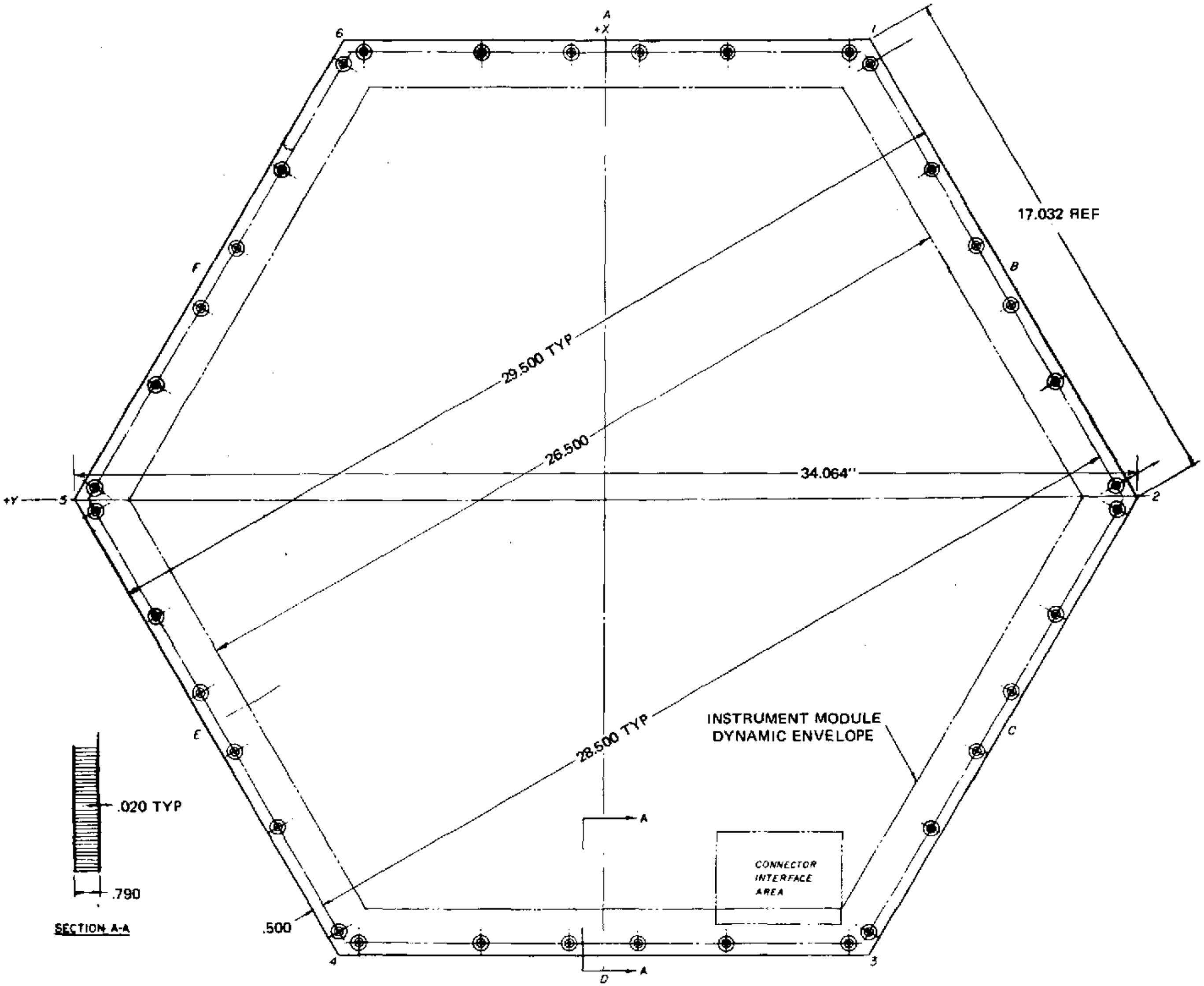


Figure 3-6. Instrument Module Deck

Other items such as antennas, booms, sensors, etc. can be mounted on the surface adjacent to the instrument or supported above the deck mounted packages.

A thermal environment can be maintained around the instrument by using multilayer blankets and thin sheets of aluminum or fiberglass as sun shields or as surfaces equipped with high emittance coatings. When necessary, active thermal control devices can be mounted to the deck to become part of the blanket/ sun shield enclosure.

The primary design goals are:

a. A simple and well defined interface with the base module;

b. Mimimum weight module structure;

c. Low center of gravity;

d. Minimal unbalance for the launch vehicle spin phase;

e. Optimal "look angle" for the instrument;

f. Adequate thermal enclosure;

g. Accessibility to the instrument.

This instrument mounting structure is made of the same aluminum honeycomb material as the base module deck and weighs 2.5 kg (5.6 lbs).

3.1.1.8 Despin System (Yo-Yo)—The Yo-Yo despin system (Figure 3-7) is similar to those systems used on previous spacecraft. The system consists of a pair of weights, cables, and release mechanisms located in geometrically opposite corners to maintain balanced torques during despin. Weight release can be initiated by either the vehicle fourth stage separation switch and delay timer or by direct command.

3.1.1.9 Solar Paddle System—The solar paddle mechanism (Figures 3-8 and 3-9) retains geometric flexibility so that two or three paddles can be used on the spacecraft and any rotation angle between ±90° from the initial deployed position can be selected. Each paddle is made of three flat segments that are self supporting during launch and deployment. Solar cells are bonded to these flat surfaces and segment hinges (Figure 3-10) couple them together forming the assembly. The main hinge (Figure 3-11) is bolted to the spacecraft just below the platform surface and attaches to the first paddle segment through a yoke.

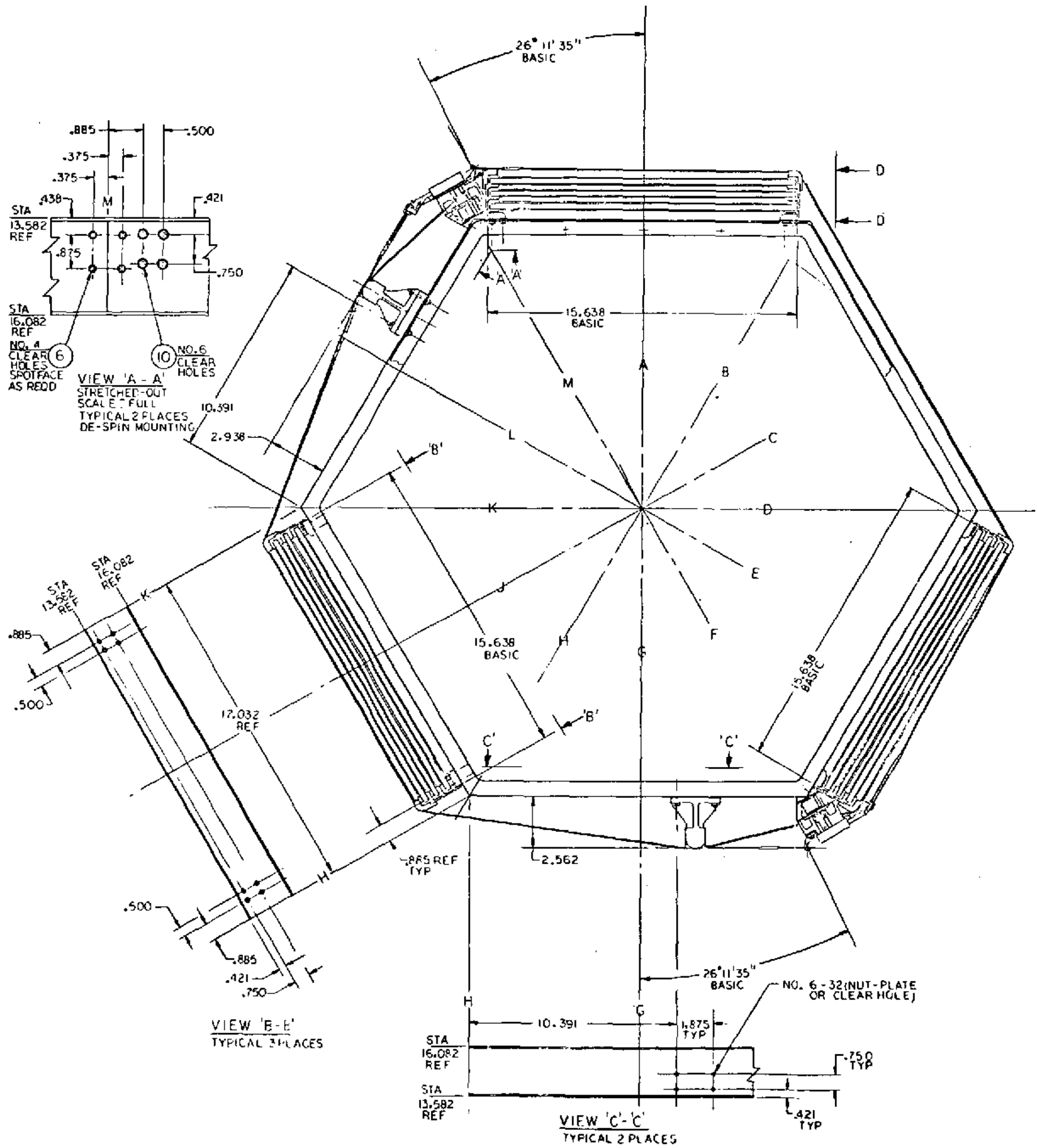


Figure 3-7. Folded Paddles and Despin System - Top View

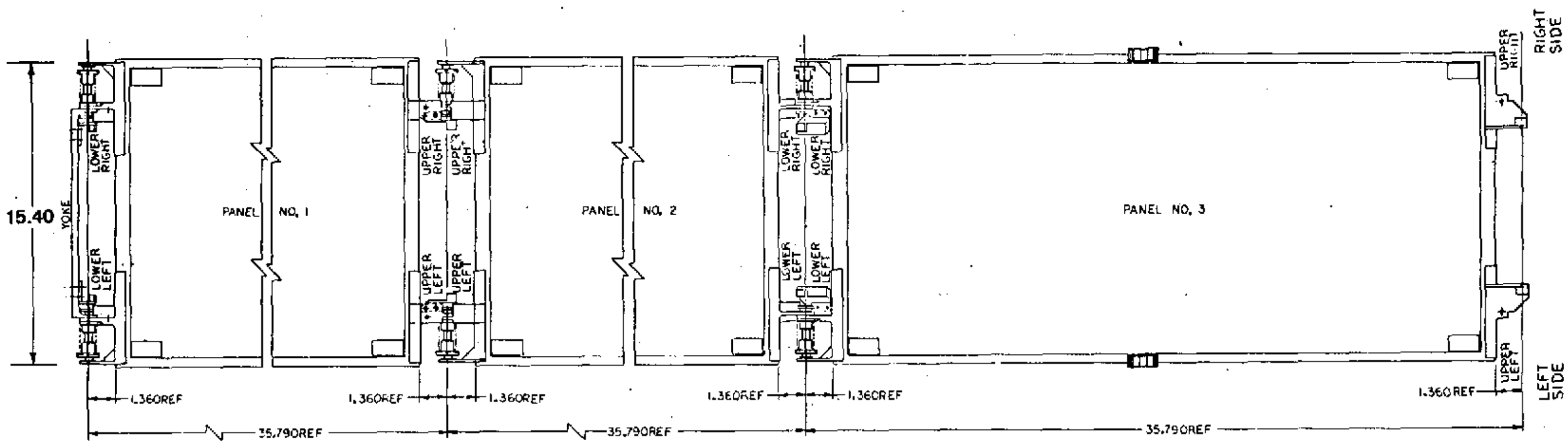


Figure 3-8. Deployed Paddle

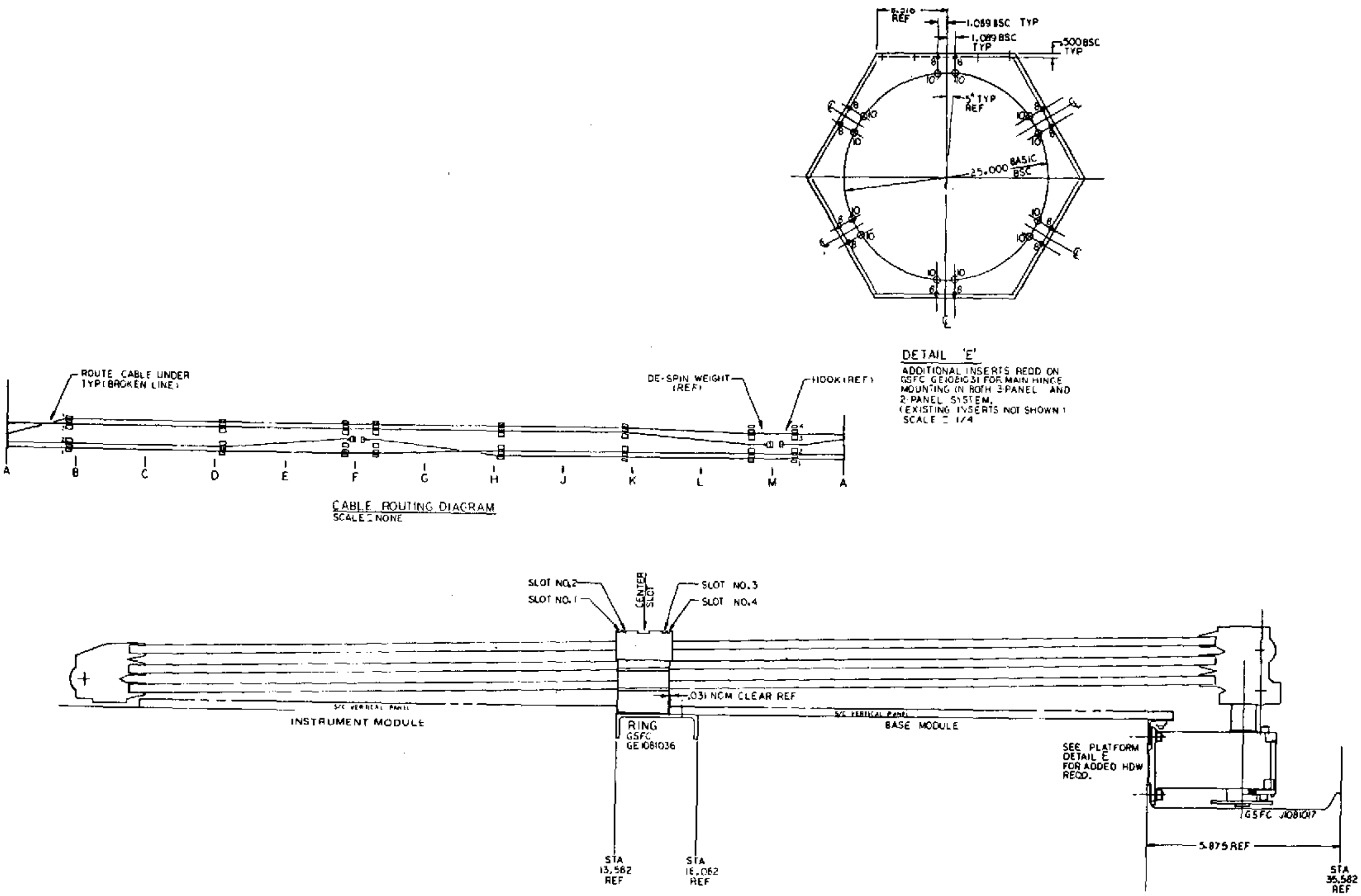


Figure 3-9. Folded Paddle

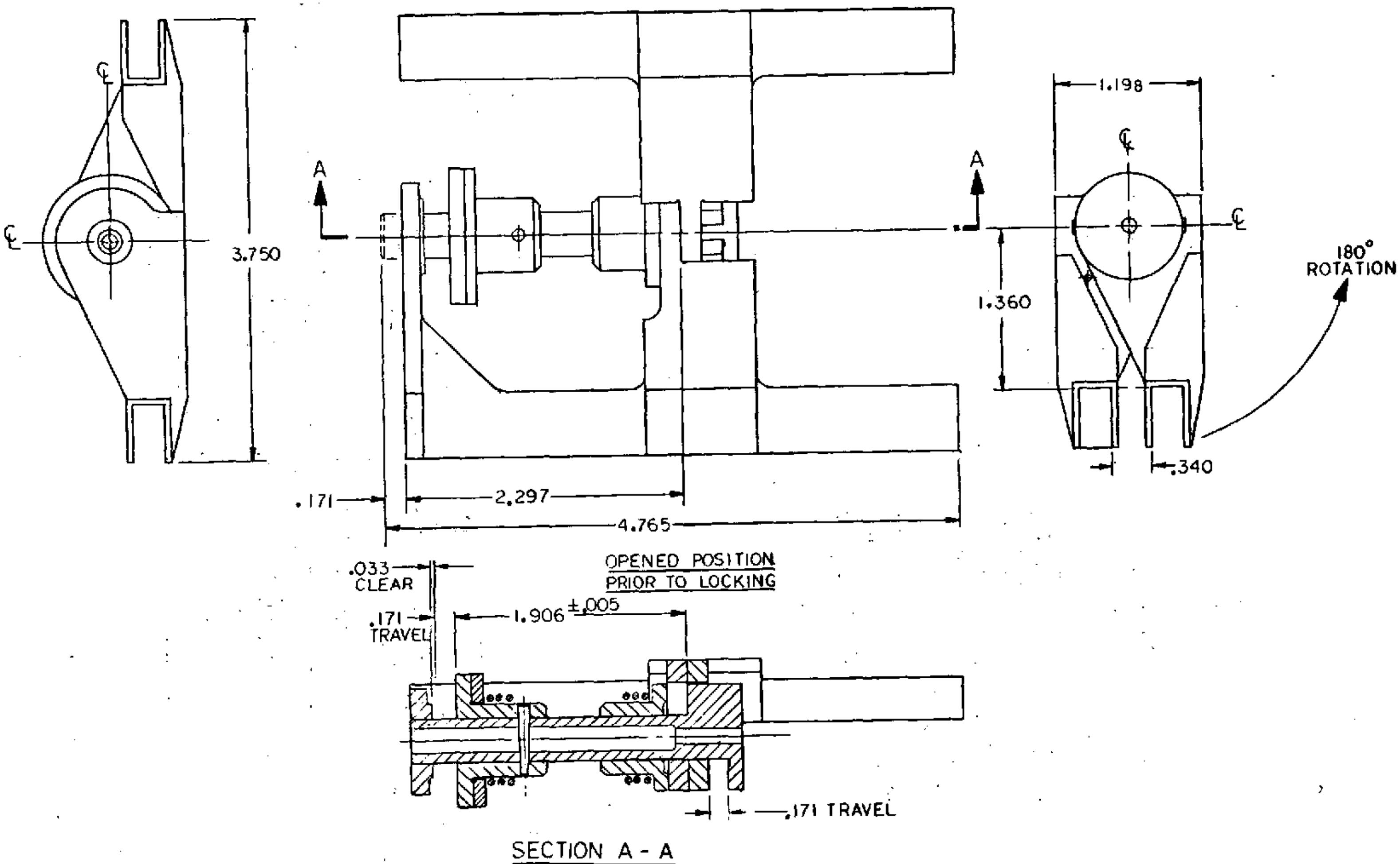


Figure 3-10. Segment Hinge

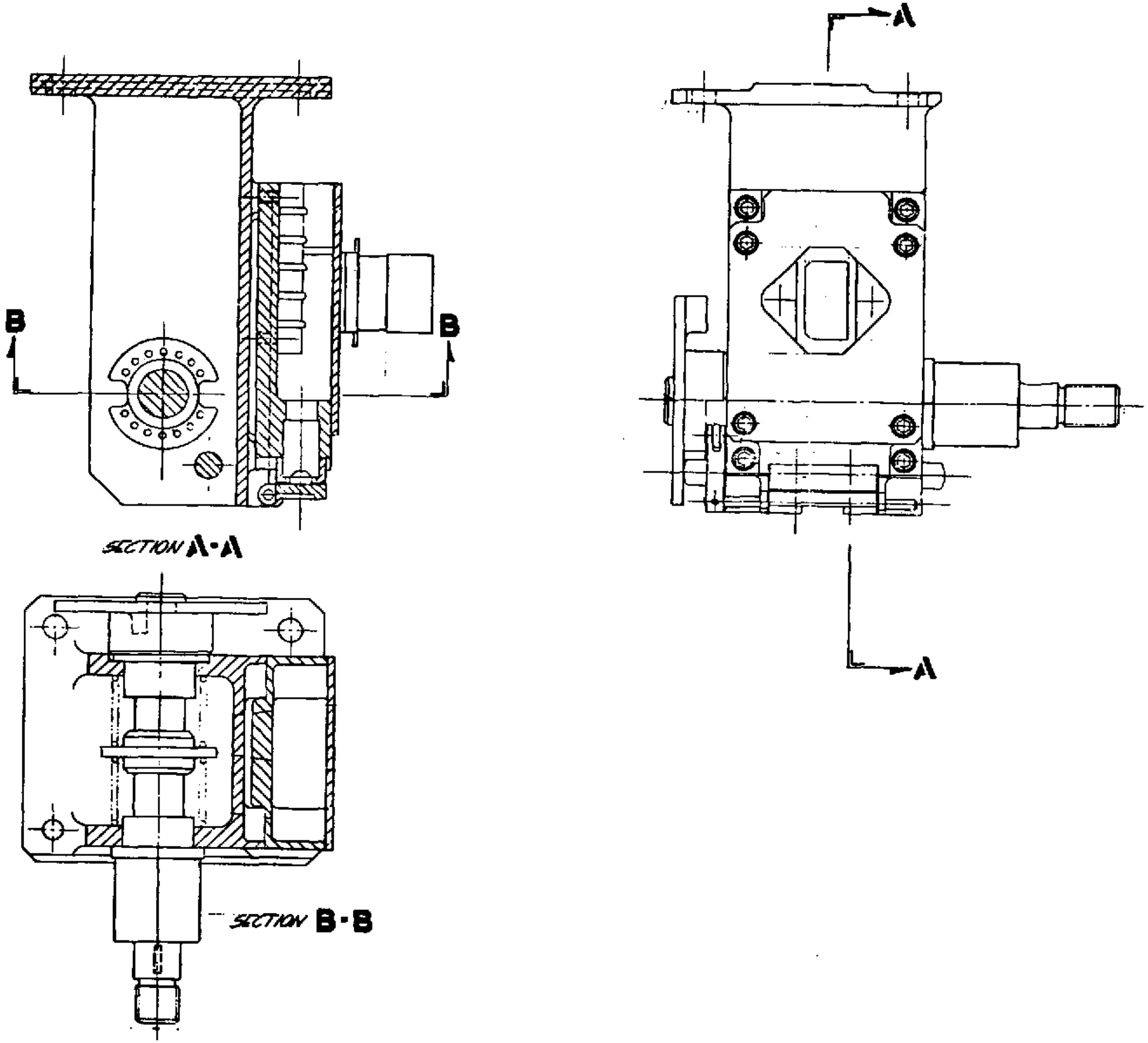


Figure 3-11. Main Hinge

Each main hinge has two dimple motors that, when fired, allow paddle rotation via redundant torsion springs. Each hinge system has redundant springs, positive stops and positive locks.

It is necessary to despin to "0" RPM before allowing paddle deployment. Otherwise, structural failure may occur from excessive lock-in loads. To reduce and distribute expected lock-in loads due to normal despin errors of ±2 RPM, each paddle segment is equipped with a deployment sequence control device.

3.2 ELECTRICAL SYSTEMS

3.2.1 General

The AEM Electrical System, shown in block form in Figure 3-12, is typical of many small spacecraft. Its standard functions include:

- Fixed solar array, the primary power source
- Ni-Cad battery for shadow and peak-load operations, such as pyrotechnic activation
- Regulated +28 VDC buss
- VHF command system
- Stored command sequence system
- VHF transmitter for spacecraft PCM telemetry
- S-band transmitter for wide-band analog or PCM data from instruments or the recorder
- Telemetry encoder
- Tape recorder
- Attitude determination and control systems
- Standard spacecraft support electronics, such as pyrotechnic sequencing and engineering instrumentation
- Mission-unique support electronics, such as clock and time code generators

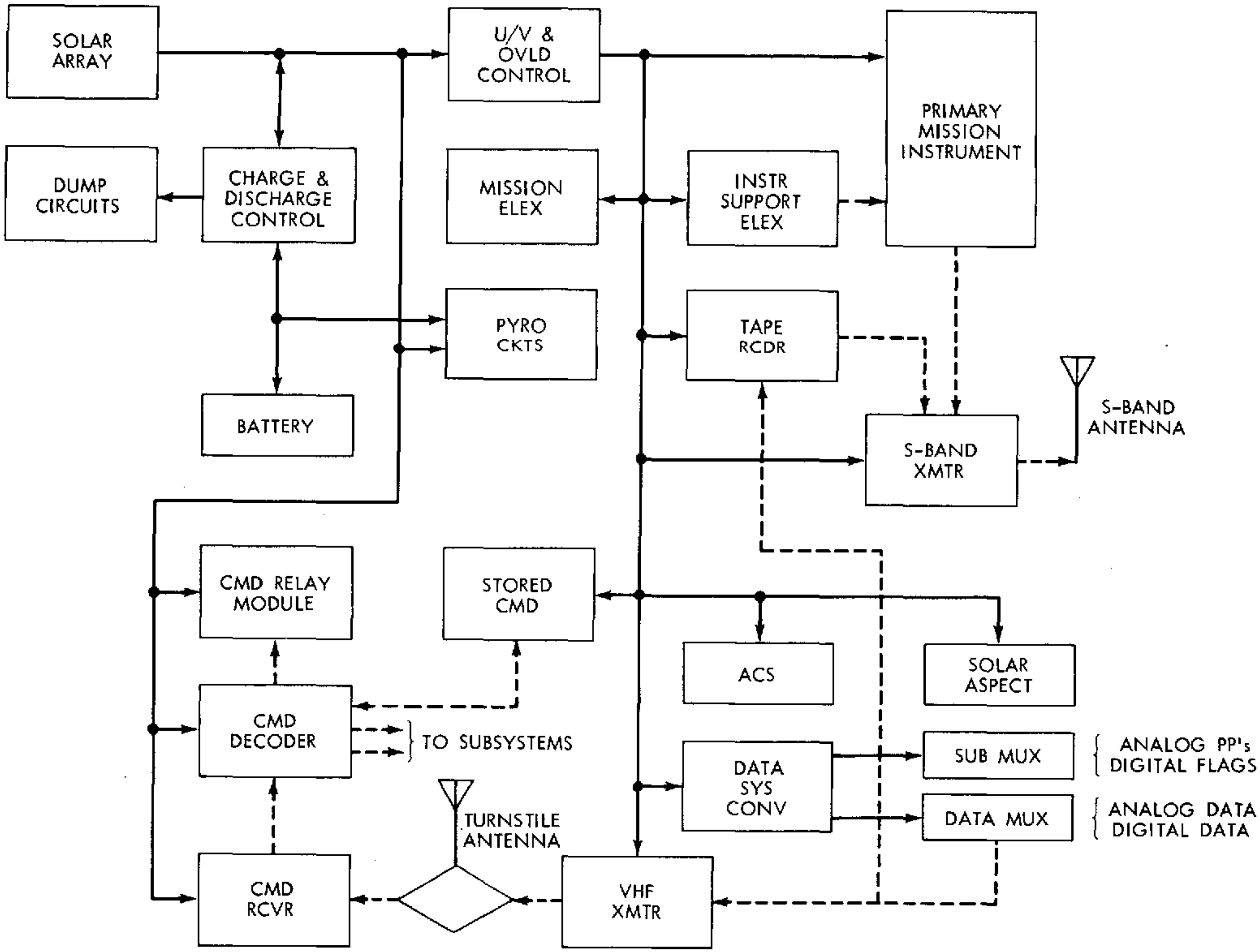


Figure 3-12. AEM Block Diagram

All AEM-user interfaces are state-of-art and straight forward, as described in the following sections.

3.2.2 Options

Typical user options include:

- Operation directly from +28 volts, conversion to other voltages, or combination thereof.
- Power/function switching internal or via the command relay module.
- Instrument telemetry generation internal or through the versatile telemetry encoder.
- Instrument sequencing, timing, or pre-positioning from internal sources from the mission support electronics, or from the command and/or encoder systems.
- Instrument data collection real time and/or remote (stored on tape recorder).

3.2.3 Grounding

The basic AEM grounding scheme is "single point," with all power and signal return leads returning to the source. All sources are then tied to a single ground reference point. The structure is not generally used as a low-frequency or DC current path.

Adaptation of the structure for implementing a "ground plane" or hybrid grounding technique is possible for special applications. However, this is not a simple task, and users must carefully weigh the trade-offs before making this a mission requirement.

3.2.4 Electromagnetic Interference (EMI)

The basic EMI specification is similar to the IUE specification. Modifications and changes to the EMI specification are incorporated for each AEM mission as required and are intended to be reasonable as well as practical.

3.3 POWER SYSTEM

The power system has been studied in depth because it affects both the spacecraft configuration and thermal design. As is the case with any power system design, certain assumptions with respect to environmental and performance parameters were made prior to the design effort. The assumptions are as follows:

a. Six month life requirement with design goal of one year.

b. Single regulated bus (+28.0 volts) with instrumenter providing all special power conditioning.

c. Consideration of all orbits practical with a Scout Launch vehicle.

d. No redundant units, minimum weight compatible with system reliability.

e. Use of conventional solar conversion/energy storage power system.

f. Maximum commonality in design for different missions.

The power system for the AEM spacecraft is a Direct Energy Transfer (DET) System. A function diagram of the system is shown in Figure 3-13. The primary source of power is the solar cell array located on paddles attached to the spacecraft base module. Power from the solar paddles is transferred directly to the spacecraft bus which is regulated at 28.0 volts ±2.0 percent. The lack of any series element between the solar array and spacecraft loads provides for a transfer of daytime power to the load at near 100 percent efficiency. Daytime power exceeding the solar array output and eclipse power is obtained from a single nickel cadmium battery through a boost regulator at 85 percent efficiency or greater. A control unit generates signals to control the shunt drive/dump circuits, battery charger, and boost regulator in the proper sequence such that the spacecraft 28.0 volt bus is operated at maximum possible system efficiency during all phases of the mission.

Figure 3-14 illustrates the average solar array power required for the DET system to maintain energy balance for various spacecraft loads up to 100 watts and for peak loads up to 200 watts and 15 minute duration in daylight. Sixty percent of that peak energy can be supported at night and still maintain energy balance. A generalized approach was taken in order to demonstrate the range of power that might be required for the various missions. The data shown for 0 Watt-Minutes (lower curve) are representative of the base module plus continuous orbital instrument power. All calculations assume energy balance is achieved each orbit, i.e., the battery will always be charged prior to entering dark. Many variations in power profiles can be accommodated, depending on various orbital parameters. An upper boundary of 120 watts (in addition to base module power) for a 15 minute maximum duration was used for the study.

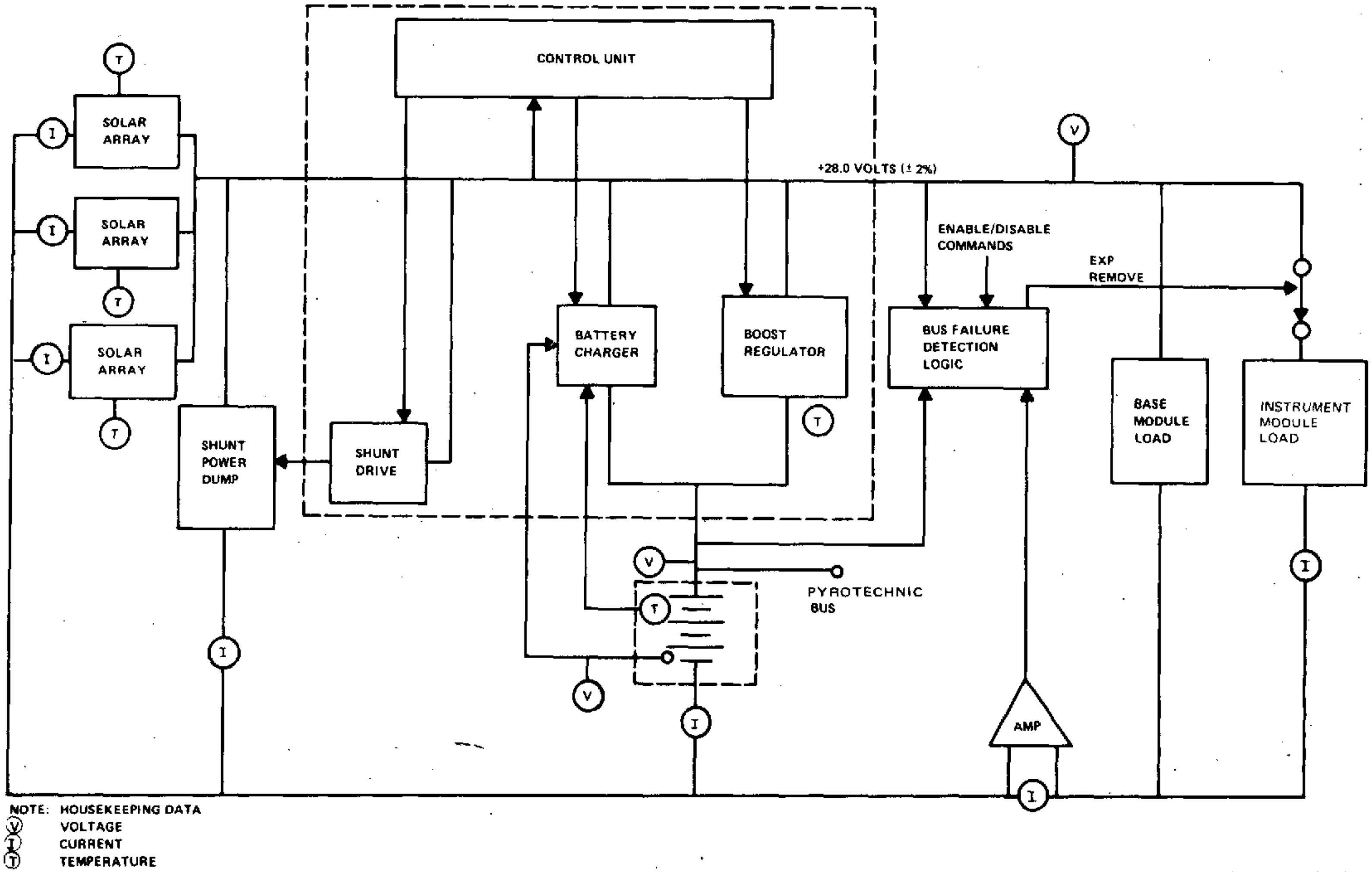


Figure 3-13. AEM Power System Functional Diagram

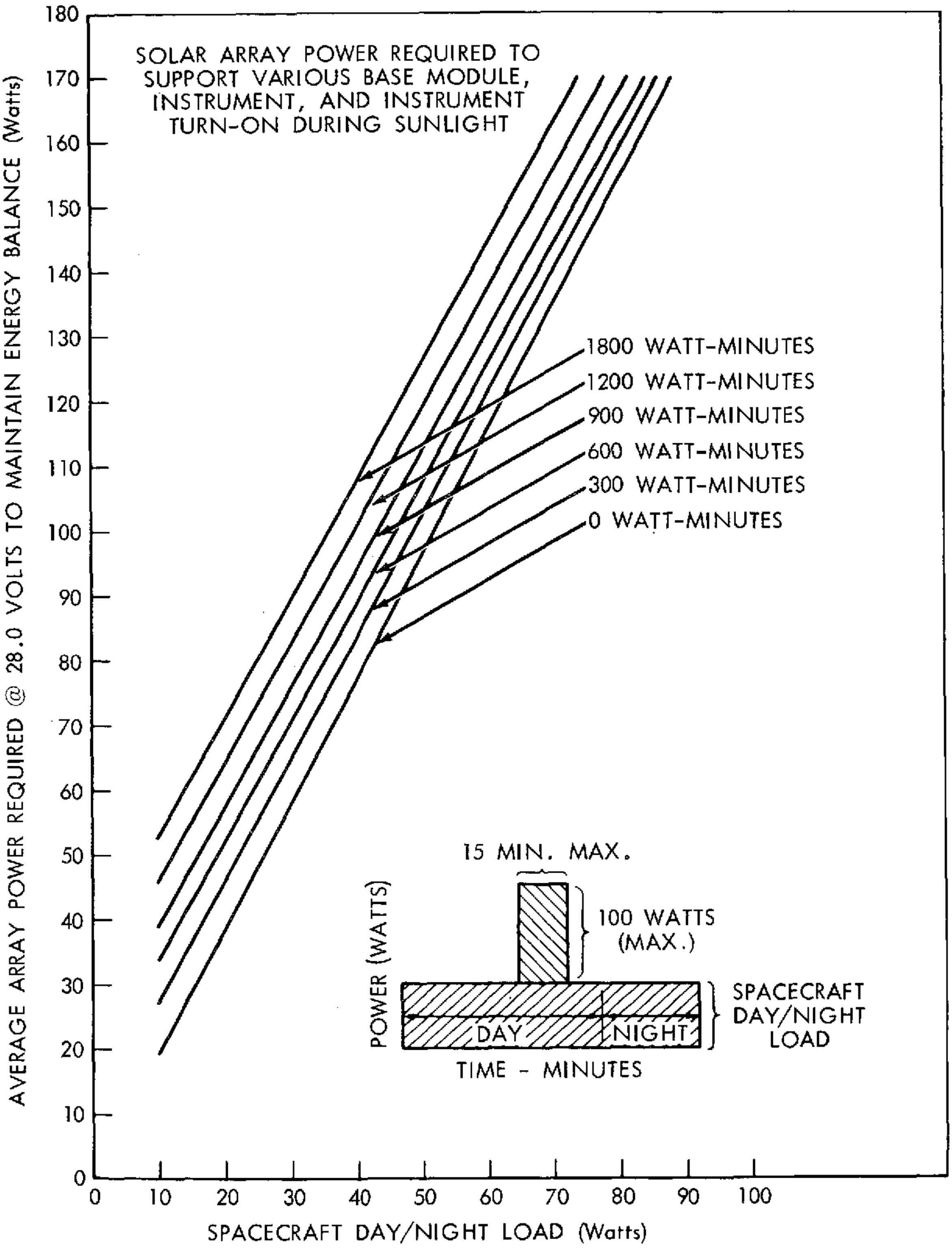


Figure 3-14. Solar Array Power Requirements (During Sunlight)

3.3.1 Battery

A six amp-hour Nickel-Cadmium battery has been elected for energy storage to supply the daytime peak power and the eclipse power for the spacecraft loads. The selection is predicated on the cycle regime imposed by near earth orbit and the demonstrated cycle capability under conditions of deep discharges.

Figure 3-15 illustrates the predicted battery depth of discharge for a 17 cell battery with various spacecraft loads up to 100 watts (0-Watt-Minutes) and for additional loads up to 120 watts for 15 minutes (1,800 Watt-Minutes) during the eclipse periods. For a six month mission, design depth of discharges of 40 percent average with peaks not to exceed 50 percent are recommended. A 17 cell battery with an estimated weight of 5.68kg (12.4lbs) meets the range of mission requirements considered.

