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BIORESEARCH MODULE DESIGN DEFINITION AND SPACE SHUTTLE VEHICLE INTEGRATION

CR # 114408

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FINAL PROGRESS REPORT

VOLUME 1 -TECHNICAL REPORT

December 15, 1971

GENERAL ELECTRIC
Space Re-entry Systems Programs
RE-ENTRY & ENVIRONMENTAL SYSTEMS DIVISION
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GE-RESD Proposal N-72469

FINAL REPORT FOR THE STUDY OF A BIORESEARCH MODULE DESIGN DEFINITION AND SPACE SHUTTLE VEHICLE INTEGRATION

VOLUME 1
TECHNICAL REPORT

15 December 1971

Prepared under Contract No. NAS2-6523
by General Electric Company
Re-entry & Environmental Systems Division
Philadelphia, Pennsylvania

for

Ames Research Center
National Aeronautics and Space Administration

GENERAL ELECTRIC
Re-entry & Environmental Systems Division

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I. INTRODUCTION

The Re-entry and Environmental Systems Division (RESD) of the General Electric Company is pleased to submit this document covering the Final Report for the Study of a Bioresearch Module Design Definition and Space Shuttle Vehicle Integration. This document has been prepared under NASA/ARC Contract NAS 2-6523 and submitted in response to contract Specification and Work Statement A-17193 and related specifications and attachments.

The objective of the study was to use the baseline preliminary design developed for the Bioexplorer spacecraft under the previous NASA/ARC Contract NAS 2-6027, and devote further study effort in areas of thermal control, attitude control and power subsystem design, and evaluate the use of the Space Shuttle Vehicle (SSV) as a potential launch and recovery vehicle for the Bioresearch Module (formerly called Bioexplorer).

The results of the study are to include: a refinement of the baseline design definition of a Bioresearch Module as a Scout-launched payload to accomplish Missions I & II as defined in the specification; an evaluation of the design impact of using the SSV to launch the Bioresearch Module for Missions I, II and III and recover Missions I and II; and a preliminary definition of the Space Shuttle Vehicle/Bioresearch Module interfaces involved in the conduct of the missions defined.

The Final Report is submitted in three (3) volumes. Volume 1 contains the results of the technical work performed during the study in accordance with the contract work statement. Volume 2 presents the updating and modifications to the Preliminary Spacecraft Development Program Plan developed under Contract NAS 2-6027 as influenced by the results of the changes or revisions to the design, development, fabrication and test programs as determined and evaluated during the conduct of this Bioresearch Module Study. Volume 3 - the Management and Funding Plan - provides a description of the proposed project organization; communications; documentation and reports; project planning, direction and control; related experience and facilities; and cost estimate data and options for the implementation of the Bioresearch Module development program.

This is Volume 1 - Technical Report. The experienced systems engineering and design and analysis personnel, as well as senior quality assurance, manufacturing and test personnel, who were available at GE-RESD from the recently completed NASA Bioexplorer Study Program were utilized to conduct and support the study phase and spacecraft development planning for the Bioresearch Module Project.

Besides fulfilling the final report requirements of the Bioresearch Module Study Contract, GE-RESD hopes that the material furnished in these three volumes can serve as a basis for continued work and planning leading to the crystallization and implementation of a viable and on-going Bioresearch Program. To any future development phases, GE-RESD offers its unique resources of experienced technical and management personnel, facilities and flight-proven hardware designs resulting from the development and flight of numerous space systems programs, including its demonstrated performance and directly applicable experience from NASA's series of Biosatellite space biology missions.

GE-RESD welcomes the opportunity to be of additional service to the Bioresearch Module Project in terms of preparing and/or conducting presentations and proposals which NASA/ARC may consider useful to the interpretation of the material furnished herein or to the contribution of follow-on program implementation planning.

II. SUMMARY

1.0 BACKGROUND

GE-RESD has conducted the Technical Study for the Bioresearch Module Definition and Space Shuttle Vehicle Integration in accordance with the requirements established in NASA/ARC Specification and Work Statement A-17193 of Contract NAS 2-6523. The study project was initiated on June 7, 1971 and two comprehensive design reviews were conducted, the first on August 24 and 25, 1971 (at GE-RESD), and the second on November 9, 1971 (at NASA/ARC).

2.0 BASELINE

This Bioresearch Module Design Definition and Space Shuttle Vehicle Integration Study is based on the preliminary design concepts determined in the Preliminary Design Study of the Bioexplorer Spacecraft developed under NASA/ARC Contract NAS 2-6027. This Bioexplorer design, referred to herein as the baseline design or the Bioexplorer design, is detailed in the Final Progress Report submitted to NASA/ARC on November 29, 1970. In order to avoid inefficient duplication and unnecessary bulk, the following sections, extracted from the Bioexplorer Final Report, summarize the baseline design.

2.1 SYSTEM AND SUBSYSTEM DESIGN SUMMARY OF THE BIOEXPLORER BASELINE DESIGN

2.1.1 Mission Analysis

The mission analysis effort of the study was devoted to the determination or orbit parameters for the various missions, launch vehicle selection criteria, and definitions of orbit lifetimes, thermal environment, communication distances, eclipse times, acceleration forces, and launch sequence of events to furnish information useful to the design of appropriate subsystems. The results of the mission analysis work have determined the following orbit parameters and launch vehicle selections:

Mission	Injection Altitude	Semi-Major Axis	Perigee	Inclination	Launch Vehicle*	Spacecraft Weight
Types I and II	290 nm ±110 nm			38° ± 0.5°	Scout with Algol III 1st Stage	375 lb.
Type III		75,300 nm	100 nm	33° ± 1°	Delta 302 or Delta 303	525 lb.

*Delta 302 is Long Tank Thor, Delta Transtage, FW 4
Delta 303 is Long Tank Thor, Delta Transtage, TE 364 (1450)

2.1.2 Configuration

A pictorial representation of the Bioexplorer spacecraft configuration is shown in Figure II-1. The basic spacecraft structural elements include the experiment compartment, the experiment mounting adapter, the service module assembly, the solar array structure, and the subsystem mounting structures for the attitude control, electrical power and distribution, thermal control, and telemetry, tracking and command subsystem modules. The experiment compartment will house the GFE experiment package and will mate with the thermal control subsystem module. The experiment package adapter is comprised of a machined breech lock ring for attachment of the experiment package to the spacecraft. The service module assembly is the main structure of the spacecraft, and houses the electrical power and distribution, telemetry, tracking and command, and attitude control subsystem modules. The solar array is made up of eight solar panel assemblies. Each panel structure contains the solar cells, interconnections, deployment springs and dampers.

2.1.3 Attitude Control

The attitude control subsystem will accomplish despin of the spacecraft and last stage of the booster, acquire and maintain sun orientation, and control spacecraft rates to

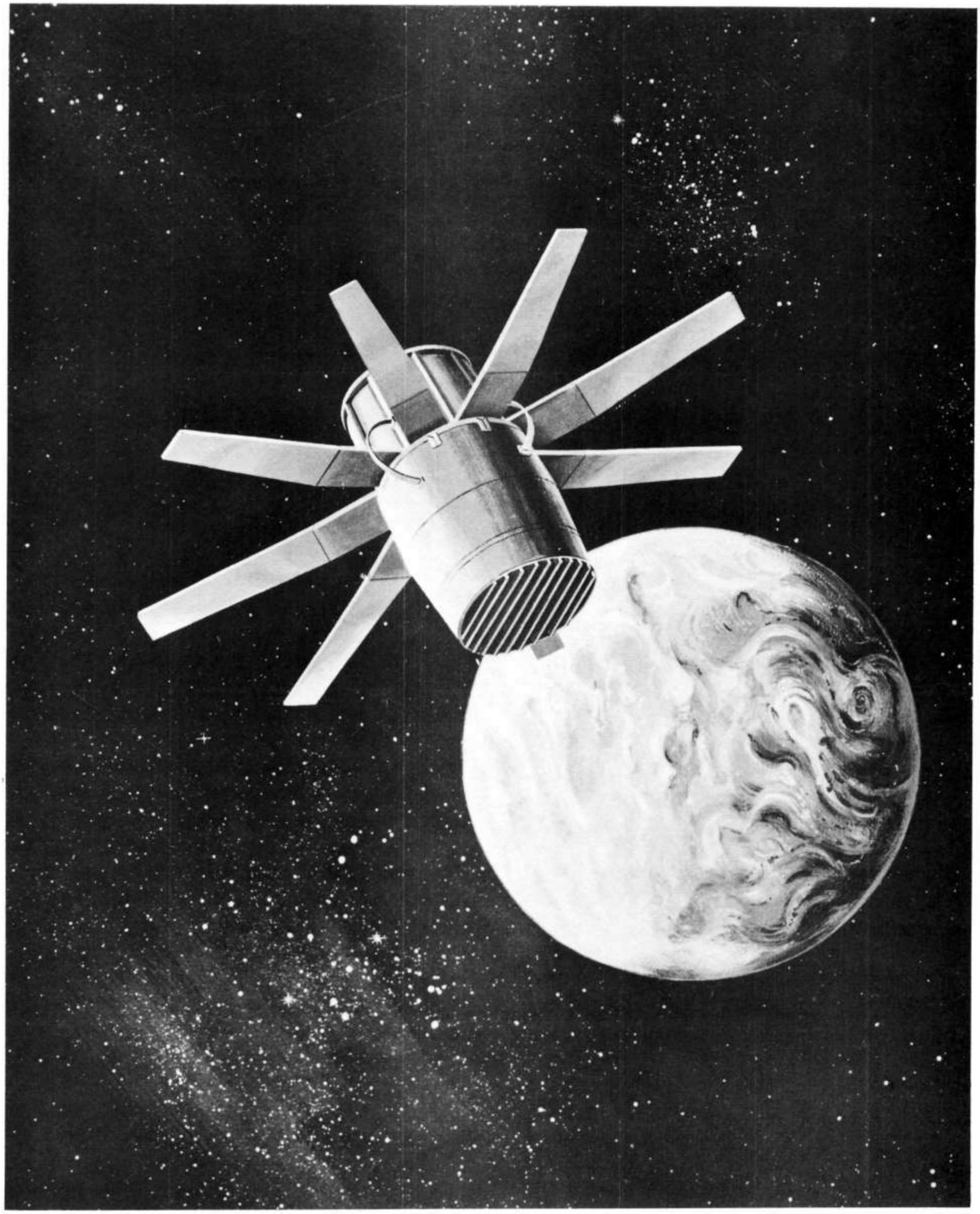


Figure II-1. Bioexplorer Spacecraft

provide the required gravity environment for each of the Bioexplorer missions. The sun sensors, pneumatic control, gas storage assemblies, redundant rate gyro packages; and the attitude control programmer/jet controller serve as common equipments for all of the planned mission types.

For the Type I missions, in addition to the basic equipments using cold gas jets, a momentum wheel is employed to provide for the required spacecraft stabilization. For the Type II mission, the momentum wheel is not required and an extendable rod package is used as the most efficient solution for the variable spin rate control. Four extendable rods, used to control spacecraft rates to the desired variable g settings are installed such that their travel is perpendicular to the spacecraft roll axis, and their motion synchronized to limit undesired variations of spacecraft inertia characteristics. For the Type III missions, only the basic sun-sensor, attitude control programmer/jet controller, rate gyro, gas storage, pneumatics assembly is necessary, and neither the momentum wheel or extendable rods package are required.

2.1.4 Data Storage, Telemetry, Tracking and Command

Digital data from the experiment will be stored and transmitted to ground stations on command along with experiment and spacecraft telemetry signals. Selectable, redundant telemetry transmitters will be provided. Also, redundant receivers and decoders are furnished to allow complete control of spacecraft operations. Redundant tracking signals will also be provided. The spacecraft data system incorporates patchboards for the experiment package analog data channels and commands. A timer is provided for timing signal functions to the experiments, for transmission of spacecraft mission time, and for back-up telemetry terminations.

The subsystem is designed for compatibility with the NASA Space Tracking and Data Acquisition Network (STADAN) and will meet the requirements of the NASA/GSFC Aerospace Data Systems Standards.

The data storage unit is common to all missions and consists of a solid-state memory with a total capacity of 8,000-six bit words. Capability for either a 4,000 word or 8,000 word mode is selectable by ground command. The telemetry transmitter is common to all missions. Capability is provided to phase modulate the carrier with the split phase from the spacecraft data handling assembly, or, in Mission III, with the tracking signal from the Range and Range Rate Receiver. Frequency is 136 mHz nominal, with output power of 0.3 watt for Missions I and II, and 10 watts for Mission III.

The antenna is common to all missions and spherical coverage is provided for telemetry, tracking and command in the 136 to 150 mHz frequency band. The gain is above -2dB in any direction and the maximum power is 20 watts.

The command subsystem design is compatible with the Tone Command Standard (NASA/GSFC Aerospace Data Systems Standards, Part II, Section I). The receivers, decoders and timer are common to all missions. The command transmitters are amplitude modulated with a sequence of four audio tones. The command receiver frequency is 150 mHz, nominal.

A Minitrack tracking beacon is employed for Missions I and II. Back-up tracking capability is provided by using the spacecraft telemetry transmitter. Tracking for Mission III is accomplished by the provision of redundant NASA/GSFC Range and Range Rate transponders. The Minitrack beacon frequency is 136 mHz, nominal, while the nominal frequency of the Range and Range Rate transponder is 150 mHz.

2.1.5 Thermal Control

Thermal control of the experiment package will be accomplished by the provision of a cold plate (heat sink) for heat rejection. The cold plate will be controlled to 40°F + 0°, -5°F during pre-launch, powered flight and orbital operations. In addition, the control set point shall be adjustable, on ground command, in increments of 1°F within the range of 35°F to 45°F. The temperature limits of the spacecraft battery will be maintained between 30 °F and 80 °F for all mission phases. Components in the attitude control, telemetry, tracking and command, and electrical power and distribution subsystems will be maintained between 0 °F and 130 °F for all mission phases.

Thermal control of the experiment package will be accomplished by controlled radiation of heat from the cold plate using thermal louvers. Sensors, a stepping motor and gear train will be used to control the opening area of the louvers. During pre-launch, the cold plate temperature will be maintained at 37 °F ± 2 °F by the circulation of a ground coolant through a tube brazed to the cold plate. Temperature control during powered flight will be accomplished by the utilization of the thermal capacitance properties of the thermal control subsystem and the experiment package. Passive thermal control techniques will be employed for the spacecraft subsystem components, including thermal coatings, radiation and conduction insulation, and heat sinks.

2.1.6 Electrical Power and Distribution

The spacecraft will provide power to the experiment package through a patchboard as follows:

Voltage	Continuous Power (Watts)	Peak Power (Watts)
+ 27.5 ± 2.5 VDC	64	89 (6 mins. of every hour)
+ 15 ± 2.5 VDC	3	3
- 15 ± 0.5 VDC	3	3
+ 5 ± 0.2 VDC	7	7

In addition to the experiment power, spacecraft operation power, including conversion inefficiencies and a 6 percent design margin, will be furnished as follows:

Mission	Voltage	Average Power (Watts)	Peak Power (Watts)
I	+28 ± 2.8Vdc	52	78
II	+28 ± 2.8Vdc	48	95
III	+28 ± 2.8Vdc	46	115

Electrical power will be supplied by an on-board solar array and battery during the mission, and ground equipment before launch. Pre-launch power, (up to launch minus 20 minutes), is unregulated, AGE-supplied, 34Vdc ± 20%, provided to the spacecraft via an umbilical. During the sunlit portions of the missions, power will be supplied by an eight-panel deployed solar array, common to all missions. The nominal voltage output of the array is 39 ± 4Vdc. On-board storage batteries consist of a 9 ampere-hour nickel cadmium battery for mission Types I and II, and two 9 ampere-hour nickel cadmium batteries for mission Type III. A power control unit will be incorporated, common to all missions, with provisions for battery charge regulation, and power distribution, conversion and control.

The spacecraft design will be in accordance with the Electro-magnetic Interference (EMI) requirements of MIL-STD-461A. Electro-explosive devices will be designed with power and firing signals separated and isolated from primary bus power, by means of separate harnesses and connectors, and protected with safety devices. Cable harnesses will be grouped according to signals or power, and will be separated where practical. Twisted conductors will be used where appropriate, and connectors shall have effective shield terminations.

A single-point ground system will be used. Components will be isolated from the their cases and separate data, command and power returns will be provided to the experiment package, isolated from spacecraft functions.

3.0 APPROACH

The overall study approach, consistent with the contract work statement, was addressed to two major areas of investigation and evaluation, in terms of interdependent system and subsystem trade-offs and design definition. The first area included: the optimization of the baseline thermal control subsystem; the determination of system and subsystem impacts resulting from specified reductions in experiment power requirements; the evaluation of the effects on the baseline TT&C subsystem of incorporating provisions for a television system for Missions I and II, and of using the Manned Space Flight Network (MSFN); and the refinement and optimization of the baseline attitude control subsystem. The results of these evaluations are presented in Sections III, IV, V, and VI of this volume. For the second area, effort was devoted to the investigation of the impact on the baseline design of the availability of the Space Shuttle Vehicle (SSV) as a launch and retrieval vehicle. The results of this portion of the study are furnished in Section VII of this volume.

III. THERMAL CONTROL SUBSYSTEM

BASELINE OPTIMIZATION

1.0 EXPERIMENT REQUIREMENTS

Thermal control of the experiment package shall be accomplished by providing a coldplate for experiment heat rejection. The plate shall be controlled to 40 ± 5°F and be capable of dissipating the following experiment thermal loads:

(1) Peak - 350 BTU/hr for 10 minutes per hour

(2) Continuous - 180 BTU/hr to 270 BTU/hr

The BRM experiment compartment shall be designed to thermally isolate the experiment package from the BRM structure, except at the coldplate thermal interface. Heat transfer between the experiment package and the BRM experiment compartment, exclusive of the coldplate, shall be limited to ±50 BTU/hr with a 70°F experiment package external structure temperature.

The BRM shall provide thermally isolated structural mounting points exterior to the experiment volume for the experiment package. These mounting points shall be located to permit connection of either the experiment package itself or its thermal interface to the coldplate. The coldplate shall be designed to permit a mechanical connection with the experiment package thermal interface for conductive heat transfer. The coldplate set point temperature shall be adjustable to increments of 1°F during orbit within a minimum range of ±5°F by ground command.

2.0 BACKGROUND

The previous Bioexplorer study concluded that the coldplate thermal control requirements could be met with louvers located on the coldplate as shown in Figure III-1.

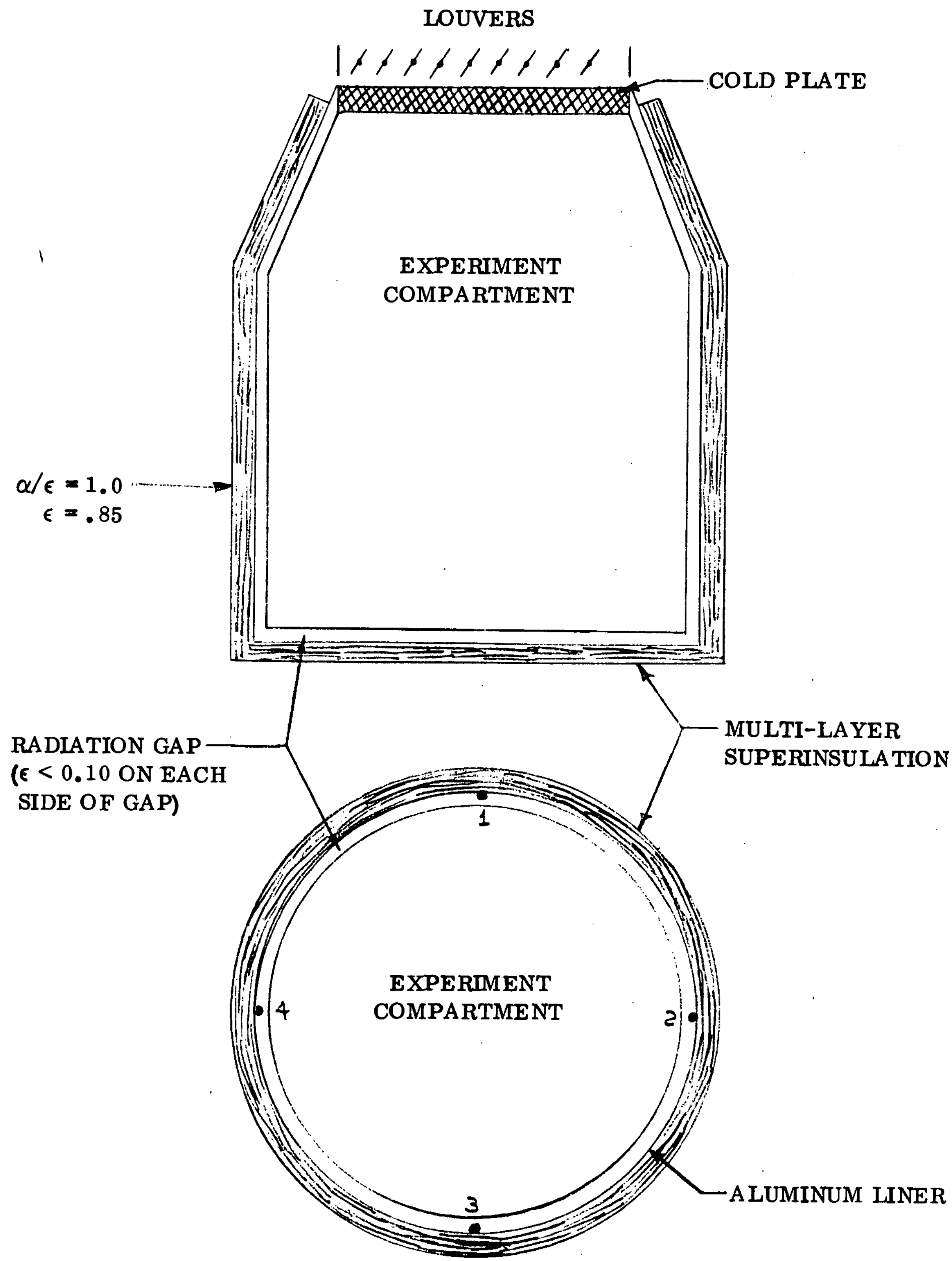


Figure III-1. Experiment Compartment Temperature Control (Louver Approach)

Although the average orbital sink temperature was low enough to maintain the experiment coldplate at the required 40°F temperature, there were time periods during the orbit in which the sink temperature would be higher than the experiment coldplate. As a result, the system heat capacity would be required to attenuate the coldplate transients. The coldplate temperature for different heat capacities is shown in Figure III-2. When only the capacitance effect of the colplate was assumed the coldplate far exceeded the maximum required temperature of 40°F (wc_p = 4.4 or 5.9). However, if one assumed that the experiment mass was part of the overall capacitance (wc_p = 35) the orbital transients were almost completely attenuated with the coldplate remaining within specification. The validity of assuming the experiment package capacitance as a damping mechanism was dependent upon the materials selected in the design of the package as well as having a low thermal resistance path to the coldplate.

Previous experience on the Biosatellite Program had indicated to GE that such a coupling would exist. However, without the details of how the experiment packages were to be designed the assumption might be invalid. Therefore, GE requested NASA/ARC to review the design of the experiment packages and to judge whether indeed the coupling effect would exist. NASA/ARC complied with GE's request, but concluded that the coldplate should be designed independent of the experiment capacitance. Thus, this section of the study is concerned with determining a method of thermal control independent of the experiment package thermal capacitance.