Battery charging is accomplished with a temperature-compensated voltage limit supplemented by third electrode over-charge control. This approach has a distinctive advantage in a system with the many possible array power profiles that can be obtained for various missions. Upon acquiring sunlight (or after battery discharge periods during sunlight), battery recharge is limited only by available array power and spacecraft loads. Consequently, early in the sunlight portion of the orbit all excess power can be used to restore a large percentage of the previously used battery night power. During periods of minimum battery use (low depth of discharge) and/or extended sunlight periods, the third electrode signal will prevent overcharge (high thermal dissipation) by switching the charger to a low rate charge mode. The advantage gained with the third electrode control is the reduction of battery thermal dissipation during the sunlight periods. The third electrode function is supplemented by a ground command to place the charger in the low current mode. Due to lack of redundancy, thermal protection features will be incorporated into the design as necessary to insure meeting mission lifetime requirements.

3.3.2 Power Supply Electronics (PSE)

The PSE is a subassembly of the DET power system and consists of a control unit, shunt drive/dump circuit, boost regulator battery charger (Figure 3-13) and adapter circuitry, which adapts these functions to the particular mission. Each component is slaved to the control unit which regulates the spacecraft bus at 28.0 ±2 percent by operating the components in the PSE in the required sequence. A deadband of ±0.5 percent exist within the ±2 percent regulation such that when spacecraft power required is equal to the solar array output, all components are off. If the bus voltage increases above +0.5 percent, the control unit enables the battery charger and turns on the shunt drive/dump circuits to dump any excess power; if the bus voltage falls below -0.5 percent, the control

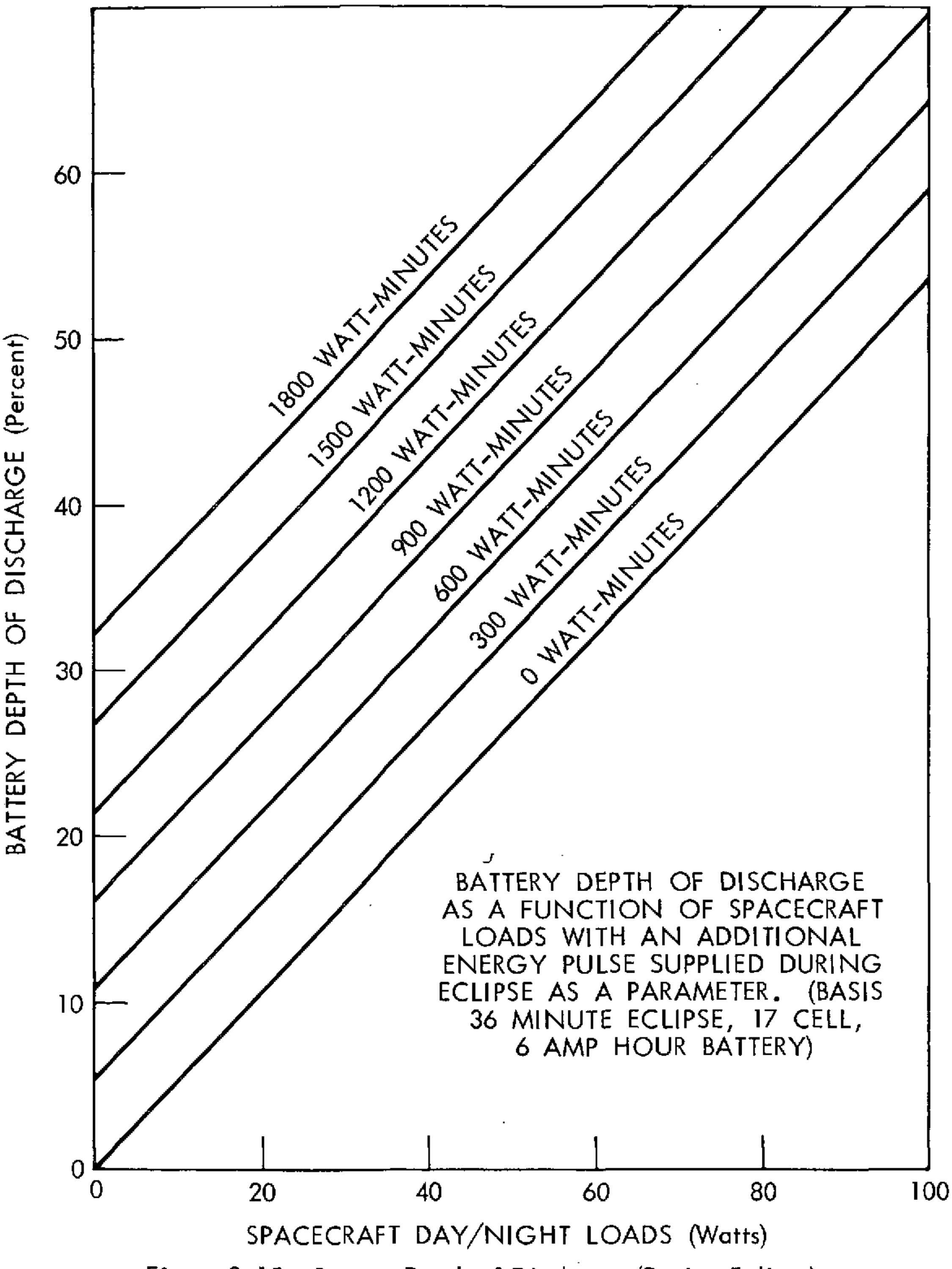


Figure 3-15. Battery Depth of Discharge (During Eclipse)

unit turns on the boost regulator to maintain the bus at 28.0 volts. Safeguards are provided to assure that components do not interact, i.e., the battery charger and shunt drive can never be in the active mode when the boost regulator is supplying power to the bus. The electronics is in two modules, the power module and the mission adapter module.

3.3.2.1 Shunt Drive/Dump Circuit—The shunt circuit consists of drivers and dump circuits capable of dissipating all excess power from the solar array. Each dump circuit is designed to dissipate approximately 7 percent of the total array power and to operate in sequence. The exact number of shunts required depends on the array capability for the various missions and the degree of redundancy required. The advantage of this approach is two fold: (1) it provides minimum instantaneous power dissipation in the dump transistors, consequently all power transistors can be centrally located in the PSE box; (2) the location of dump resistors (largest dissipating elements of the shunt circuit) can be determined primarily by thermal considerations.

3.3.2.2 Battery Charger—The battery charger is a pulse width modulator (PWM) type to minimize power losses. Current limiting is incorporated to the extent that prevents the battery from pulling the 28.0 volt regulated bus out of regulation. The current limit is determined by the control unit which allows the battery charger to pass all excess array power to the battery when it can accept the charge. Battery voltage control is temperature biased by a temperature sensitive element located in the battery. A threshold detector determines when the battery third electrode voltage exceeds a pre-selected value and switches the charger to a constant current mode. The constant current mode can also be selected by ground command.

3.3.2.3 Boost Regulator—The boost regulator supplies battery power to the spacecraft when the load exceeds the solar array output or during eclipse. The regulator is designed to handle continuous and peak power loads. The basic regulator employs a power inverter paralleled with the battery to provide a voltage boost. Since the inverter is required to handle only the boost power and not the entire load power, typical conversion efficiency of 90 percent can be anticipated for the load range projected.

3.3.3 Solar Array

Due to the wide range of solar aspect angles anticipated for various AEM missions, a parametric study was initiated to define a solar array configuration to provide the greatest orbital average power for a variety of missions. Specifically, the configurations considered are as follows:

Case	Configurations
1	Two paddles, (pointed in velocity vector) twilight to noon orbit, instrument module earth pointing, cells one side of paddles.
2	Two paddles, (pointed in velocity vector) twilight to noon orbit, instrument module earth pointing, 1 (Note: this case a pre-launch option of Case 1) cells one side of paddle.
3	Three paddles (one paddle in velocity vector) instrument module earth pointing, cells both sides of two paddles, one side of paddle in velocity vector. Orbits from noon to ±45 degrees (±3 hours).

Specific guidelines used as basis for the array analysis are as follows:

a. Both two paddle and three paddle configurations are possible, depending on solar aspect angle.

b. Paddle pitch angle (paddle displacement) is optimized for a particular mission prior to launch.

c. Stowed array configuration meets envelope requirements of the vehicle.

d. Obtain maximum solar cell effective area 35.6 cm x 78.9 cm (14" x 31") consistent with AEM-Scout envelope with minimum consideration for weight/power optimization.

e. Design for minimum complexity of deployment mechanism.

f. Power calculations are based on 0.0097 watts/cm^2 (9 watts/ft^2). Degradation not considered due to variation in missions considered.

g. No consideration given to series/parallel arrangement of solar cells or array temperature.

CASE 1: Two paddles were considered (cells on one side only) since there are some obvious advantages for this configuration. For the twilight orbit with the instrument module earth pointed, the effective paddle area is two. As illustrated in Figure 3-16 the day one power output is approximately 170 watts and decreases with the cosine of the sun angle as the orbit plane changes towards a

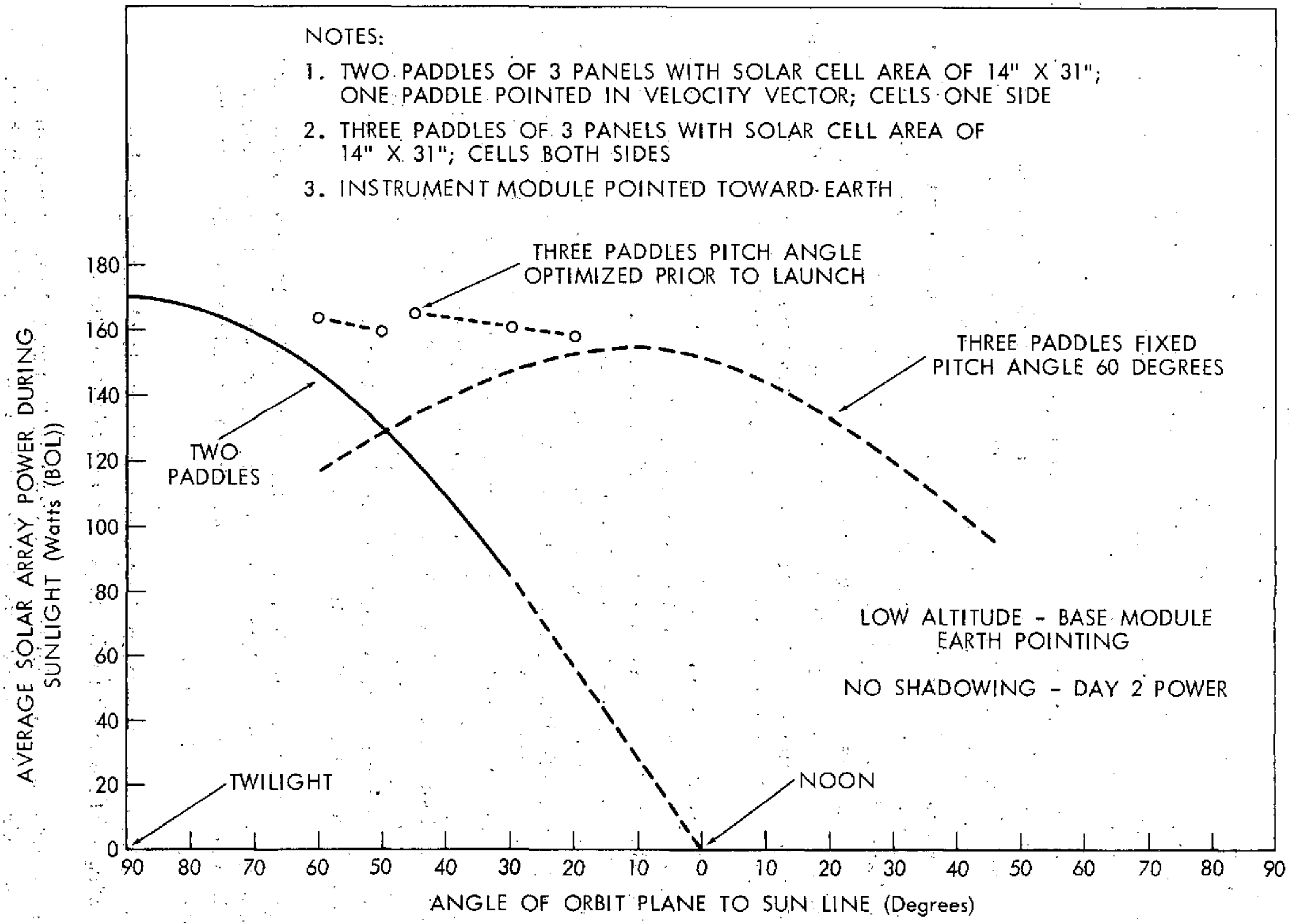


Figure 3-16. Solar Array Configuration Study

noon orbit. However, orbits up to 10AM (±60° from twilight) can be accommodated while maintaining an average solar array power of 85 watts. The effective watts per unit weight varies from a maximum of 28.5 watts/kg (12.9 watts/lb) for the twilight orbit to 14.1 watts/kg (6.4 watts/lb) for a 10AM orbit (Figure 3-17). Since solar cells are not required on both sides of the paddles, this configuration represents the greatest effective watts per unit weight of the cases considered during the study.

CASE 2: (Power output normal to sun same as Case 1): Case two is a variation of Case 1 in that the paddles are rotated (prior to launch) about the X axis by 90 degrees. This configuration was considered as a possible solution for the loss of effective paddle area in CASE 1 as the orbital plane approaches a noon orbit. It is obvious that as the spacecraft passes over the equator (noon orbit) the effective paddle area is two with a power output of 170 watts and decreases as a function of the cosine of the sun aspect angle towards both the North and South poles. Consequently the average effective paddle area for an orbit is 1.47 and provides an average, effective power of 125 watts per orbit. For this calculation it was assumed that the array could not maintain an effective operating voltage (±28.0) for sun aspect angles greater than 75 degrees.

While the average effective paddle area appears to be good, the problem encountered is due to the fact that the array is illuminated for less than 50% of the total orbit period. Consequently, the orbital average power is considerably lower than if the array was illuminated throughout the entire sun period. Figure 3-18 illustrates for CASE 2 the results of calculation for energy balance and battery depth-of-discharge for various continuous base module and instrument module loads. Due to the decrease in effective sun time (38 minutes used for calculations), the required average power to maintain energy balance is approximately twice that required for conditions where the array is illuminated for the full sun light period. For example, the average array power required for the twilight orbit (CASE 1) for a 60 watt spacecraft load is approximately 95 watts. The same array configured for the noon orbit requires approximately 200 watts to support a 60 watt spacecraft load.

CASE 3: A three paddle design with the spacecraft instrument module earth pointing and with one paddle in the velocity vector was analyzed. The paddle in the velocity vector has cells on one side only, while the other two paddles have cells on both sides.

The omission of cells from the one paddle is due to shadowing on the earth viewing surface during a large percentage of the sunlight.

Power for CASE 3 configuration is shown in Figure 3-16 for a noon to ±45 degrees. The maximum day one power obtained for a fixed paddle pitch angle of

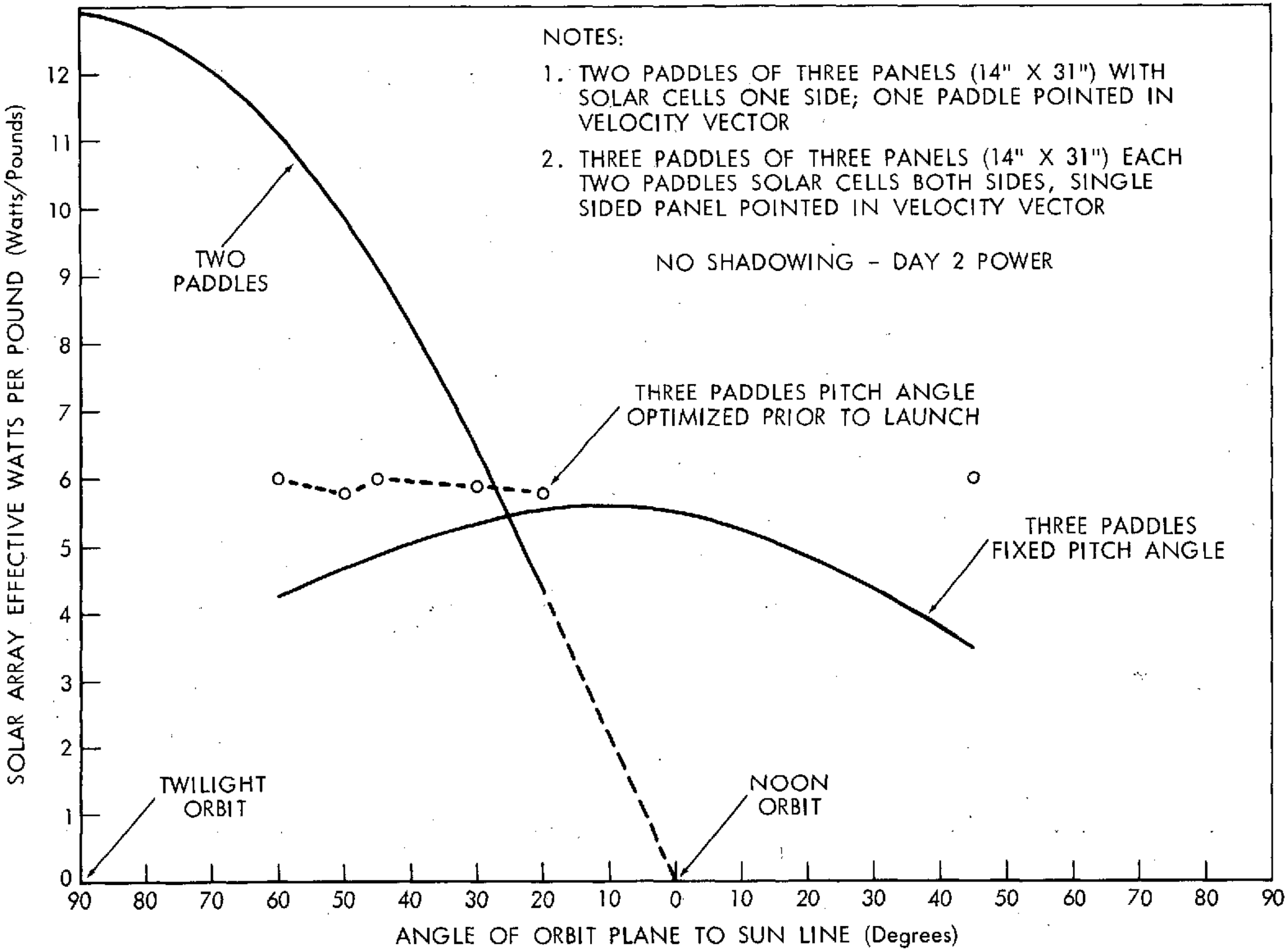


Figure 3-17. Solar Array Configuration Study

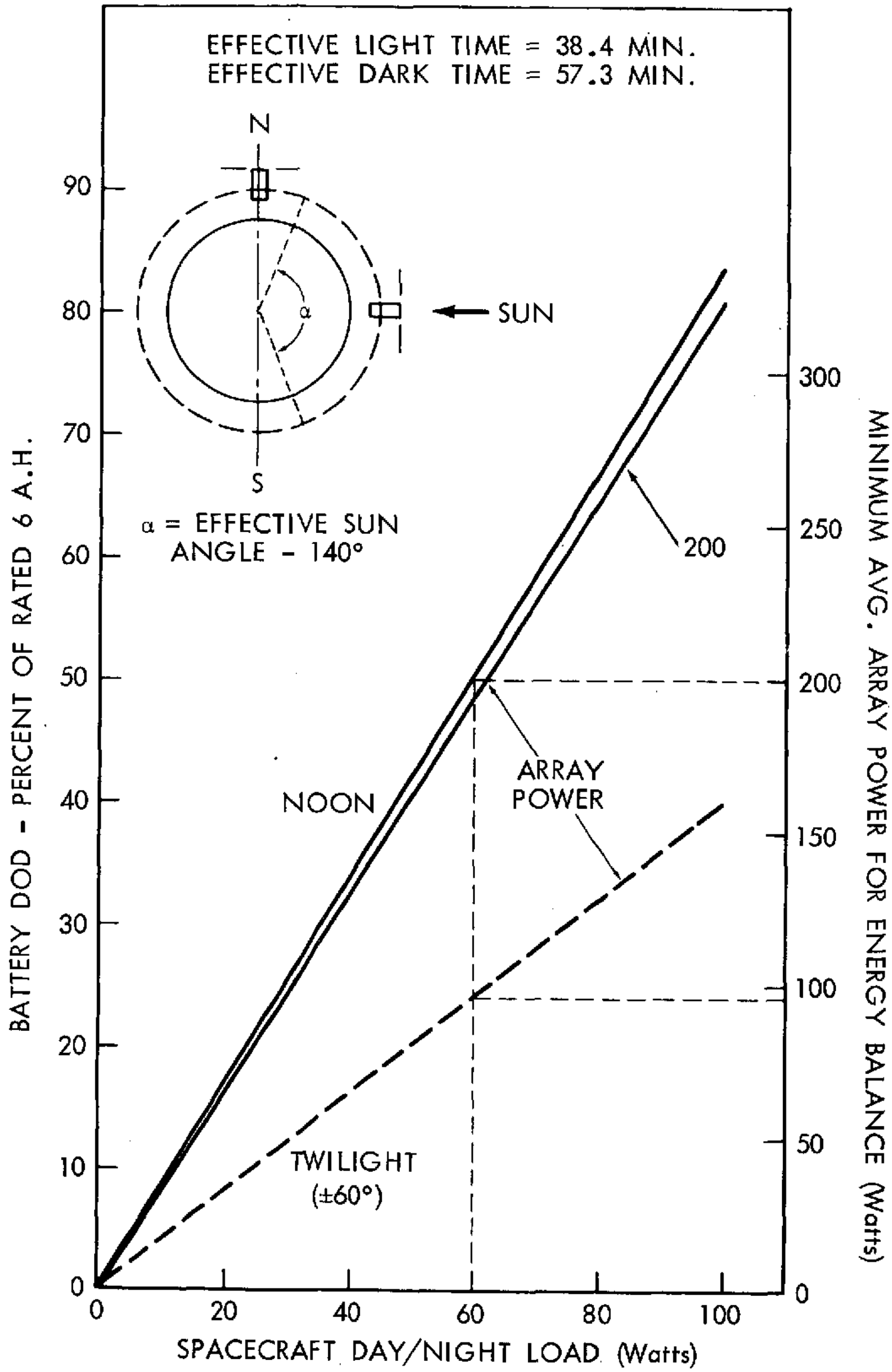


Figure 3-18. Noon Orbit Two Paddle Configuration

60 degrees is 155 watts. It is noted that the power output is not symmetrical about a noon orbit (0 degrees). This is due to the initial definition of the pitch angles for the two trailing paddles. The power profile illustrated for the various orbit plane to sun line angles can be reversed by reversing the pitches of the two trailing paddles. A typical source profile for the noon orbit is illustrated in Figure 3-19.

The maximum power from the solar array can be maximized for a specific orbit-plane to sun-line angle by optimizing the paddle pitch angle. The selection of the pitch angle is made prior to launch as determined by specific orbital parameters. The results of calculations made to optimize the pitch angle are also illustrated in Figure 3-16. It should be noted that the data points represent the power available with the spacecraft in the specific orbit plane (with respect to sun line). A decrease in power similar to that shown for the 60-degree fixed pitch angle will occur as the spacecraft drifts in either direction from the position indicated.

This configuration provides up to 13.2 effective watts per kg (6 watts/pound) when the pitch angles are optimized prior to launch as shown in Figure 3-17. The estimated paddle weight is 12.5 kg (27.4 pounds) based on the solar cell arrangement described above. In adhering to the commonality of module design, cells may possibly be required on both sides of all three paddles. The total estimated weight of this configuration is 14.2 kg (31.2 pounds).

3.4 TELEMETRY

3.4.1 Data Multiplexer System (DMS)

The DMS performs the function of gathering instrument and engineering or housekeeping data from all subsystems in the spacecraft and formatting these data into a PCM serial bit stream suitable for transmission to the ground stations. The system provides all timing and control signals necessary to accomplish this task.

The DMS was designed to satisfy the commonality requirements among several different spacecraft. The resulting design uses read-only memories and variable format memories to generate the telemetry format. The contents of a memory, read out in consecutive order, controls the sequence followed by the multiplexer in the sampling of data. Thus, the DMS can be adapted to any of a variety of missions by using memories tailored to the mission requirements. The basic portion of the DMS, called the data-plexer, contains the main analog and digital multiplexers, analog-to-digital converter, spacecraft clock, and timing and control signal logic all in one box. Provision is also made for block

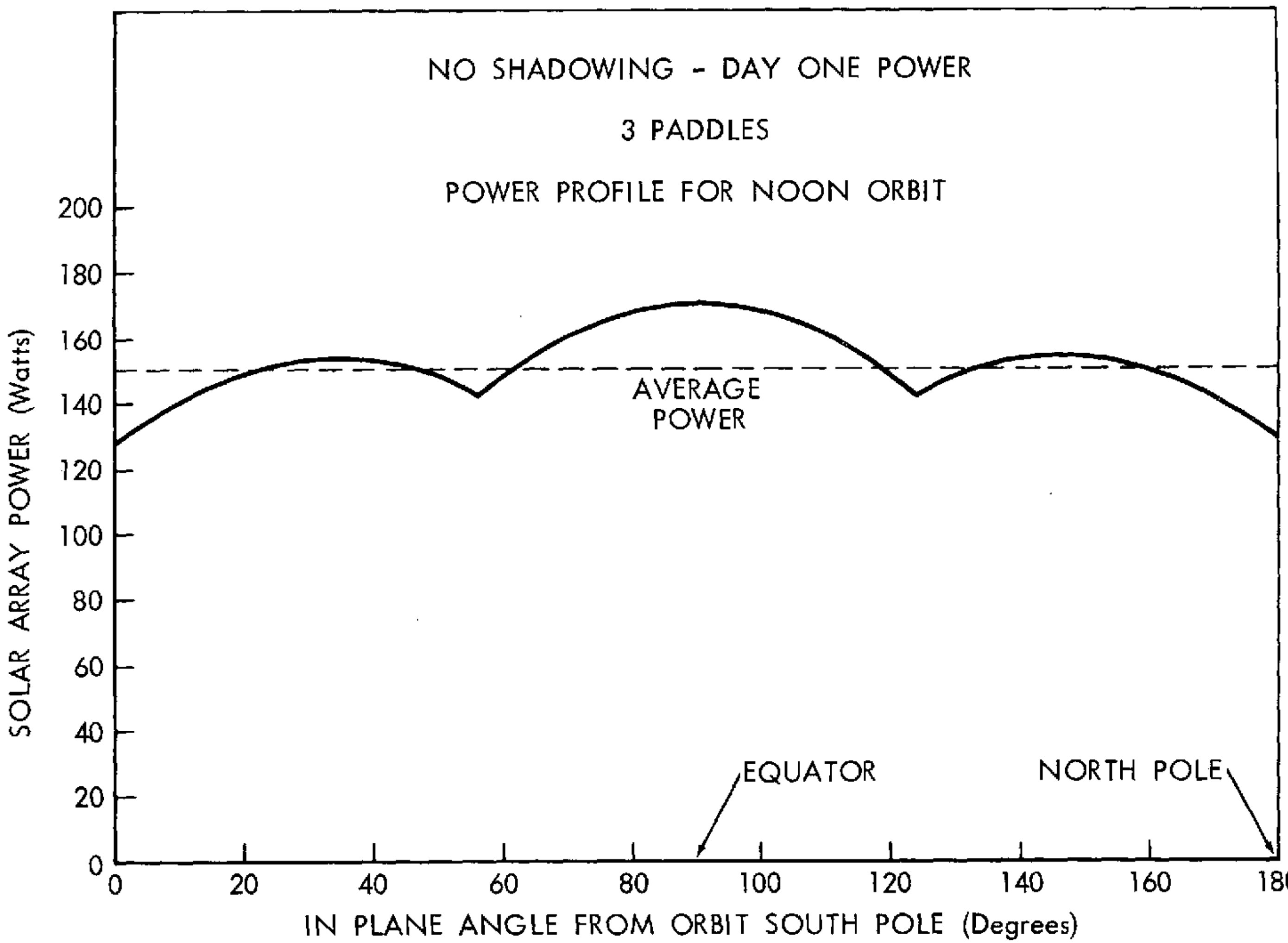


Figure 3-19. Solar Array Three Paddles Configuration Study

code or convolutional encoding where needed. One or more submultiplexer units, called Subplexers, each in its own box, can be added as required to expand the data handling capacity of the DMS. Figure 3-20 shows the standard AEM DMS.

The system is fabricated primarily from P-channel MOS circuits, with some linear integrated circuits.

The multiplex unit generates a 128-by-8 bit word minor frame having 9 fixed and 119 variable word positions. The fixed word positions include a frame synchronization pattern, sub-multiplex position, spacecraft time, etc.

The word time at which any gate is sampled is selected by one of three 128-by-8 bit format memories. Two of these are read-only fixed format telemetry memories, and the third is a variable format memory that is loaded by ground command. The format memories are selectable by command and could include (1) engineering only, (2) engineering plus instrument and (3) instrument formats. The instrument formats would probably include a minimum amount of submultiplexed engineering data for attitude determination and spacecraft health status. Exact frame formats are selected after a mission is defined.

The spacecraft sub-system and instrument data sources are assigned gate numbers. The sampling and word position within a frame of data is then controlled by the selected format memory. The incorporation of fixed formats for a given mission, or for follow-on missions, requires only the changing of the affected read only memory (ROM) flat packs within the DMS. No layout or wiring change is required. Thirty-two digital gates, 32 analog gates and at least two 32-gate sub-multiplexers may be selected by the format programs in any of the word positions not occupied by fixed words.

Timing and data rate is selectable by ground command. Different bit-rates may be selected in binary steps from 1 kbps to 80 kbps. The output logic then generates a PCM split phase signal for transmitter modulation.

A typical DMS data interface might consist of an analog data input (0-5.1 volts), a digital serial input (8-bits per word), digital word gate pulse and gated bit time clock line for readout control.

The system includes a convolutional encoder, switched into use by ground command, if one is required on a particular mission. Also, the DMS is designed to be fully compatible with an on-board computer, such as will be used on the IUE spacecraft.

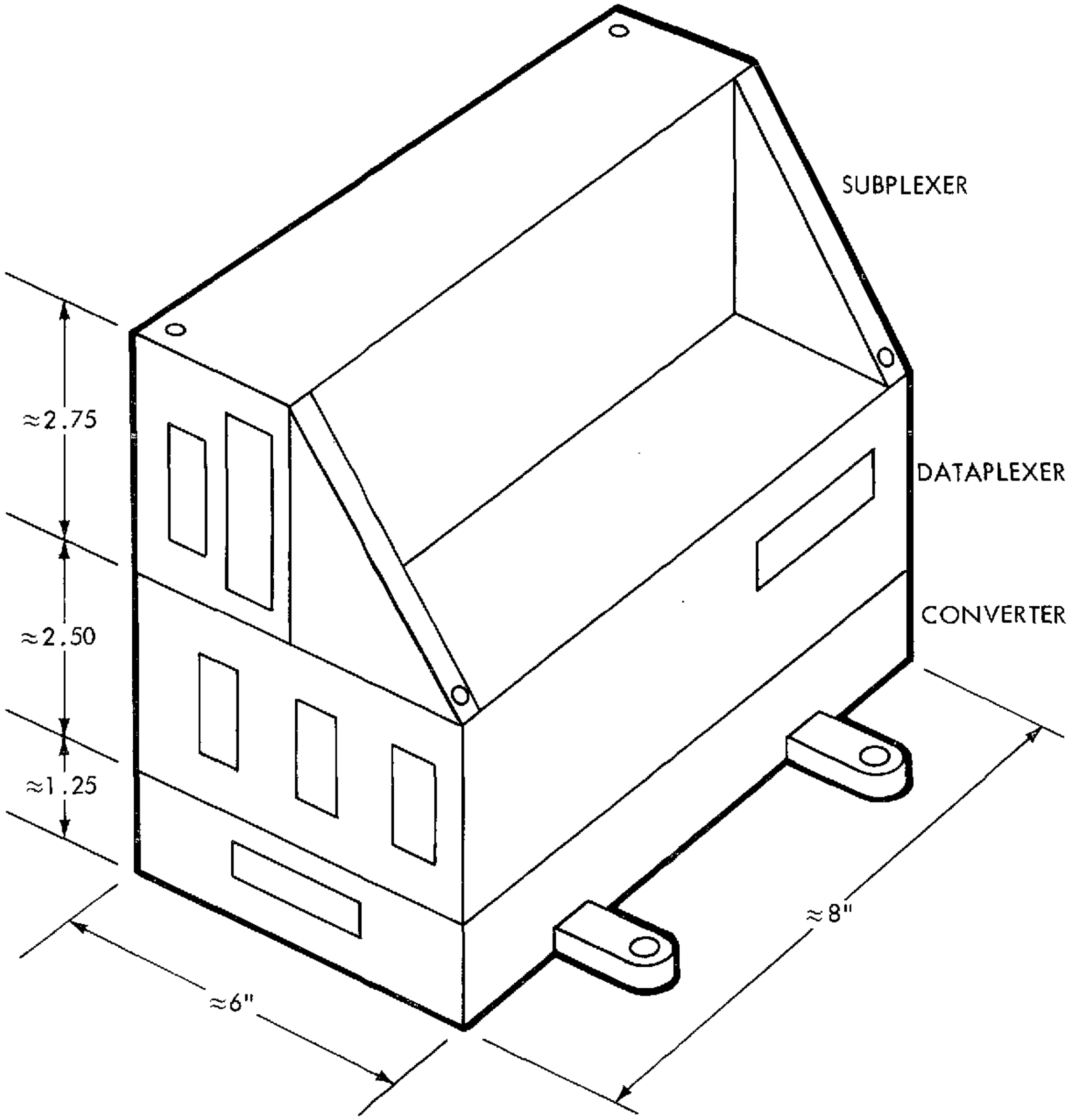


Figure 3-20. AEM Data Multiplexer System

The DMS uses medium scale integrated MOSFET circuits that have been designed, developed and tested long enough to have established a confidence in their reliability. Moreover the units will possess an inherent compatibility with the proposed encoding system which will also feature 22 pin MSI MOS units.

3.4.1.1 Data Multiplexer Subsystem

a. The Dataplexer selects digital or analog data samples in a time sequence controlled by a format memory. Each data sample is transformed into an 8-bit data word and transferred to a serial data bit stream. One complete sequence is called a minor frame and is 128 words in length. Each minor frame contains words dedicated to fixed parameters that always appear in the same locations independent of format memory. These parameters include frame sync words and information such as the contents of the frame counter, the spacecraft status bits. Figure 3-21 shows the typical telemetry minor frame format and Table 3-4 lists the parameters associated with each dedicated word. A major frame is defined as one complete sequence including all submultiplexer data samples. The major frame height is dependent on the number of times the subcom gates are sampled on the dataplexer, but would be no longer than 64 minor frames.

b. The Dataplexer includes a 32-input analog multiplexer, and a 32-input digital multiplexer. Each digital input has a companion word-gate output multiplexer and a gated shift pulse output multiplexer to control serial digital data transfer from the data source connected to the selected input. Figure 3-22 shows the timing of these signals.

   A particular input to the dataplexer is selected and the information present is transmitted through the multiplexer when the appropriate combination of enabling voltages is applied to the multiplexer's address lines. These address lines are driven by the contents read out of a location in the selected format memory. Format memories contain 128 locations of 8 bits each. Format generation is accomplished by stepping through successive memory locations and transferring data from the addressed dataplexer input at each location. Any desired sequence of dataplexer addresses may be loaded into the memory to compose a format. It should be noted that at the times dedicated to the fixed parameter words, mentioned earlier, the required information is inserted into the bit stream internally, and no input information is processed in the Dataplexer. This eliminates the possibility of having an error in format composition jeopardize the data reduction process on the ground. As indicated earlier, 4 fixed-format and 2 variable format memories are supplied for telemetry formatting.

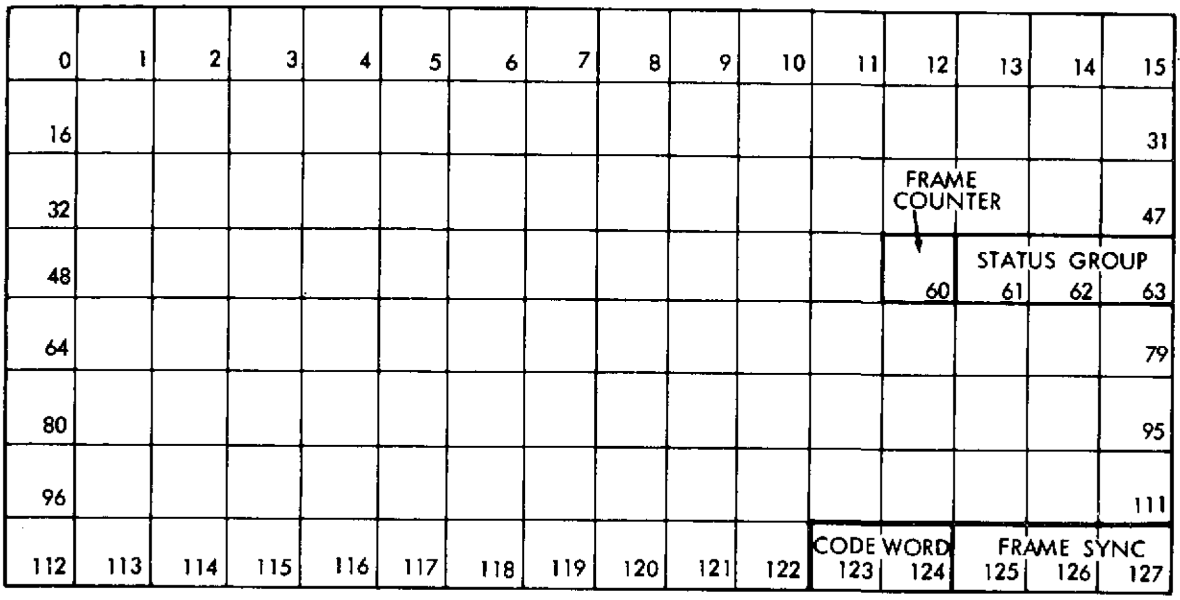


Figure 3-21. Data Multiplexer Telemetry Format

Table 3-4

Fixed Word Parameters

Word 60 (FRM CTR) 2 LSB	Word 61	Word 62	Word 63
0	Spacecraft Clock 8 MSB's	Spacecraft Clock Bits 9-16	Spacecraft Clock 8 LSB's
1	Variable Memory Readout	Command Register Bits 1-8	Command Register Bits 9-16
2	Status Register Bits 1-8	Status Register Bits 9-16	Status Register Bits 17-24
3	Variable Memory Readout	Command Bits 17-24	Execute Address Readout

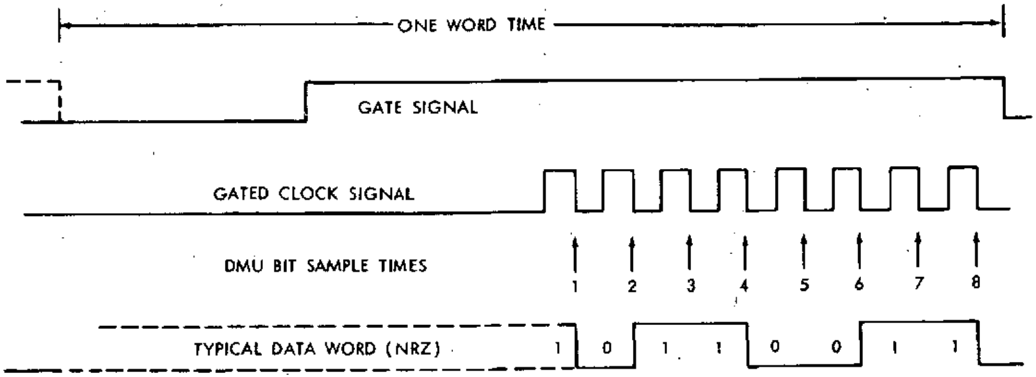


Figure 3-22. Digital Interface Signal

The ROM memories mentioned earlier are used for the fixed formats. The contents of these memories cannot be altered after fabrication, hence the term "fixed" formats. These memories are random-access, P-channel devices. They are each actually 256 word memories but a bit from the commandable memory selection decoder will permit only half to be selected at a time by putting a "1" or a "0" on the most significant memory address line. Thus, the four telemetry fixed format memories require two chips. In addition, two fixed format memories for the computer's use are supplied on a single chip. Memory location selection is controlled by a counter, incremented once per read out. Figure 3-23 shows a typical fixed format memory configuration.

Each of the two variable format memories is formed by the parallel combination of 8 serial shift registers 128 bits long. Figure 3-24 illustrates this concept. Each 128 bit serial shift register is a single P-MOS chip so that 8 chips are required per memory for a total of 16 chips for both. Memory location advance for these memories is achieved by supplying a single pulse on the shift inputs and circulating the contents. Note that this configuration is not a random-access memory.

Figure 3-25 is a simplified block diagram showing the multiplexers and format memories. It combines the concepts described above. The total number of analog and digital inputs is 64 requiring 6 address bits. The 7th address bit shown is used to enable word gate pulses when required. The 8th bit is used to generate an indirect address in the fixed formats. It is reserved for parity checking of the variable format memory output.

c. There are 32 digital data input lines available in the dataplexer. An additional 128 submultiplexed lines will be provided by the subplexer. Two control lines can be supplied to the user for each digital input, the word

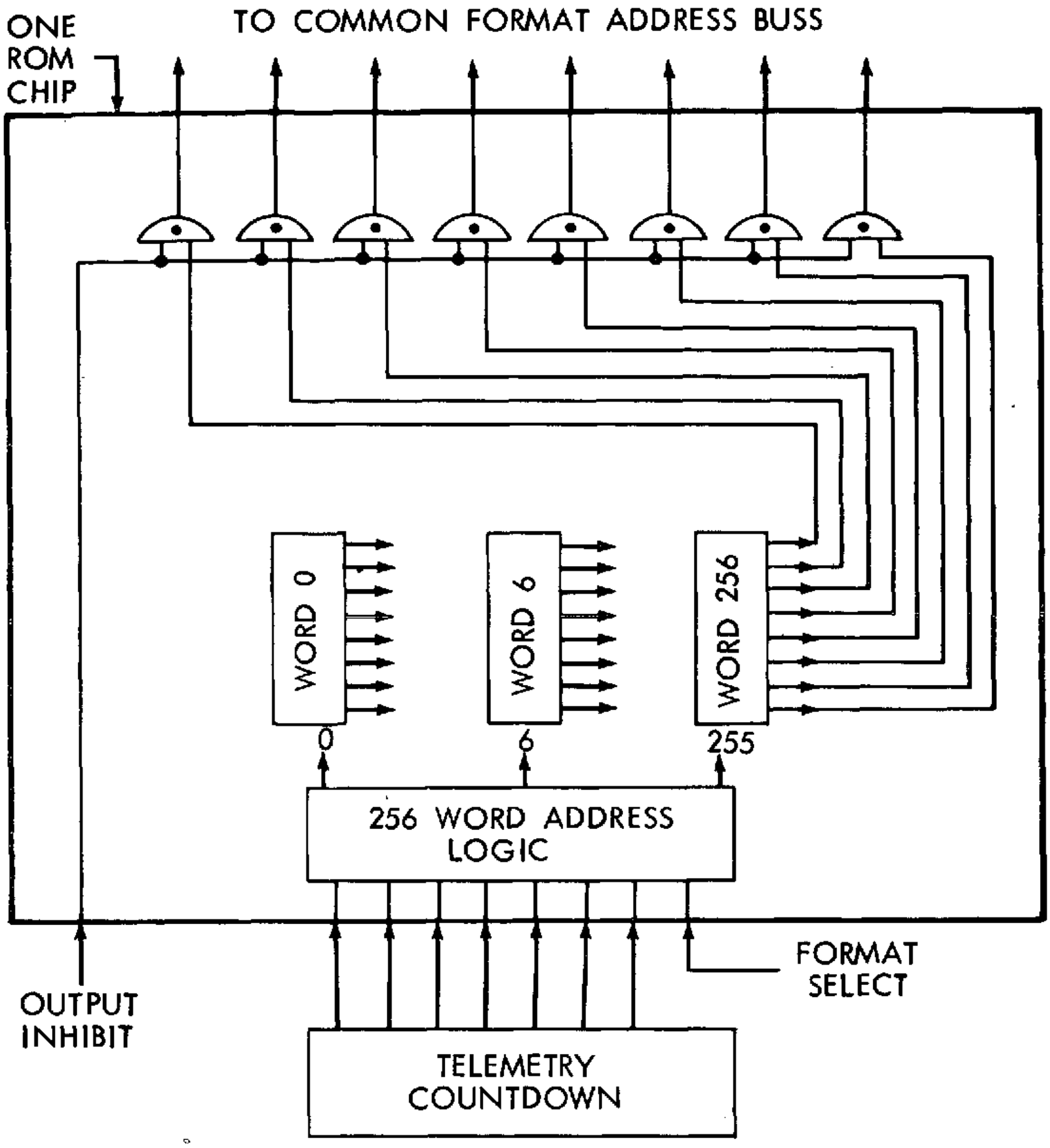


Figure 3-23. Fixed Format During Camera Readout

gate and shift pulse signals referred to in the previous sub-section and shown in Figure 3-22.

Digital output signal characteristics at 80 kHz sample rate are:

Word Gate Length	75 microseconds
Shift pulses	8 pulses per word at 160 kHz

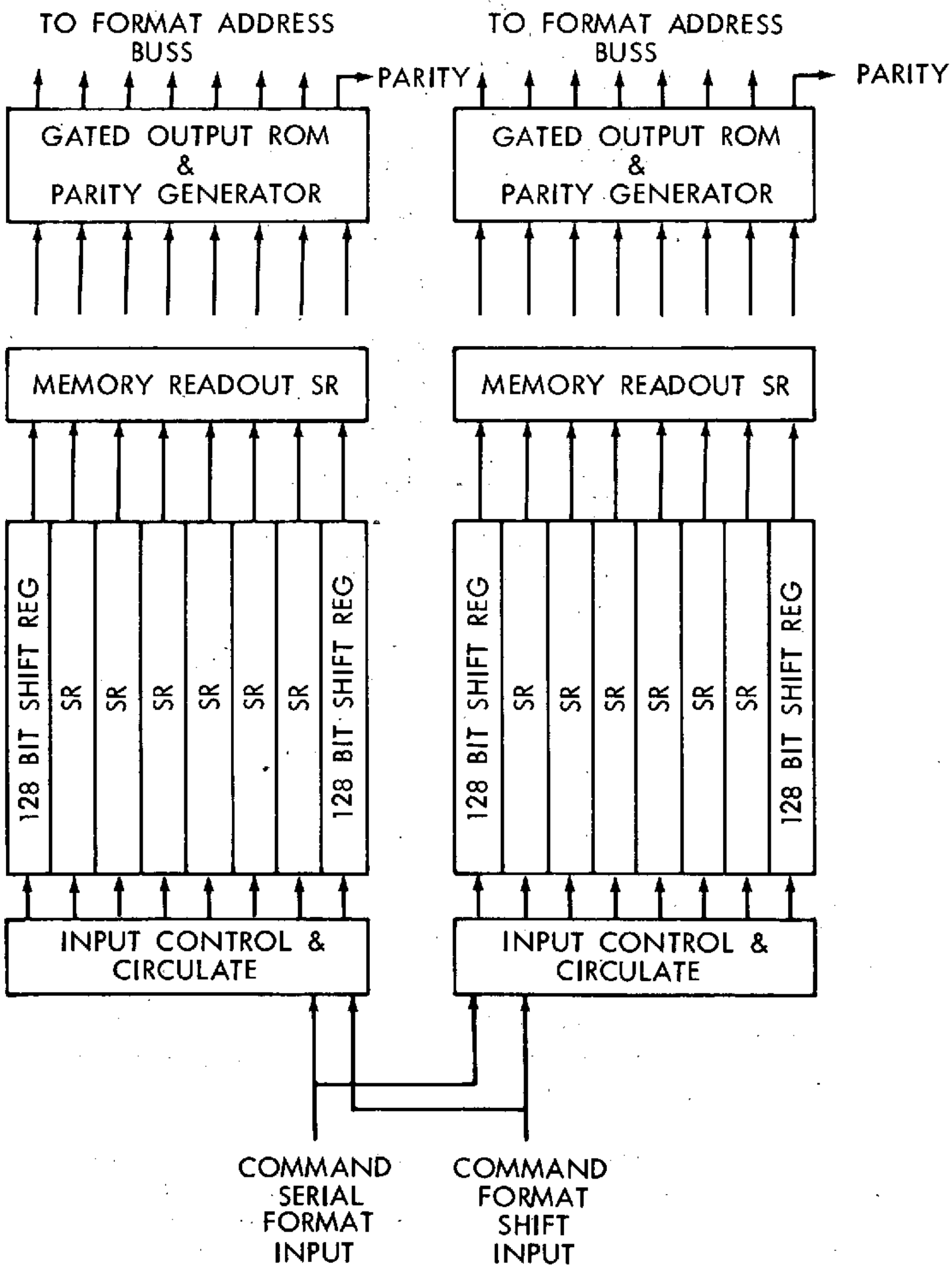


Figure 3-24. Variable Format Memories

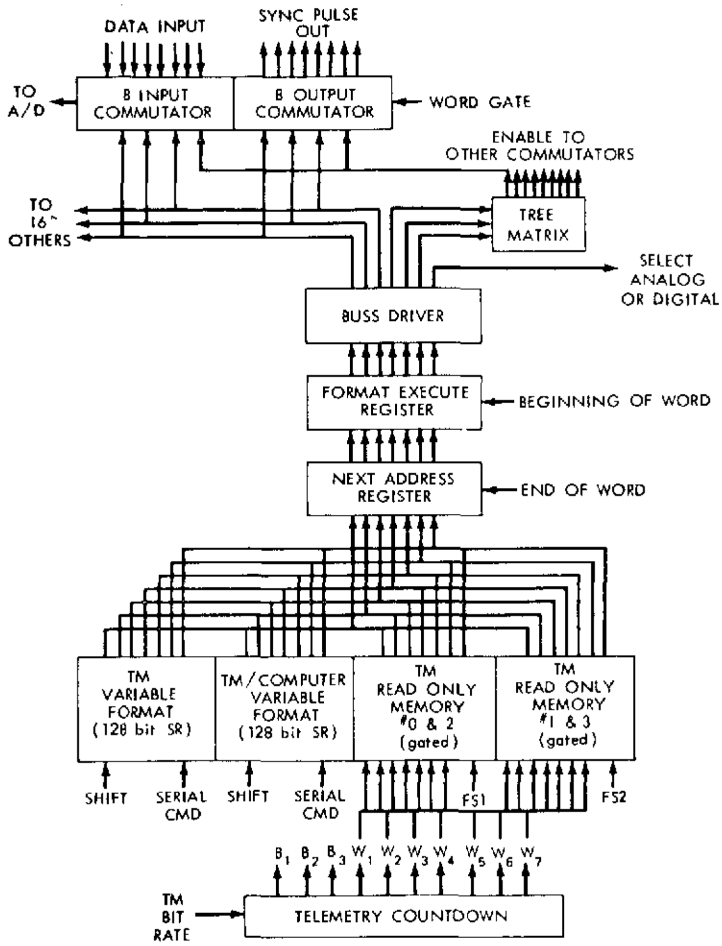


Figure 3-25. Simplified Block Diagram

A partial schematic of a typical digital output circuit is given in Figure 3-26. Rise and fall times are functions of circuit resistances at both ends and stray wiring capacitance.

Digital input signal characteristics to the multiplexer should be similar to those given for the outputs to guarantee successful data transfer. A partial schematic of the input circuit is shown in Figures 3-27 and 3-28.

d. The analog data inputs are routed to an 8-bit analog to digital (A/D) converter. The A/D converter is a successive approximation type, running at the rate up to 160,000 comparisons per second. The conversion time for a word is as short as 50 microseconds. The maximum conversion word rate is 10,000 words per second, which is identical to the maximum word transmission rate through the Dataplexer. The range of analog signal voltage input is from 0 volts to 5.10 volts. The analog input circuit is the same as the digital input circuit as shown in Figure 3-28.

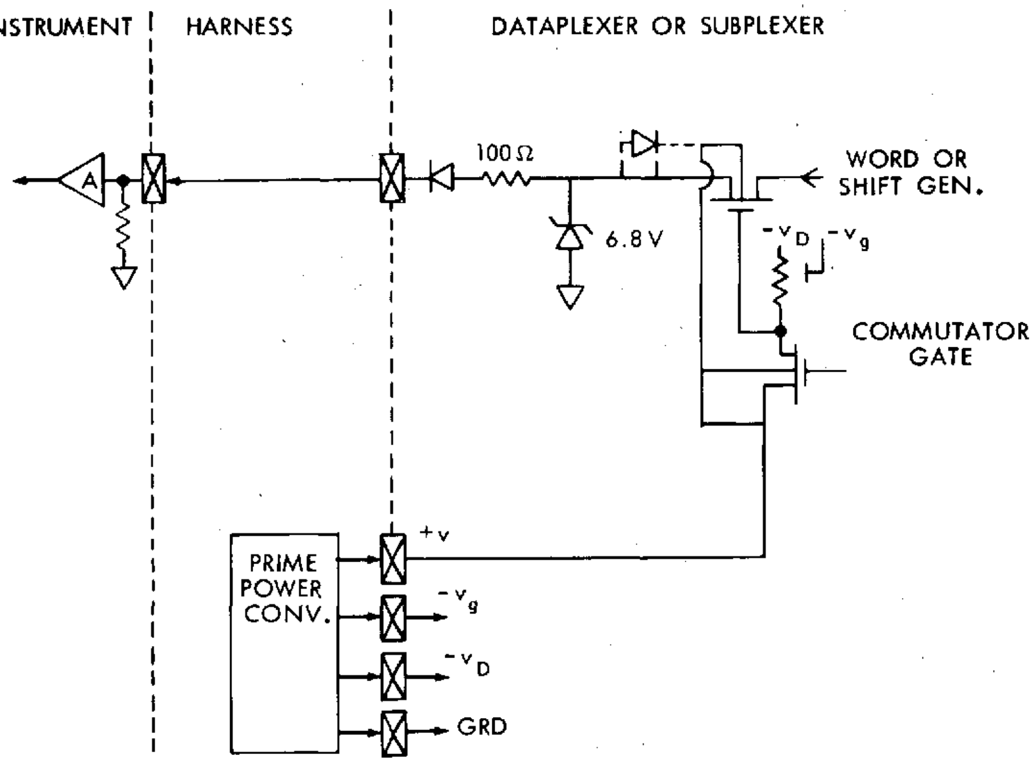


Figure 3-26. Digital Output Interfaces

e. Submultiplexing of 128 analog or subplexer inputs, 32 analog, and 52 digital inputs is handled by the subplexer, a box containing four 32-input multiplexer groupings. The subplexer is designed so that combinations of these units can be interconnected internally as digital or analog submultiplexers. For example, a 32-input analog submultiplexer and a 32-input digital submultiplexer with their associated 32 word gates and 32 shift pulses could be selected for a digital subplexer. The specific allocation of inputs between analog and digital signals is readily accomplished by the installation of jumper wires in otherwise identical boxes. If more inputs are needed, two subplexers could be used with one Dataplexer. The Dataplexer supplies all control signals needed by the Subplexer.

   The Dataplexer can be hard wire programmed so that 16 of its analog inputs can be subcommutated and/or 16 of its digital inputs can be subcommutated.

f. The Command System interface provides the means of controlling format, bit rate, multiplex ratio, and other DMS functions as shown in

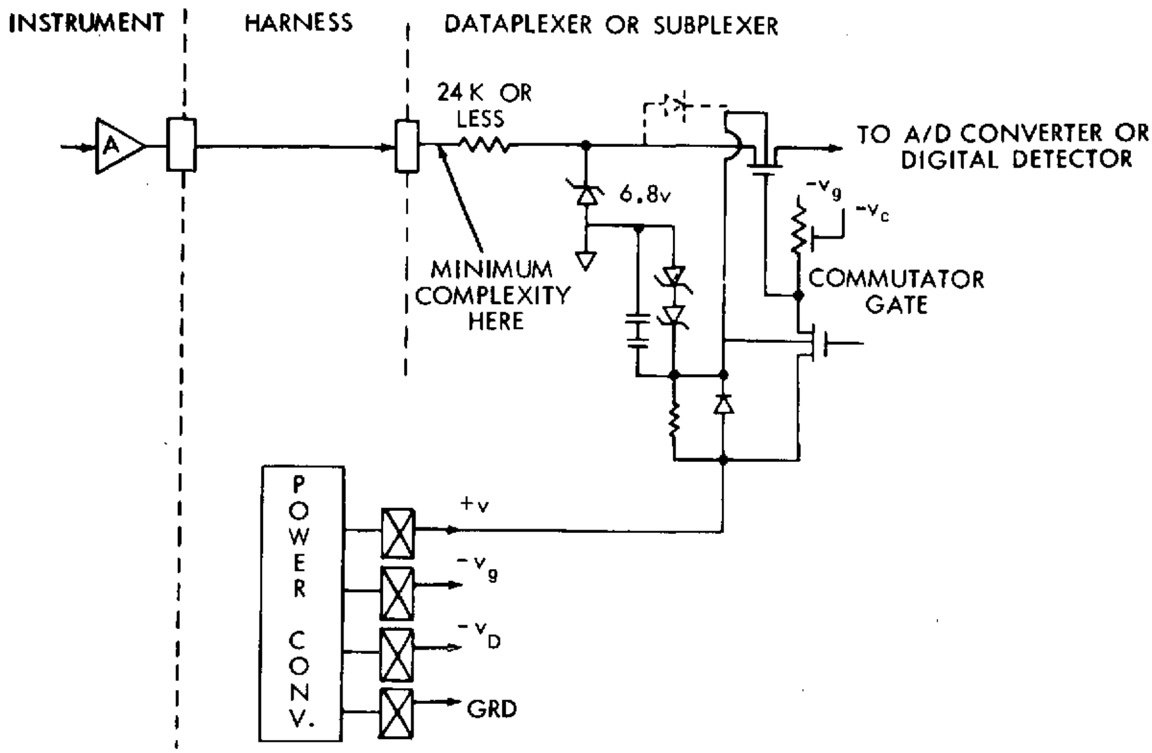


Figure 3-27. Digital Input Interfaces

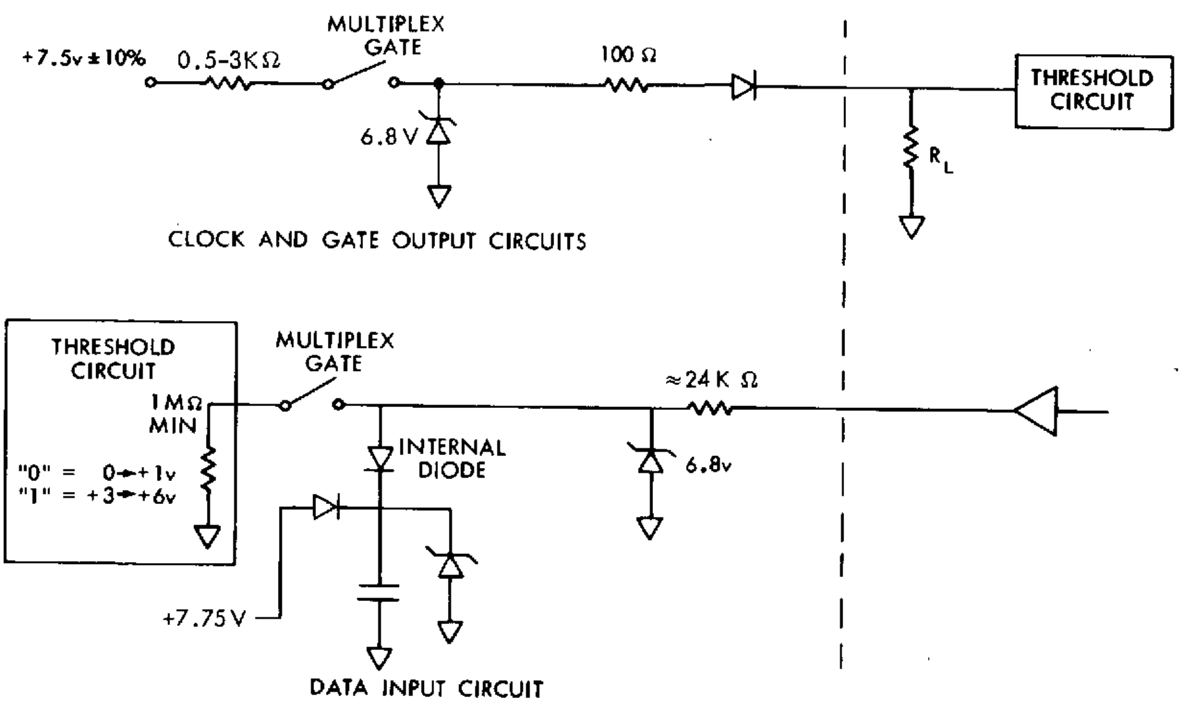


Figure 3-28. Digital Interface Equivalent Circuit

Figure 3-29. A 24-bit command word is shifted serially into the command input register under control of shift pulses supplied from the command system. After the "execute" envelope has ended, the new command word is parallel transferred into the command control register at the end of the current minor frame. The new commands are then decoded and take effect at once. The command word bits and their related functions are listed in Table 3-5.

g. All spacecraft time is derived from a crystal oscillator and countdown chain included as an integral part of the dataplexer. These circuits comprise the spacecraft clock. Twenty-four bits of the spacecraft clock are telemetered to the ground in every fourth minor frame as shown earlier by Figure 3-21 and Table 3-4.

A redundant crystal oscillator is employed with binary dividers to obtain the desired drive frequency for the clock. The crystal oscillator frequency is 640 kHz. Oscillator stability, to ensure that the ground data reduction bit synchronizer can maintain reliable synchronization

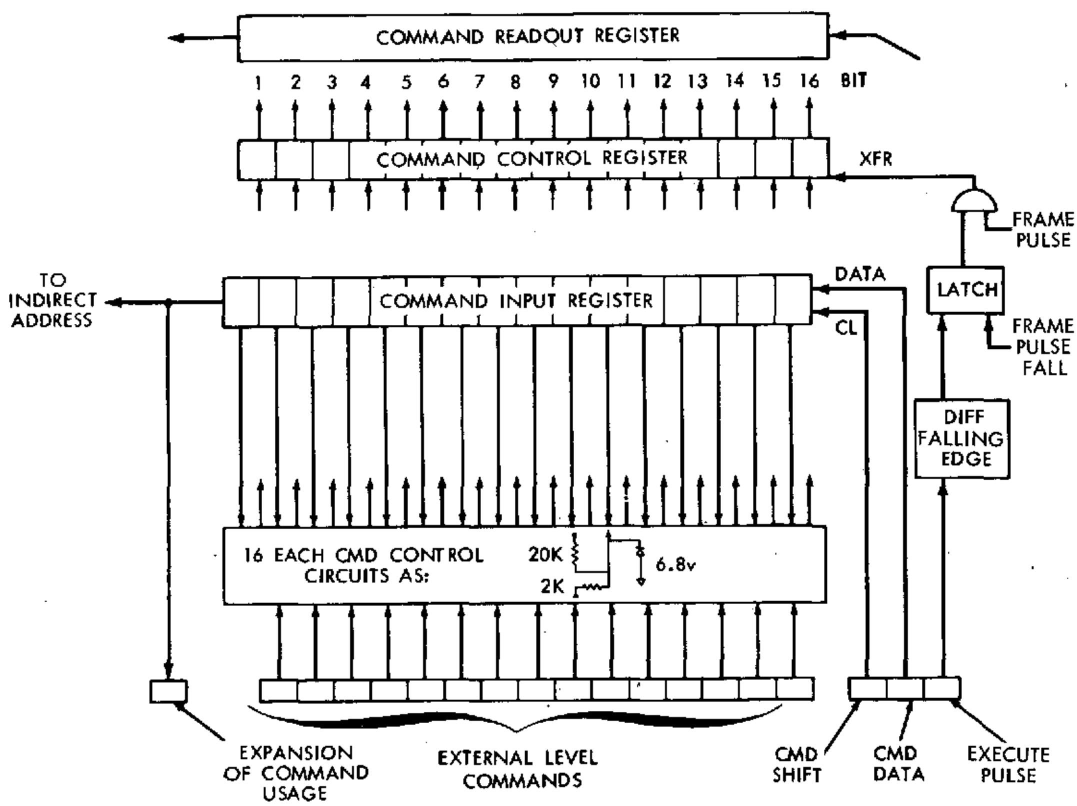


Figure 3-29. Command and Control Registers

Table 3-5

Command Word Bits

Command Register Bit	Function Name	Function Operation
1	Ext.	Used outside of dataplexer
2	VAM Load	"1" = Load VAM No. 1 "0" = Load VAM No. 2
3	Slave to Redundant	"0" = Use back-up Xtal and slave "1" = run on internal Xtal
4	Alternate Fixed Formats	"1" = Alternate fixed formats in ROM "0" = Address only one format
5	N/A	
6	N/A	
7	VF	"0" = TM variable format "1" = TM fixed format
8	FS1 (Format Select)	"1" = Select low address "0" = Select high address
9	FS2 (Format Select)	"1" = Select ROM #1 "0" = Select ROM #2
10	Convol/Block	Selects Convol = 1 or Block Code = 0
11	MR3	Most significant bit multiplex ratio
12	MR2	Multiplex ratio (Faster = 0)
13	MR1	Least significant bit multiplex ratio
14	SR3	Most significant bit sample rate
15	SR2	Sample rate (Slower = 1)
16	SR1	Least significant bit sample rate

even when noise in the telemetry link masks the synchronization information, is ±0.02%. A Pierce oscillator was chosen because:

(1) The circuit does not require weldable inductors, difficult to procure with high reliability.

(2) The circuit is self-starting and reliable over a wide range of crystal frequencies and circuit parameters.

(3) Since oscillation requires the inductance of the crystal, spurious modes of oscillation are minimized.

The telemetry rate countdown binaries are synchronized to the constant-rate satellite clock countdown chain. Commands from the command system determine the telemetry rate.

Three binaries (B1, B2, B3) in the telemetry countdown chain define the 8-bit times required for each telemetry word. The next 7 binaries (W1 to W7) define the 128 words of each minor frame. The last 8 binaries (F1 to F8) define the 256 minor frames which make up the maximum major frame. These binaries are set in the proper phase to insure agreement with the satellite clock binaries running at the same rates. Figure 3-30 shows the countdown block diagram.

h. Power estimate for the DMS, including converter inefficiency, is 4.3 watts maximum. Weight is estimated at 6 pounds per system.

3.4.1.2 Special Signal Conditioning Requirements—Up to 32 analog multiplexer and submultiplexer input lines can be modified to service thermistor circuits. The modifications consist of adding jumper wires on the mother board in such a way that the remotely located thermistors receive the necessary operating voltage from the DMS. The output voltage of one of these circuits is proportional to the temperature of the environment where the associated thermistor is mounted. This voltage is in the 0 to +5.10 volts range and is switched to the A/D converter. Figure 3-31 shows the type of circuits used for this purpose.

3.4.2 Tape Recorder

The NASA Standard Low Capacity Tape Recorder can be used for on-board data storage. This recorder has great flexibility, adaptability and is designed for long life. It uses no gears, clutches, or belts and features redundant bearings for extended life.

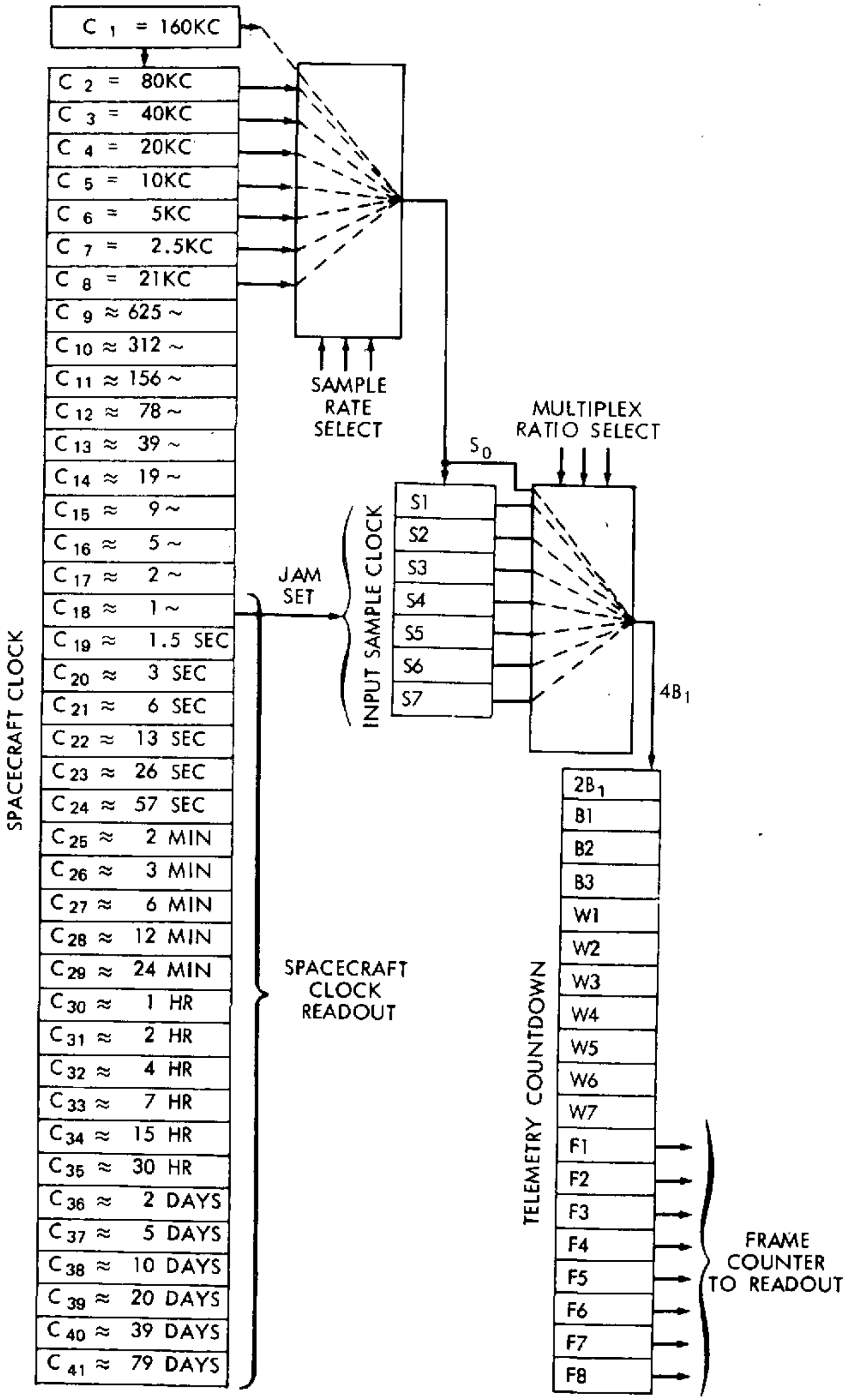


Figure 3-30. Spacecraft Clock and Countdown Block Diagram

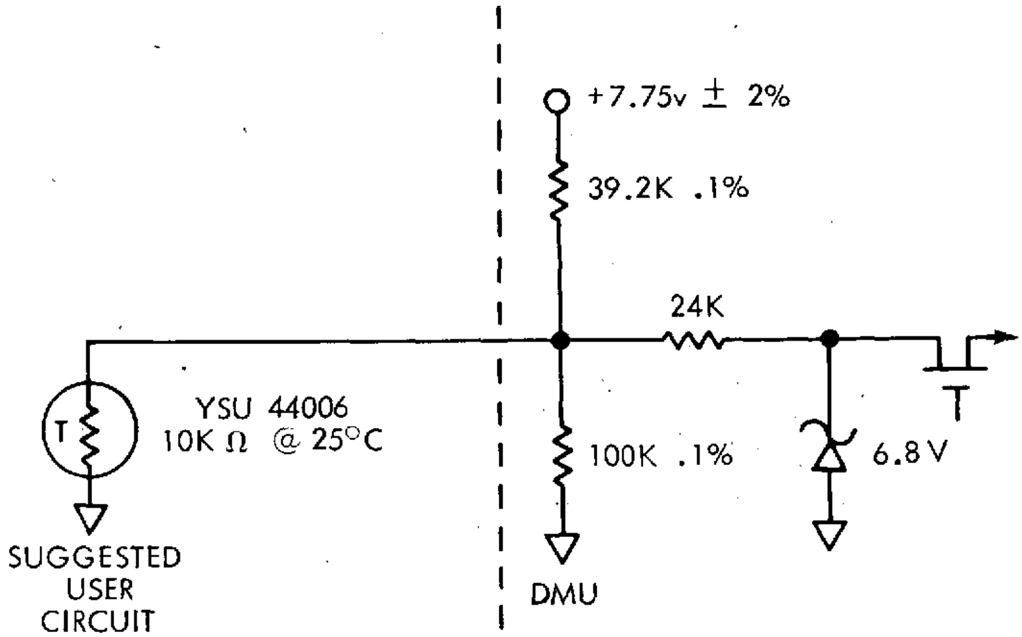


Figure 3-31. Thermistor Interface Circuit

3.4.2.1 General Description—The recorder is a coaxial reel configuration using a single brushless DC capstan motor. Tape guiding is achieved by crowned rollers while tensioning is obtained from negator springs attached directly to the reels. The recorder electronics is packaged separately from the hermetically sealed transport container. The recorder has a total storage capacity of 10^8 bits. Figure 3-32 is the recorder transport.

3.4.2.2 Recorder Parameters

Configuration - Coaxial Reel to Reel

Tape Tensioning - Negator

Tape - 1/4 in. width x 1 mil thick (nominal) x 400 usable ft long

Required Tape Passes - 25,000

Standard Number of Tracks - Four data (minimum)

Playback to Record Speed Ratio Range - Any ratio of speed increments for either function electronically switchable

Record & Playback Time Range - 2.50 minutes (4 tracks simultaneously) to 26.66 hours (4 tracks in sequence)

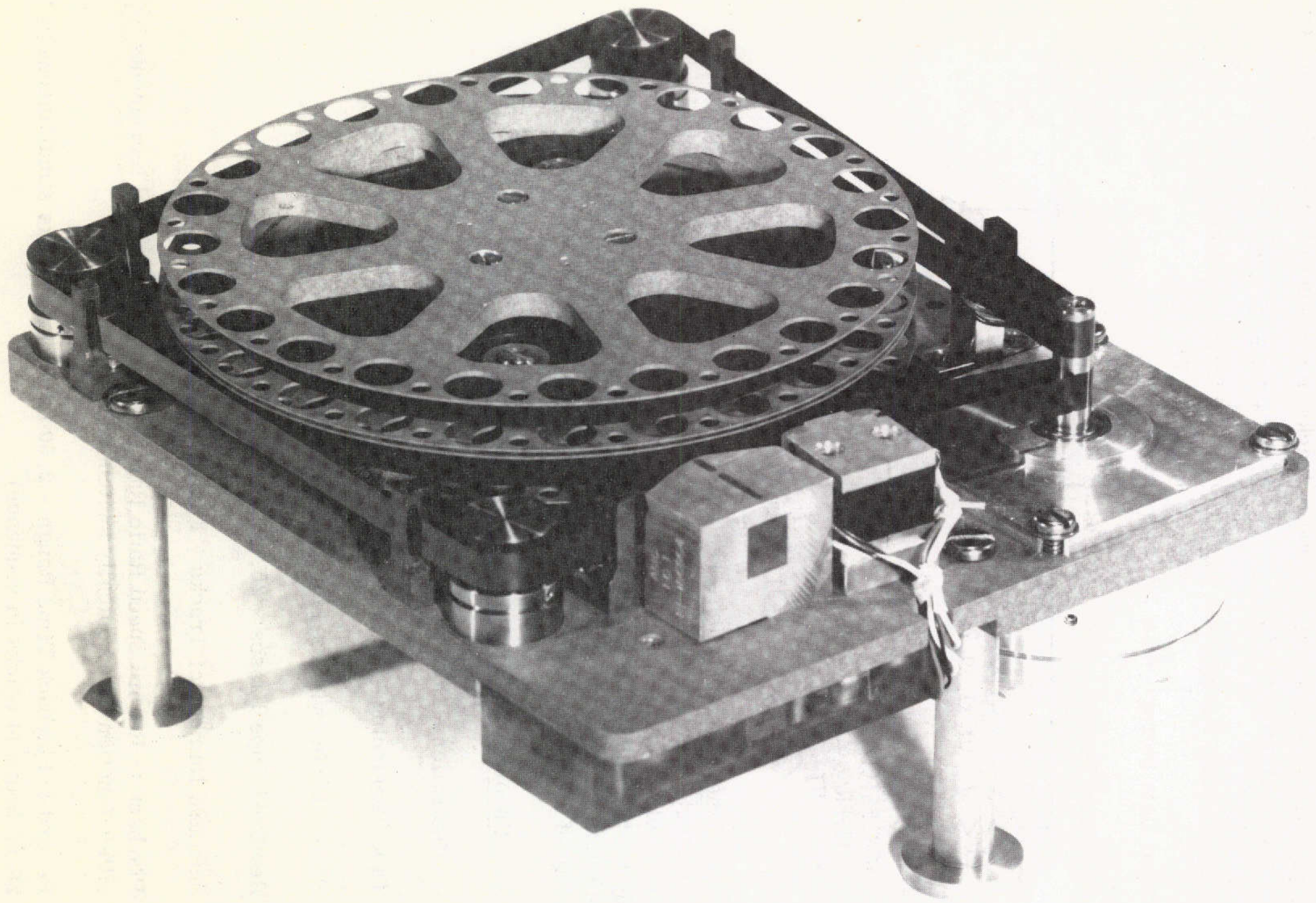


Figure 3-32. Tape Recorder Transport

Input/Output Data Rates - 1,000 (4 tracks in sequence) to 640,000 bits/sec (4 tracks simultaneously)

Tape Packing Density - 5,000 bits/in. per track (maximum)

Total Data Storage - 1×10^8 bits (4 tracks)

Input Data Format - Serial, NRZ or Bi-Phase-Level PCM

Output Data Format - Serial, Bi-Phase Level PCM

Error Rate (max) - 5 in 10^6 bits (Beginning of Life)

Error Rate (max) - 1 in 10^5 bits (End of Life)

Jitter - (electronically dejittered) 0.01% peak-to-peak

Input Voltage - +28 V ±35%

Input Power (Record) - 2.5 to 10 watts (depending upon speed for input data rate)

Input Power (Playback) - 2.5 to 10 watts (depending upon speed for output data rate)

Transport Size (Including Container) - 6 x 6.5 x 3 in. (design goal)

Electronics Size - 6 x 6.5 x 2 in. (design goal)

Total Weight - 10 lb (design goal) (electronics, transport, and container)

Tape Speed Range - 0.2 ips to 32 ips in 23 increments (see Table 3-6)

If the required bit rate is not one of the standard rates listed, it can be accommodated if a coherent clock at the desired bit rate is provided.