3.0 DESIGN APPROACH

3.1 WAX HEATSINK

To obtain the required WC_p of 33 BTU/°F-hr., it would be necessary to increase the weight of the coldplate by almost 70 lbs. Considering the overall weight limitations of the system, this weight penalty is unacceptable. A more practical solution would be to employ a phase change material which would melt during the hot portion of the orbit and resolidify during the night cycle. Figure III-2 shows the effect of adding 2.2 lbs of wax

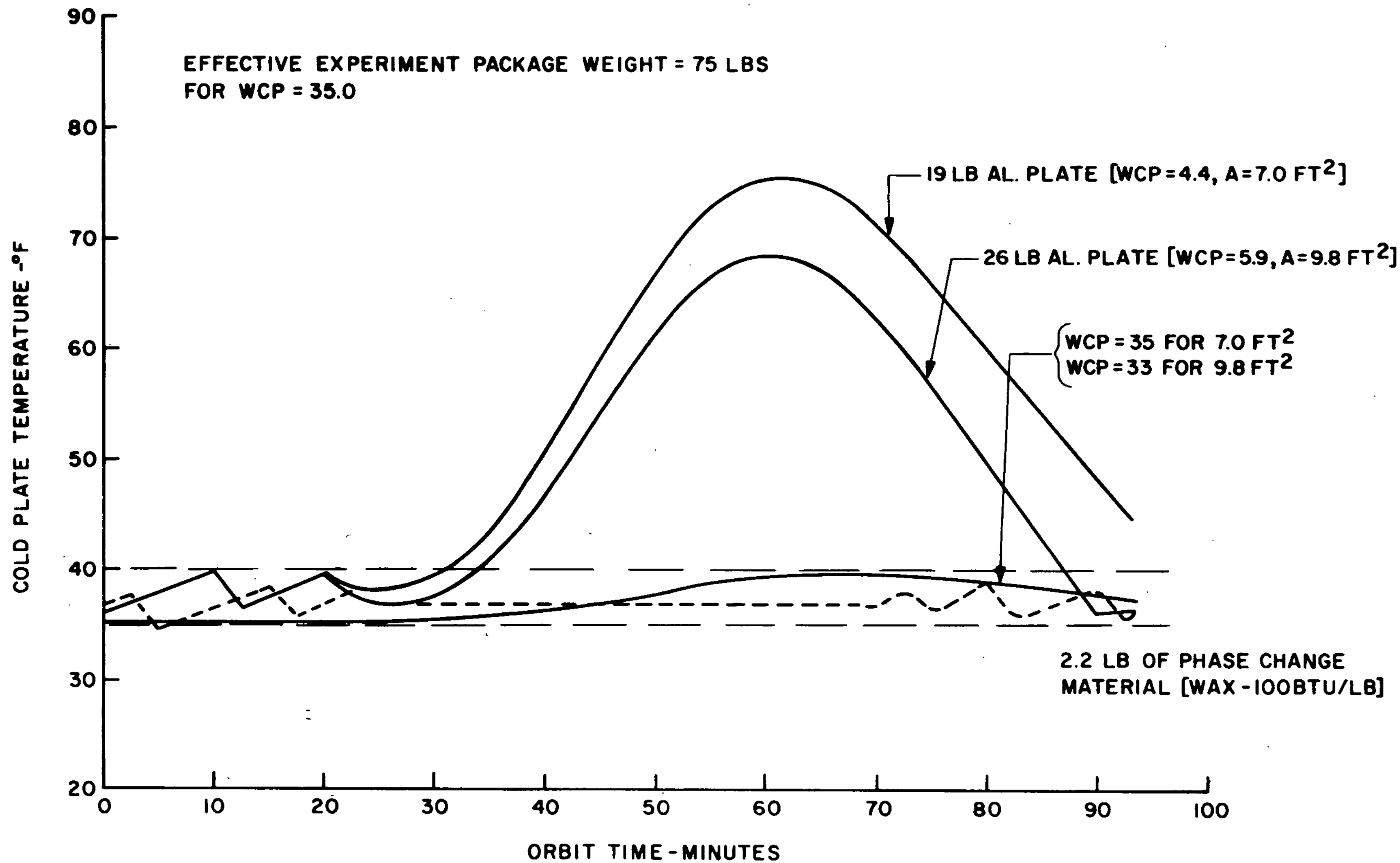


Figure III-2. Aluminum Coldplate Transient Temperature Response into the coldplate. As can be seen, by using the fusion heat of the wax, the orbital transients can be greatly reduced. Tetradecane has a melting temperature of 42°F. In tests conducted at GE, it has demonstrated repeatable performance with varying heat loads for over 150 cycles. The main problem with using the heat of fusion as a control mechanism is that once set, it cannot be changed since the melt temperature will remain constant. Thus, such a system will not be capable of adjusting the coldplate temperature of ±5°F in 1°F increments by ground command.

3.2 RELOCATION OF HEATSINK

A second approach, which meets all the requirements and has been selected as the design solution is to use the conical section surrounding the experiment package as the heat rejection area. The heat could be conducted directly from the experiment package to the conical surface or into the top circular section and then finned to the conical section as shown in Figures III-3A and B.

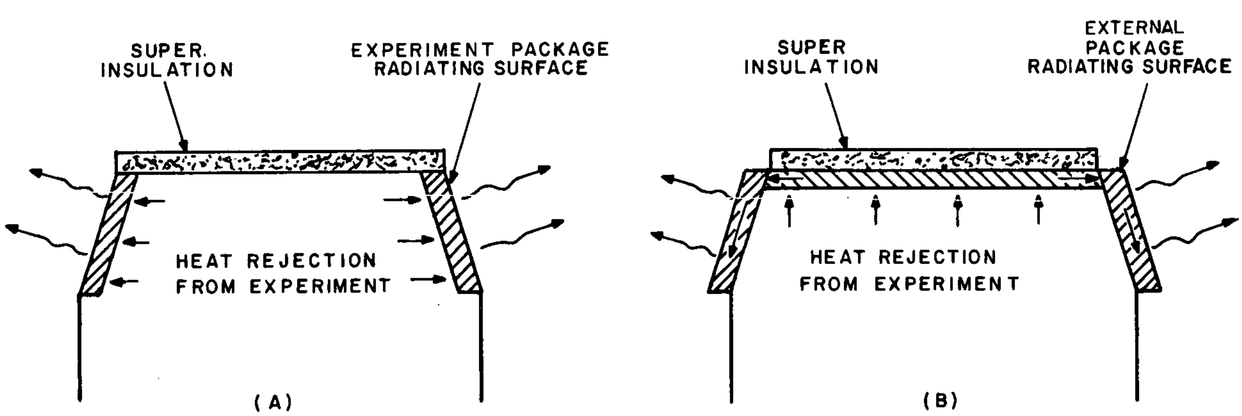


Figure III-3. Heat Flow

Figure III-4 gives the maximum and minimum orbital sink temperatures on the conical surface for two different variable emittance devices. The $\epsilon = 0.85$ curves show the resultant sink temperatures for the variable radiation cover device shown in Figures III-5 and 6. The maximum sink temperature at $\epsilon = 0.70$ is for a louver system as shown in Figures III-7 and 8. Sink temperatures will be higher for the louver case because of the resultant higher solar absorbtivity of this mechanism. In either case, the maximum sink temperature which will be encountered will be -16°F well below the temperature which had previously been calculated for the top surface.

Figure III-9 presents the required radiating area if the experiment heat load were conducted directly into the conical surface. To maintain the experiment interface at 40°F, the cover mechanism would require 6 ft^2 of radiating area while the louver device would require 8.2 ft^2.

If the interface between the experiment package and the coldplate remained at the front surface as shown in Figure III-3b, the required radiating area in the conical section would increase. The conical surface in this case would be acting as a radiating fin with a resultant decrease in performance due to the temperature gradient which would exist.

Figure III-10 shows the resultant fin effectiveness as a function of thickness for the conical section. The required radiating area would be 8.5 ft^2 for the cover and 13 ft^2 for the shutters.

With the interface at the front surface, a large radial temperature differential could exist. Figure III-11 presents this temperature differential as a function of weight and the distribution of the heat load. It can be seen that if the heat input is assumed uniform over the entire plate rather large temperature differentials will exist even at high plate weights. However, if the heat input is concentrated along the outer circumference, the penalties both from the weight and temperature differential would be less. An annular heat distribution of two inches would be ideal from this standpoint.

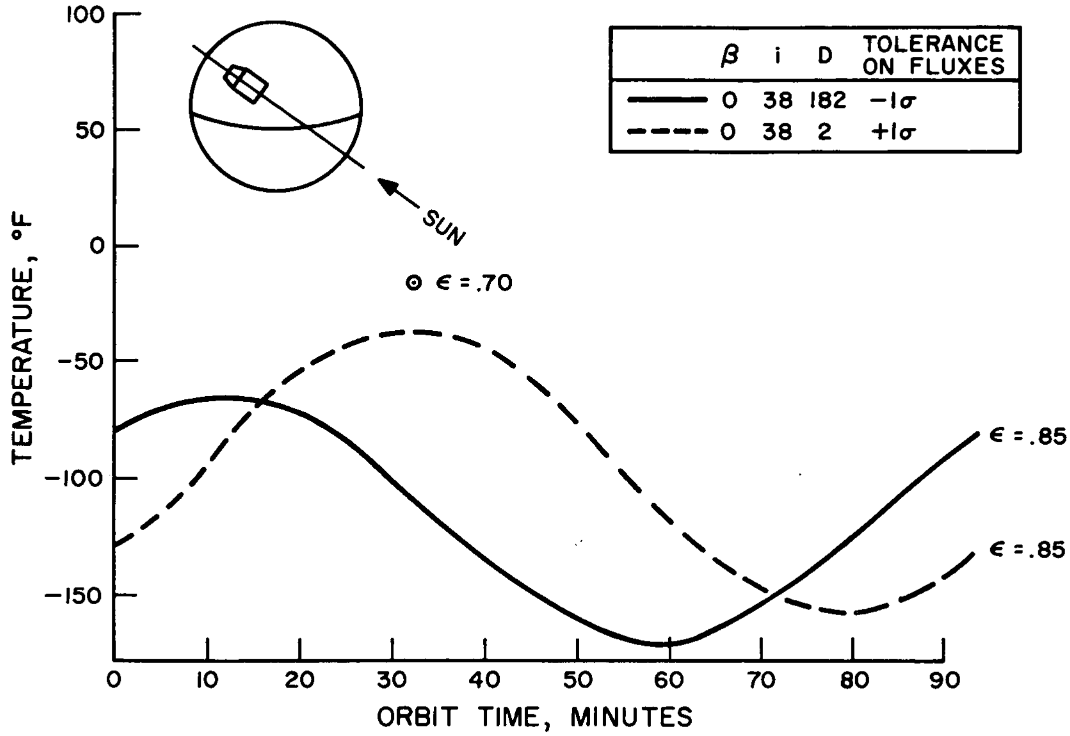


Figure III-4. Average Sink Temperature History For 12.5° Cone

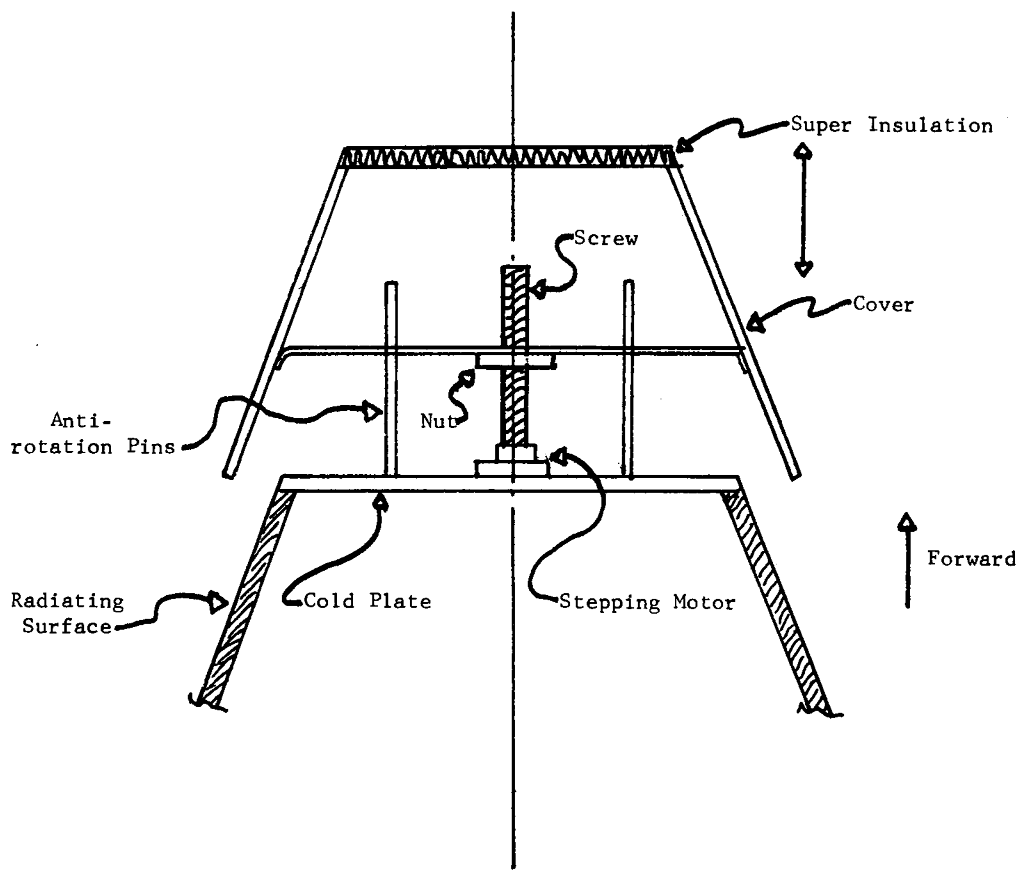


Figure III-5. Variable Radiation Cover - Aft of Coldplate

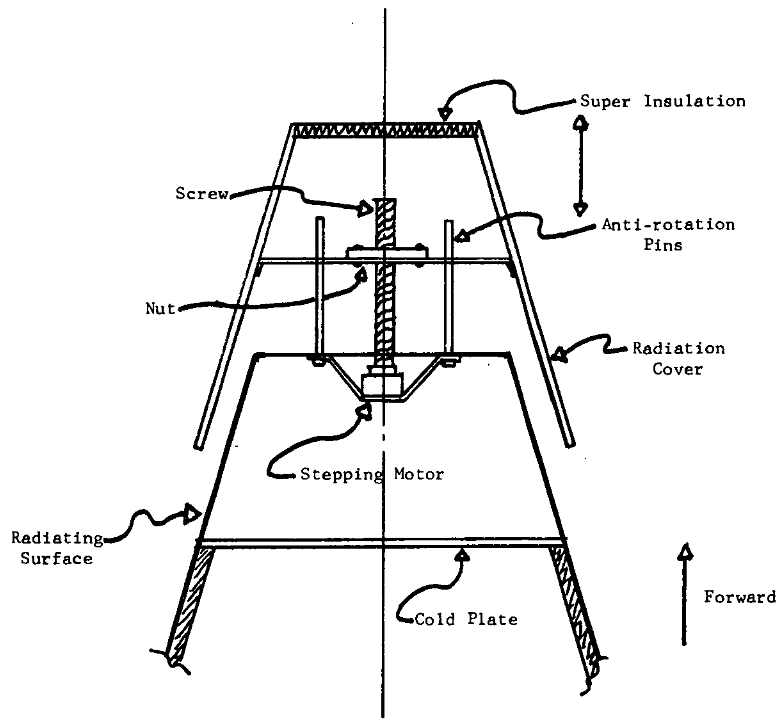


Figure III-6. Variable Radiation Cover - Forward of Coldplate

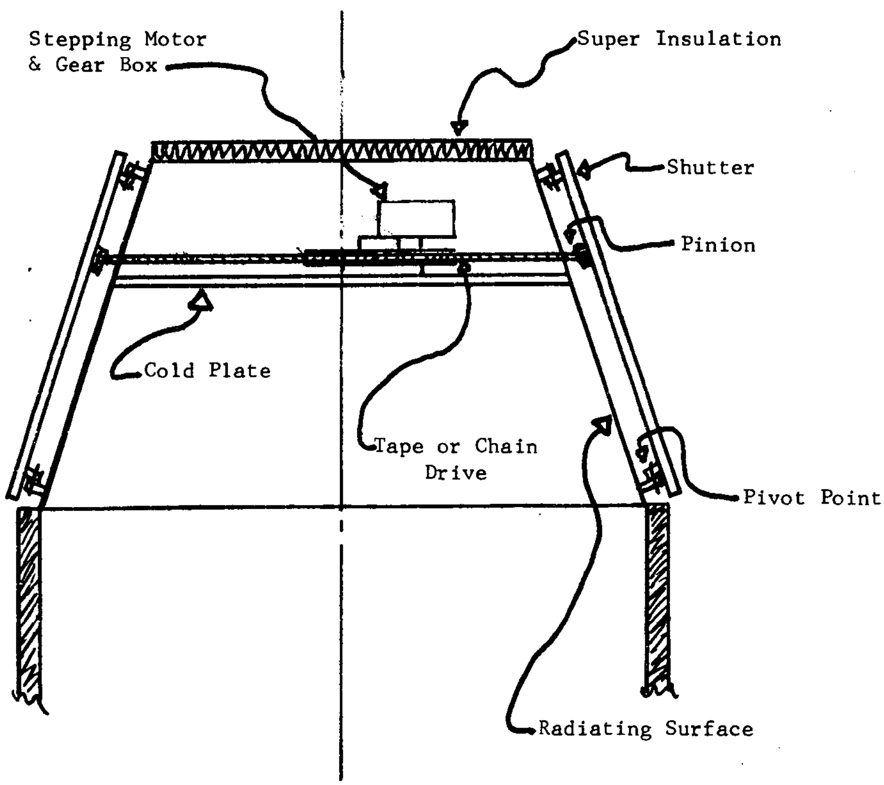


Figure III-7. Shutters - Aft of Coldplate

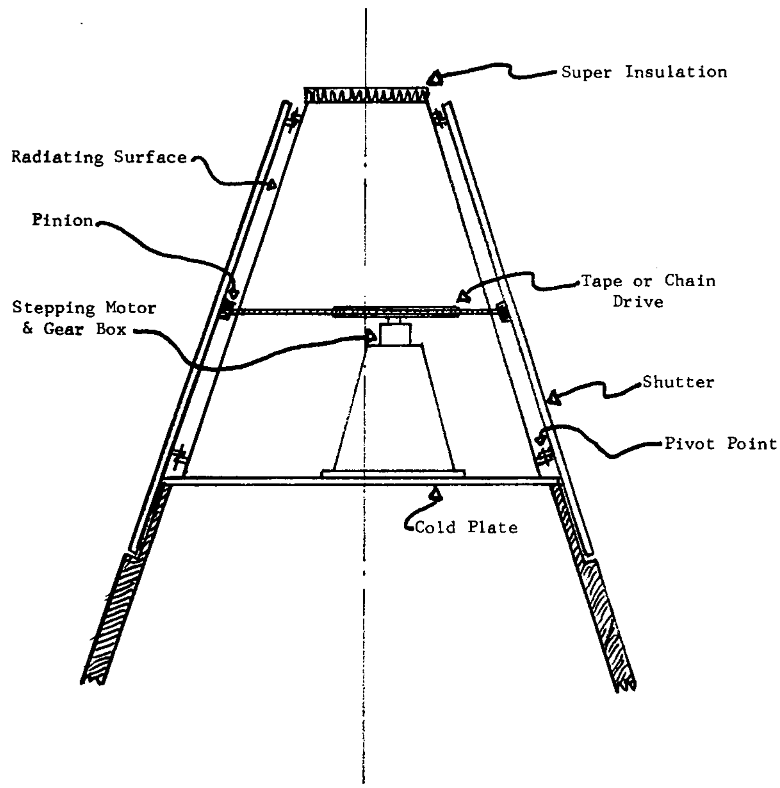


Figure III-8. Shutters - Forward of Coldplate

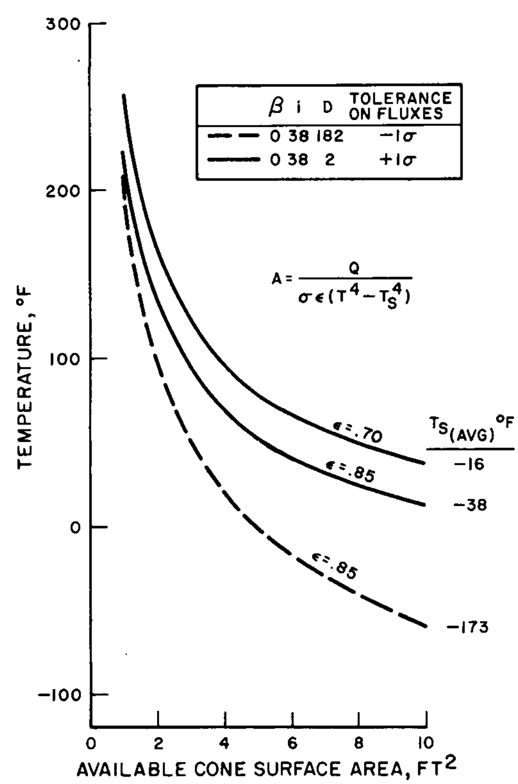


Figure III-9. Average Cone Temperature For Experiment Heat Load = 270 BTU/HR

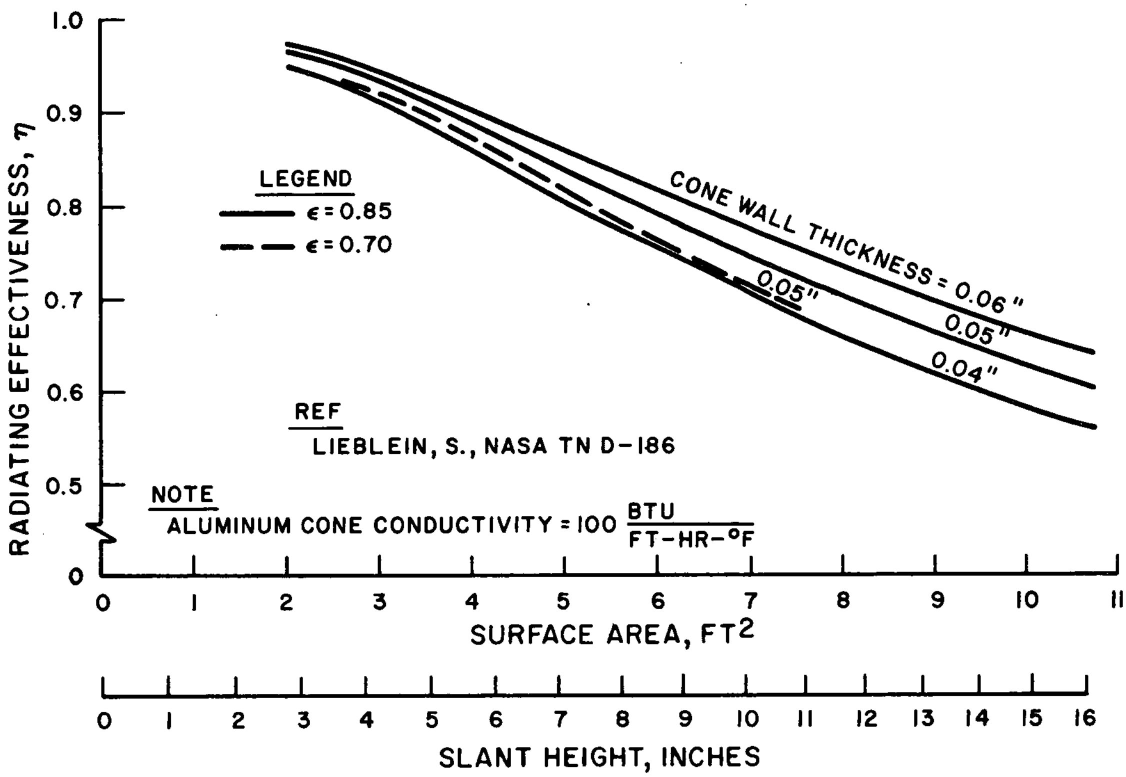


Figure III-10. Fin Effectiveness of 12.5° Cone

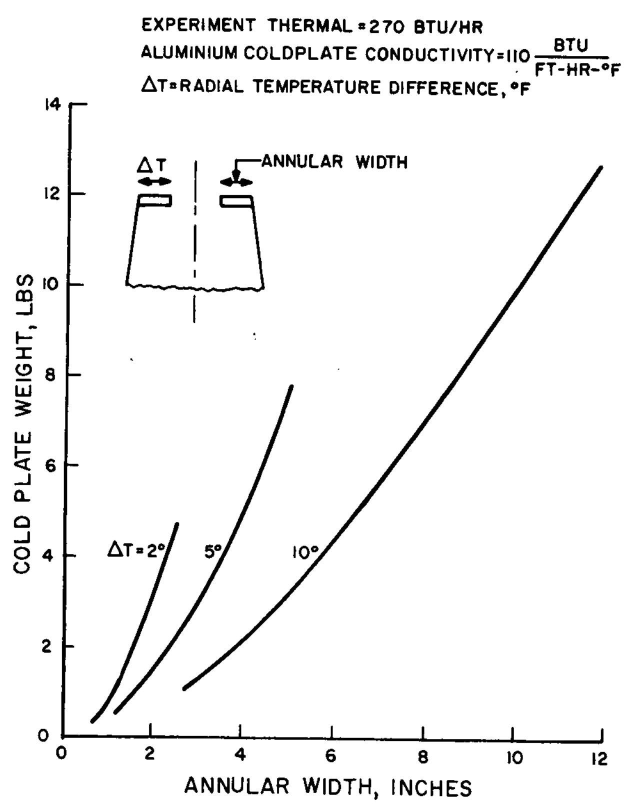


Figure III-11. Coldplate Weight as a Function of Annular Width and Radial Temperature Difference

4.0 THERMAL CONTROL MECHANISMS

Mechanical implementation of the variable emittance devices under consideration are shown in Figures III-5, -6, -7, and -8. The variable radiation cover offers the advantage of requiring less radiating area and no change in the experiment package. It can be either placed aft of the coldplate as shown in Figure III-5 or forward as in Figure III-6 depending upon the required radiating area. A screw drive would position the cover as a function of the coldplate temperature. The stepping motor, actuated by a digital control system using the coldplate temperature sensor, would be used to position the screw. A similar control system was employed on the Biosatellite modulating valve with a high degree of sucess.

The louver system can also be employed aft or forward of the coldplate (Figures III-7 and III-8). However, if it is positioned aft of the coldplate, the experiment conical section diameter must be reduced by one inch to obtain the required clearance with the Scout Shroud. If it is positioned forward, no experiment change would be required. The control system would be the same as for the cover with the exception that a tape or chain drive would be used.

All four methods discussed above would provide the required coldplate thermal control. The methods utilizing the aft conical sections would probably result in a lighter weight and also maintain the current vehicle length. The actual selection is dependent upon the final experiment interface requirements as well as the results of a detailed mechanical design trade-off.

5.0 EFFECTS OF REDUCED HEAT LOAD

The reduced experiment thermal load of 136 BTU/hr would not change the above results exeept to reduce the required radiating area by approximately 50%. This would simplify the design of the variable emittance system since its control area could also be reduced by 50%. In addition, as can be seen from Figure III-12, any temperature differential problem existing in the coldplate will be eased.

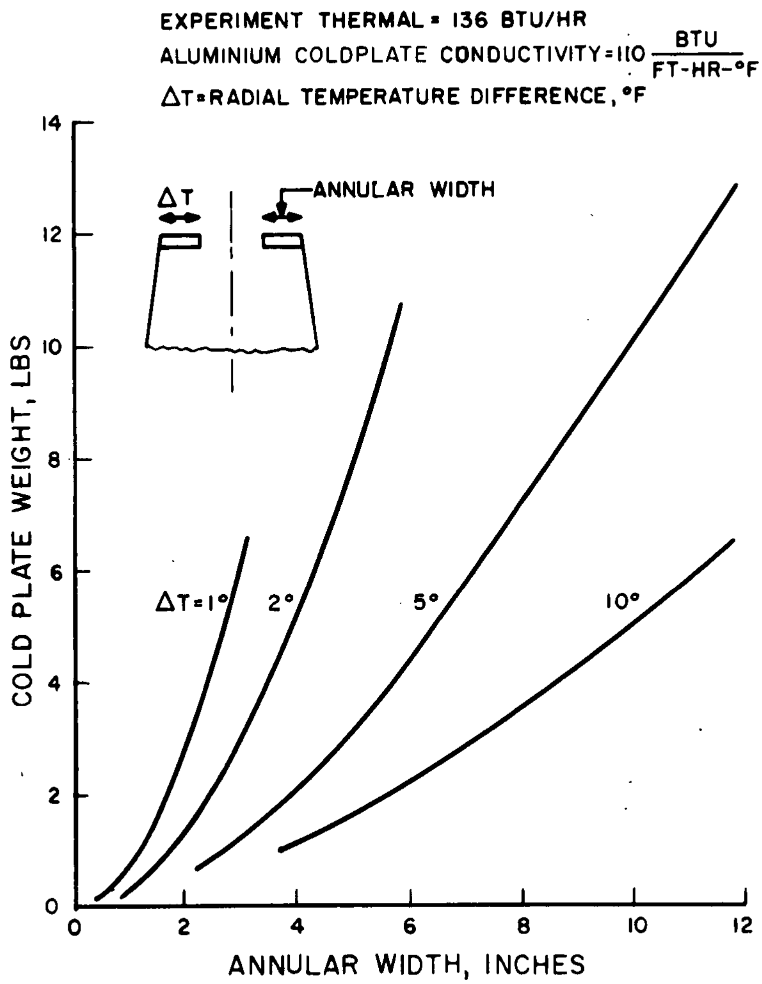


Figure III-12. Coldplate Weight as a Function of Annular Width and Radial Temperature Difference

IV. REVISED POWER PROFILE EFFECTS AND CONSIDERATIONS

1.0 EFFECTS OF A REDUCTION IN EXPERIMENT POWER