3.4.2.3 Commands—The Standard Recorder is adaptable to a variety of command requirements. A special card is used for command interface for each mission requirement. The exact command functions are not established at this time but the recorder is capable of responding to almost any command a partial mission may require.

Table 3-6

10^8 Recorder Standard Speeds and Bit Rates

Tape Speed I.P.S.	KBPS Per Track	KBPS Per 4 Track
32.0	160.0	640.0
25.6	128.0	512.0
20.0	100.0	400.0
16.0	80.0	320.0
12.8	64.0	256.0
10.0	50.0	200.0
8.0	40.0	160.0
6.4	32.0	128.0
5.0	25.0	100.0
4.0	20.0	80.0
3.2	16.0	64.0
2.5	12.5	50.0
2.0	10.0	40.0
1.6	8.0	32.0
1.25	6.25	25.0
1.0	5.0	20.0
0.8	4.0	16.0
0.625	3.125	12.5
0.50	2.5	10.0
0.40	2.0	8.0
0.3125	1.5625	6.25
0.25	1.25	5.0
0.20	1.0	4.0

3.4.2.4 Telemetry—The Standard Recorder has extensive housekeeping telemetry monitors. Although the exact telemetry requirements have not been established, the following is a list of possible monitors.

a. Motor Speed
b. Motor Direction
c. Motor Current
d. Motor Voltage
e. Motor Servo Error Voltage
f. Top Reel Speed
g. Bottom Reel Speed
h. Temperature
i. Humidity
j. Pre-Amp Signal Level (all tracks)
k. Light Emitting Diode Currents
l. End of Tape Sensor Outputs
m. Container Pressure
n. Tape Tension
o. Command Status Flags
p. Tape Position (±50 feet)

This list can be modified to suit a user's needs and telemetry capabilities.

3.5 COMMAND SYSTEM

3.5.1 Command Decoder and Relay System

The AEM Command Subsystem shown in Figure 3-33 consists of redundant command decoders and a command relay unit. Spacecraft commands from the ground can be processed by either decoder through either VHF receiver. Additionally, all commands can be issued from the command storage processor through either decoder. Command conflict or priority establishment between ground generated and stored commands is avoided by use of time-shared control of the decoder execution logic. Figure 3-34 indicates the format for real time and stored commands.

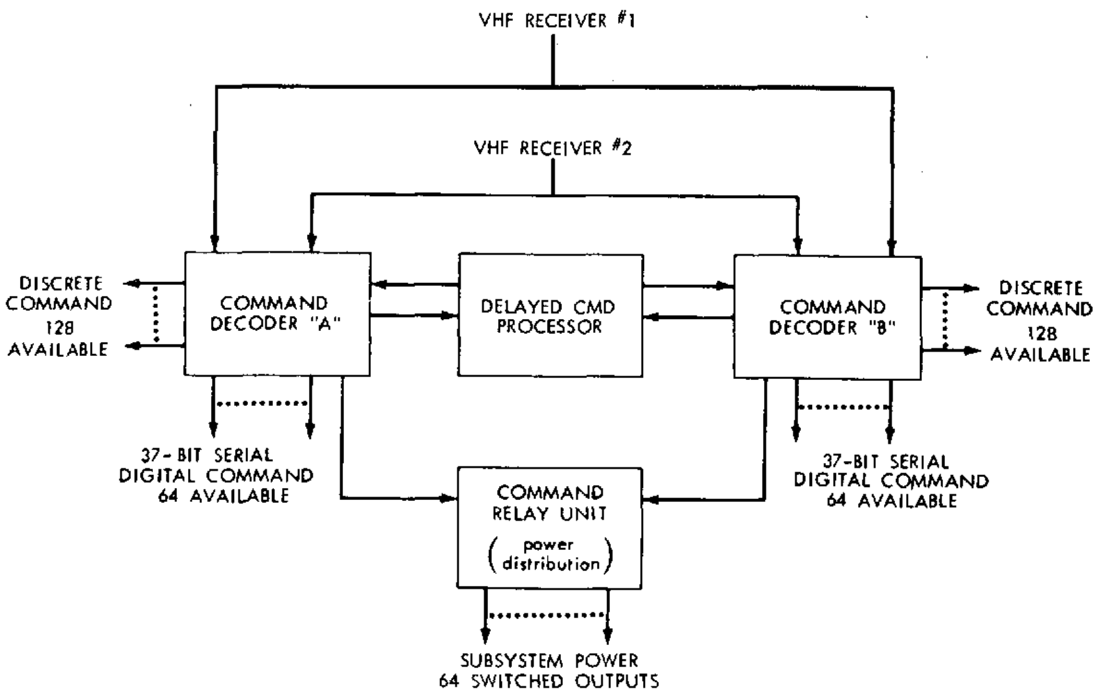


Figure 3-33. Block Diagram AEM Command Subsystem

The redundant decoders are individually identifiable by spacecraft/decoder address. Each ground command message to the spacecraft identifies one of these decoders—the redundant decoder loads a false address inhibiting further command data processing. A false address flag in telemetry indicates this state. Commands continue to be processed by the active decoder until the message is complete or until a false parity check is detected. False parity also interrupts command processing with a telemetry flag to indicate this state. A decoder can be reset from false address or false parity by removal of the subcarrier FSK modulation. An 8-bit command execution counter is incremented for each ground command processed.

The contents of this counter is also telemetered. Verification of receipt and execution of an entire command message is possible by an initial and final status of this 8-bit counter.

The command relay unit receives serial commands from either decoder to command relay on/off status. A capacity of 70 relays is available and may be either latch or nonlatch. Subsystem power distribution and pyrotechnic circuits are typical functions the relay unit handles.

The command storage processor will receive 37-bit serial commands to program the desired delayed sequence and will generate a 44-bit serial input to a command decoder for each command to be issued.

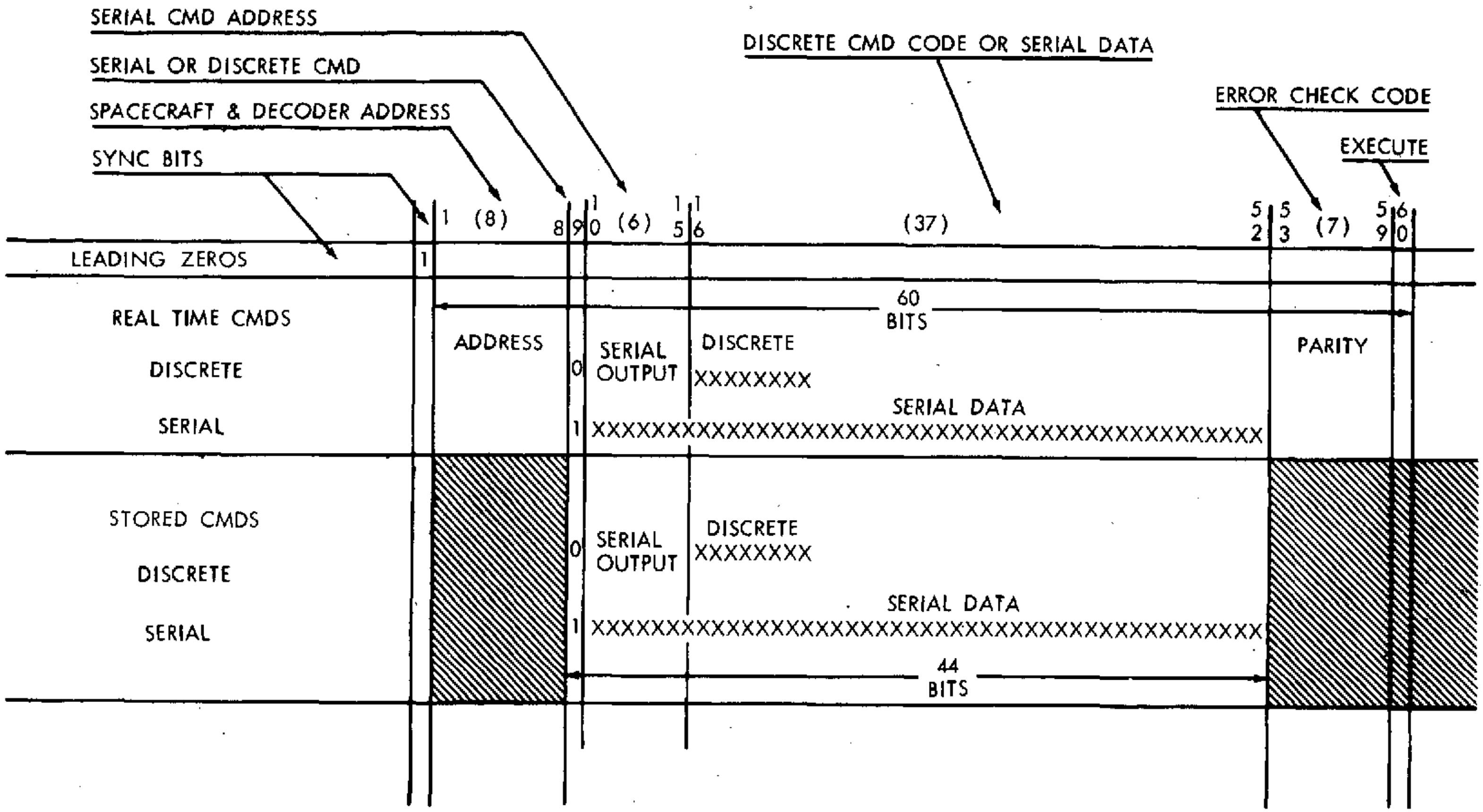


Figure 3-34. Command Format

3.5.1.1 Command Detector and Decoder Design

3.5.1.1.1 General Characteristics. The command capacity of each decoder is 128 discrete, or impulse, commands and 64 serial digital commands. Serial commands contain 37 bits each.

The real time command execution rate is 20 commands per second once the initial uplink synchronization is complete. The maximum rate at which delayed commands can be executed by the decoder is 33 per second.

Each decoder including DC/DC converter in a 15.2 x 20.3 x 8.9 cm housing, weighing 1.1kg, and consuming 0.5 watts average power. CMOS logic is used for the digital section of the decoder.

3.5.1.1.2 Electrical Design. Input signal conditioning is used to select one of the receiver signals for processing. In the absence of an RF carrier signal, broadband noise is present on the decoder input, and the input circuit searches the two receivers for an FSK/AM signal with sufficient signal to noise characteristics. Whenever the subcarrier presence threshold is exceeded, the search mode ceases for as long as this threshold remains positive. The FSK/AM signals are assigned in the 7 to 12 kHz spectrum.

Bit synchronization, Figure 3-35, is derived by AM detection of the subcarrier envelop which is 50% modulated at the 1200 Hz bit rate. The detected signal is input to a phase-lock loop where a voltage controlled oscillator (VCO) is locked to the bit rate. The VCO clock is used in the data detection filters as well as to clock the digital data processing.

Bit detection or data detection is accomplished by applying the FSK subcarriers to a pair of bandpass filters—one filter is tuned to data "0" frequency and the other filter is tuned to the data "1" frequency. The bit rate clock is used to dump these filters each bit interval, thus, destroying any energy present from the preceeding data bit. These filter outputs are detected, integrated by an RC circuit, and supplied to a differential amplifier input. The differential amplifier determines which filter has the stronger signal and whether a "one" or a "zero" was transmitted. Outputs from the bit synchronizer/detector circuitry are data, clock, and data presence signals. These signals are converted to levels compatible with the digital logic used to process the command data.

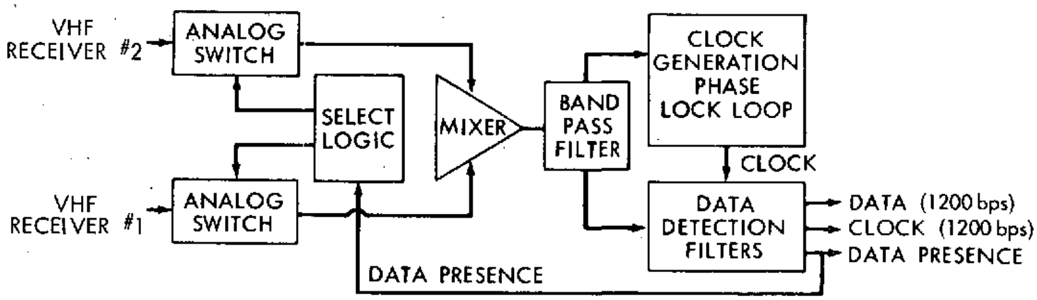


Figure 3-35. Bit Detector/Synchronizer

Real time command data enters the command process logic as shown in Figure 3-36. The data are examined for proper spacecraft and decoder address, the command bits stored in the data register, and the command word examined for errors. The data format logic is the control circuit used to process each command message.

The command execution logic, shown in Figure 3-37, is controlled by the time share logic and executes serial or impulse commands as supplied from either real time or stored inputs. The time share logic of the redundant decoders is synchronized to insure command execution intervals in the decoders are compatible.

Figure 3-38 indicates the timing requirements for the time share logic. The basic time period is 15 milliseconds during which command execution or command data load may occur. All necessary timing is derived from the 4.27 kHz clock.

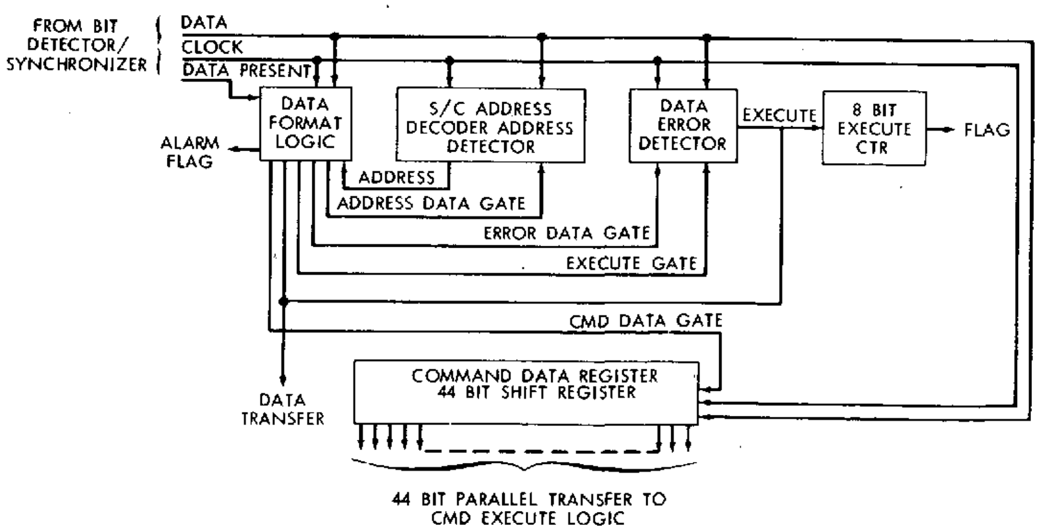


Figure 3-36. Real Time CMD Process Logic

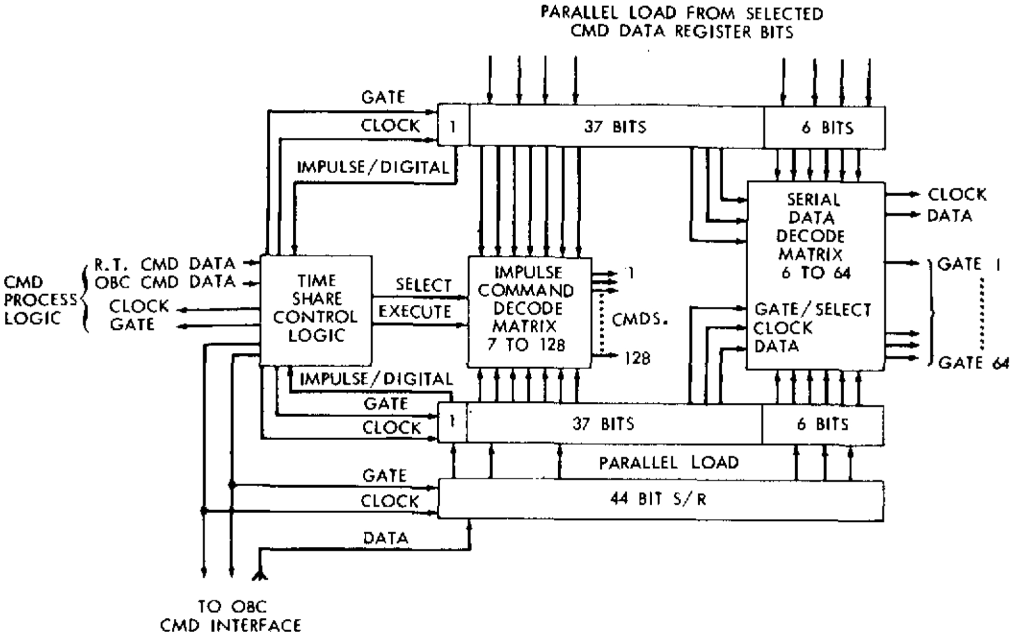


Figure 3-37. Command Execution Logic

Data input and buffer registers are provided for real time and stored command data. This data storage is required as a consequence of the asynchronous nature of the real time and stored command data. Timing for data transfer from input to buffer registers is determined by the time share logic.

Command execution is provided by an impulse matrix or a serial matrix. An identification bit is present in the command data which determines the type of command to be issued. The time share logic monitors the control bit and enters seven bits to the impulse matrix or six bits to the serial matrix. If an impulse command is selected, the seven bits generate one of 128 possible discrete outputs for the 15 millisecond interval. Selection of a serial command results in the six bits determining one of the 64 gated serial output terminals for the 37 NRZ data bits to be clocked to a subsystem. A serial output terminal consists of a three line interface containing clock, NRZ data, and data envelop.

3.5.1.1.3 Subsystem Interfaces. The electrical interfaces are as follows:

Command Input (VHF Recvr)

Subcarrier Frequencies f_0 = TBD Hz

f_1 = TBD Hz

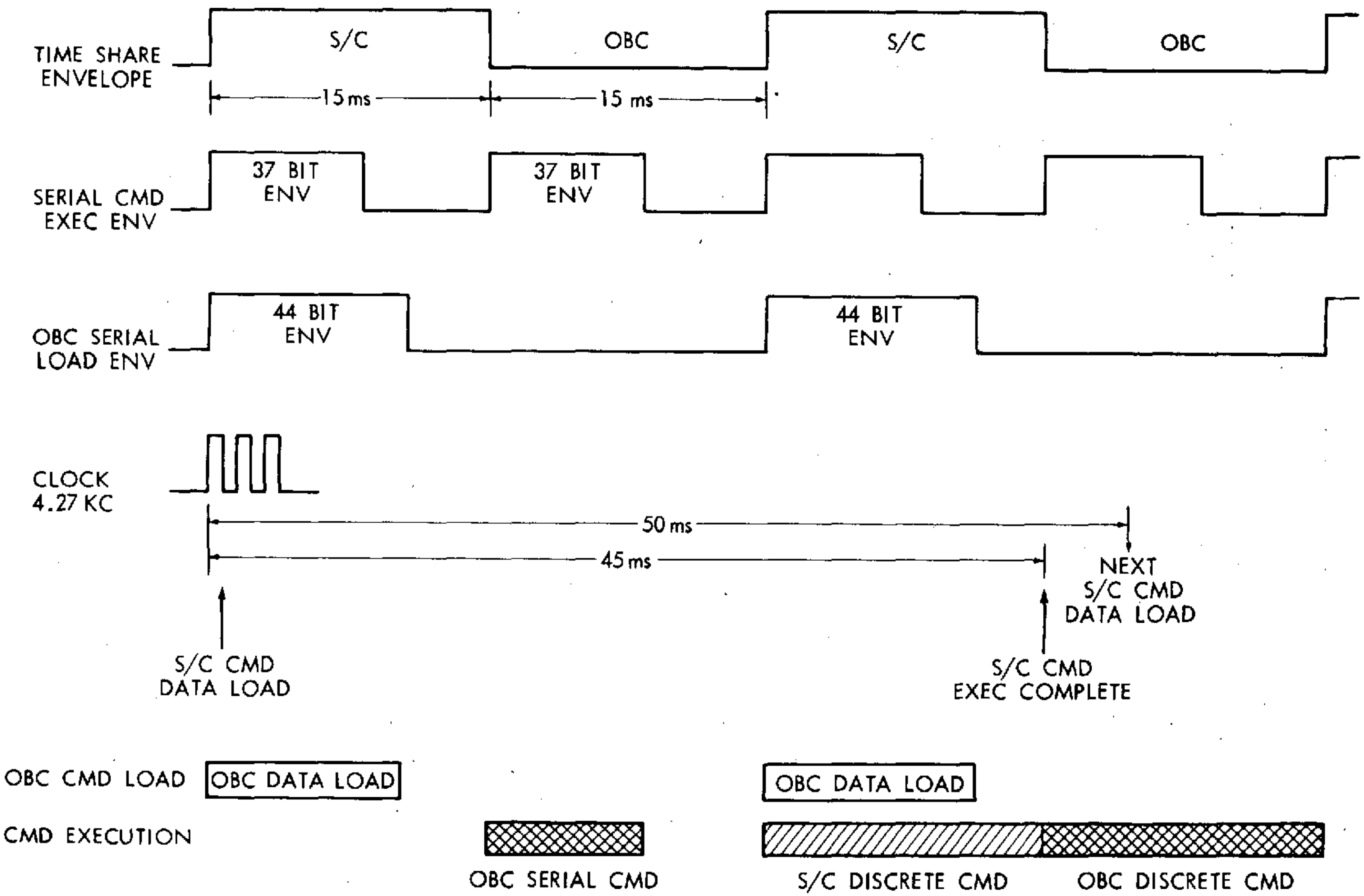


Figure 3-38. Timing Diagram, Time Share Logic

Command Input (VHF Recvr) (continued)

Bit Rate 50% AM = 1200 Hz

Composite Signal Amplitude = TBD $\pm$ TBD VRMS

Composite Signal Source Impedance = TBD Ohms

Delayed Command Input

Each decoder has available two separate inputs to process delayed commands. See timing diagram, Figure 3-39.

Data (44 Bits NRZ at 4.27 kHz)

Clock (44 Bit Duration at 4.27 kHz)

Envelop (Positive 44 Bit Duration)

Input Enable (Positive Voltage enables Input Port-1 of 2)

Parity Disable (Positive Voltage prevents command execution if parity test fails following data transfer)

Decoder Power Input & Converter Supply

+28 Volts S/C Bus

S/C Common

+12 VDC

+10 VDC

Signal Common

S/C Telemetry Monitors

Digital

Execution Counter (8-Bit Word)

False Address (Flag)

False Parity (Flag)

Analog

+10 Monitor (0-5 V)

-10 Monitor (0-5 V)

Temperature (Thermistor)

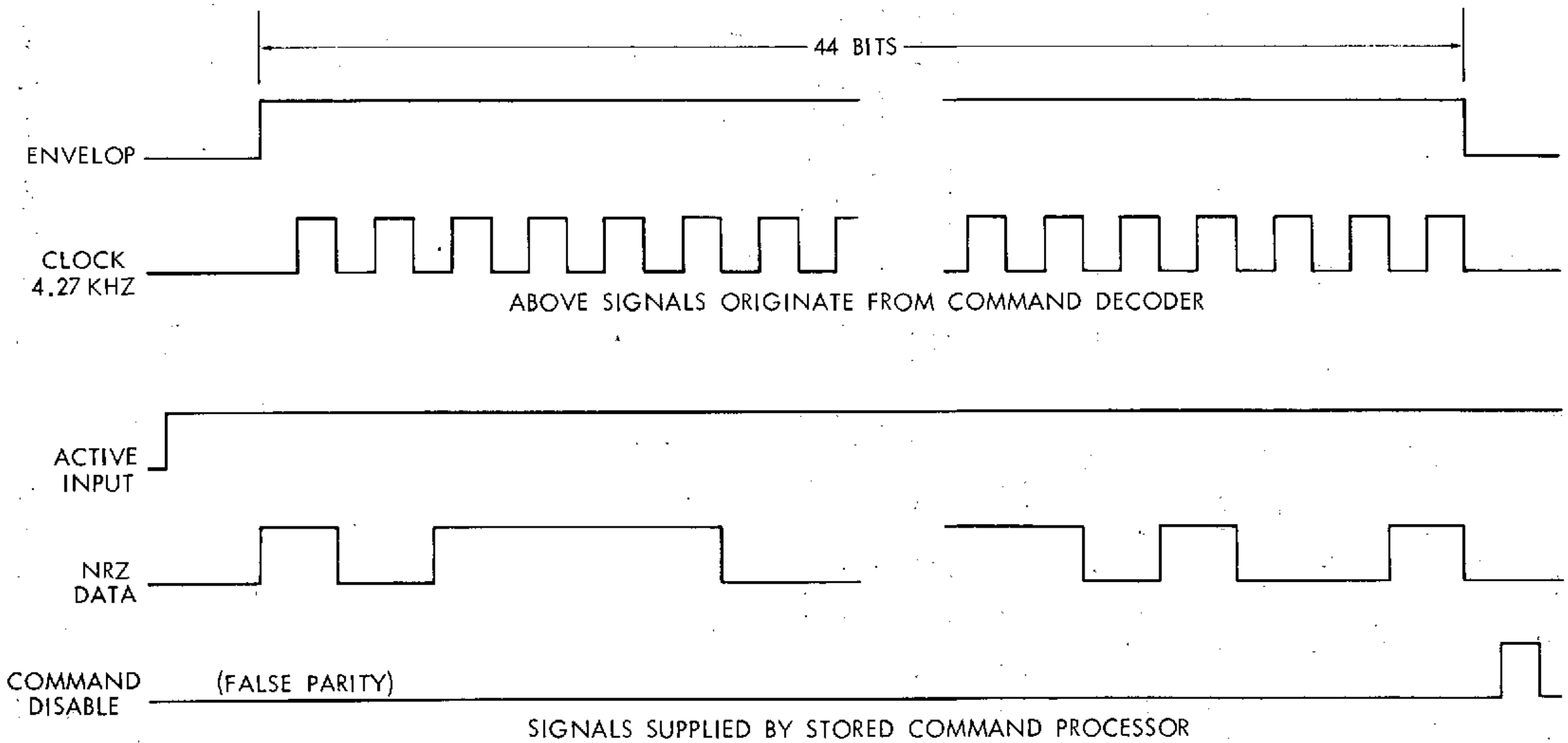


Figure 3-39. Stored Command Timing Diagram

Impulse Commands (See Figure 3-40)

128 Available

Serial Commands (See Figure 3-41)

64 Available

Envelop (37 Bit Positive Duration)

Clock (37 Bit 4.27kHz)

Data (37 Bit NRZ)

S/C Decoder Address

8 Bits (Programmable Connector)

3.5.1.2 Command Relay Unit Design

3.5.1.2.1 General Characteristics. The Command Relay Unit has a design capacity of 70 relays. The majority of these are used for distribution of subsystem power. Pyrotechnic circuitry, controlled by commands either real time or by delayed command stored and timed in the command storage processor, is part of the relay unit. Commands

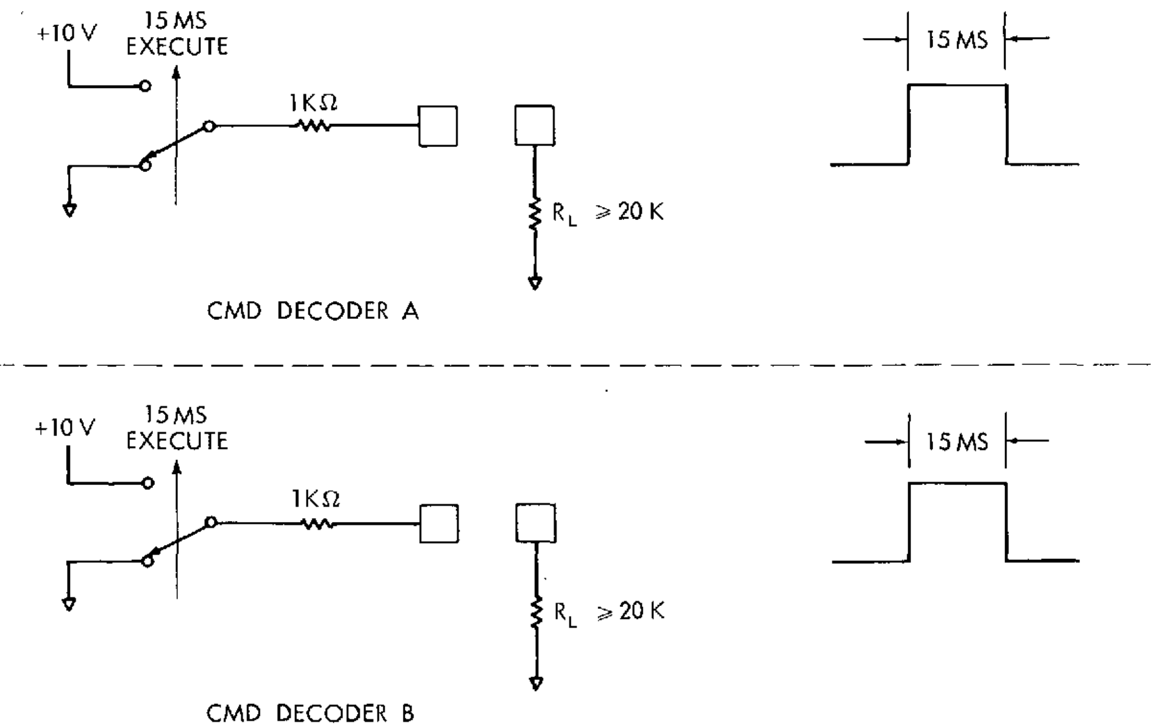


Figure 3-40. Impulse Command Output Circuits, Redundant Commands Shown

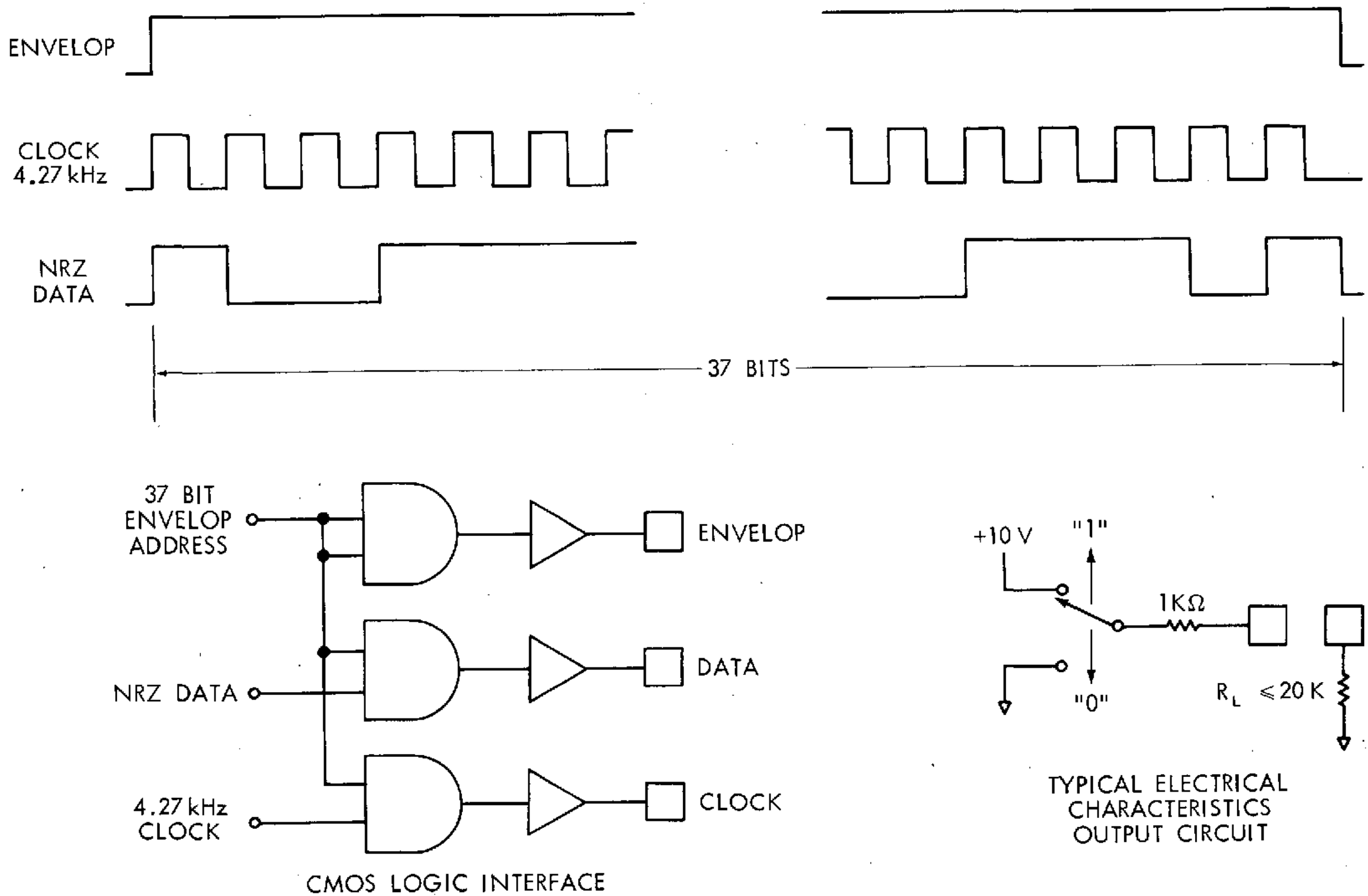


Figure 3-41. 37 Bit Serial Command Interface & Timing

to the relay unit are 37-bit serial from each decoder. Inputs are available to interrupt subsystem power distribution by signals originating from undervoltage or overcurrent detecting circuits.

The command relay unit is packaged in a 15.2 x 20.3 x 12.7 cm housing, and a full relay complement weighs about 3.6 kg pounds. Power consumption depends on usage (command rate) but is estimated to average 0.1 watt.

3.5.1.2.2 Electrical Design. The command relay unit is shown in block diagram, Figure 3-42, to consist of 70 commandable switch outputs. Command control is provided by a 10 by 14-bit matrix where the 10 bits identify 1 to 10 relays, and the 14 bits identify 7 on and 7 off functions. A module of 10 relays is addressed by 12 bits to control each relay "on" or "off". The logic which interfaces to the decoders is redundant; therefore, 24 bits are required to each module of 10 relays.

The relay drivers are all identical, and a typical circuit is shown in Figure 3-43. This type of relay driver has several distinct advantages for spacecraft use, such as stored energy for minimum surge or transient currents on power bus, command pulse width requirement is small, and single-part failure modes do not result in high steady state power dissipation. The repetitive on/off command rate is limited by the RC charge time constant of the storage capacitors but this limitation is not usually significant.