The baseline experiment power requirements are:

(1) 27.5 ±2.5 Vdc - 89 watts peak for 6 min/hr
64 watts, continuous
(2) 15 ±2.5 Vdc - 3 watts continuous
(3) -15 ±0.5 Vdc - 3 watts continuous
(4) +5 ±.02 Vdc - 7 watts, continuous

The option which GE-RESD was asked to evaluate as part of the Bioresearch Study reduced the experiment requirements as follows:

1.1 EXPERIMENT (ALL MISSIONS)

(1) 27.5 ±2.5 Vdc - 111 watts for 3 min/hr
31 watts continuous
(2) 15 ±2.5 Vdc - 2 watts continuous
(3) -15 ±2.5 Vdc - 2 watts continuous
(4) 5 ±0.02 Vdc - 5 watts continuous

1.2 SPECIAL EXPERIMENT (MISSION I)

(1) 27.5 ±2.5 V - 45 watts peak 2 min/orbit
115 watts peak 15 sec/orbit
20 watts continuous
(2) 15 ±2.5 V - 2 watts continuous
(3) -15 ±2.5 V - 2 watts continuous
(4) 5 V ±0.02 V - 5 watts continuous

The effects of the reductions in the experiment power requirements were found to have little effect on the design of the EP&D subsystem with the obvious exception of the solar array and batteries. Basic trade-off studies which were conducted to determine, voltage and charge regulation as well as power distribution are still valid and have been documented in the Bioexplorer Final Report. The resultant electrical interface schematic derived during the previous Bioexplorer Study is shown in Figure IV-1. The effort in the Bioresearch Module was based on this electrical system and concentrated on the system effects resulting from changes in the solar array design.

The baseline Bioexplorer design is shown in Figure IV-2. The solar array consisted of eight panels, deployed perpendicular to the spacecraft roll axis, oriented to face the sun. Preliminary analysis indicated that, considering the spacecraft power profile and the Scout shroud limitation, insufficient spacecraft body area was available as a solar array substructure to provide the required power output. Therefore additional area would have to be obtained by either deploying panels or by telescoping the additional array into the main vehicle structure. The panel approach was selected on the basis of its flight proven concept, greater simplification of the vehicle structural design and lower weight.

Preliminary analysis of the reduction in experiment power however indicated that a body mounted solar array should be reconsidered. Such a design change would not only result in a change to the basic vehicle configuration but would also require revisions in the attitude control since the vehicle spin axis would now have to be perpendicular rather than parallel to the sun vector. The resultant change in vehicle attitude would result in a redistribution of the incident orbital heat fluxes thus requiring modifications in the thermal control design.

The study effort was directed towards trading off the paddle configuration considering the reduced power profile versus the body mounted array. Both concepts are shown in Figure IV-3.

2.0 SPACECRAFT POWER PROFILE

The spacecraft power requirements are as follows.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