3.5.1.2.3 Subsystem Interfaces

Serial Commands (Redundant 37 Bit)

Data (37 Bit NRZ)

Clock (37 Bit 4.27 kHz)

Envelop (37 Bit Duration)

Auxiliary Off Inputs (Redundant 3 Each)

+10 Volts for 20 Milliseconds from Source Impedance

1 K Ohm or Less

Power Input

+10 V Supplied from Each Decoder

+28 V for Relay Drivers from S/C Bus

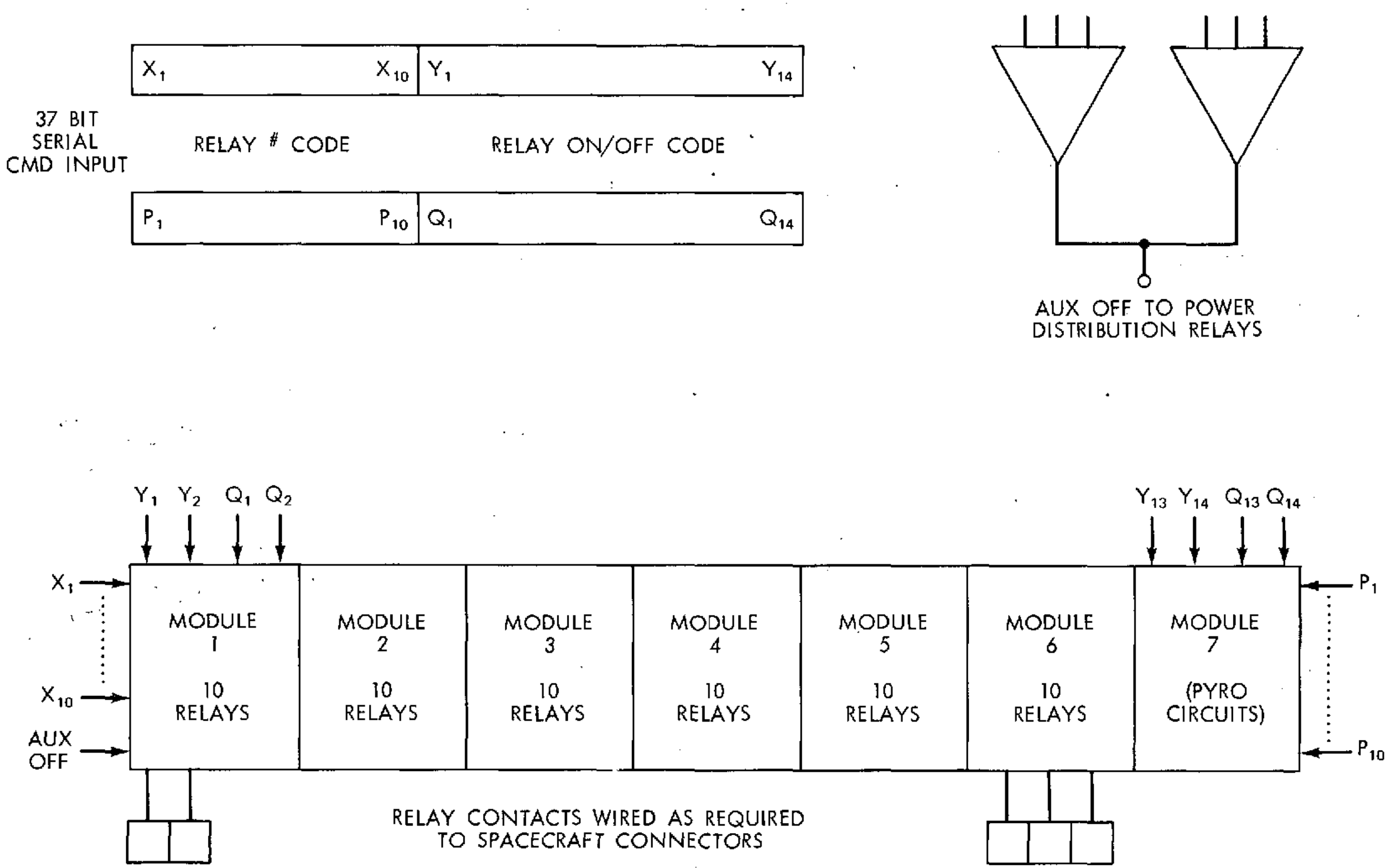


Figure 3-42. Command Relay Unit Block Diagram

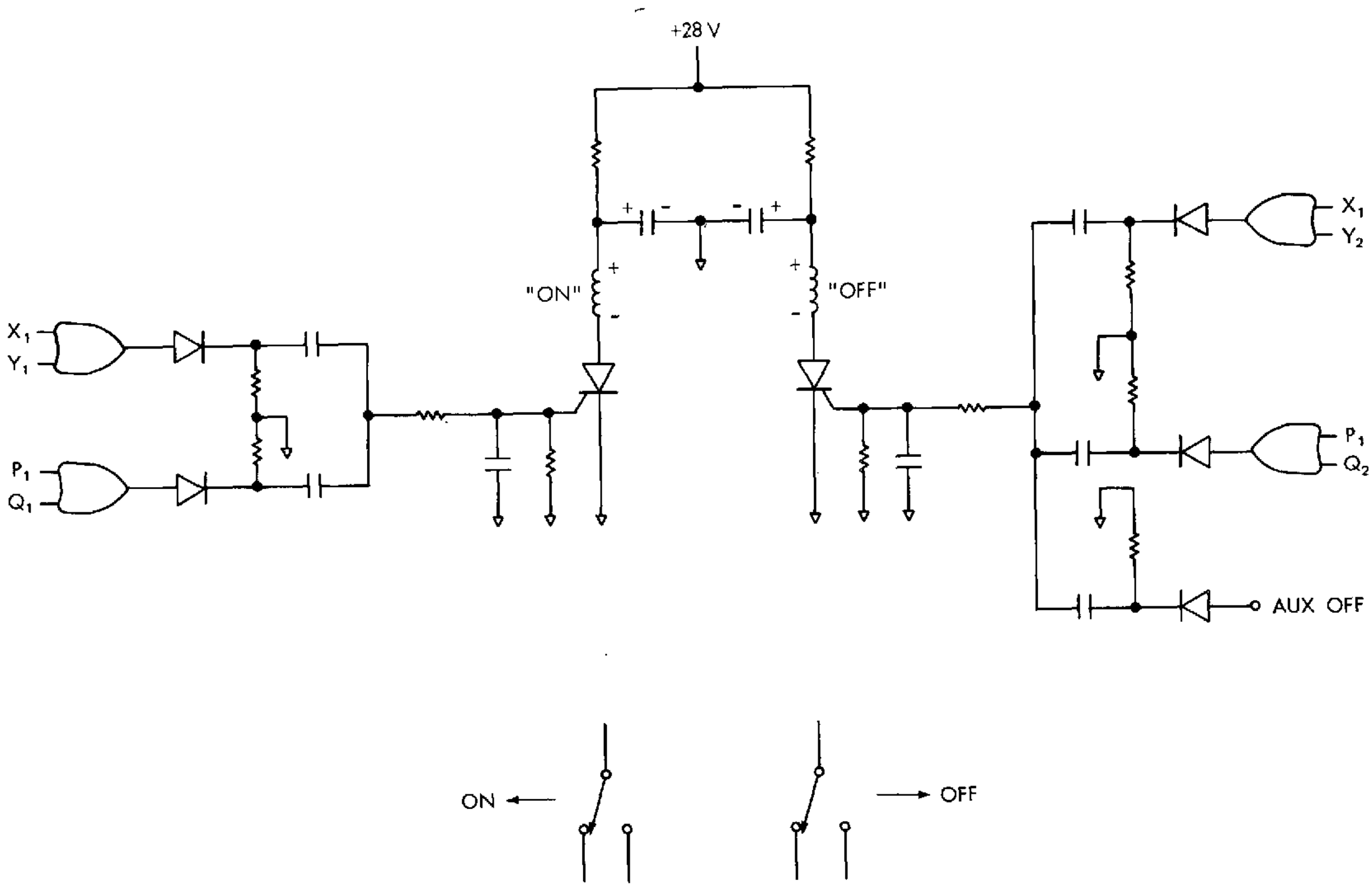


Figure 3-43. Typical Relay Driver

Relay Outputs

DPDT Two Amp Resistive Contacts

Available (Maximum of 70 Relays)

Pyrotechnic Circuits to be Determined

3.5.2 AEM Operations Electronics

The AEM Operations Electronics incorporates a number of operations and engineering functions that are not otherwise provided by the standard equipment. These functions include:

- Stage IV Telemetry Signal Conditioning
- NASA 36-bit Time Code Generator
- Auxiliary Timers
- Pyrotechnic and Deployment Sequence Parameters
- Spin Rate Parameter
- Orbit Timer

These functions are modularized and, therefore, optional for each mission. A typical application of the option exercise is deletion of the 36-bit time code generator when not required for a particular mission. Of particular importance is the optional Microscope minicomputer, discussed in section 3.5.2.4. Other options are as follows:

3.5.2.1 Stage IV Telemetry Signal Conditioning—This function is relatively simple, providing switched power to accelerometers and a head pressure transducer. If required, signal-buffering for these instruments can be implemented.

3.5.2.2 Auxiliary Timers—The auxiliary timers are used in parallel with or in place of the Microscope remote command capability. They are actuated by critical functions, such as "S-Band ON," and provides "OFF" commands after a pre-selected period. This period is established as 20% longer than the anticipated maximum use period for the mission. Thus, for a 600 KM mission, one timer is preset by design for 12 minutes and provides back-up "OFF" functions in the event that the spacecraft goes over the horizon in a high power mode.

3.5.2.3 NASA 36-Bit Time Code Generator—This optional function uses digita logic to generate the standard 36-bit time code with two outputs:

- 100 BPS standard format for transmission on a 70 kHz side-band of the S-band link.
- 36-bit serial data within the spacecraft PCM telemetry link.

Deviation is not to be greater than one second per day. Reset to ground time ±20 milliseconds can be accomplished by ground command as required.

3.5.2.4 AEM Operations Microscope Minicomputer—This versatile device pro vides command memory and variable-sequence programming. Some of the com ponents include:

- Read-Only-Memory (ROM), programmed prior to fabrication, and is non-destructible.
- Random-Access-Memory (RAM) which is used by the ROM programs as a volatile memory.
- Parallel and serial input/output (I/O).
- Basic Arithmetics
- Interrupts and Timing

The internal programs can react to command, an external event, an internal event, or an internal calculation. Interfaces are directly compatible with the AEM command decoder, data multiplexer, and other digital functions. Typical applications are as follows:

1. Pyrotechnic and Deployment Sequence Parameters - Digital flag inputs and memory are used to define the time sequence of paddle deployment and other pyrotechnic actuated operations. These are telemetered via a standard digital data interface.

   The memory stores information from 9 to 12 paddle deployment flags every few milliseconds commencing with deployment initiation. Memor load is inhibited after 2.5 seconds. Memory readout then commences, requiring minor frames for a full readout. This process repeats until the function is disabled. Thus, this vital data is available even though real time coverage of the event is not.

2. Spin Rate Parameter - The Microscope is used, in conjunction with an X-axis magnetometer signal, to measure spin rate. The last known spin rates following spin-up and also prior to despin are retained until the program is no longer required. Spin period is determined by monitoring a magnetometer signal with a level detector and by measuring the time between crossovers of a preselected level. (Note: The common practice of using sun-slit pulses for spin period information is not relative to AEM, as some orbit insertions occur at night.)

3. Orbit Timing - This function employs memory registers, set up by ground command, to provide on/off actuation of selected equipment. It requires three (3) serial-digital commands to establish:

   - Orbit period (in seconds)
   - On-command from orbit "zero" (in seconds)
   - Off-command from orbit "zero" (in seconds)

   Additional commands are required to start the timing operation, therefore establishing orbit "Zero", to select the on/off functions, and to define operations for several successive orbits. Status of all logic registers in memory is telemetered via a standard digital data interface. Missions requiring complex remote operations, either impulse or serial-digital, are easily accommodated.

   Typical applications of the timer are:

   - Actuation of a device, such as velocity increment propulsion, at apogee or perigee.
   - Operation of on-board instrumentation over a site, such as Madrid, that does not have VHF command capability.
   - Prepositioning of an instrument relative to orbit position or an external event.

3.5.2.5 AEM Pyrotechnic Control—The AEM Pyrotechnic control card is located in the Command Relay Module and provides the following critical functions:

- Spacecraft despin (Yo-Yo)
- Solar paddle deployment

- Stage IV separation
- Paddle rotation to selected mission angle

All functions but paddle rotation are controlled through electronic sequencing that commences with Stage III Separation as shown in table. A number of safeguards are incorporated in the subsystem, such as:

- Disarming through blockhouse, not requiring spacecraft power
- Primary arming by ground command prior to launch, with telemetry verification a launch requirement
- Momentary "Chattering" of the Stage III separation switches ("D-Switches") cannot actuate the sequence
- Sequence can be initiated despite total failure of any one "D-Switch"
- Yo-Yo release failure inhibits the automatic sequence
- Back-up ground commands, armed by another command, are provided for all functions

Table 3-7 is a typical pyrotechnic control sequence.

Table 3-7

Normal Pyro Sequence

• T-6 secs:	Stage IV/Spacecraft Spin-up
• T-0 :	Stage III Separation
• T+192 :	Yo-Yo Despin
• T+256 :	Solar Paddle Deployment
• T+320 :	Stage IV Separation
• T+XXX :	Solar Paddle Rotation by Direct Command

3.6 COMMUNICATIONS

3.6.1 VHF Command Receiver

The VHF Receiver is a single conversion superheterodyne AM receiver operating in the 148 MHz band. When interfaced with the PCM decoder, the nominal system sensitivity is -110 dbm at a bit rate of 1200 bits per second of the standard PCM/FSK/AM modulation.

The receiver block diagram is shown in Figure 3-44. The signal enters through the preselector filter that has a 3 MHz bandwidth at 148 MHz. After amplification at 148 MHz, the first mixer converts the signal to the IF which is 15.5 MHz. The signal is amplified, filtered, and detected for PCM subcarrier and internal AGC operation.

The physical dimensions of the basic receiver module are 8.9 x 11.4 x 2.9 cm. This module is designed to be mounted in a standard spacecraft module such as the RAE module which is 12.7 x 17.8 x 3.2 cm. Overall weight of the receiver module is 0.55 kg.

The receiver bandwidth is 40 kHz to accommodate the received signal bandwidth plus allowances for doppler shift and local oscillator instability. All spurious responses are greater than 60 db below the system threshold. The power requirement is 350 mw at +12 volts to be supplied from the decoder converter.

3.6.2 VHF Transmitter

The VHF (136 MHz) transmitter will provide 250 milliwatts of power at the transmitter output for transmission of S/C housekeeping parameters and, possibly low bit rate instrument data. The transmitter is phase modulated (PCM/PM) similar to one flown on S^3, RAE and the IMP series spacecraft.

It will be crystal controlled, with a frequency stability of ±0.002% and will have a phase modulation capability of greater than ±1 radian. As shown in the block diagram, Figure 3-45, the oscillator is followed by a buffer amplifier to improve its short term stability. The signal is then modulated by the linear phase modulator and then doubled to the output frequency. After amplification to 250 milliwatts, the signal will be filtered to reduce harmonics to an acceptable level.

The modulation input will have a linear phase filter to limit the modulation bandwidth to about 12 kHz. This will insure that modulation harmonics will be within the assigned frequency channel.

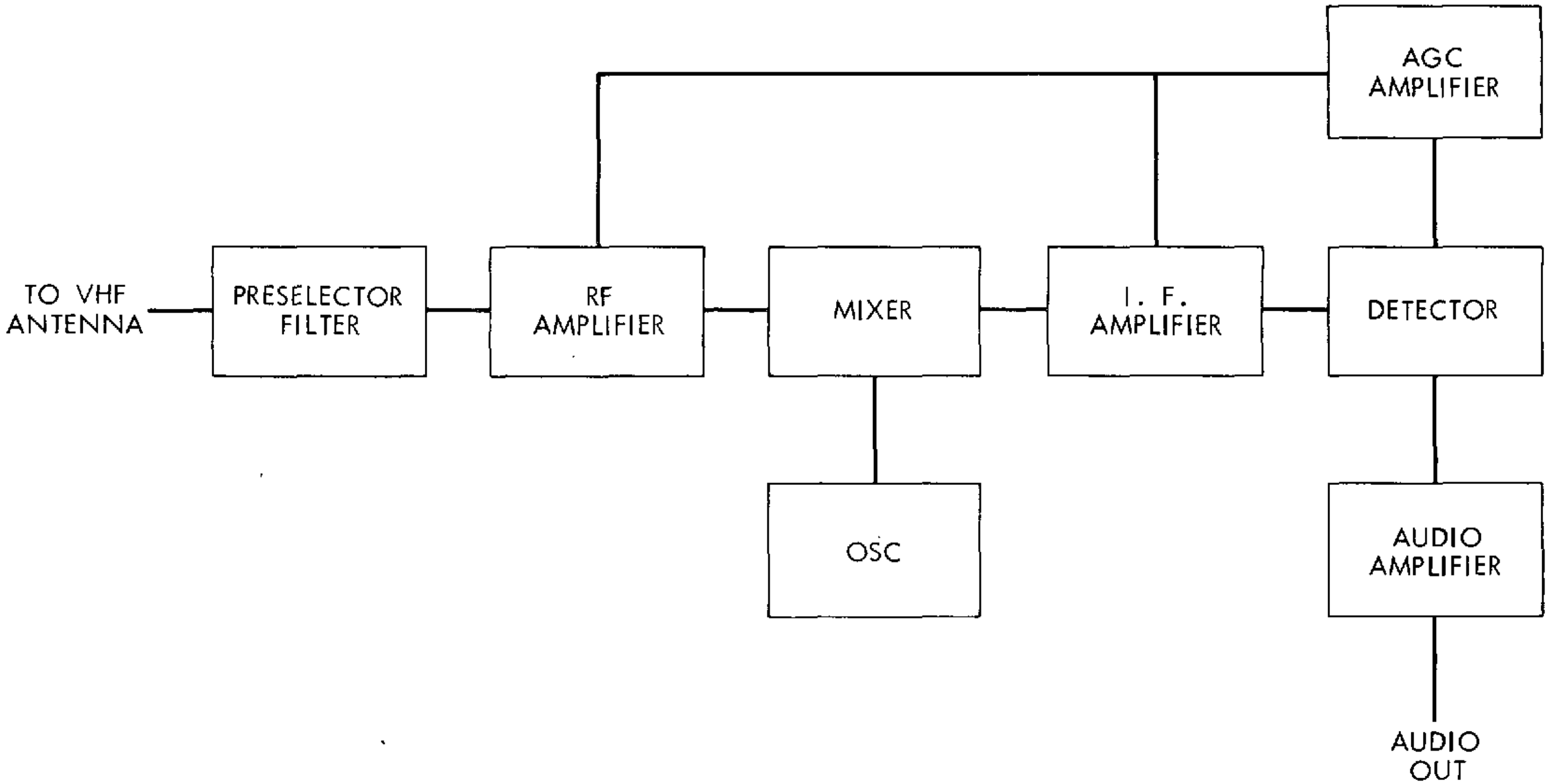


Figure 3-44. VHF Receiver Block Diagram

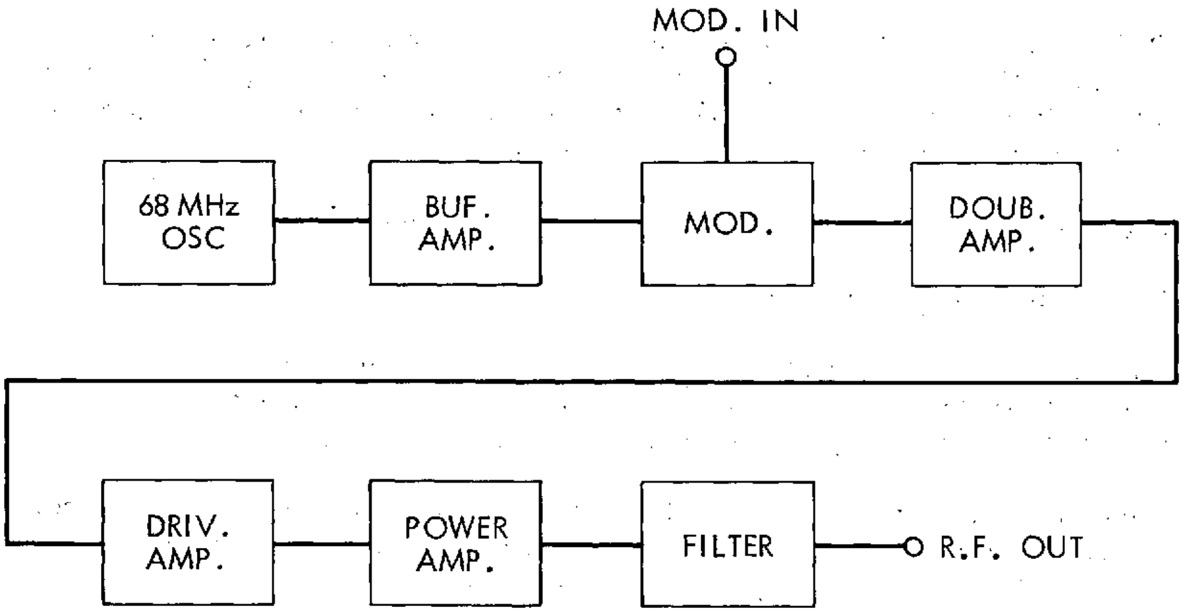


Figure 3-45. VHF Transmitter Block Diagram

The transmitter will operate off the 28 volt ±2% supply and have an efficiency of at least 30%. The weight is 1.0kg and the size is 12.7 x 17.8 x 3.2 cm.

3.6.3 S-Band Transmitter

The S-band transmitter will provide transmission of instrument data to ground receiving stations (STDN). The two watt power output at the transmitter terminal provides an adequate link margin for both PCM digital and wide band analog data. A linear phase modulator with a 2 MHz modulation bandwidth will transmit the PCM analog/PM data.

The transmitter is built in three functional subassemblies. These are: auxiliary oscillator, multiplier modulator, and power amplifier. This breakdown provides maximum interchangeability with other programs. The auxiliary oscillator develops a crystal controlled, 19 MHz signal at a 2-1/2 milliwatt level. This signal is multiplied to S-band, amplified to 250 milliwatts, and phase modulated. This output drives the power amplifier to a two watt output level. An isolator at the output terminal provides protection to the power amplifier in the event of an extreme load mismatch.

The transmitter operates from 28 volts ±2% using 8.0 watts nominally.

Its dimensions are 13.5 x 6.6 cm on the base and is 9.4 cm in height. The weight is 1.0kg.

With one exception the subassemblies are identical to those used on IUE. This is the transistor type used in the power amplifier. The IUE amplifier has four watts output. To use this transistor at a two watt level would be less efficient than using a similar device with fewer junctions.

Figure 3-46 is an artist's conception of the transmitter.

3.6.4 VHF Antenna System

The HCMM VHF Antenna System will be of the modified turnstile type. Four quarter wave monopoles will be mounted on the lower cone structure of the spacecraft and will be fed in phase quadrature.

A coaxial hybrid and diplexer will feed both receiver and transmitter to a single antenna system. The antenna pattern is essentially omnidirectional in shape with generally circular polarization. Hybrid isolation between transmitter and receiver is about 25db.

3.6.5 S-Band Antenna

The S-Band Antenna will be a "Z" slot configuration on a hemispherical ground plane as shown in Figure 3-47. Overall size of mounting plate is about 4" in diameter with a 1" radius hemisphere. The antenna will be fed by a quadrature

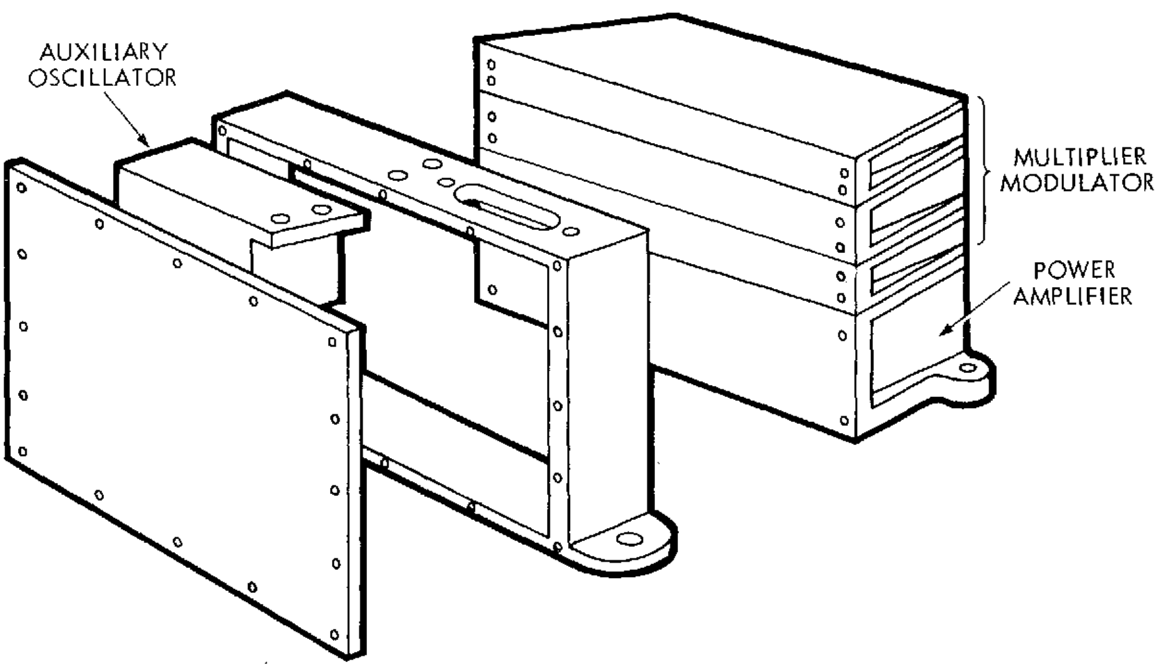


Figure 3-46. HCMM S-Band Transmitter

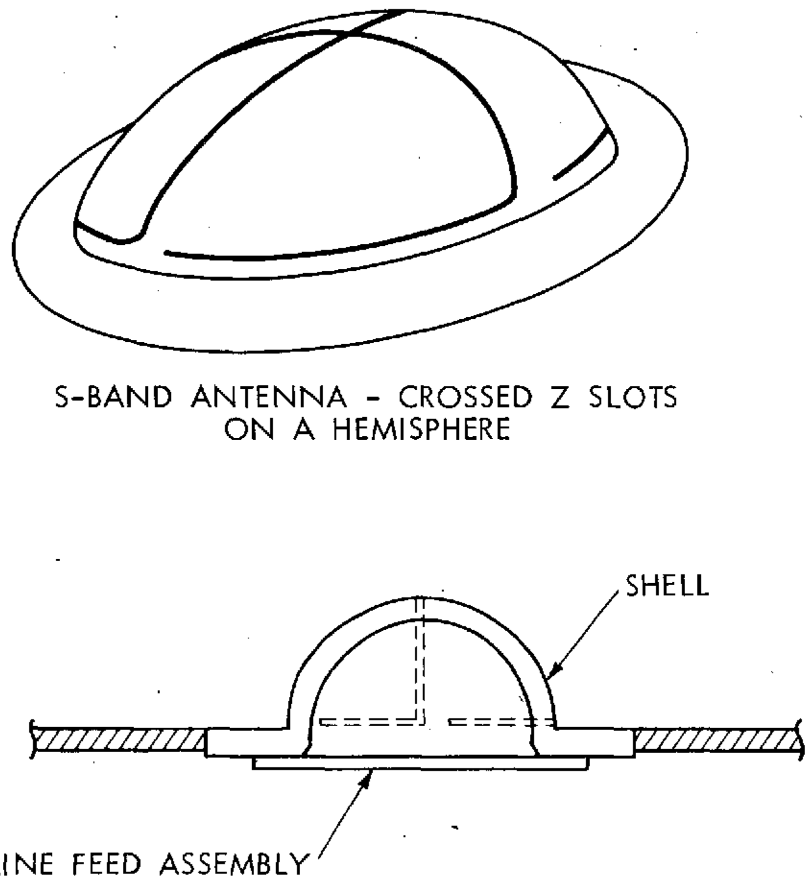


Figure 3-47. S-Band Antenna

hybrid to achieve circular polarization. The antenna will be mounted on the earth facing side of the spacecraft and have a maximum gain of +3 db and a 3 db beamwidth of about 160°. The antenna pattern is shown in Figure 3-48.

3.7 ATTITUDE CONTROL SYSTEM

3.7.1 Description

As always, choice of an attitude control system design is guided by a number of constraints, both technical and subjective, associated with the spacecraft program. For the AEM program the guidelines being used include the following points: Three-axis stabilization; earth-orientation; employment of existing components; low weight, power, and cost; and self contained operation. The performance objective for attitude control is nominally 1° about all three axes, with body rates of 0.01°/sec about all axes. The design, however, favors pitch and roll, whose errors will be a little less than that of yaw. The orbital reference axes for the control system are as follows: (1) Roll axis: The vehicle velocity

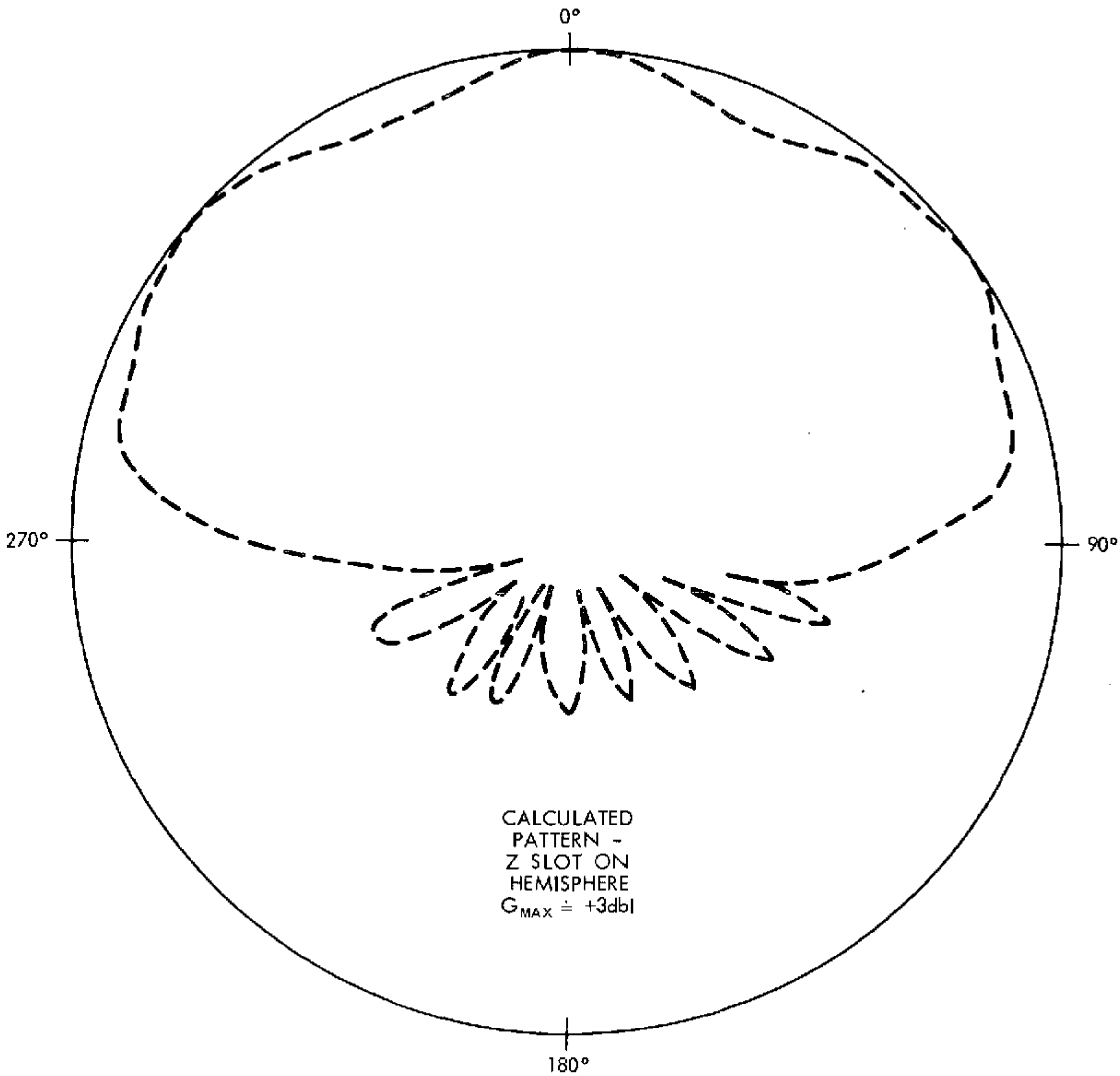


Figure 3-48. AEM S-Band Antenna Pattern

vector. (2) Pitch axis: Normal to the orbit in the opposite direction from the vehicle orbit rate vector; (3) Yaw axis - along the local vertical, positive toward the geocenter. These reference axes are earth-oriented and body attitude errors are measured with respect to these axes. The ACS schematic diagram is shown in Figure 3-49.

The attitude control concept for AEM is a combination of momentum bias and active reaction wheel control loops. The need for momentum orientation and unloading is satisfied by a magnetic torquing system. This momentum orientation and unloading is performed actively on-board the spacecraft.

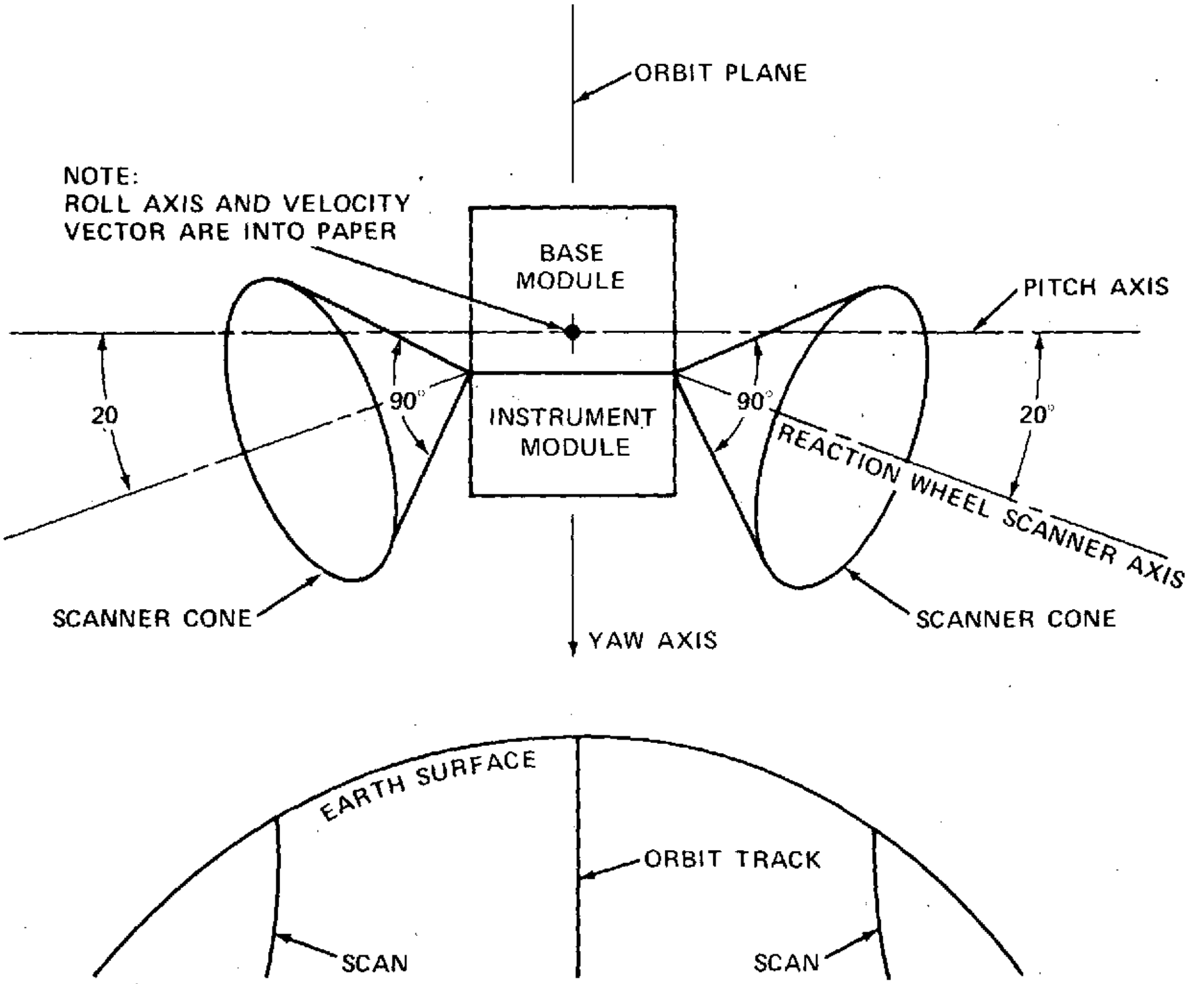


Figure 3-49. ACS Schematic Diagram

The AEM control system configuration is as follows: Two reaction wheel scanners have their axes of rotation (and scan axes) tilted in the pitch-yaw plane. The reaction wheels are operated about a bias speed which provides a net momentum bias along the spacecraft pitch axis. The orientation of the reaction wheels allows generation of control torques and storage of momentum along both the spacecraft pitch and yaw axes.

A separate reaction wheel may be added along the spacecraft roll axis to provide control torque and momentum storage along that axis for missions expected to experience larger environmental disturbances. The reaction wheel scanners provide attitude error sensing capability by employing the infrared horizon of the earth to generate attitude error signals about both the pitch and roll spacecraft axes. A three axis magnetic dipole moment generating system allows the control system to produce inertial control torques by generating a magnetic moment which interacts with the earth's magnetic field. The remaining system components include a three axis magnetometer and control logic electronics.

The control system operation about each axis is basically as follows: About pitch, the horizon scanner produces a pitch attitude error signal which is used to activate the pitch component of reaction wheel torque to reduce the error signal. The pitch component of wheel momentum is used to absorb periodic disturbances and temporarily store accumulated momentum. About roll, the control system operates similarly. While both pitch and roll control are active in nature in that an error signal is sensed and a control torque applied, control about the spacecraft yaw axis is primarily passive. No attitude error is sensed in yaw and no direct torque about the yaw axis is supplied to reduce yaw error. Yaw control is achieved passively by means of the system momentum bias tending to remain inertially fixed. Because of the orbit rate frequency interchange in attitude error between yaw and roll, yaw attitude errors may be limited by the roll control system. The yaw component of wheel momentum capability is used to temporarily store accumulated momentum. The 3-wheel system provides for active interchange of stored momentum between the roll and yaw axes without introducing spacecraft attitude errors. This capability allows a reduction in required momentum bias (and consequently weight and power) to achieve a given yaw accuracy, in the presence of a given disturbance torque environment. Three types of control system performance remain to be discussed: momentum unloading, nutation damping, and initial acquisition.

Disturbance torques acting on the spacecraft will, over time, accumulate momentum on the spacecraft. This momentum accumulation will appear as increased reaction wheel speeds or precession of the system momentum bias introducing a yaw error. The control system must be capable of exerting control torques to counteract the disturbance torques and unload any accumulated momentum. Since disturbance torques are external inertial torques, the control system must generate inertial control torques. The control system produces the desired control torque by generating a magnetic dipole to interact with the earth's magnetic field. The required dipole will be determined by the stored momentum levels, the spacecraft error signals, and the earth's magnetic field. The earth's field will be measured by use of a three axes magnetometer. It is the nature of magnetic torquing that the desired control torque may not be producible over a period of the orbit because of the local magnetic field. The spacecraft parameters are chosen so as to account for this phenomena. The magnetic momentum unloading system performs actively on-board the spacecraft.

A characteristic dynamic behavior of a control system with momentum bias system is nutation. Nutation is a coning motion of the spacecraft due to misalignment of the body rate vector from the system momentum vector. The system controls nutational motion by using the reaction wheel scanner roll error information to activate the roll and yaw torque components to damp the nutational motion. The active nutation damping employing existing control system components alleviates the need (and weight) of a passive nutation damper.

To this point, we have described control system performance in a fine control mode, i.e., a region about the control system null. The control system must also be capable of initial acquisition, i.e., proceeding from the initial conditions provided by the launch vehicle to motion about the control system null. Initial acquisition is performed as follows: Initial spin rate (100-200 rpm), typical of the Scout launch vehicle, is reduced by a Yo-Yo despin device to a few revolutions per minute. From this point, the control system actively magnetically despins the body to rates low enough to permit activation of the active control loops which allows the body to fully acquire the control system null attitude without saturating the momentum storage capability of the reaction wheels.

The ultimate performance capabilities for the control system involve the following factors. Weight and power limitations produce an upper bound for system bias momentum. Once this upper bound is reached, control system performance is a function of disturbance torque levels; to achieve maximum performance, it will be necessary to minimize disturbances torques. A lower limit of attitude pointing is ultimately placed on the control system by the accuracy of its error sensing devices. Improvement beyond this point requires improvement to the error sensing reaction wheel scanners. The performance objectives for attitude control are 1° about all three axes with body rates of 0.01°/sec about all axes. The error sensors to be used will not limit the performance in this case. The performance of the system is expected to be limited by the disturbance torques.

Initial acquisition performance is a function of the initial spin rate conditions. Despin for initial acquisition may require a number of hours, but in any case time to despin and capture should not exceed a day.

3.7.2 Subsystem Hardware

The hardware used to implement this control system is all flight proven equipment and fits well into the modular packaging concept. It should be noted, however, that while the majority of the control system electronics processing is flight proven, some of the yaw control law techniques entail new developments. The reaction wheel scanners are the Bendix-Ithaco scanners flown on both Nimbus D, Delta-Pac, Nimbus E and F, and ERTS A and B. The signal processing and control loop about the wheel axis are nearly identical to those of the mentioned spacecraft. The roll reaction wheel is the same as flown as the Nimbus D yaw wheel. The reaction wheels are sealed, high speed (1000 rpm) units capable of long life spacecraft performance. The signal processing and control logic electronics hardware are well understood and flight proven with no problems. The three axis magnetometer requirements are easily met by the Schonstedt triaxial flux-gate magnetometer, essentially the same model as flown on OSO, GEOS, and various sounding rockets.

3.7.3 Attitude Determination

Three axis attitude determination capability is inherent in the control system hardware. The attitude determination system is as follows: The reaction wheel scanners sensing the earth's infrared horizon will determine two axis attitudes (roll and pitch) from the local vertical. The accuracy of this hardware at this time is 0.25° for each axis. Attitude information about the local vertical (yaw) may be gained by using the magnetometer readings with knowledge of position in orbit and earth's magnetic field. Accuracy for attitude determination about yaw using this system may be 1° - 2°. Digital aspect sun sensors of the flight - proven Adcole type are employed to provide somewhat better yaw determination (0.5°) in daylight portions of the orbit. The sun sensors and magnetometer can provide three axis attitude determination capability during initial acquisition where the scanners may not provide attitude error information. The capability of such a system would be 1 to 2°, which is acceptable in the acquisition mode.

3.8 THERMAL SYSTEM

The thermal system is composed of two independent thermal subsystems - the instrument module and the base module. They are to be separate and distinct with thermal isolation between one another and provide individual thermal control. The rationale for thermal isolation lies in the modular concept for the program where the base module can be used interchangeably with any instrument module.

Different instruments mean differing powers, duty cycles, orbital environments and thermal requirements, all of which are unique to a particular mission and require individual thermal design for the instrument module.

The base module, on the other hand, will have common components and associated thermal requirements for all missions. With the base module isolated from the instrument module, the only variation the base module sees from mission to mission is a changing orbital environment. It is useful then to provide a base module thermal design that can be utilized for all proposed missions, thus eliminating base module re-design for each mission.

The base module design has been made common to all missions by minimizing the sensitivity to orbital environments using an active design composed of an insulated base module with thermal louvers. Insulation decouples the base module interior from varying orbital effects, while the thermal louvers provide a stable interval thermal environment by modifying base module dump capability as interval power varies. This provides adequate bulk temperature control for the base module.

Internal base module temperature distributions and gradients are controlled by utilizing thermal control paints and coatings on all components and structure. An alternative approach using grooved wall heat pipes can be adopted if required. Thermal greases, indium or other heat sink materials can be employed under high powered components if necessary.

The bimetallic actuated thermal louver system is located on the bottom of the base module where a maximum unobstructed view of space is available. Because periodic solar illumination is unavoidable, a low absorbing highly specular base (i.e., under the louver blades) such as silverized teflon becomes necessary to minimize adverse effects on louver performance.

The independent temperature control of the instrument module will be dictated by the thermal requirements of the instrument. Depending on requirements, the design can be active utilizing heat pipes and louvers, passive using paints, vapor deposited coatings and multilayer insulation blankets, or a combination of both. The design can be tailored such that the absorbed energy from all orbital inputs coupled with the internal power dissipation can be regulated by the net power dumped to the space sink to provide an acceptable temperature environment for all instrument components. Internal temperature distributions can be modified in a manner similar to the base module (i.e., using high emittance coatings, heat sink materials and heat pipes if required).

3.9 ORBIT ADJUST SYSTEM

An option being considered for the AEM spacecraft is an orbit adjust propulsion system. The system is a relatively simple single-thruster one.

The system weight fully loaded is estimated to be 8.43 kg (3.63 dry, 4.80 N_2H_4 & GN_2). This is based on a straightforward, simple design with redundancy only in the propellant valve. The design incorporates a single 0.5 lbf thruster under the assumption that the spacecraft can be oriented in yaw 180° from the normal mission orientation. This allows thrusting along or against the velocity vector. The initial thrust vector misalignment error can be initially set within 1/2 degree of a known spacecraft center of gravity location. The change in center of gravity with propellant usage and the effect on disturbance torques must be determined for each application.

The design is based on a three-axis stabilized spacecraft weighing 117 kg without the propulsion system. The system is required to provide 200 km altitude change (70 m/s velocity change) to circularize the orbit in the three-axis stabilized mode. The following assumptions apply to the design:

- Thrust vector alignment error with respect to spacecraft center of gravity = 1/2°.

- Total impulse per orbit = 9.1 lbf-second (based on 1/2° thrust vector misalignment).
- Maximum thrust (start of blowdown) = 0.50 lbf.
- Minimum thrust (end of blowdown) = 0.25 lbf.
- Approximate firing time per orbit = 18 seconds.
- Use qualified off-the-shelf components to minimize cost.
- Use available propellant loading cart (RAE-B, ERTS, ATS).
- Design for minimum weight.

Figure 3-50 is a block diagram of the system. Figure 3-51 shows the thruster. Table 3-8 is a weight breakdown estimate for the system.

3.10 LAUNCH VEHICLE SYSTEM

The AEM launch vehicle will be the Scout-F, a four-stage, solid-fuel rocket system procured from Ling-Temco-Vought (LTV) Aerospace Corporation. The propulsion motors are arranged in tandem with transition sections between the stages to tie the structure together and to provide space for instrumentation and controls as shown in Figure 3-52. Data for the launch vehicle are given in Table 3-9.

In addition to strapped-down gyro sensors the guidance and control system contains a relay unit for power and ignition switching; an intervalometer to provide precise scheduling of events during flight; a programmer to provide torquing voltages to the pitch or yaw, an electronic-signal conditioner to convert the gyro outputs to proper control signals; and the associated 400-cycle inverter and dc batteries.

3.10.1 Guidance and Control

The guidance and control system provides an attitude reference and the resultant control signals and forces necessary for stabilizing the vehicle in its three orthogonal axes (corresponding to pitch, yaw, and roll).

Normally, the yaw and roll axes are maintained at the launch reference and the pitch axis is programmed through a pre-selected angle corresponding to the desired vehicle trajectory.

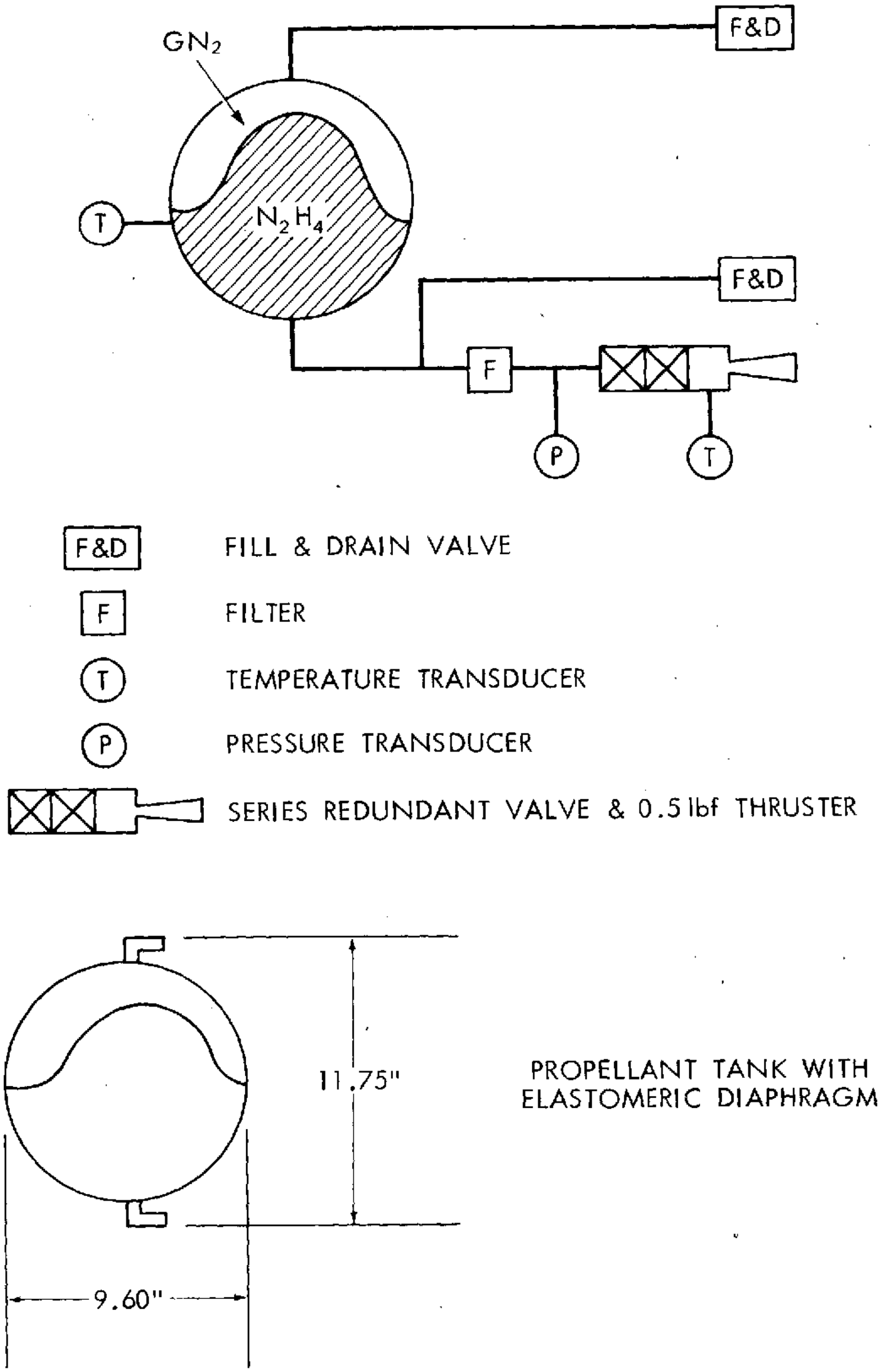


Figure 3-50. Propulsion System Block Diagram

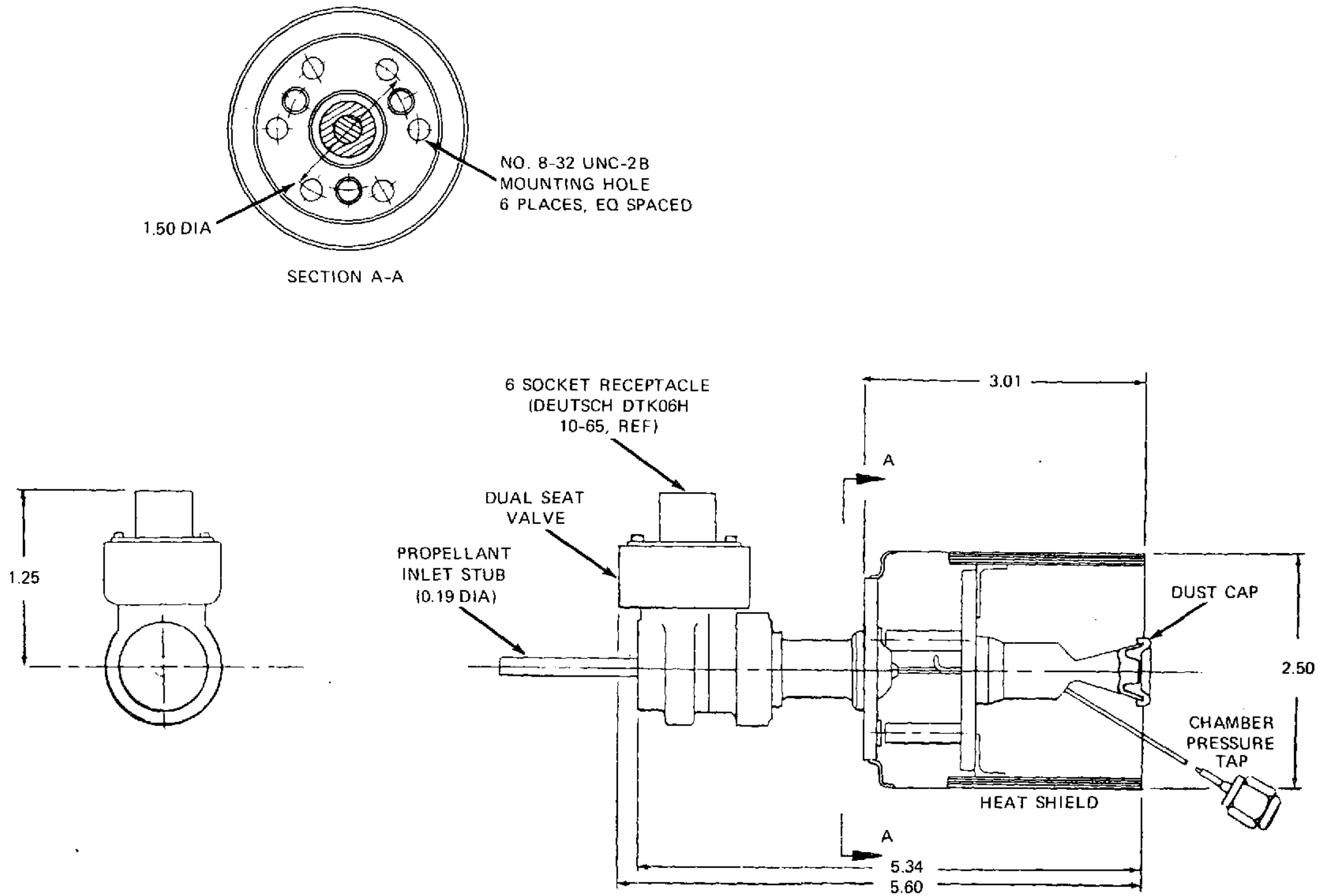


Figure 3-51. Thruster and Valve

Table 3-8

Propulsion System Weight Breakdown

Material Description	Unit Wgt (lbm)	Qty	Total (lbm)	Total Kg
Thruster & Valve	0.90	1	0.90	
Fill Valve	0.30	2	0.60	
Filter	0.10	1	0.10	
Pressure Transducer	0.25	1	0.25	
Temperature Transducer	0.20	2	0.40	
Connector & Wiring	1.00	1	1.00	
Lines & Fitting	0.50	1	0.50	
Tube Support & Brackets	0.25	1	0.25	
Miscellaneous Hardware	1.00	1	1.00	
Tank	3.00	1	3.00	
Dry Total			8.00	3.63
N_2H_4			10.50	4.75
GN_2			0.10	0.50
TOTAL			18.60	8.43

Note: Mounting structure if required.

The first stage is controlled by a combination of jet vanes and aerodynamic tip control surfaces. The jet vanes provide most of the control force during the thrust phase; the aero-dynamic tip controls provide all the control force during the coast phase following burnout of the first-stage motor.

The second and third stages are controlled by hydrogen-peroxide reaction jets operated as an on-off system.

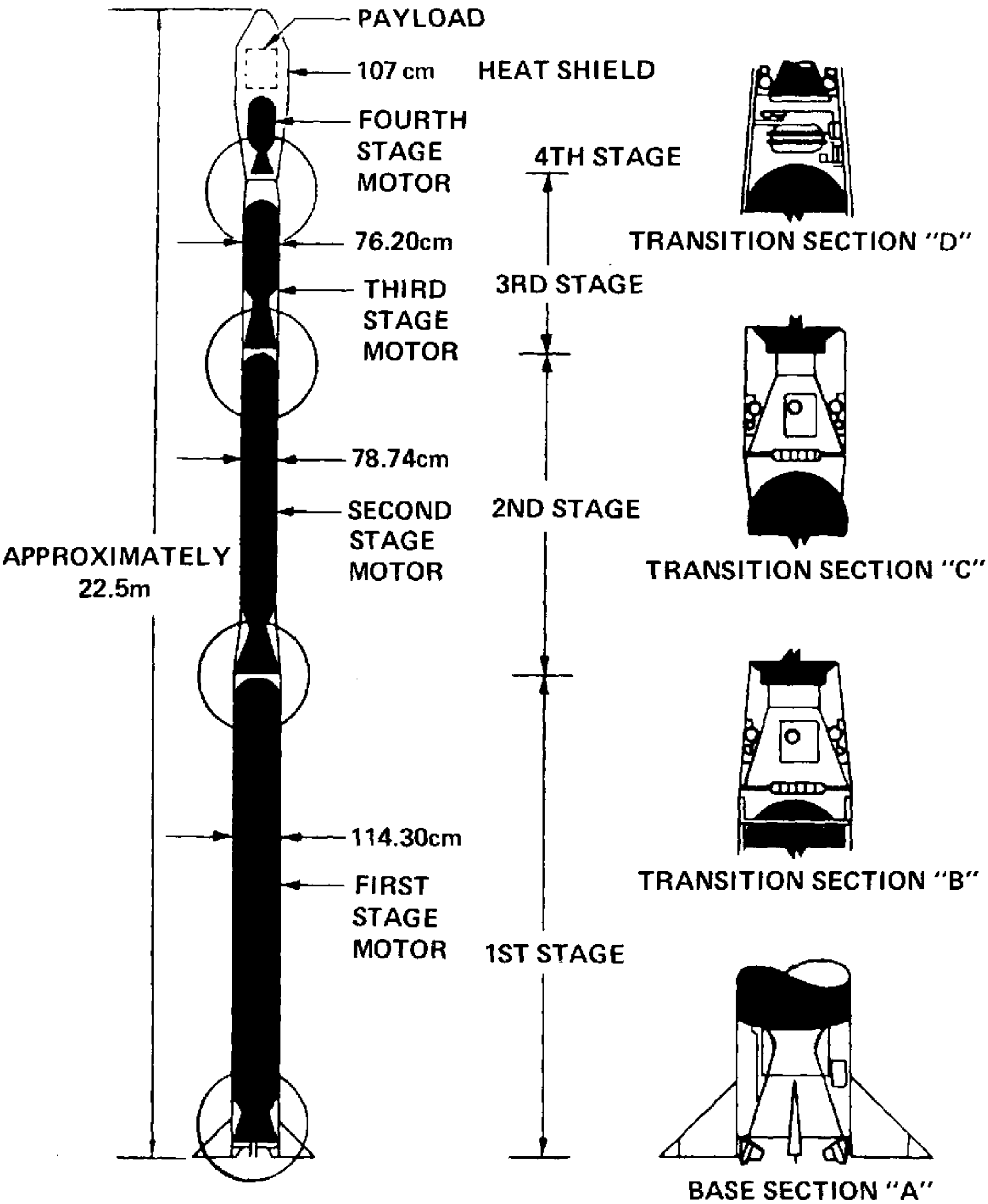


Figure 3-52. Scout Launch Vehicle

Table 3-9

Scout Launch Vehicle Data

Scout Launch Vehicle Data	First Stage Algol III	Second Stage Castor II	Third Stage Antares II	Fourth Stage Altair III (FW-4S)
Impulse (Newton-sec)	32,142,784	10,304,114	3,257,937	762,972
Weight, Total (Kg)	14,175	4,433	1,272	301

The fourth stage, which includes the payload, receives proper spatial orientation from the control exerted by the first three stages; subsequently, it is spin-stabilized by a combination of four impulse-spin motors.

3.10.2 Stage Separation

The Scout vehicle's four rocket motors are joined by interstage structures referred to as "transition sections." (See Figure 3-52.) Each transition section is divided into lower and upper portions at the stage separation plane. A frangible-diaphragm separation system is used in both transition section B and transition section C; a spring-ejection system is used in transition section D.

3.10.3 Spacecraft Separation

A separation system timer is available to initiate the separation sequence of the spacecraft. Sufficient time is allowed after fourth-stage burnout to ensure that any residual burning of the fourth-stage propellant will not cause additional acceleration. A spring ejection system separates the spacecraft from the fourth stage motor case.

3.10.4 Radio Command Destruct System

The UHF radio command destruct system is provided as a means of destroying the vehicle should a malfunction cause the vehicle to present a hazard to inhabited land areas or navigation and commerce. The destruct system is designed to comply with the requirements of the Test Range. The vehicle has two completely independent destruct systems consisting of antennas, power supplies, command receivers and pyrotechnics. The system is compatible with the two command destruct transmitters which have a nominal r-f output of up to 1 kw. The actual destruct command requires the command destruct transmitter to be modulated by three IRIG channels in the proper sequence, thus reducing the probability of an inadvertent destruct command as well as the probability of extraneous signals causing destruct.

3.10.5 Telemetry System

The telemetry system of the Scout vehicle is a standard IRIG PAM/FM/FM system capable of handling 18 standard IRIG subcarrier channels. The normal vehicle operating data are obtained. The vehicle telemetry transmitter operates in the 225-260 MHz band. The nominal r-f power output is seven watts delivered to two diametrically opposed external quarter-wave stubs located in lower "D" section.

3.10.6 Radar Tracking Beacon System

The radar beacon has a nominal peak r-f power output of 500 watts, single pulse. The beacon antenna is an H-plane sectoral horn mounted externally on lower "D" section.

Figures 3-53 and 3-54 show the Scout capabilities for circular orbit performance for a two and three paddle AEM configuration.

3.11 MISSION OPERATIONS

This section provides descriptive material for HCMM mission control. A summary of the planned ground support is provided and the control center and mission management concepts are described in detail. Though the Tracking and Data Acquisition (Section 3.12) and Data Management Plan (Section 3.13) functions are an integral part of Mission Operations, and, as such are discussed in this section, specific areas of responsibility of the Mission and Data Operations and Networks Directorates are delineated in Sections 3.12 and 3.13.

3.11.1 Ground Support Plan

The HCMM Project Operations Control Center (POCC) will be the focal point for all project unique mission operations beginning with the pre-launch simulations, through launch, and throughout the mission lifetime. The ground support plan is described to provide an understanding of the interfaces between the HCMM POCC and the other ground support systems.

The ground support concept includes the capability for real-time spacecraft health evaluation; spacecraft command and control; capability for real-time evaluation of the acquired instrument data by the instrumenters for planning of the operation of the instruments; and a data management system that provides timely data handling, processing, and distribution to the instrumenters. Figures

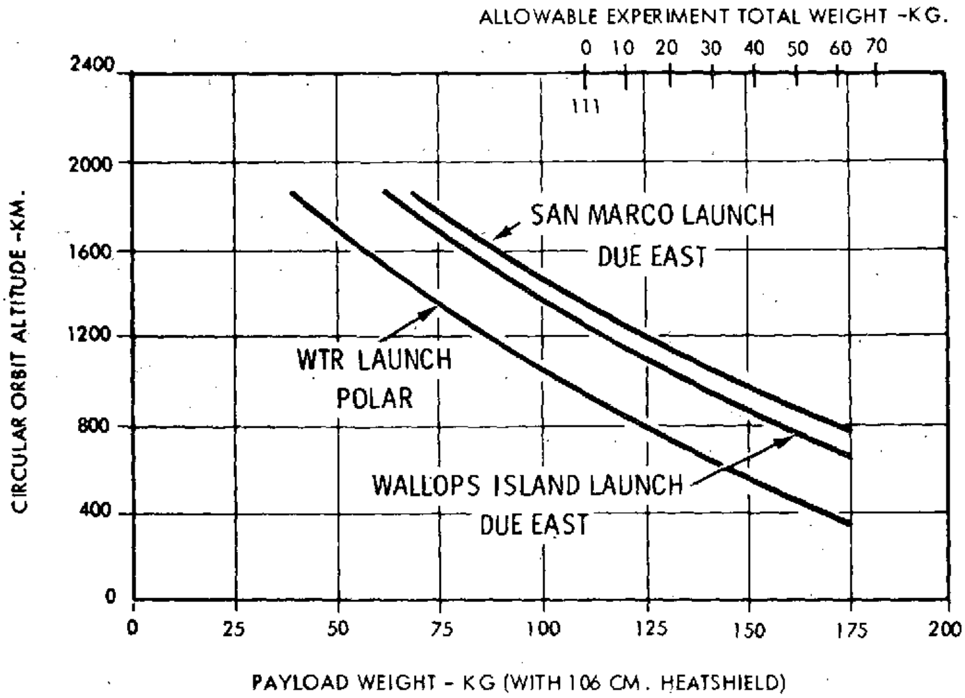


Figure 3-53. Two Paddle Configuration, Circular Orbit Performance with Scout F Using 106 cm Heatshield.

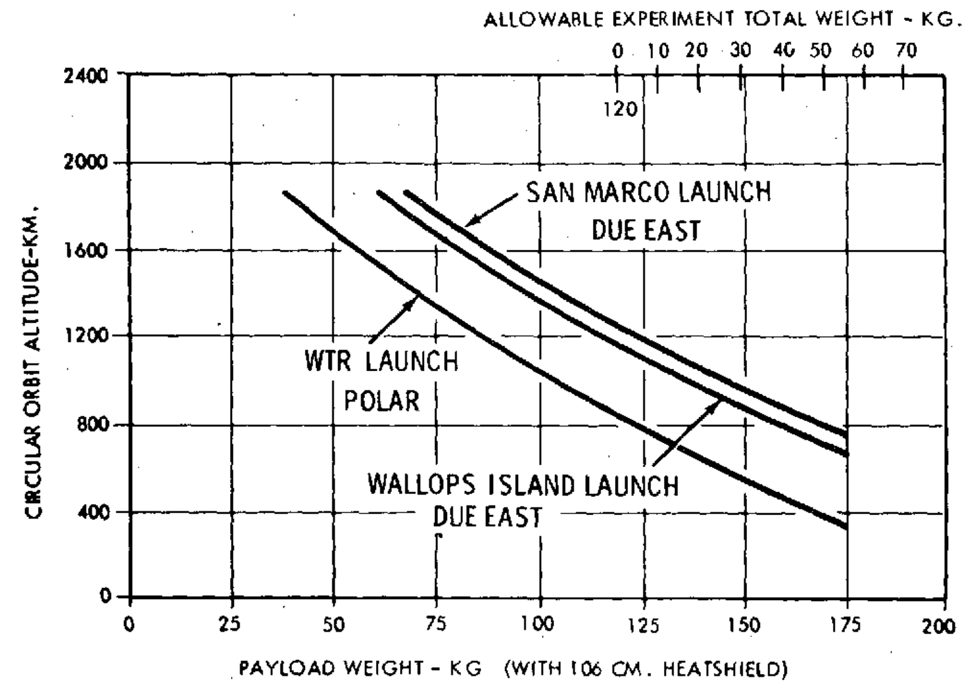


Figure 3-54. Three Paddle Configuration, Circular Orbit Performance with Scout F Using 106 cm Heatshield.

3-55, 3-56, and 3-57 are functional block diagrams that show the proposed HCMM ground support system, data flow, and management concepts.

3.11.2 Mission and Network Scheduling and Control

The HCMM POCC is responsible for developing mission and operations control network support schedule requests and forwarding them to the network scheduling and control activity. This activity, Figure 3-55, includes the Mission Scheduling Operations Center (MISSOC) and the Network Operations Control Center (NOCC). The HCMM POCC forwards all network support schedule requests through MISSOC who integrates it with the mission support requests of all the POCCs into a single support request and forwards it to the NOCC who integrates it with all other network support requests and schedules the network stations. Required network support is normally scheduled on a weekly basis (emergencies as required) commensurate with spacecraft priorities that are established for the network.

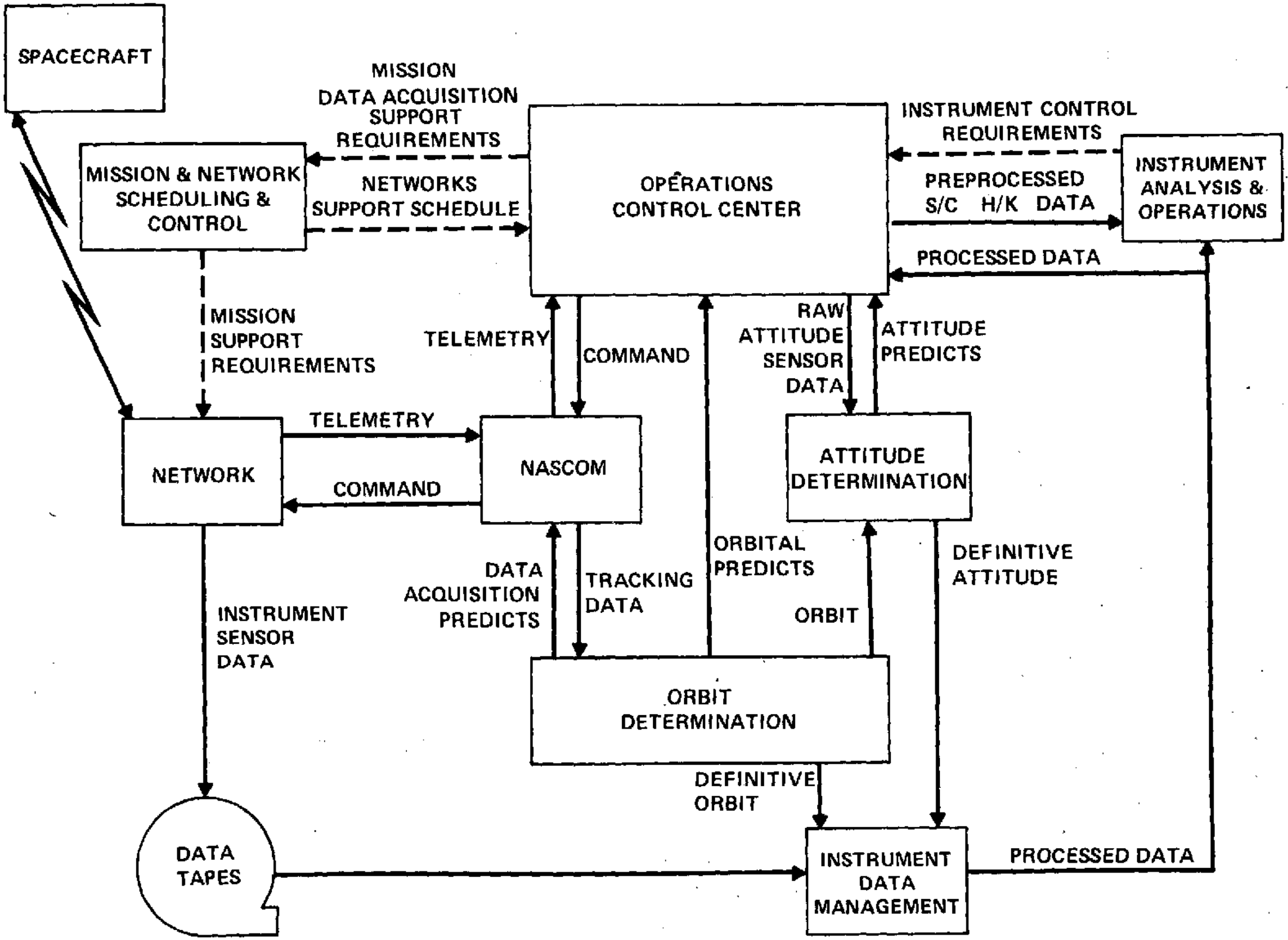
3.11.3 Orbit Determination and Attitude Computations

The operational orbit determination function takes the operational tracking data from the GSFC STDN stations and determines the necessary operational spacecraft orbit data. The attitude determination function is one of taking the telemetered spacecraft attitude sensor data along with spacecraft orbit data and generating spacecraft attitude as a function of time. This capability exists at GSFC as a general support capability that can provide both attitude predictions (future spacecraft attitude) and definitive attitude information (after-the-fact-spacecraft attitude).

Requirements for the HCMM mission are within the capabilities of existing GSFC orbit determination programs.

Prior to launch, a support plan will be formulated providing for: (a) early orbit computations, (b) mission phase orbit computations for project support and mission scheduling, and (c) definitive orbit ephemeris generation. Preparations will include tuning the orbital support system to the specified requirements of the mission and the generation of aids to conduct various mission scheduling simulations.

Following launch, orbits will be updated as required to provide a reasonably accurate basis for mission planning and scheduling. Orbits will be refined if necessary to meet project accuracy requirements in the definitive ephemeris data.


SPACECRAFT
MISSION & NETWORK SCHEDULING & CONTROL
MISSION DATA ACQUISITION SUPPORT REQUIREMENTS
NETWORKS SUPPORT SCHEDULE
OPERATIONS CONTROL CENTER
INSTRUMENT CONTROL REQUIREMENTS
INSTRUMENT ANALYSIS & OPERATIONS
PREPROCESSED S/C H/K DATA
PROCESSED DATA
MISSION SUPPORT REQUIREMENTS
TELEMETRY
COMMAND
RAW ATTITUDE SENSOR DATA
ATTITUDE PREDICTS
NETWORK
TELEMETRY
COMMAND
NASCOM
ATTITUDE DETERMINATION
INSTRUMENT SENSOR DATA
DATA ACQUISITION PREDICTS
TRACKING DATA
ORBITAL PREDICTS
ORBIT
DEFINITIVE ATTITUDE
ORBIT DETERMINATION
DEFINITIVE ORBIT
DATA TAPES
INSTRUMENT DATA MANAGEMENT
PROCESSED DATA

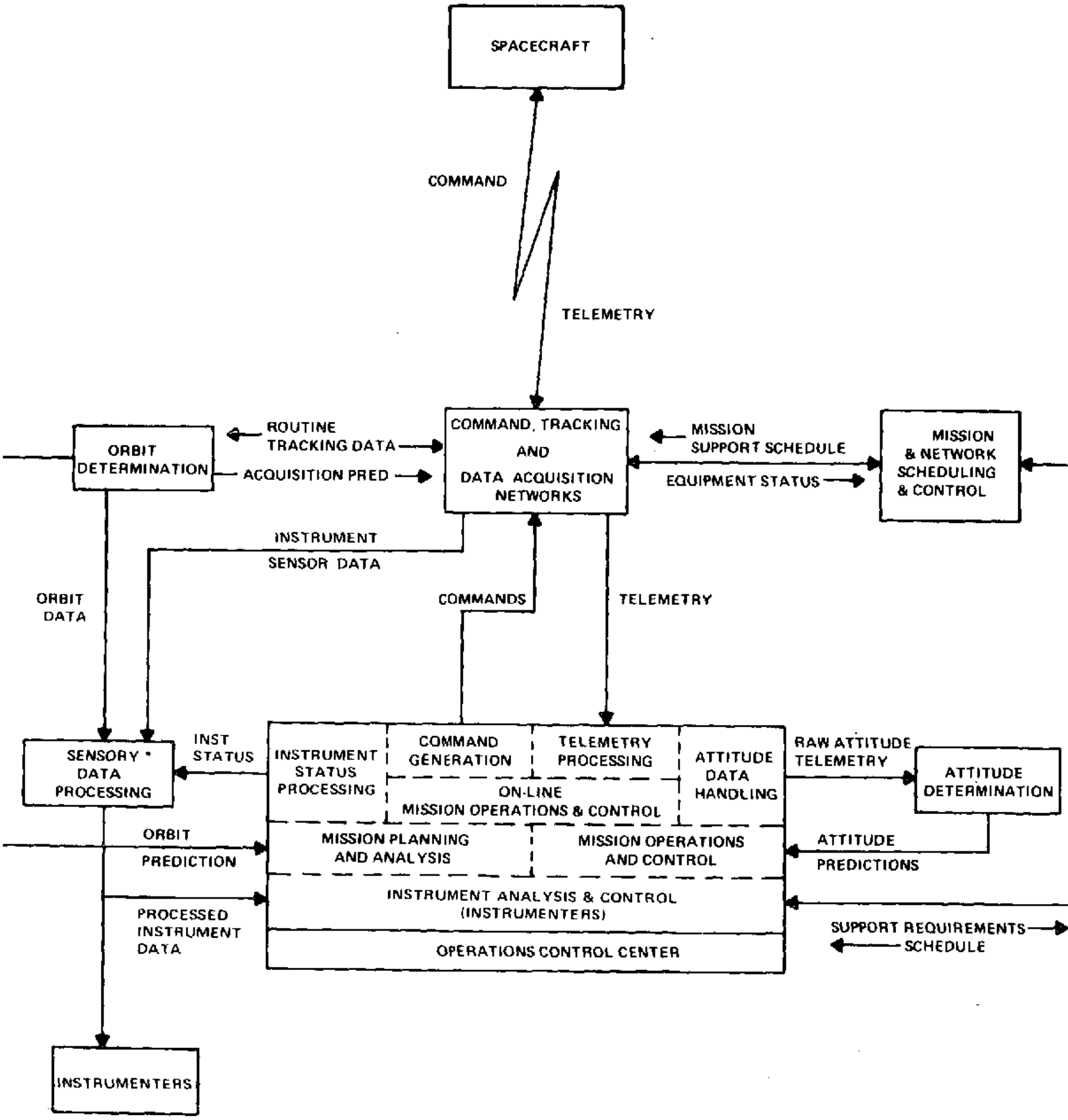


Figure 3-56. Ground System Functional Interfaces

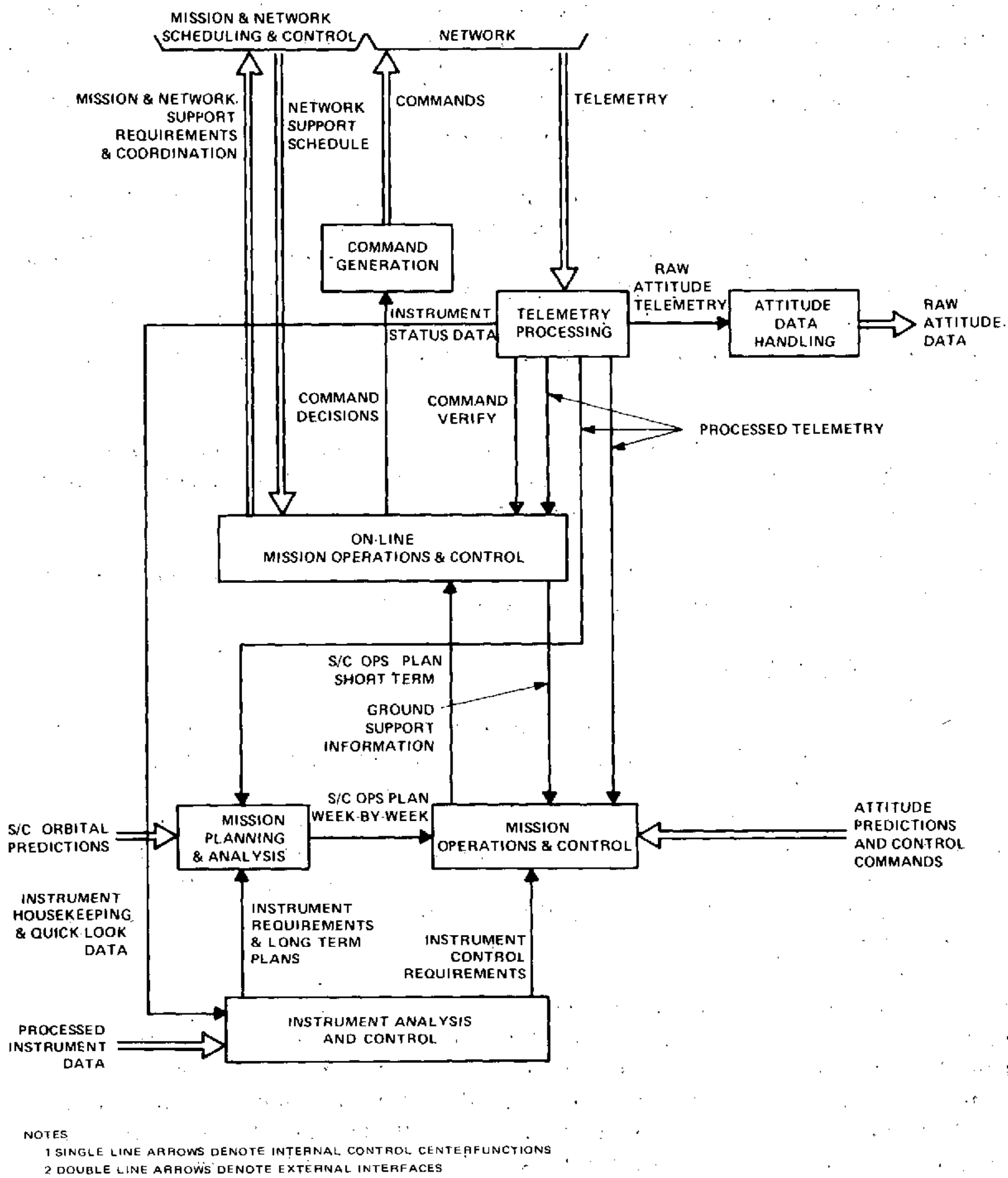


Figure 3-57. Operations Control Center Data Management Interface

The attitude determination function is considered to be a control center function, but since a large computer is required, the support is obtained from a time shared general support computer facility. The processed attitude information will then be transmitted back to the POCC for command and control of the HCMM spacecraft. The POCC will also receive orbit data from the orbit determination areas. The orbit data will be utilized to plan spacecraft operations and to schedule ground stations for operational and mission support.