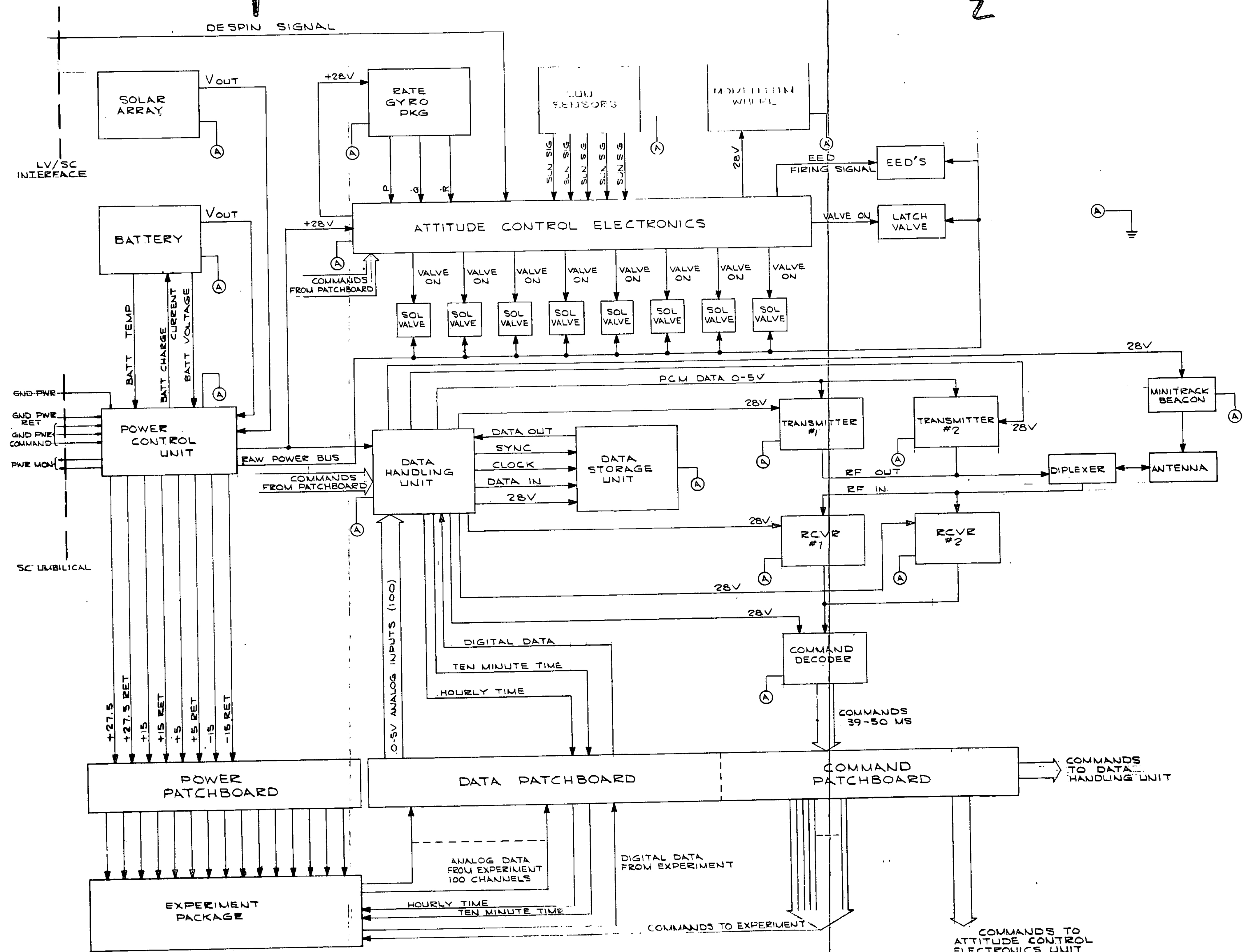


Figure IV-1. Block Diagram: Mission I Electrical Interfaces

I

FOLDOUT FRAME 1

FOLDOUT FRAME 2

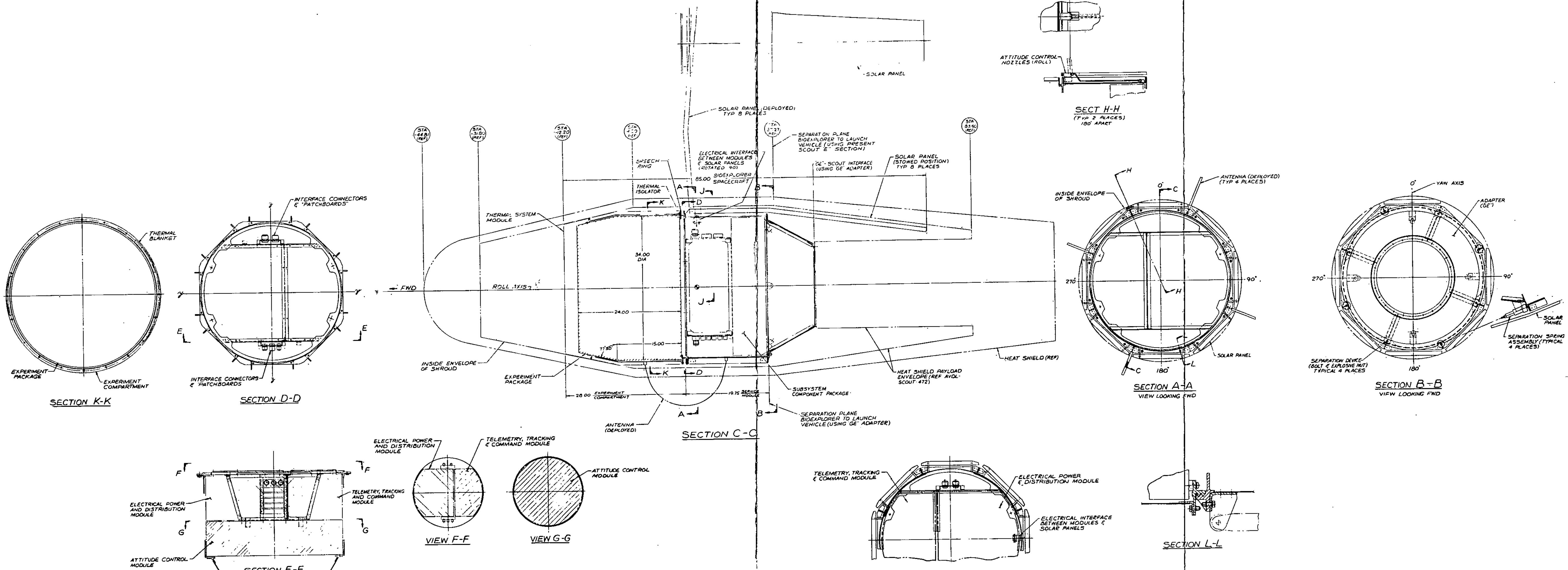


Figure IV-2. Bioexplorer Inboard Profile

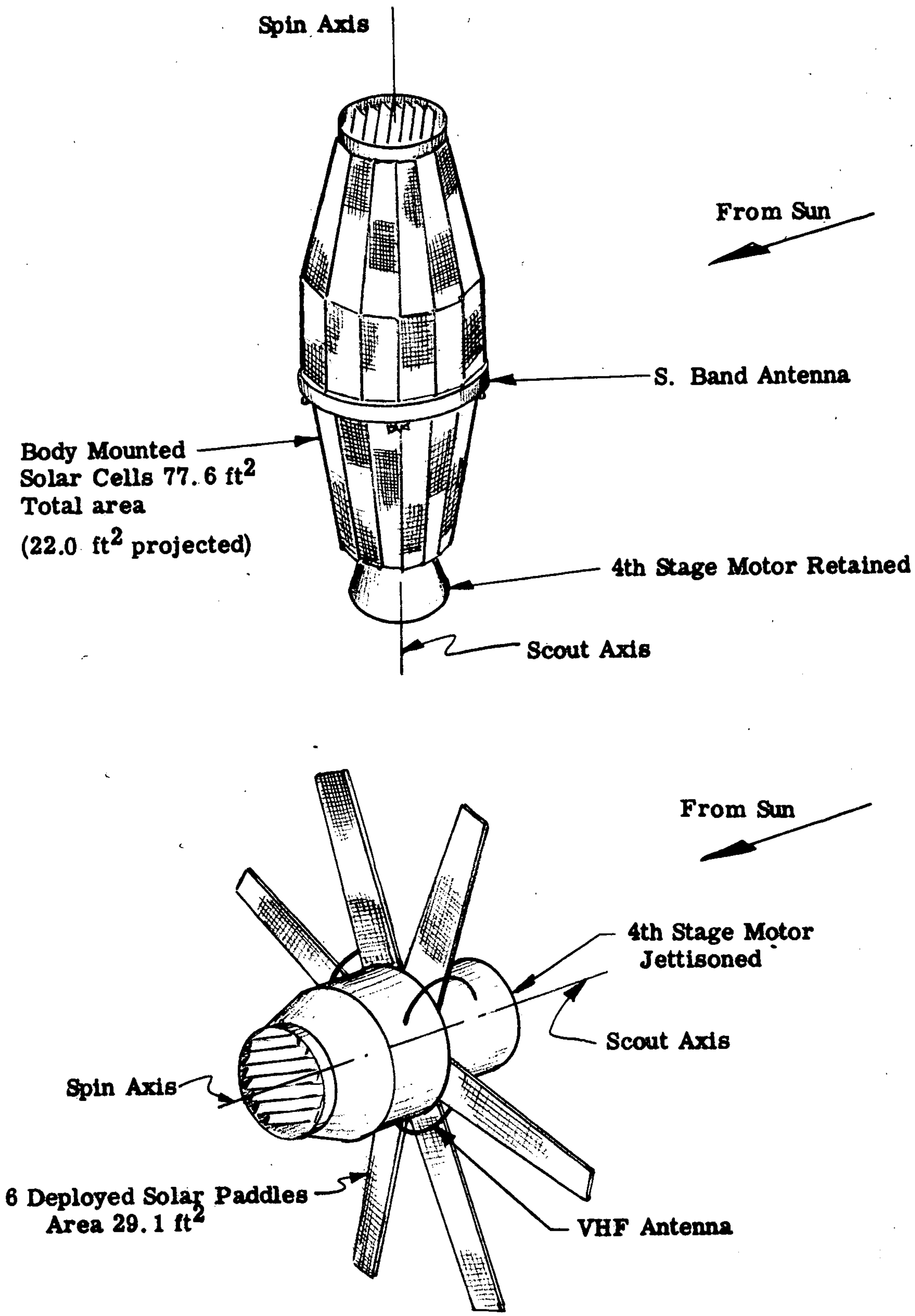


Figure IV-3. Paddle Configuration and Body Mounted Array

2.1 EXPERIMENTS (ALL MISSIONS)

Continuous	40 watts
Peak	120 watts for 3 min/hr

SPECIAL EXPERIMENT (MISSION I ONLY)

Continuous	29 watts
Peak	54 watts for 2 min/orbit
	159 watts for 15 sec/orbit

2.2 ATTITUDE CONTROL

Mission I

Rate Gyros	11 W Cont.	18 W Peak
Electronics	2.5 W Cont.	5 W Peak
Momentum Wheel	3.1 W Cont.	3.1 W Peak
Pneumatic Valves	Negligible	
Total	16.6 W Cont.	26.1 W Peak

Mission II

Electronics	2.5 W Cont.	5 W Peak
Deployable Rods	0 W Cont.	16.2 W Peak 7 min/day
Pneumatic Valves	Negligible	
Total	2.5 W Cont.	21.2 W Peak

Note: Lack of rate gyros on Mission II assumes a body-mounted solar array with rates derived from sun sensors as discussed in Section VI-6.

2.3 TT&C

The TT&C power requirements are based upon the S-band system with the capability of utilizing the MSFN net, transmitting television data, and being adapted to the frog otolith experiment rather than the VHF system which was originally specified for the Bioexplorer.

S-Band (All Missions)

Components	Continuous Power	Peak Power
Memory	1.0	1.0
Multicoder	0	8.4
Clock	1.6	1.6
Command Receiver	4.2	5.6
Command Decoder	0.1	4.0
Transmitter	0	25.0
	6.9	45.6

2.4 THERMAL CONTROL

5 watts cont. - 5 watts peak

2.5 EP&D SUBSYSTEM

2.5.1 Voltage Conversion Power Estimates

The above subsystem power requirements are based upon the regulated voltage requirements to the components. For purposes of sizing the EP&D S/S, an estimate of raw power (before conversion) is necessary. Based on past experience an overall average conversion efficiency of 80% was assumed. This includes primary regulation, voltage conversion and any secondary regulation necessary. Analysis of voltage regulation schemes in the Bioexploer study indicated that this is a realistic, but conservative value. The resultant conversion power requirements are:

	Mission I	Mission II
Experiment	10 W Cont. 30 W Peak	10 W Cont. 30 W Peak
Attitude Control	4 W Cont. 7 W Peak	0.6 W Cont. 5.3 W Peak
Thermal Control	1 W Cont. 1 W Peak	1 W Cont. 1 W Peak
TT&C	1.6 W Cont. 11.4 W Peak	1.6 W Cont. 11.4 W Peak
Total	16.6 W Cont. 49.4 W Peak	13.2 W Cont. 47.7 W Peak

2.5.2 Battery Charging Requirements

The previous Bioexplorer study has shown that at the minimum vehicle altitudes of 150 nautical miles the maximum eclipse time would be 37 minutes with an orbital period of 89 minutes. A review of the analysis based upon orbit decay and the Scout injection tolerances have shown an insignificant effect due to the change in vehicle configuration (body mounted or reduced solar paddles). Thus the system parameters discussed in Section A of the initial study are still valid and were used for the sizing of the battery and array requirements.

The maximum battery charging requirements were based upon a 37 minute eclipse time consisting of 29 minutes of continuous operation plus three minutes of peak experiment power and a peak TT&C mode occurring during a five minute station pass.

Mission I	Mission II
29 Minutes of Continuous Mode	
Experiment - 50 watts	50 W
Thermal - 6 watts	6 W
Attitude Control - 21 watts	3.1 W
TT&C - 7.5 watts	7.5 W
84.5	66.6 W
6% Margin - 5.1 watts	4 W
89.6 watts for 29 Minutes	70.6 W
= 43.3 WHR	= 34.1 WHR
3 Minutes of Peak Mode -	
Experiment - 150 watts	150 W
Thermal - 6 W	6 W
Attitude Control - 21 W	3.1 W
TT&C - 7.5 W	7.5 W
184.5 W	166.6 W
6% Margin - 11.1 W	10 W
195.6 Watts for 3 Min (.05 Hr)	176.6 W
= 9.8 WHR	= 8.8 WHR

The average battery charging rate during the 52 minute sunlit portion of the orbit is:

Mission I	Mission II
75.2 Watts	61.6 Watts

But energy storage has inefficiencies associated with it, and an estimate was made that 67% efficiency was realistic for this case, based on the following formula:

$$W_{input} = W_L \ (1.21)\ (1.10)\ (1.08)\ (1.04) = 1.5\ W_L$$

Where:

W_{input} is power into battery

W_L is power out of battery to loads

1.21 = charge voltage efficiency
1.10 = amp hour efficiency
1.08 = charge regulator efficiency
1.04 = diodes efficiency

With this 1.5 factor considered, the EP&D subsystem must provide 112.7 watts for Mission I and 92.5 watts for Mission II to the battery during sunlight.

Thus far, two of the three power usages by the EP&D S/S have been calculated. The third, power to energize sensing and control circuits associated with charging the battery, was estimated at 3 watts for all missions, based on predicted charge times and charge technique implementation.

The total EP&D requirements are therefore:

	Mission I		Mission II
Conversion	16.5 W 49.6 W	Cont. Peak	13.1 W 47.9
Battery Charging	112.7 W		92.5 W
Charging Circuits	3 W		3 W
Total	132.2 W 165.3 W	Cont. Peak	108.6 W 143.4 W

2.6 TOTAL SPACECRAFT POWER REQUIREMENTS

2.6.1 The sunlight power requirements are:

	Mission I		Mission II
Experiment	40 W 120 W	Cont. Peak	40 W 120 W
Attitude Control	17 W 26 W	Cont. Peak	2.5 W 21.2 W
Thermal Control	5 W 5 W	Cont. Peak	5 W 5 W
TT&C	6 W 46 W	Cont. Peak	6 W 46.4 W
Subtotal	68 W 197.4 W	Cont. Peak	53.5 W 192.6 W
+6% Margin	4.1 W 12.0 W	Cont. Peak	3.2 W 12.0 W
Subtotal	72.1 W 209.4 W	Cont. Peak	56.7 W 204.6 W
+EP&D (Includes 6% Margin)	132.2 W 165.3 W	Cont. Peak	108.6 W 143.4 W
Total	204.3 W 374.7 W	Cont. Peak	165.3 W 348 W

2.6.2 Eclipse Energy

Mission I	Mission II
65.1 WHR	53.3 WHR

Time to First Sun Energy (38 minutes) is assumed to be equal to eclipse energy.

2.7 COMPARISON WITH BIOEXPLORER REQUIREMENTS

	Mission I Std. Exper.		Mission I Spec Exp.	Mission II	
	BRM	BIOEX		BRM	BIOEX
Continuous Sunlight Power	204.3 W	292 W	172.2 W	165.3 W	289 W
Peak Sunlight Power	374.7 W	348 W	270.1 W	348 W	361 W
Eclipse Energy	65.1 WHR	89 WHR	55.2 WHR	53.3 WHR	87.1 WHR

3.0 BODY-MOUNTED SOLAR ARRAY

Based upon the previous derived power profile the solar array end of life design requirements are governed by the continuous power required in Mission I, 204.3 watts. Peak power requirements in excess of the array output will be supplied by the battery (in a load sharing mode if such peaks occur during sunlight).

The experiment rate requirements necessitating a high spin rate for Mission II and a very low spin rate for Mission I have a major effect on the design of the body mounted array. Mission II requirements dictate that the array be continuous around the spacecraft circumference. For the purpose of maintaining commonality between the array design for Missions I and II it would also be necessary to supply a continuous array for Mission I (see Figure IV-3 top).

3.1 ARRAY THERMAL ANALYSIS

Prior to sizing the array it was necessary to determine the solar array temperatures to obtain the resultant array efficiency. The analyses were performed for the following flight and heat flux conditions.

- Angle between earth-sun line and orbit plane = 0°
- Inclination - 38°
- Day 182 (launch hour 0842)
- Solar flux (-1 σ) = 420.0 BTU/hr ft^2
- Earth flux (+1 σ) = 74.87 BTU/hr ft^2
- Albedo flux (+1 σ) = 0.44 BTU/hr ft^2

Although selecting solar aphelion will not result in a maximum array temperature, the selected orbital conditions represent the minimum sunlit orbit and, therefore, the worst case array sizing conditions. Two spacecraft orientation cases were analyzed:

(1) Roll axis normal to sun spacecraft line and the ecliptic plane; zero spin.

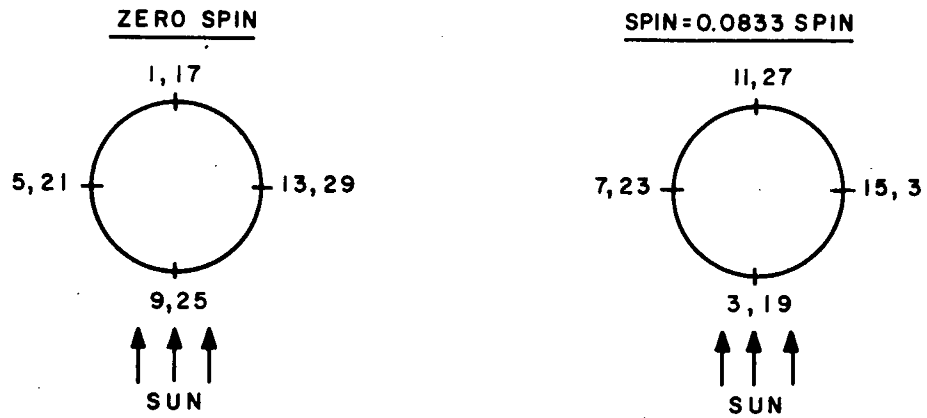
(2) Same as (1) except spin = 0.0833 rpm

The solar cell temperature responses for the two cases evaluated are shown in the figures indexed in Table IV-1.

TABLE IV-1. INDEX OF SOLAR CELL TEMPERATURE FIGURES

Spacecraft Attitude	Figure Number
• Roll axis normal to Sun spacecraft line and ecliptic plane	
• zero spin	IV-5 through IV-8
• spin = 0.0833 spin	IV-9 through IV-12

The orientation of the nodes is identified in Figure IV-4 for each case.



Notes:

1. Spin is in the direction of orbit rotation.

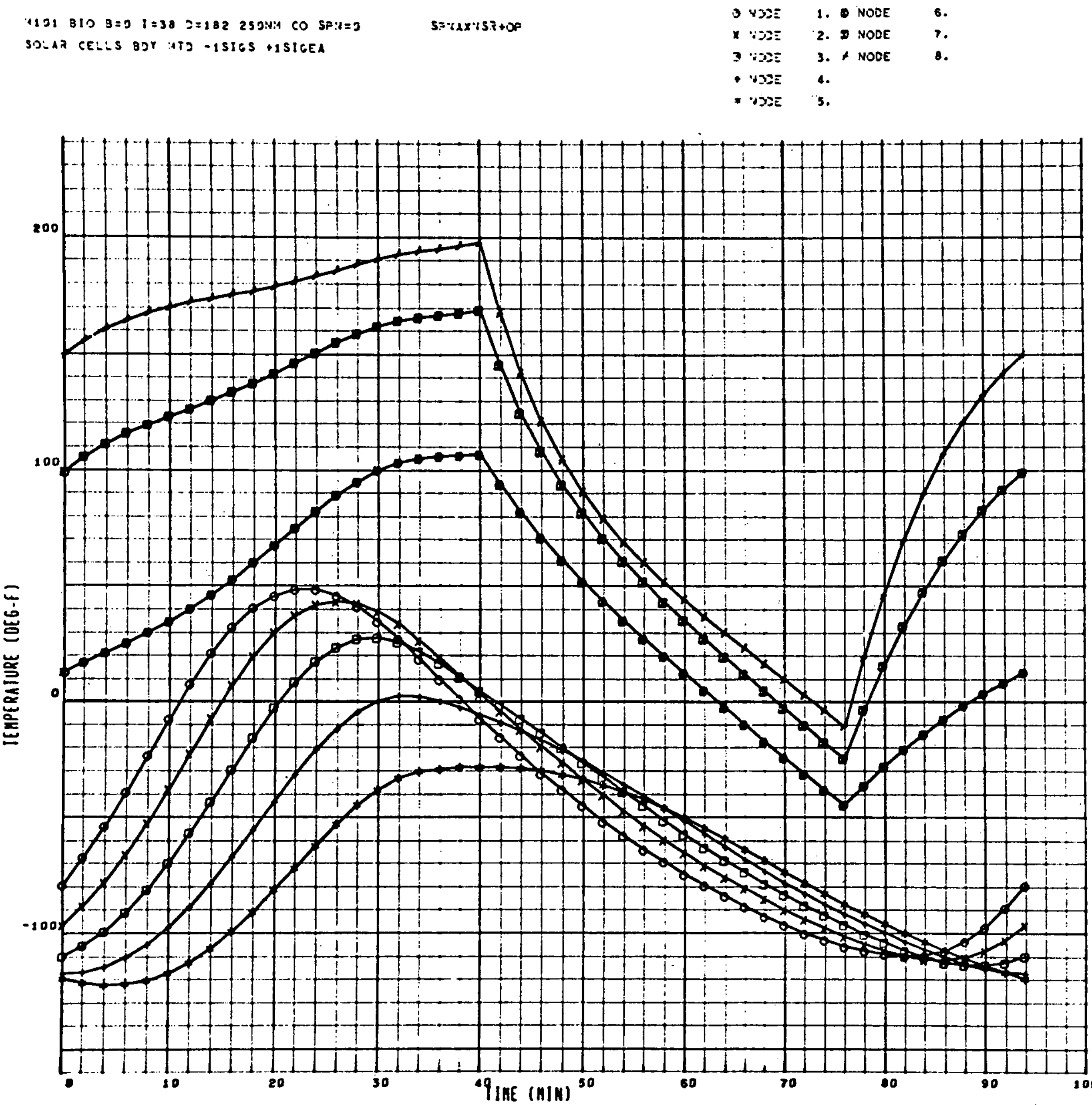
2. Nodes 1 to 16 are located on the conical surface.
   Nodes 17 to 33 are located on the cylindrical surface.

Figure IV-4

As can be seen from these figures, the zero roll case results in a very large circumferential temperature gradient. Although, the gradient itself does not represent a problem, the resultant high temperatures in the subsolar section of the array would result in a significant decrease in solar cell performance. However, if a low residual spin rate of 0.0833 rpm is imparted to the vehicle, the circumferential gradient is almost eliminated with a corresponding decrease in the array subsolar temperature. The 0.0833 rpm spin rate is considerably lower than the maximum allowable 0.035 rpm required to meet the minimum experiment requirement. Since the array size would be approximately 15% lower with the residual spin rate, the temperatures calculated for this condition were used in the sizing of the array. If later attitude control studies indicate that the vehicle roll rate could stabilize out at zero, it would be a simple matter to add a lower threshold limit to the rate control system to provide a minimum roll rate.