The spacecraft will be stabilized in earth orientation mode at one revolution per orbit. The HCMM will use inertia wheels in an on-board closed loop system. This system will require little or no ground support once the mission configuration has been achieved. The attitude will be monitored during the initial acquisition phase using sun sensor and magnetometer data. Necessary control support will be provided during this time to ensure that the desired mission configuration is achieved.

In addition to the above functions, definitive attitude determination will be provided. The definitive requirement will include the generation of 3-axis attitude results to be provided to the HCMM Instrumenter.

All orbit and attitude determination support requirements are the responsibility of the M&DOD and are within existing M&DOD capability.

3.11.4 Communications

The responsibility for determining the type and amount of communications required to support the HCMM Project resides with the Mission Operations Systems Manager (MOSM). The responsibility to ensure implementation and operation of the required support is assigned to the Network Support Manager (NSM) of the Networks Directorate. The communications function will be supported by NASCOM which provides communications support to the STDN Network sites under the cognizance of the Network Directorate. This function includes: transmission of real-time and near real-time spacecraft housekeeping data from the ground stations to the HCMM POCC; real-time coordination of commands and, if required, real-time command capability from the HCMM POCC to the ground stations; transmission to the HCMM POCC of selected real-time or near real-time instrument data for real-time or near real-time instrument evaluation and control; transmission of routine tracking data from the tracking stations to the orbit determination areas for generation of the HCMM operational orbits; transmission of operational orbit predictions to the ground stations for spacecraft data handling operations between the HCMM POCC, Network Control, NASCOM control, and those supporting ground stations that are in the STDN Network.

There are no special communications support requirements unique to HCMM. All required support is within the capability of existing or planned facilities during the proposed HCMM lifetime.

3.11.5 HCMM POCC Implementation

Based on the HCMM requirements as understood to date from participation of the Mission Operations Systems Management team in the HCMM study working group, HCMM requirements were categorized as being among the type normally supported by the GSFC Multi-Satellite Operations Control Center (MSOCC). MSOCC's GEOS-C POCC support termination is scheduled nearly coincident with the date of initiation of HCMM POCC implementation. As such, the bulk of both the GEOS-C POCC facilities and POCC O&M personnel would be available to support the HCMM mission control requirements.

A study of the MSOCC support workload, inclusive of computer support, during the HCMM spacecraft lifetime was conducted and it was concluded that HCMM can be adequately supported in the MSOCC.

Mission control center capabilities and equipments for the GSFC control centers are detailed in GSFC X-530-70-454, Space Tracking and Data Acquisition Manual where MSOCC capabilities are described in Section 5.3.2. An updating of this publication is in process.

3.12 TRACKING AND DATA ACQUISITION

3.12.1 Network Support

Network support is based on the HCMM spacecraft being compatible with existing or planned network facilities. Additionally the HCMR instrument will require the use of Nimbus Ground Station equipment located at Madrid (MAD), Alaska (ULA), Orroral (ORR), Rosman (ROS), Goldstone (GDS). All tracking, command, and data acquisition functions will be supported by the GSFC STDN stations under the cognizance of the Networks Directorate. This will provide adequate support of all HCMM requirements.

3.12.2 Range Support

Range support for tracking data and/or monitoring of flight event parameters (launch to spacecraft separation) will be within the normal depth of launch support.

3.12.3 Telemetry

The GSFC Network will support the HCMM spacecraft VHF telemetry and S-band telemetry downlinks with the facilities listed in Table 3-10.

Specific Network Support will be fully defined when all support requirements are known. However, the HCMM telemetry system will utilize NASA data standards and be compatible with all network stations.

3.12.4 Command

Since the spacecraft will fly a VHF command system, Network command support can be provided by each station or a nearby alternate station.

The Network will be equipped with the Spacecraft Command Encoder (SCE) system for the HCMM mission with which the HCMM command system will be compatible.

3.12.5 Tracking

Network tracking will be provided by the Minitrack stations as required.

3.13 DATA MANAGEMENT

3.13.1 Instrument Data Processing

The Information Processing Division (IPD) being the central telemetry data facility for GSFC's scientific and applications spacecraft will serve the processing requirements for the HCMM mission telemetry and special purpose data recorded by the Spaceflight Tracking and Data Network (STDN). All such data will be forwarded to the IPD for subsequent reduction.

Telemetry signal processing operations for the HCMM project will include analog tape evaluation and analog-to-digital conversion. Additional data reduction operations will be performed on a UNIVAC 1108 or IBM 7010 computer. After initial computer processing, a decommutation program is used to separate spacecraft housekeeping data and instrument data. HCMR data will be sent to IPD or the Nimbus Ground Station at GSFC for quicklook processing for quality prior to selective data reduction.

Instrument data analysis is performed by the investigator except in unusual instances which have prior management approval. Instrument data will be processed and distributed by IPD according to project requirements as outlined in the HCMM Support Instrumentation Requirements Document (SIRD).

Table 3-10

GSFC STDN Facilities

	VHF			S-BAND		
	TRKG	TLM	CMND	TRKG	TLM	CMND
ULA	*X	X	X	X	X	X
ACN		X	X	X	X	X
BDA (1)		X		X	X	X
CYI				X	X	X
GDS	**X	X	X	X	X	X
GWM		X	X	X	X	X
HAW		X	X	X	X	X
MAD				X	X	X
WNK	X	X	X			
AGO	*X	X	X	X	X	X
MIL		X	X	X	X	X
ORR	*X	X	X	X	X	X
QUI	X	X	X	X	X	X
ROS	**X	X	X	X	X	X
TAN	*X	X	X	X	X	X
NTTF (2)	**X	X	X	X	X	X
VAN (2)				X	X	X

*MINITRACK AND RANGING
**RANGING ONLY

(1) LAUNCH SUPPORT ONLY
(2) SPECIAL PURPOSE SITE

Dissemination of processed instrument data will be done in accordance with investigator requirements.

The data requirements of the HCMM are consistent with the capabilities of the Networks Directorate Data Processing Systems (DDPS) and Mission and Data Operations Directorate Telemetry On-Line Processing System (TELOPS). HCMM data requirements will therefore be considered a candidate for this system combination.

3.13.2 Spacecraft Data Processing

All spacecraft health data will be processed as described above and will be disseminated as required as shown in Figures 3-55, 3-56, and 3-57.

4.0 MISSION PLANNERS GUIDE

This section is intended to aid the instrumenter in determining the applicability of the AEM spacecraft to the applications instrument. Refer to subsystems descriptions where needed.

4.1 ORBIT

Define the needed orbit. As a first cut determine:

a. Altitude

b. Inclination

c. Circularity constraints

d. If sun synchronous, the constraints on local time of ascending node, drift/month

e. Needed operational lifetime

4.2 ENVELOPE

Using the module concept determine if the envelope is within the constraints of the Scout heatshield. Refer to Figure 3-3.

4.3 WEIGHT

Next, determine if the envisioned spacecraft can be placed in the desired orbit by the Scout vehicle. This section is the first cut at determining this.

To determine the Scout Mission Capability, the total payload weight including the launch vehicle adaptor must first be determined. The base module has a number of options. These include a solar paddle configuration of 2 or 3 paddles depending on orbit inclination and power required. This is discussed in detail in the power section in the book. For general planning purposes refer to Table 4-1.

Another option is the addition of a third momentum wheel (in addition to the two reaction wheel scanners). Table 4-2 gives the control characteristics of the 2 and the 3 wheel systems.

Table 4-1

Solar Paddle Options

No. of Paddles	Orbit Application	System Weight (kg)
2	Sun synchronous with equator crossing times from 3 to 6 p.m. and 6 to 9 a.m.	21.0
3	Non-sun synchronous at all inclinations and sun synchronous with equator crossing times from 9 a.m. to 3 p.m.	28.7

Table 4-2

ACS Options

No. Wheels	Control Characteristics	System Weight (kg)
2	$\pm 1°$ pitch & roll, $\pm 2°$ yaw	16.5
3	$\pm 1°$ pitch & roll, $\pm 2°$ yaw (third wheel required dependent on altitude and S/C configuration)	19.2

A third option is the use of a spacecraft tape recorder for digital data up to 40kbps. Provisions are made for the base module to house two of these GSFC standard 10^8 bit recorders. One or two recorders may be flown. Each recorder with its electronics weighs 3.5kg.

The fourth option is concerned with the degree of tracking accuracy required. The standard system employs the VHF transmitter with the STDN Minitrack System. If greater accuracy requires the range and range-rate system, this can be done by deleting the S-band transmitter from the base module and replacing it with an S-band transponder.

A fifth option is the use of an orbit adjust system to be used for orbit altitude changes.

These are the standard options available in base module. Any other unique equipment needed by the instrument must be provided by the instrumenter in the instrument module.

The following weight breakdown worksheet can be used to determine the approximate payload weight.

<u>Base Module</u>	<u>(kg)</u>
Structure	12.6
Thermal	2.8
Cable Harness and Miscellaneous Brackets	7.4
Telemetry	
Standard (2.7)	
Additional Subplexer (0.7)	( )
Tape Recorder (4.5)	( )
Command	
Standard (4.3)	
Mission Operations Electronics (1.0)	
Redundant Decoder (1.1)	( )
Communications	
Standard (4.1)	
Redundant Receiver (0.6)	
S-band Transponder (3.0)	( )
Power	
Two-paddle (21.0)	
Three-paddle (28.7)	( )

	(kg)	
Attitude Control		
Two-wheel (16.5)		
Three-wheel (19.2)	( )	
Orbit Adjust (8.4)	( )	
Base Module Subtotal	( )	
Contingency (10%)	( )	
Base Module Total	( )	
Instrument Module		
Structure	5.0	
Thermal	1.5	
Instrument	( )	
List Supporting Hardware		
	( )	
	( )	
	( )	
Instrument Module Subtotal	( )	
Contingency (10%)	( )	
Instrument Module Total		( )
Launch Vehicle Adaptor (Scout F)		11.9
Total Payload Weight		( )

Scout performance is shown in Figures 4-1, 4-2, and 4-3. Figure 4-1 indicates the deliverable payload capabilities from the Wallops Station (WS) launch site into a 37.5° inclined orbit and Figure 4-2 from Vandenberg Air Force Base (VAFB) into a polar orbit.

Figure 4-3 shows the effects of requiring other orbit inclinations out of WS and Figure 4-4 out of VAFB. This relationship will be referred to as a weight penalty from the due east or polar orbits.

Once the payload weight (including vehicle adaptor, etc.) is determined, select the inclination and refer to Figure 4-3 or 4-4 and determine the inclination penalty. Add that penalty to the payload weight. Using that sum in reference to Figure 4-1 or 4-2, determine the achievable orbit.

4.4 POWER

Power availability depends on:

a. Options chosen for S/C, e.g., tape recorder, ACS, etc.

b. Type of operation - frequency of tape recorder dumps, S-band transmitter use, etc.

c. Orbit:

   Sun synchronous:

   - 2 paddle for twilight and near-twilight (3 p.m. to 6 p.m. and 6 a.m. to 9 a.m. ascending node times).

   - 3 paddle for 9 a.m. to 3 p.m. ascending node times.

   Non-Sun synchronous

   - Power is greatly dependent on orbit inclination. However consider the three paddle case to be the general case.

Included as an appendix is the detailed study for array capability. However, the user should probably not attempt to perform detailed power calculations. He is advised to use 30 watts (orbit average) as the available power for instrument use.

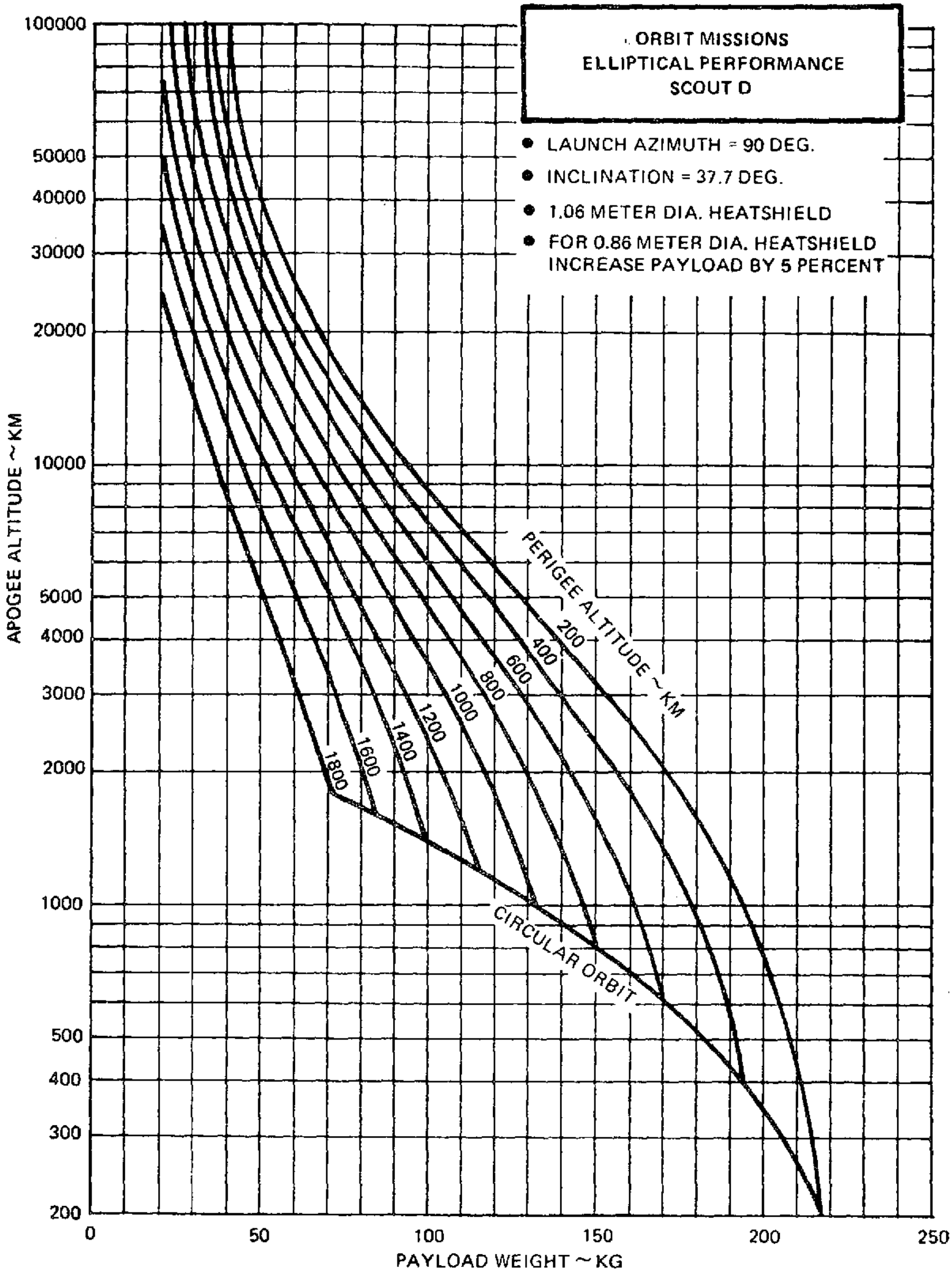


Figure 4-1. Elliptical Orbit Performance - Wallops Island

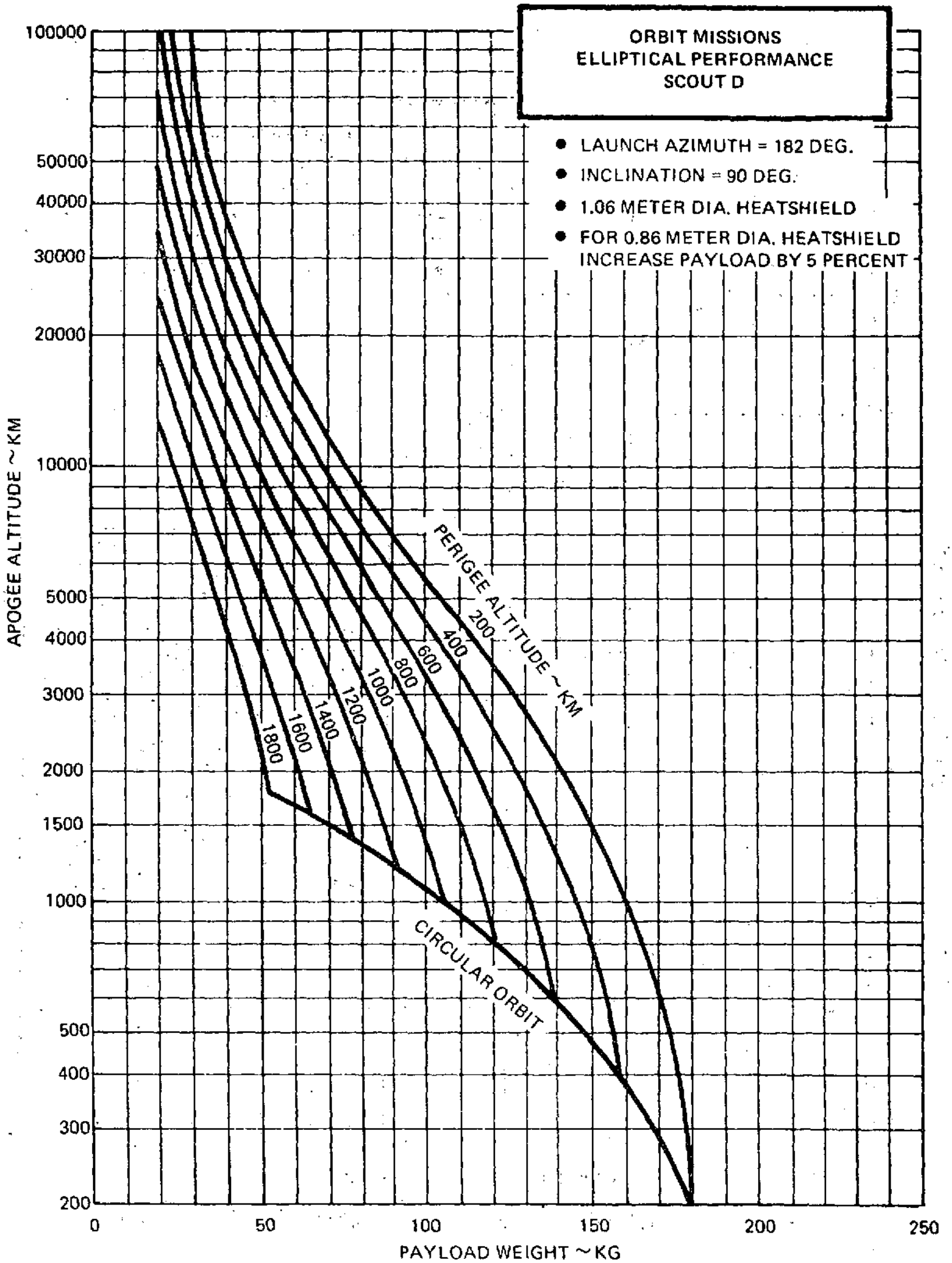


Figure 4-2. Elliptical Orbit Performance - Vandenberg AFB

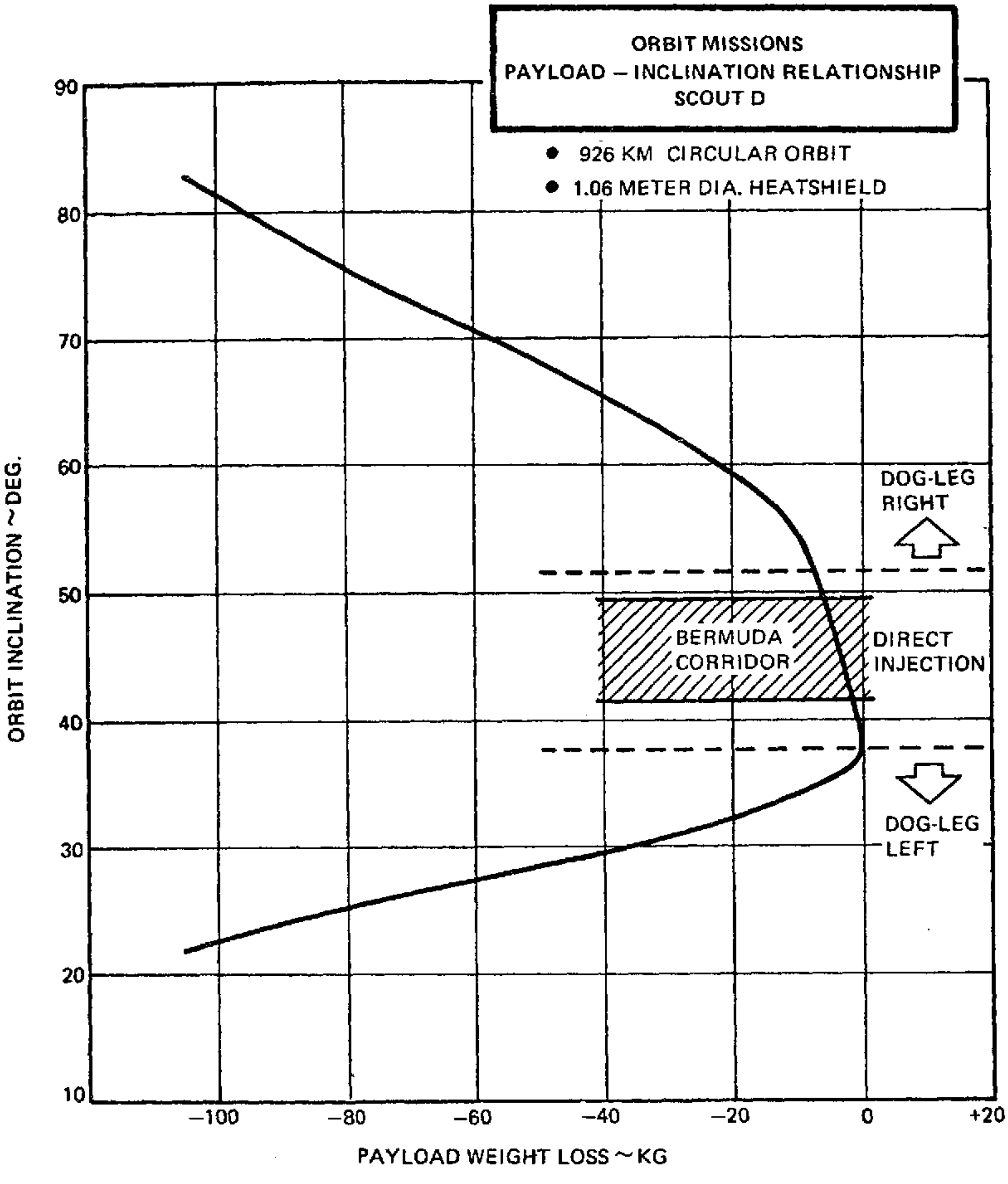


Figure 4-3. Payload-Inclination Relationship - Wallops Island

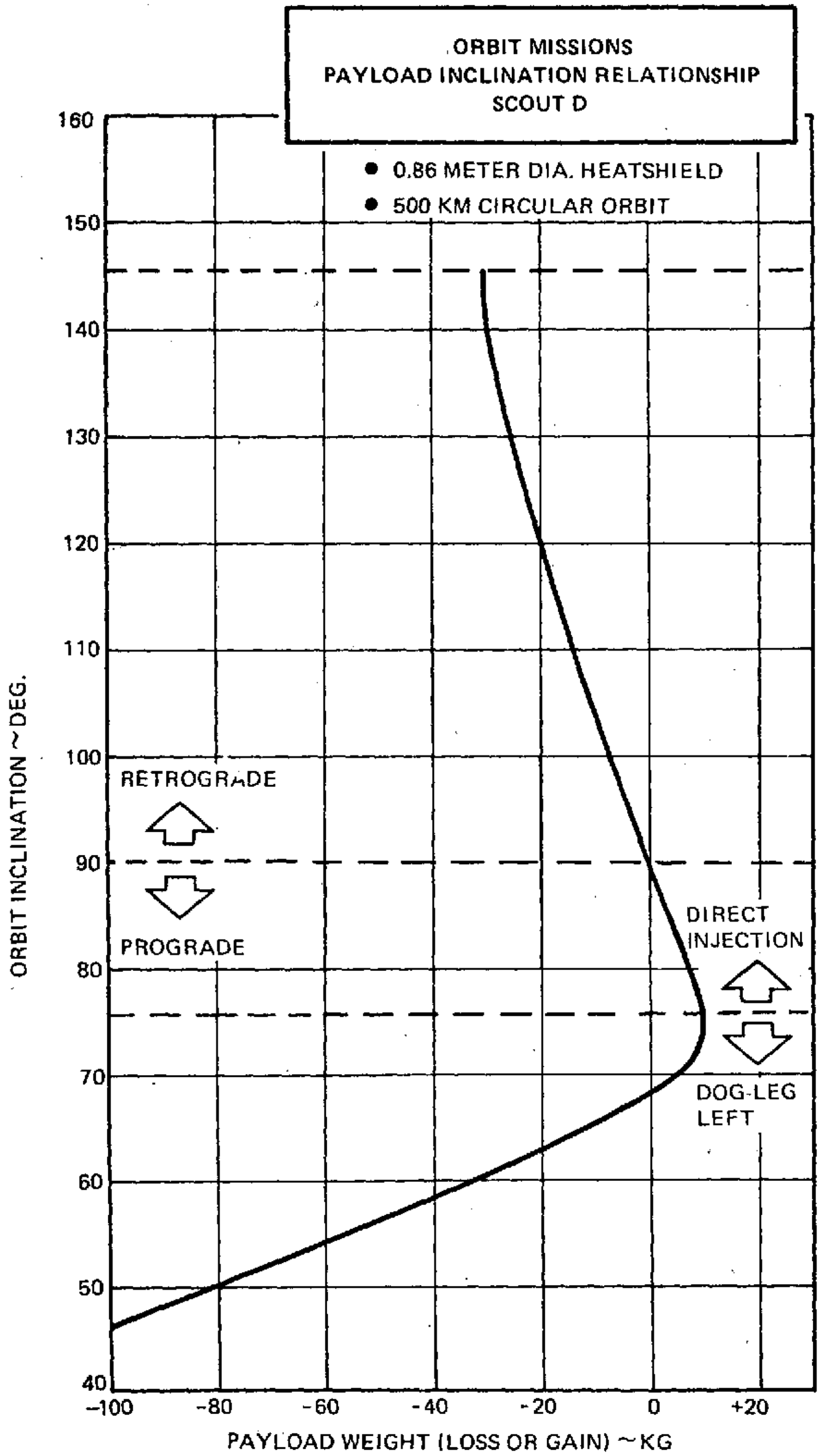


Figure 4-4. Payload-Inclination Relationship - Vandenberg AFB

This is not to infer that if requirements are considerably higher the instrument cannot be accommodated. Only a detailed study of the situation can explore the tradeoffs available.

4.5 DATA HANDLING

The data handling system as described earlier is versatile for instruments with data rates up to 40 kilobits per second. The format is programmable and changeable in orbit.

If data storage is required the GSFC Standard 10^8 bit recorder is available. It is possible to fly 2 of these recorders if needed. For planning purposes consider one tape recorder dump per orbit. For a nominal 600km circular orbit, assume the usable station contact time to be 6 minutes. Therefore to dump 10^8 bits would mean a bit rate of approximately 280 kilobits per second. This data rate is easily within the playback capability of the recorder and bandwidth of the S-band transmitter.

If mass data storage requirements are small, a solid state memory could possibly be employed. No specific one is described here but such a device is within the state of the art and the design would probably be available. If the total bit storage is small, the need for the S-band transmitter may not exist, thus helping weight and power constraints. For example, if the total bit storage were 1×10^5 bits per orbit, dumping once per orbit (assuming a 6 minute pass time) would require a bit rate of 280 bits per second, easily within the bandwidth of VHF. Thus power, weight, and money could be saved by not flying the S-band system.

4.6 FEASIBILITY CHECKLIST

If at this point the investigator determines that the AEM spacecraft is probably compatible with his mission needs, a more in-depth feasibility study should be initiated. He should contact the AEM Project Office to discuss his needs. One of the tools needed is the completed AEM Feasibility Checklist found in Appendix A. Copies of this form are available on request from the AEM Project Office.

It should be noted that the modular design and random component packaging concepts of the AEM spacecraft make changes on the component level easily accommodated. For this reason the Project Office does entertain requested minor changes to tailor the spacecraft to specific mission needs. These changes should be kept to a minimum since they tend to stretch schedules and drive up costs.

4.7 VEHICLE DISPERSIONS

The investigator should realize that the Scout vehicle performance is subject to relatively high dispersions when compared to other vehicles. The greater the accuracy requirements needed for a successful mission the higher the risk. Investigators should recognize that AEM provides an opportunity for a low cost mission but the risks could be high.

There are three basic deviations from the nominal orbital parameters that can affect the orbit and mission. The in-plane uncertainties are the apogee and perigee deviations. The out-of-plane deviations are inclination errors. All three work in combination to affect the mission in many ways. Many users need a sun synchronous orbit with a particular nodal crossing time that remains fixed in local time. This means the orbit plane must precess at a rate equal to the precession of the earth around the sun (about 1°/day). This precession is a function of apogee, perigee, and the inclination of the orbit. Therefore, a deviation from the nominal in any one of these parameters affects this precession rate, causing the local time of nodal crossing to drift away from the desired one. The rate is determined by the amount of the deviations working in combination.

Obviously on low orbits orbit lifetime is affected if a low perigee dispersion is encountered.

There is another lifetime that can be affected. This is referred to as the operational lifetime and can be defined in different ways for different missions. A particular mission may require a sun synchronous orbit with a particular local time of equator crossing with specified tolerances. Operational lifetime is therefore defined as the time the nodal crossing time stays within that limit. Thus, deviations from the nominal orbit can greatly shorten the operational lifetime even though the orbit lifetime is quite long.

Another important consideration of sun synchronous operational lifetimes is in power constraints. Depending on the expected nodal crossing times and other parameters such as weight and power requirements, the spacecraft array will be optimized for that particular orbit. In some situations precession of the orbit plane will greatly decrease the power available, shortening the mission operational lifetime. For example, a two-paddle configured spacecraft launched into a nominal 4 p.m. ascending node orbit could have adequate power between 3 p.m. and 6 p.m. If the dispersions caused the orbit plane to drift to earlier times, the operational lifetime would be determined by when the nodal crossing time drifted earlier than 3 p.m.

Among other parameters power balance is influenced by spacecraft eclipse durations. If a mission is predicated on a critical power balance situation, the operational lifetime could be affected by dispersions that lengthened eclipse durations.

In order to determine the optimum orbital parameters for launch that consider all the uncertainties and their effects, a study must be performed by the AEM Project Office (GSFC) and the Scout Project Office, Langley Research Center (LaRC). The following information must be furnished by the prospective instrumenter to the AEM Project Office:

a. Apogee altitude and constraints.

b. Perigee altitude and constraints.

c. Constraints on inclination (if not sun synchronous).

d. If sun synchronous, bounds of equator crossing times within a period of time such as: must stay between 2 p.m. and 4 p.m. for the first 6 months.

e. Time constraints on above parameters.

f. Operational lifetime.

g. Ground track constraints.

h. Minimum station contact time of "x" contacts per orbit.

i. Any other constraints that could affect injection parameters.

The Scout Project Office will be asked to calculate and furnish the parameters for the optimum nominal orbit and the probability for achieving this orbit.

5.0 HEAT CAPACITY MAPPING MISSION

5.1 INTRODUCTION

The first mission in the AEM series will be the Heat Capacity Mapping Mission (HCMM). The objectives of the mission are to:

- Produce thermal maps at the optimum times for thermal inertia measurements for discrimination of rock types and mineral resource locations.
- Measure plant canopy temperatures at frequent intervals to determine the transpiration of water and plant stress.
- Measure soil moisture effects by observing the temperature cycle of soils.
- Map thermal effluents, both natural and man-made.
- Investigate the feasibility of geothermal source location by remote sensing.
- Provide frequent coverage of snow fields for water runoff prediction.

The basic radiometer instrument has already been developed in the High Resolution Surface Composition Mapping Radiometer (HRSCMR) for Nimbus E. The spare flight radiometer, with minor modification to the basic instrument, will serve as the Heat Capacity Mapping Radiometer (HCMR). The HCMR will have a small instantaneous geometric field of view, less than 1 x 1 milliradians, high radiometric accuracy and a wide enough swath coverage on the ground so that selected areas are covered within the twelve-hour period corresponding to the maximum and minimum of temperature observed. The instrument will operate in two channels, 10.5 to 12.5 micrometers (IR) and 0.8 to 1.1 micrometers (visible). The latter is matched to the ERTS MSS Band 4.

5.2 ORBIT

The HCMM will place the Heat Capacity Mapping Radiometer (HCMR) instrument into a 600km circular sun synchronous orbit with a nominal 2 p.m. ascending node. A six-month minimum operational lifetime is planned. The vehicle will be a Scout-F launched from the Western Test Range (WTR). Constraints on the mission include:

- Sun synchronous orbit, nominal 600km altitude, with a 2 p.m. ascending node.

- Ascending node drift to be not greater than ±1 hour from initial clock angle in six months.

- Minimum operational lifetime six months.

5.3 SPACECRAFT DESCRIPTION

5.3.1 General

The HCMM spacecraft will be as described in Section three of this manual. Mission peculiar requirements dictate changes from the configuration as described. This section will describe these deviations from the standard AEM spacecraft and the options chosen.

5.3.2 Structure

The described base module structure will be used. Figure 5-1 shows the HCMM Spacecraft in the Scout heatshield.

5.3.3 Attitude Control System

The two-wheel system is adequate for this mission.

5.3.4 Power

The three-paddle solar array system will be used. A detailed study will be performed to determine the optimum pitch angle for each paddle for the nominal orbit and possible dispersions. A preliminary study shows adequate power will be available with all paddles at 45°.

5.3.5 Telemetry

The digital data handling needs for this mission include base module functions and instrument housekeeping parameters only. The described DMS system of a multiplexer and one subplexer will adequately handle all of these needs.

The scientific data from the instrument are two channel analog, and as such will not use the DMS.

The mission will be a real-time one and as such will not require a tape recorder.

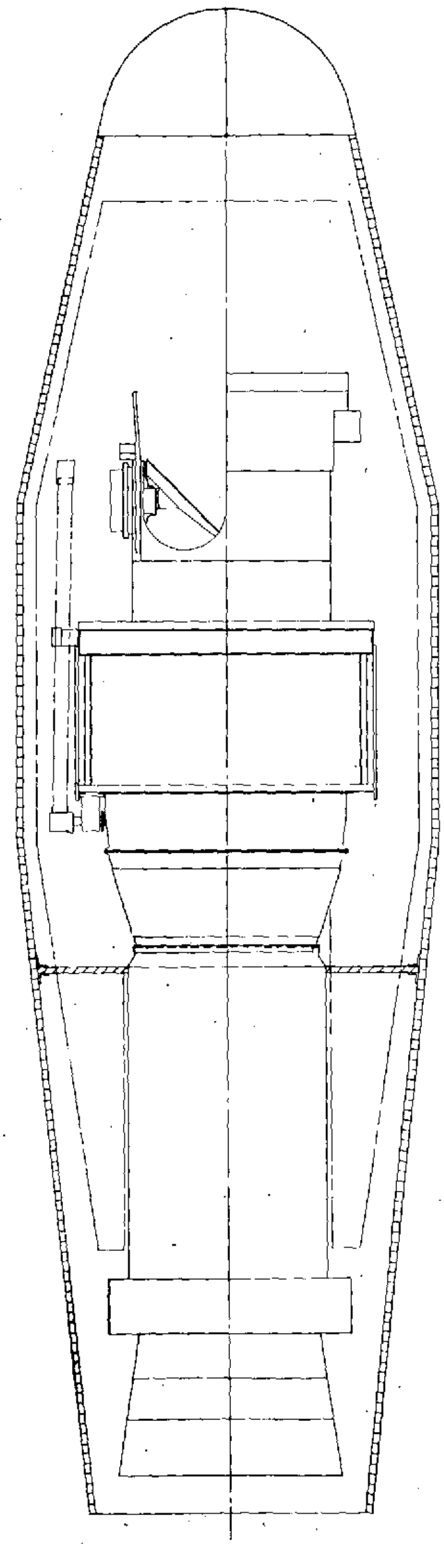


Figure 5-1. HCMM in Scout Heatshield

5.3.6 Command

Because of cost restraints it was decided to fly a non-redundant command receiver and decoder. The system to be used is as that described earlier but without redundancy. The orbit timer and pass timer functions described under AEM Operations Electronics will be used to provide the few command functions needed when no command link can be established, e.g., turning on the S-band transmitter at Madrid where no VHF system exists and turning off the transmitter after each pass should the spacecraft be out of range of the VHF command site.

5.3.7 Communications

The described VHF system will be used. The receiver will be used for commands and the transmitter will be used for the PCM housekeeping data. The S-band transmitter will be used for the scientific data being fed directly from the analog multiplexer.

5.3.8 Orbit Adjust System

The Orbit Adjust System described in section 3.9 will be used to correct for vehicle dispersions to achieve the desired orbit.

5.3.9 HCMM Analog Multiplexer

The HCMM S-band downlink is dedicated to transmission of HCMR analog data and a timing clock. Combination of these signals for simultaneous transmission is accomplished through the analog multiplexer (MUX) on three subcarriers as follows:

- 70 kHz sub: 36-bit NASA time code in 100 bps format
- 480 kHz sub: HCMR thermal channel analog data
- 800 kHz sub: HCMR daylight channel analog data

Except for deletion of a 200 kHz subcarrier tape recorder signal, this multiplex arrangement is essentially the same as used on Nimbus 5 for HRSCMR. On HCMM, the MUX is powered only during S-band operation. Power consumption is 3 watts.

Table 5-1 is the weight breakdown for HCMM. Table 5-2 is the power profile.

Table 5-1

Weight Breakdown For HCMM Components

Base Module	HCMM*
Structure	12.6
Attitude Control System	15.8
Power	28.7
Telemetry	2.7
Command	5.3
Communications	4.1
Thermal	2.8
Cable Harnesses & Misc. Brackets	7.4
Orbit Adjust	8.4
Base Module Weight	87.8
Instrument Module	
HCMR	
Radiometer & Electronics	23.2
Multiplexer	0.7
Power Converter	1.0
Thermal	1.4
Structure	7.0
Instrument Module Weight	33.3
Spacecraft Weight	121.1
L/V Adaptor	4.6
L/V Timer Switches	0.9
L/V - Payload Weight	126.6
L/V - Payload Capability	138.0
Weight Margin	11.4

*All weights in kg.

Table 5-2

HCMM Power Profile

Subsystem	Power Requirements (Watts)	
	Continuous	Data Pass
Command	1.1	
DMS	3.4	
Communications (VHF)	1.4	
ACS	16.5	
Mission Ops Electronics	1.0	
Thermal	0.5	
Shelf Heater	6.0	
Motor Heater	6.0	
S-Band	0.0	Δ 8.0
HCMR	11.9	Δ 15.0
Total	47.6	Δ 23.0

NOTE: Assuming one 10-minute pass per orbit, requires a 112-watt array with a battery depth of discharge of 29%.

5.4 SCIENTIFIC OBJECTIVES

The purpose of the HCMR is to conduct a thermal mapping experiment with high spatial resolution and in an orbit optimized for earth resources sensing rather than meteorological sensing.

The thermal inertia of a material is determined by the heat capacity, thermal conductivity and density of the material and the diurnal temperature variation. Figure 5-2 shows the variation in temperature of solid materials in mid-latitudes as a function of their thermal inertias. The optimum times for sensing temperatures are about 1:30 p.m. and 2:30 a.m. when the temperature variations are the greatest. The variations shown in Figure 5-2 are for a normal day where surface temperature variations are determined only by the normal atmospheric

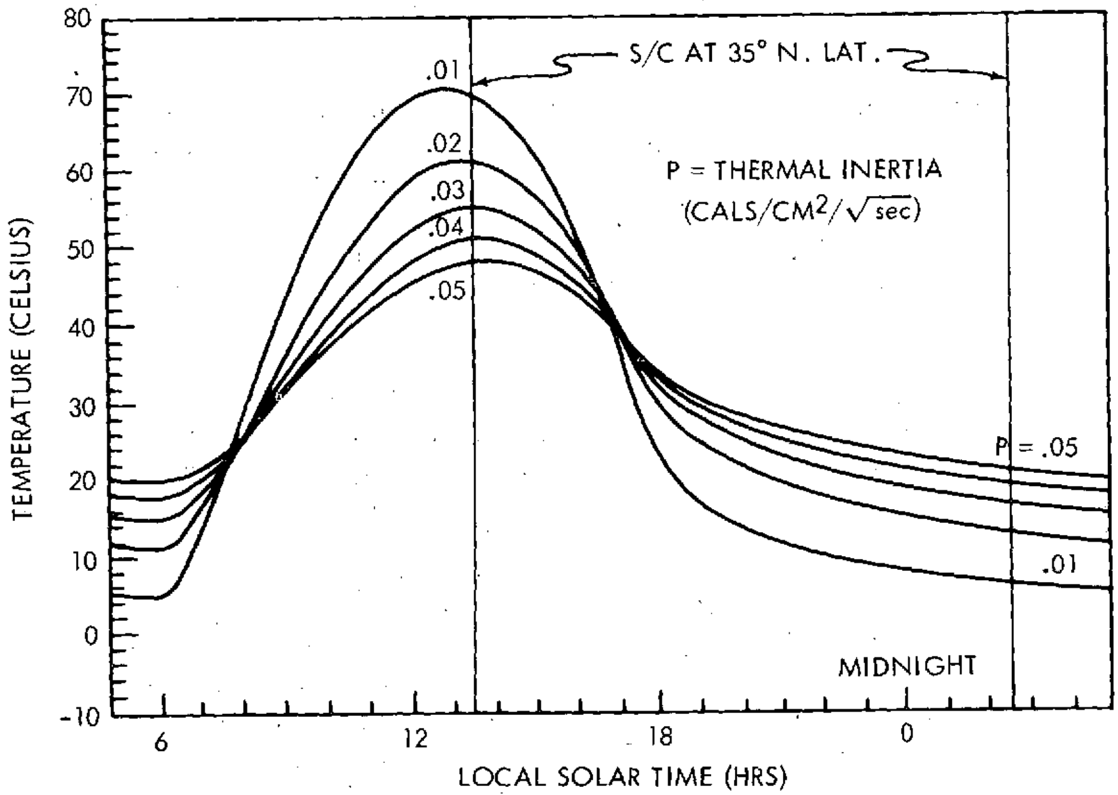


Figure 5-2. Diurnal Surface Temperature Variation as a Function of Thermal Inertia

cooling and heating, heat gain by solar energy absorption and heat loss by thermal emission. Effects such as a cold front passing over an area or a rainstorm followed by evaporative cooling would alter the temperature variation pattern shown.

Underground water will also effect temperature variations depending upon its depth and quantity. Over vegetated areas temperature variations will be affected by the transpiration of water from vegetation and can be used to estimate the volume of water transpired.

Thermal inertias for several rock types have also been determined. These thermal inertias are indicative of the variations of thermal inertia found in rocks that can lead to their discrimination by remote sensing of temperature variations. Observation at sufficiently frequent intervals and at the proper time of day offer the opportunity to measure the thermal inertias of surface materials on enough "normal" days, that is days without anomalous weather changes such as cold fronts or rain, to map thermal inertias over large areas and determine rock types as a guide to resources. Since surface temperature is, to varying degrees, affected by subsurface materials, thermal inertia sensing also provides an opportunity to determine something of the subsurface material unlike reflected solar energy that is representative of only the top most layer.

HCMM offers the opportunity to conduct an experiment with high spatial resolution (0.5 x 0.5km) and in a near optimum orbit. An orbit with an ascending daylight node, an inclination of 98° to 100° and an equatorial crossing time of 2:00 p.m. will provide a 1:30 p.m. and 2:30 a.m. crossing time over middle Northern latitudes. This orbit is very near optimum for sensing of surface thermal effects and also allow for reflectance measurements during the daylight pass.

The proposed spatial resolution (0.5 x 0.5km) is not as good as will be achieved by the ERTS MSS (0.25 x 0.25km) but the ERTS measurement will be made only in daylight with a 9:30 a.m. orbit and only every 18 days. These restrictions prevent the ERTS MSS from making thermal inertia measurements.

For the HCMR instrument a 600km orbital altitude will provide the necessary high spatial resolution. Measurements of surface temperature will be made during successive day and night passes for determination of surface temperature variation. Satellite measurements by temperature sounders (ITPR, VTPR) will provide information on the occurrence of cold fronts and mappers such as Nimbus THIR and NOAA VHRR will provide information on cloud cover and probable rain.

For "normal" days a map will be made of the gridded day and night temperature maps and comparisons made to determine surface thermal inertias

5.5 INSTRUMENT DESCRIPTION

A simplified block diagram of the HCMR is shown in Figure 5-3.

The HCMR is comprised of four major subassemblies mounted in a common housing. These subassemblies are scan mirror and drive, optics, electronics, and radiant cooler. The scan mirror drive assembly provides cross-course scanning of the instantaneous field of view with reference to the sub-satellite ground track. The optical subassembly provides increased ground resolution and spectral definition of the three channels. The electronics subassembly contains the data amplifiers and housekeeping telemetry and formats the analog sensor data such that it is compatible with the HCMR data system. The radiant cooler subassembly provides detector operating temperatures of approximately 115° K.

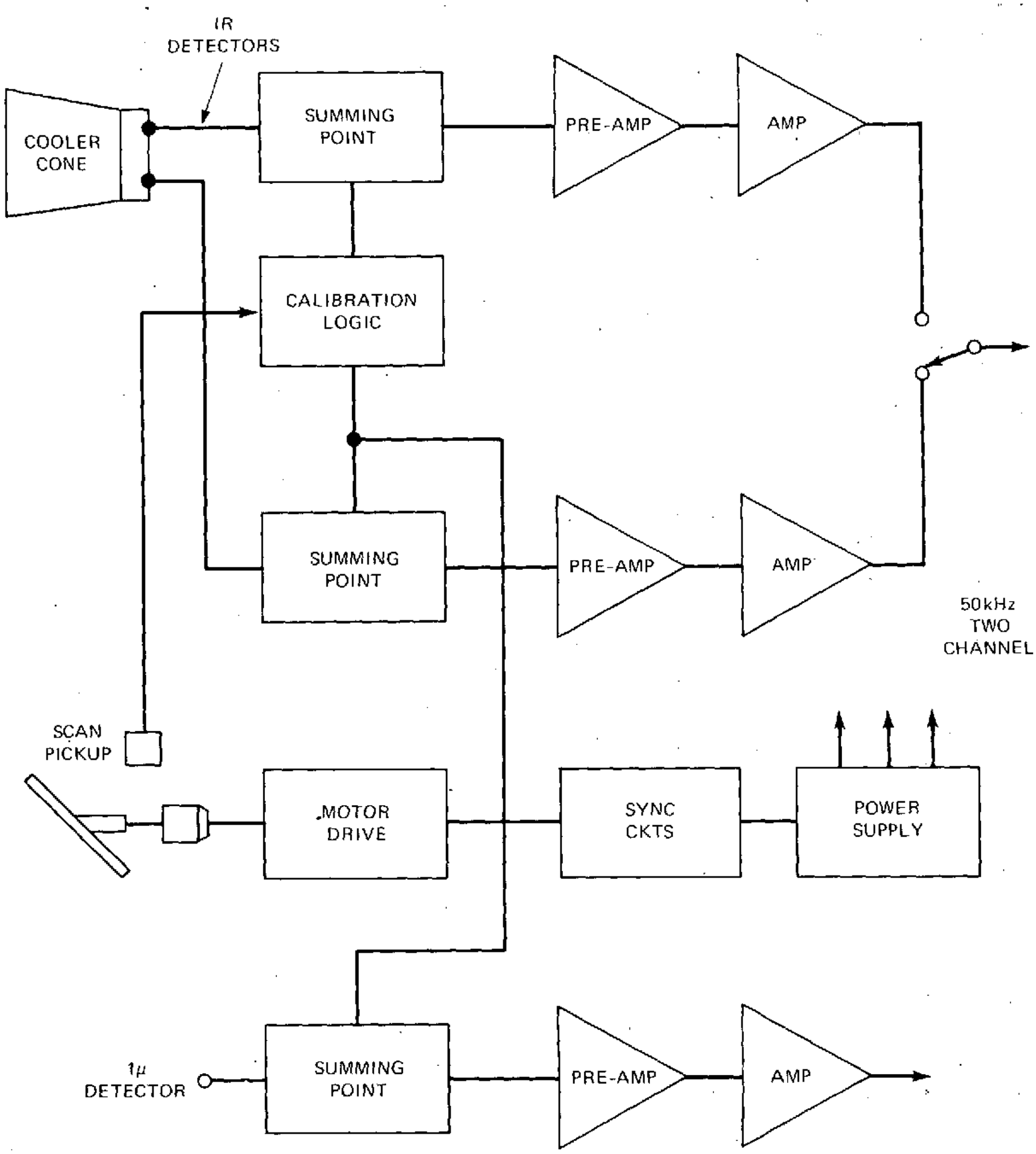


Figure 5-3. HCMR Block Diagram

5.5.1 Scan Drive Subassembly

The scanner design uses an elliptically shaped plane mirror set at 45° to the axis. The scan mirror is fabricated from beryllium and is Kanogen coated. The mirror is driven by an 80-pole Schaeffer motor which is synchronized to the spacecraft two-phase clock. Angular momentum compensation of the scan mirror is provided by a separate motor driving a compensation mass. Scan mirror position is monitored once each revolution by a magnetic pick-up.

5.5.2 Optics Subassembly

The optical subassembly (See Figure 5-4) is catadioptric collecting with an afocal reflecting telescope. The telescope is a modified Dall-Kirkham configuration which reduces the optical beam from an eight inch to a one inch diameter. Spectral separation is provided by a dichroic beam splitter positioned in the collimated beam from the secondary mirror which acts as a folding mirror for the 10.5 to 12.5 micrometer band and transmits energy at shorter wavelengths.

The daylight channel optics consist of a long wavelength-pass (>0.8 um) interference filter, a parabolic focusing mirror, and an uncooled silicon photo diode. The long wavelength cut-off of the silicon detector limits the band pass to wavelengths of less than 1.1 um. Sensitive area of the detector is approximately 0.15 mm square.

The two 10.5 to 12.5 IR channels are provided by inserting a second beam splitter in the reflected beam of the first. The two beams are focused onto the Hg-Cd-Te detectors using germanium lenses. Final focusing and spectral trimming is accomplished by germanium band pass filters and germanium aplanats located at the detectors.

5.5.3 Electronic Processing

The detectors produce a small ac electrical signal which is proportional to the difference in radiant energy between the scene and space. The electrical signals from the detectors are amplified in each video amplifier to a level required for processing. Each video amplifier contains a low noise preamplifier, video filter and postamplifier. A space clamping technique is also used which establishes the dc zero level once every rotation of the scanner by clamping the output to zero when viewing cold space and holding this level for the duration of the scan. The overall video amplifier gain will be such that the highest energy scene will produce a 6-volt output signal. Calibration signals consisting of a 6-step staircase waveform will be inserted at the amplifier input as well as at the amplifier output on every scan line to provide constant calibration and complete assessment of the amplifier performance. At the amplifier output, synchronizing

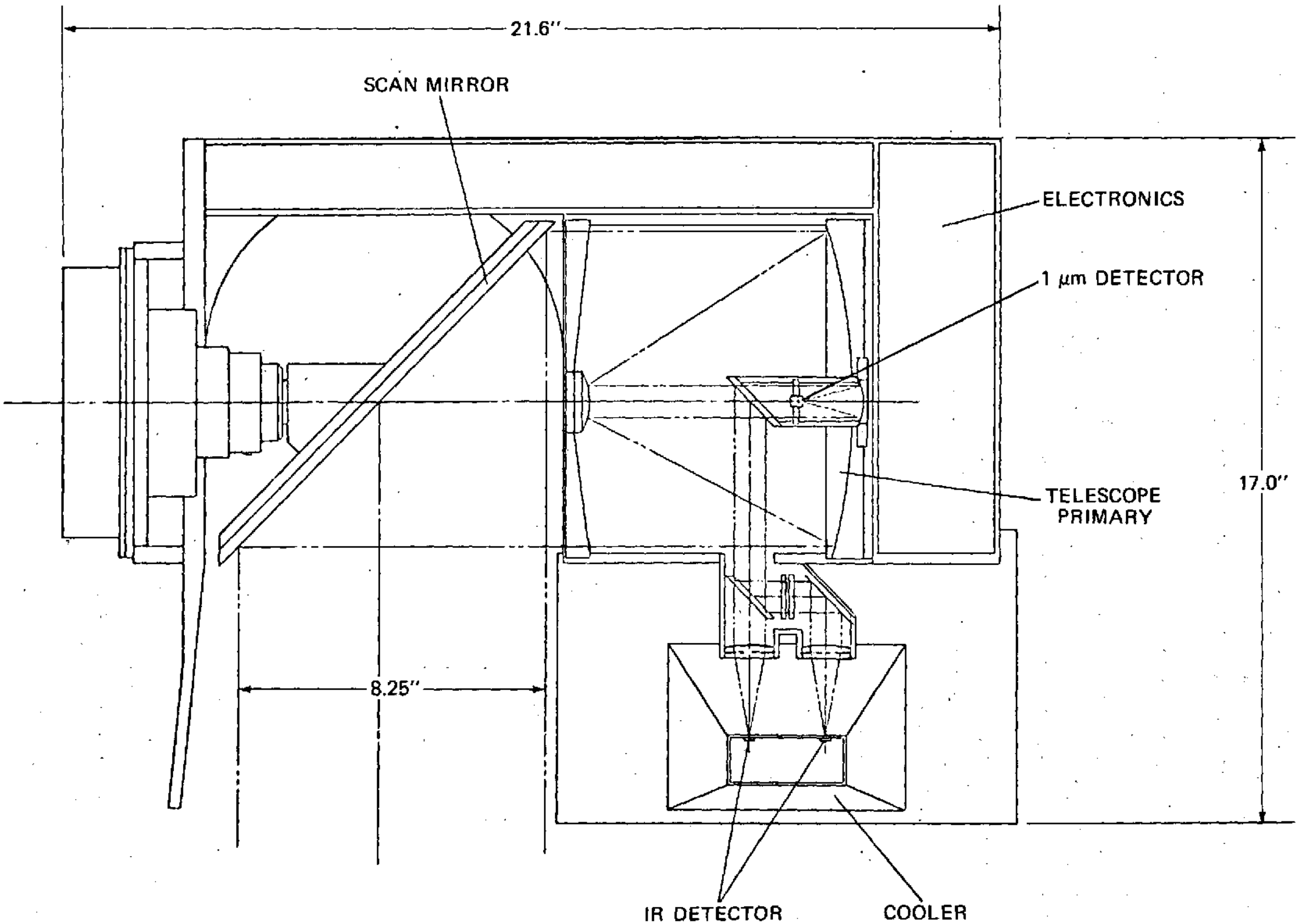


Figure 5-4. HCMR Optics Subassembly

pulses are also gated in along with the output calibration to make up the composite video. Output buffer amplifiers with unit gain and low output impedance are used for the output interface to the data system.

The timing and control logic generate timing signals for generating the synchronizing pulses, the input and output calibration waveforms and the space clamp signals. A spacecraft clock signal is used as the basic counting frequency by the logic and a magnetic pickup mounted near the periphery of the rotating mirror initiates the counters once every rotation.

The calibration circuitry consists of an accurate, stable digital-to-analog converter which will generate a staircase of six one-volt steps for insertion at the amplifier input and output.

The instrument utilizes two dc to dc converters. One converter, energized by either Motor ON or Electronics ON commands provides ±15 volt power for the telemetry conversion circuitry not directly associated with the data channels. The other converter which is energized by the Electronics ON command only supplies ±15 volt power to the data channels and +5 volts to the digital logic.

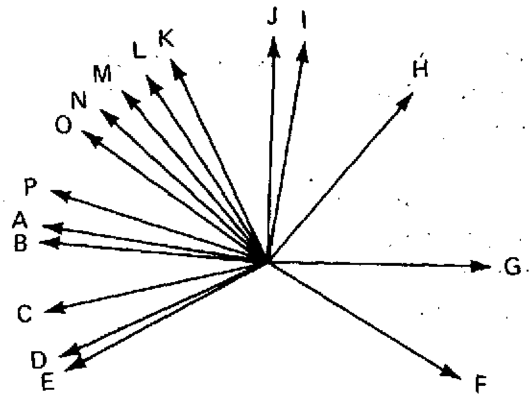
The scan motor power supply has two operating modes. A high power mode used for in-air operation in which the regulated bus voltage is applied directly to the motor drive circuitry. A low power mode is used for vacuum operation. In the low power mode the regulated bus voltage is applied to a switching regulator which reduces the reg bus voltage to an appropriate value. The motor drive signals are synchronized to the spacecraft two-phase motor drive signals.

5.5.4 Radiant Cooler

The radiant cooler is designed to cool the patch to 110° K. The patch will be controlled in temperature to 115° K by a temperature control circuit which monitors temperature with a thermistor and supplies heat to the patch.

5.5.5 Scan Sequence

The SCMR scan sequence is illustrated diagramatically in Figure 5-5. The scan sequence is initiated with the occurrence of sync pulse No. 1. This sync pulse is generated by a ferrous slug, attached to the scan mirror, rotating past a magnetic pickup. The slug and magnetic pickup are located such that the sync pulse occurs at the instant the instrument field of view clears the housing and "looks" at space. The sync pulse is used to reset the logic utilized to generate the temperature telemetry, sync pulse No. 2 and the precursor burst. The leading edge of sync pulse No. 1 defines the 0° angular position and zero time for events occurring in the scan.



Reference Letter	Angle (degrees)	Time (msec)	Event
A	0	0	Begin Sync Pulse #1
B	3.6	1.0	End Sync Pulse #1
C	21.6	6.0	Begin Input Calibration
D	34.2	9.5	End Input Calibration
E	38.2	10.6	Begin Earth Scan
F	158.0	43.9	End Earth Scan
G	189.0	52.5	Begin Output Calibration
H	239.4	66.5	End Output Calibration
I	270.4	75.1	Begin Internal Target View
J	278.3	77.3	Complete Internal Target View
K	304.2	84.5	Begin Internal Target Temp. Telemetry
L	311.4	86.5	End Internal Target Temp. Telemetry
M	318.6	88.5	Begin Sync Pulse #2
N	325.8	90.5	End Sync Pulse #2
O	333.0	92.5	Begin Precusor Burst
P	351.0	97.5	End Precusor Burst

NOTE: Scan sequence shown is for Nimbus SCMR Instrument.
This will be modified for HCMM.

Figure 5-5. HCMR Scan Sequence

5.6 DATA PROCESSING PLAN

5.6.1 Data Collection

Data will be collected in real time only, when the spacecraft is within the range of one of the stations equipped to receive HCMM data. Stations presently equipp are Rosman, N. C., Mojave, Calif., Gilmore Creek, Alaska, Honeysuckle, Australia and Madrid, Spain. Data will be collected in analog form only. Data tapes will be mailed to GSFC in all cases and some direct recording at GSFC ca be accomplished using the wide band link from Rosman to GSFC.

Data will be collected at each station for every pass where the station is operating and where cloud cover is not so extensive as to cover 70 to 80% of the area that would be seen.

Figure 5-6 shows the data flow.

5.6.2 HCMR Survey Pictures

Pictures from each data pass will be produced directly from the analog tapes to determine if the data quality is sufficiently good to merit digitization and further processing. The analog pictures will be made on the EIS machine where a black and white positive and negative are made simultaneously at a rate of three minutes per frame. In no case will a station pass result in more than three frames.

The black and white positives will be scanned to determine if the data is of good quality and the cloud cover sufficiently small to merit further processing. The negatives from this uncalibrated picture will be washed and stored for possible further use. In cases where some useful interpretation can be made from the uncalibrated survey picture, prints will be made from the negative in the amount required.

The positive survey pictures will be utilized to prepare instructions for the digitization line in the Information Processing Division. Instructions normally contain the tape register number, the number of the tracks containing the data of interest and the start and stop time from the spacecraft time code recorder on the tape. The tapes and the instructions for processing will be transported to the digitization facility in the event that the survey picture facility and the digitization line are not co-located.

5.6.3 HCMR Data Digitization

Digitization of the analog data will be carried out with the digitization line established for the Nimbus 5 Surface Composition Mapping Radiometer. The data will be digitized in the format utilized for the SCMR so that the software developed for the SCMR can be utilized.

5.6.4 HCMR Processing

The digitized data will be processed first into thermal maps utilizing the instrument internal calibration for thermal calibration and the spacecraft altitude and attitude information to correct the data to a uniform scale and grid the data. Based on past Nimbus experience, it is estimated that 20 to 25 usable frames of data will be acquired each day. The processed data will be converted into calibrated imagery in the facilities of the Information Processing Division.

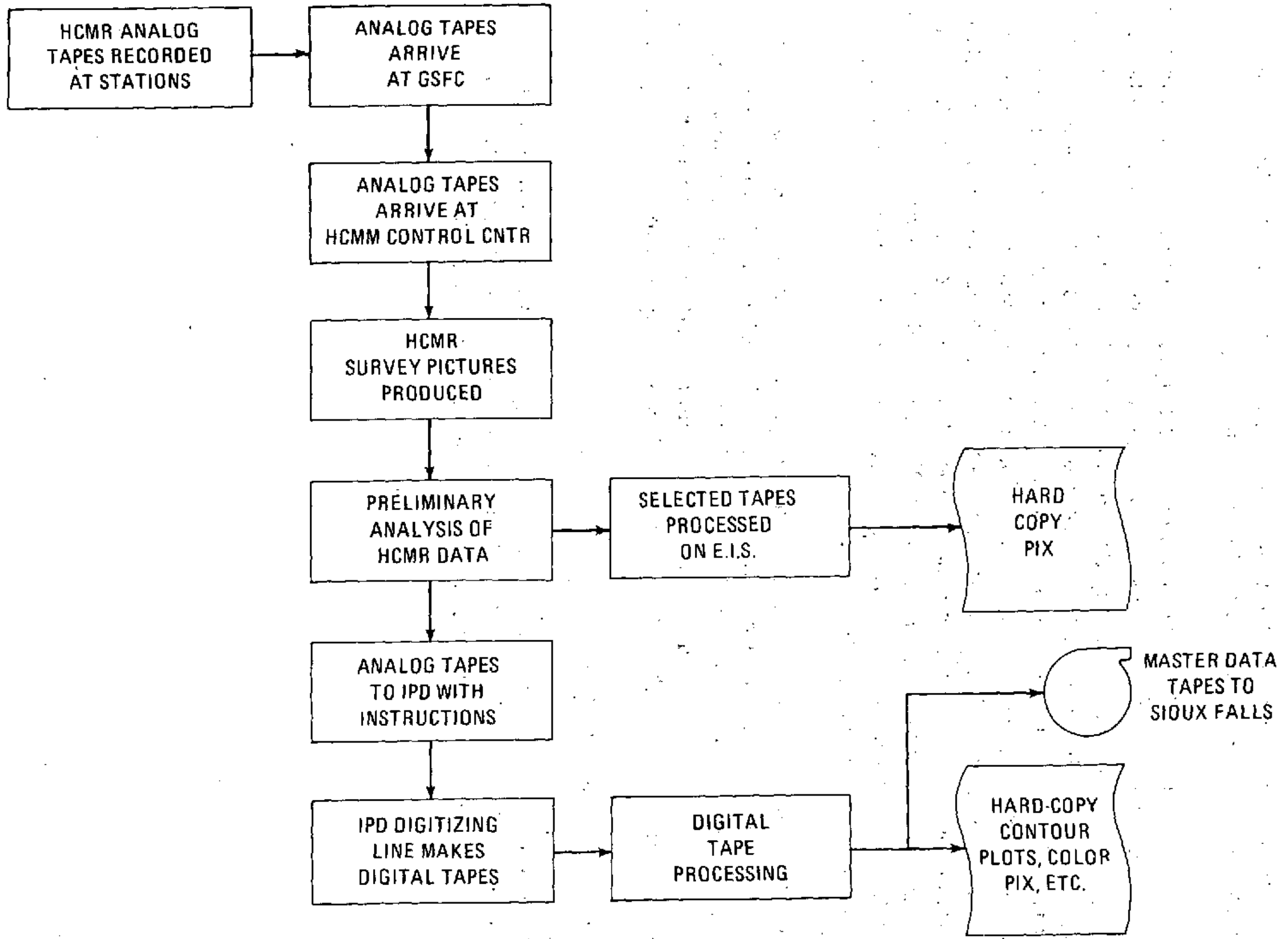


Figure 5-6. HCMM/HCMR Analog Data Flow

Duplicates of the processed tapes will also be produced in the facilities of the Information Processing Division. With the exception of scale correction that may be necessary because of the non-circular orbit possible with a Scout launch, software for all operations up to this point already exists.

Processing for thermal inertia mapping will require registration of data from overlapping orbits from a day and night overpass of the same area. The scale correction and location gridding already applied will be utilized as the first step in registration. Since the day and night orbits will cross at an angle of approximately 30° one of the two scenes will be rotated to coincide with the other. Software for this operation will be developed in cooperation with the Information Processing Division. The registered frames will be processed with an algorithm to extract thermal inertia from the temperature measurements and the daytime albedo measurements. From the 20 to 25 frames of useful data per day an estimated 2 pairs will be sufficiently cloud free and occur in an area where thermal inertia measurements will be of use.

The processed thermal inertia data will be produced to provide imagery and magnetic tapes. The imagery will be produced in black and white with gray scale annotation. Annotation will also include gridding and the times at which the frames were taken. Data tapes will be duplicated for the primary users in the IPD facilities. Further duplication will be carried out at a data depository.

5.6.5 Data Dissemination

Processed data in the form of pictures or magnetic tapes will be sent to the EROS Data Center, Sioux Falls, S. D. Distribution of the data to investigators other than at GSFC or the USGS will take place through the Data Center.

APPENDIX A

AEM FEASIBILITY CHECK LIST

INSTRUCTIONS FOR COMPLETING AEM FEASIBILITY CHECK LIST

This Check List was developed to be used as a working document to be completed when a candidate instrument is to be studied to determine the feasibility of flying the instrument on the basic AEM spacecraft. The standard philosophy applies, that is, the spacecraft will consist of two modules, the base module that contains the standard spacecraft support instrumentation with a standard electrical and mechanical interface with the instrument module that contains all the instrument unique hardware including sensors, power converters, command interface, and unique data handling conditioners. All equipment in the instrument module will be the responsibility of the instrumenter.

This Check List is designed as a working document and as such is designed to be completed by both the proposing instrumenter and the project staff. The left margin contains two columns that lists which party has the responsibility for completing that item. In general the instrumenter must complete his items first. The right margin contains two columns labelled yes and no. A number of questions require only such an answer which can be indicated by a check in the appropriate column. Other items require a written statement and should be completed on a separate sheet referencing the item number.

If the instrument proposer cannot fully complete an item, he should contact the project staff for assistance. As many items as possible should be completed for the initial submission.

AEM FEASIBILITY CHECK LIST

Date:

Instrument Name:

Proposer's Name, Address, Phone:

Instru-menter	Proj-ect Staff		Yes	No
X		1.0 Instrument Envelope Description		
X		1.1 Any Booms or Appendages?		
X		1.2 Any Pyrotechniques or Shutters?		
X		1.3 Instrument Look Angles?		
	X	1.4 Is the envelope compatible with the base module and Scout Heatshield?		
	X	1.5 Are the look angles compatible with the earth oriented instrument module?		
		2.0 I/M Weights:		
		Structure 5.0 kg		
		Thermal 1.5		
X		Instrument ( )		
X		Support Equip.* ( )		
		( )		
		( )		
X		Total Instrument		
X		Module Weight ( ) kg		

*Equip. that is not furnished by the base module that is needed to support the instrument, e.g., an analog multiplexer.

Instru-menter	Proj-ect Staff		Yes	No
		3.0 Base Module Weight		
		Structure 12.6 kg		
		Telemetry 2.7		
		Thermal 2.8		
		Cable Harness/Misc. 7.4		
		Subtotal 25.5 kg		
		Command		
		Standard (4.3)		
		Miss Ops Elec (1.0)		
		Redundant Decoder (1.1)		
	X	Communications ( )		
		Standard (4.1)		
		Redundant Receiver (0.6)		
		Transponder (3.0)		
	X	Power ( )		
		2-Paddle (21.0)		
		3-Paddle (28.7)		
	X	Attitude Control ( )		
		2-Wheel (16.5)		
		3-Wheel (19.2)		
	X	Tape Recorder ( )		
		Single (4.5)		
		Redundant (9.0)		
		Orbit Adjust (8.4) ( )		
	X	Base Module Subtotal ( ) kg		
	X	Spacecraft Subtotal ( )		

Instru-menter	Proj-ect Staff		Yes	No
	X	Contingency (10%) ( )		
	X	Spacecraft Total ( ) kg		
		Launch Vehicle Adaptor 11.9		
	X	5.0 Total Payload Weight ( )		
		6.0 Orbit Parameters:		
X		6.1 Desired:		
X		Apogee (km):		
X		Perigee (km):		
X		Inclination:		
X		Period:		
X		Operational Lifetime:		
		6.2 Achievable:		
	X	Apogee (km):		
	X	Perigee (km):		
	X	Inclination:		
	X	Period:		
	X	Operational Lifetime:		
		7.0 Power Requirements:		
		7.1 Instrument Module:		
X		Can the instrument operate directly off the +28V bus? If not, a converter is required in the I/M.		
X		Continuous: ______ watts		
X		Standby: ______ watts		
X		Peak: ______ watts		
X		Average: ______ watts		

Instru-menter	Proj-ect Staff			Yes	No
X			Power Profile vs Time:		
			Include applicable information such as:		
			1. Orbital profile		
			2. Short term profile if peaks occur for short times in an operating cycle.		
			3. Note if instruments will be operated in only certain parts of an orbit such as during sunlight only, night only, sunrise or sunset only, or only during station contact.		
X			4. During standby will additional power be needed to maintain thermal control?		
X			Will the spacecraft tape recorder be used?		
X			If so, how often will playback be required?		
	X	7.2	Calculate S/C power balance.		
		8.0	Attitude Stability and Control Requirements:		
X		8.1	Control: _____ ° Pitch		
X			_____ ° Roll		
X			_____ ° Yaw		
X		8.2	Allowable body rates: _____ °/sec Pitch		
X			_____ °/sec Roll		
X			_____ °/sec Yaw		
X		8.3	Are there any moving instrument masses that could affect attitude control, e.g., a movable telescope?		
X		8.4	Are there any instrument magnetic torques that could affect attitude control?		
X		8.5	Are there any mass expulsion devices?		

Instru- menter	Proj- ect Staff		Yes	No
	X	8.6 Determine which is applicable: A. ACS 2-wheel B. ACS 3-wheel C. Standard system inadequate		
		9.0 Data Requirements:		
X		9.1 Does the instrument contain its own encoder producing a serial bit stream?		
X		9.2 If yes, what is the bit rate? _____ bps.		
X		9.3 Can the system be slaved to an external clock and run at a different bit rate?		
X		9.4 Does this data include housekeeping parameters?		
		9.5 If S/C data handling system is to be used for instrument data,		
X		9.5.1 How many analog channels need to be sampled?		
X		9.5.1.2 What accuracy is needed for the analog-to-digital conversion? (How many bits?) What is the desired sample rate?		
X		9.5.2 How many serial digital channels?		
X		9.5.2.1 How many bits/channel?		
X		9.5.2.2 Desired sample rate?		
X		9.5.3 How many parallel digital channels?		
X		9.5.3.1 Bits/channel?		
X		9.5.3.2 Desired sample rate?		
X		9.5.4 What is the total bit rate required including instrument housekeeping data?		
X		9.6 Instrument Video (Analog) Data: Does the instrument have video output? If so, describe.		

Instru-menter	Proj-ect Staff		Yes	No
	X	9.6.1 Is it compatible with the base module S-band system? Is it compatible with STDN ground station equipment?		
		10.0 Mass storage:		
		10.1 Digital storage requirements:		
X		10.1.1 Input data rate? ______ bps		
X		10.1.2 Total bits per orbit?		
	X	10.1.3 From orbit parameters, what is the average station playback time? Are the record rates, total storage, and playback rates compatible with the standard GSFC 10^8 recorder? (Use as a base, one playback per orbit.)		
	X	10.1.4 If total number of bits is small, could a solid state mass storage be used?		
		10.2 Analog Data Storage Requirements:		
X		10.2.1 Is there a requirement for on-board analog tape recording?		
X		10.2.2 If so, describe the proposed analog tape recorder system required for the instrument module.		
	X	10.2.3 Are the playback characteristics compatible with the S-band transmitter, ground stations, and station contact times?		
		11.0 Command Requirements:		
X		11.1 Number of commands required by the instrument?		
X		11.2 Description of command interface.		
X		11.3 Typical command sequence (critical timing, if applicable).		
X		11.4 Are commands required when not in view of a station?		

Instru-menter	Proj-ect Staff		Yes	No
	X	11.5 Are command requirements compatible with base module command system?		
X		12.0 What are the definitive attitude requirements?		
X	X	12.1 Is Minitrack accuracy adequate?		
X	X	12.2 If not, is S-band range and range-rate adequate? (Feed into Weight Breakdown Statement.)		
		13.0 From orbit, determine station coverage --		
	X	13.1 What stations?		
	X	13.2 Average station contact time.		
X	X	13.3 Are the stations adequate for desired coverage? (Location)		
	X	13.4 Are the contact times adequate for data acquisition (such as tape recorder dumps)?		
X		13.5 Is there any special station equipment required? If so, describe.		
X		13.6 Is any real-time data required at the POCC?		
X		13.7 If so, what are the requirements?		
		14.0 Data Processing Requirements:		
		14.1 Digital Data Requirements:		
X	X	14.1.1 What are the requirements? Do they fall within the standard functions of the Information Processing Division (IPD) which include: a. Digitization of station T/M tapes. b. Formatting and merging with attitude/orbit data. c. Distribution of data to users.		

Instru-menter	Proj-ect Staff		Yes	No
X		14.1.2 Is image processing required? If so, it is considered special purpose. IPD can be involved but special equipment may be needed and project will have to fund it and the production of data.		
		14.2 Analog Data Requirements:		
X		14.2.1 What are the requirements for the processing of these data?		
X		14.2.2 Is any special equipment needed?		
	X	14.3 Specify any additional equipments or procedures needed?		
	X	15.0 Obtain link calculations.		
	X	15.1 Is the standard spacecraft equipment compatible?		
	X	16.0 Does the instrument require any special calibration procedures?		
X		16.1 How do these affect operations?		
X	X	16.2 Is special equipment needed at the stations?		
X		17.0 Are there any special handling requirements or precautions?		
X		17.1 Are any special handling fixtures required?		
X		18.0 What are the thermal requirements? Specify operating and not operating temperature limits.		
X		19.0 Does the instrument require stringent magnetic cleanliness? If so, specify.		
X		20.0 Does the instrument require stringent electro-magnetic interference (EMI) control? If so, specify.		
X		21.0 Is there a requirement for satellite-to-satellite-communications?		
X		22.0 Other requirements or items that are applicable.		