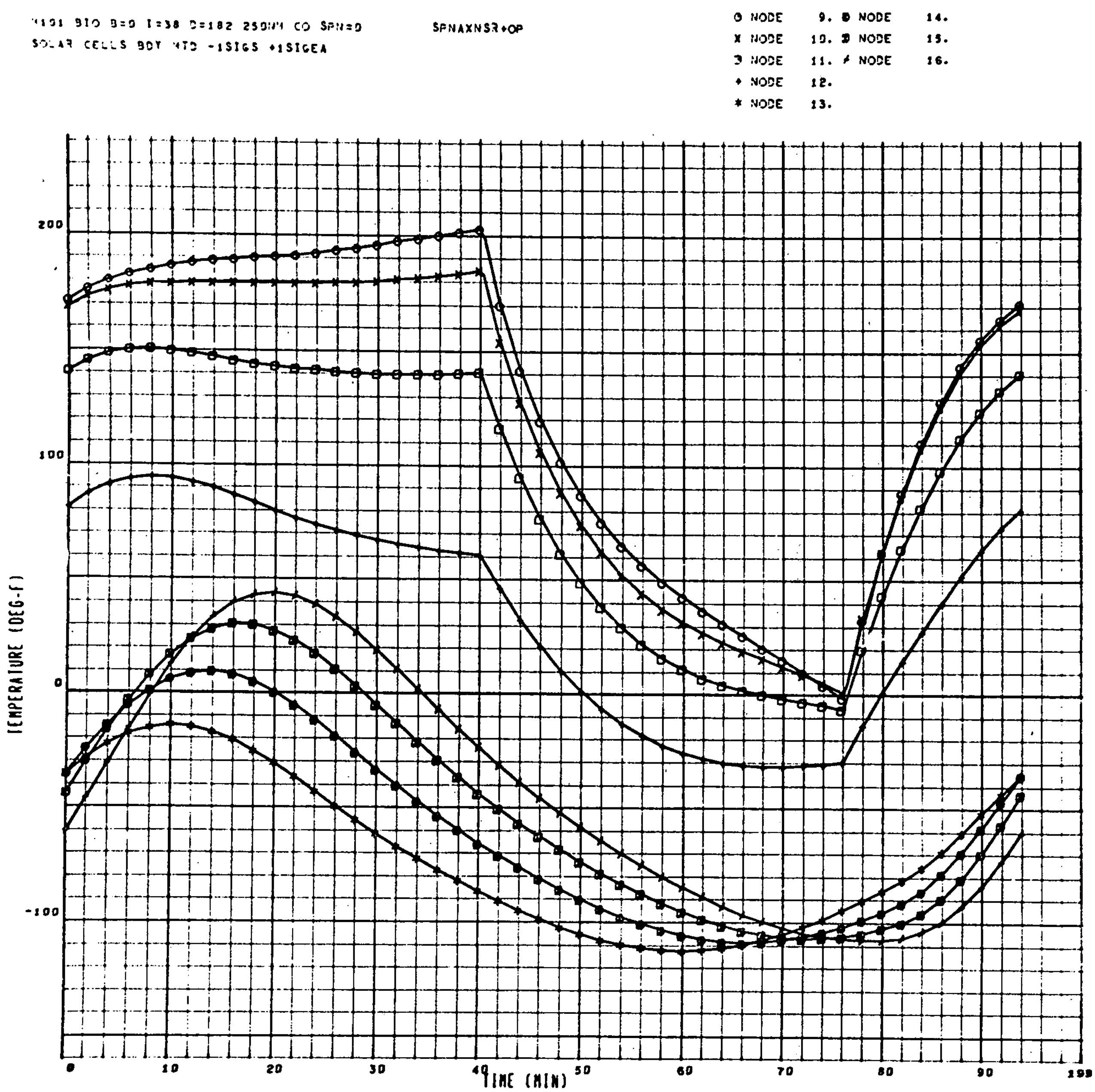
3.2 SIZING OF BODY-MOUNTED ARRAY

The following solar cell parameters were used to determine the required solar array area.



NODE 6.
NODE 7.
NODE 8.
TEMPERATURE (DEG-F)
200
100
0
-100
TIME (MIN)
10
20
30
40
50
60
70
80
90
100

Figure IV-5



SPNAXNSR+OP
BDY
-1SIGS +1SIGEA
NODE 9.
NODE 10.
NODE 11.
NODE 12.
NODE 13.
NODE 14.
NODE 15.
NODE 16.
200
100
0
-100
TEMPERATURE (DEG-F)
0
10
20
30
40
50
60
70
80
90
TIME (MIN)

Figure IV-6

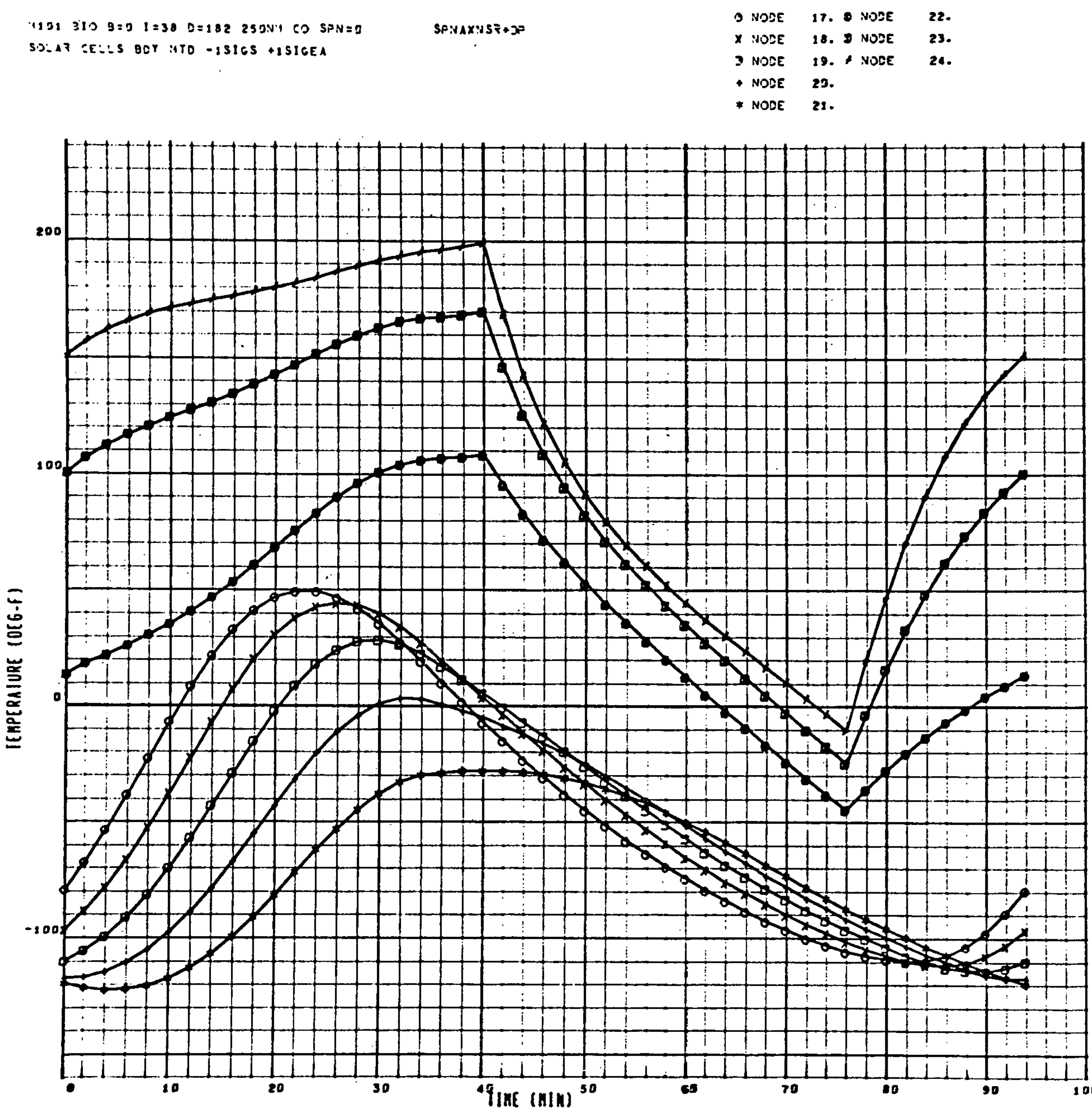
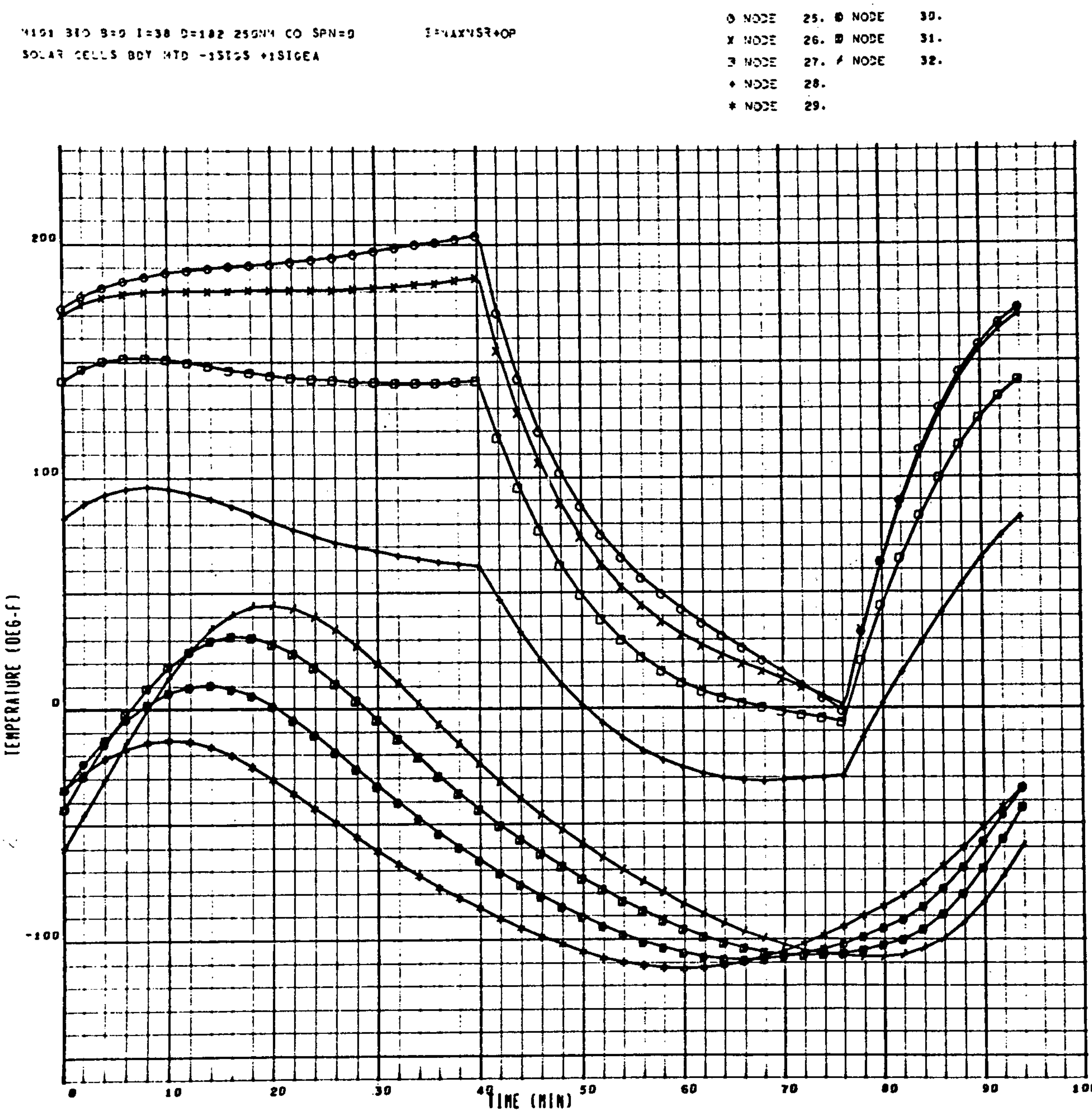
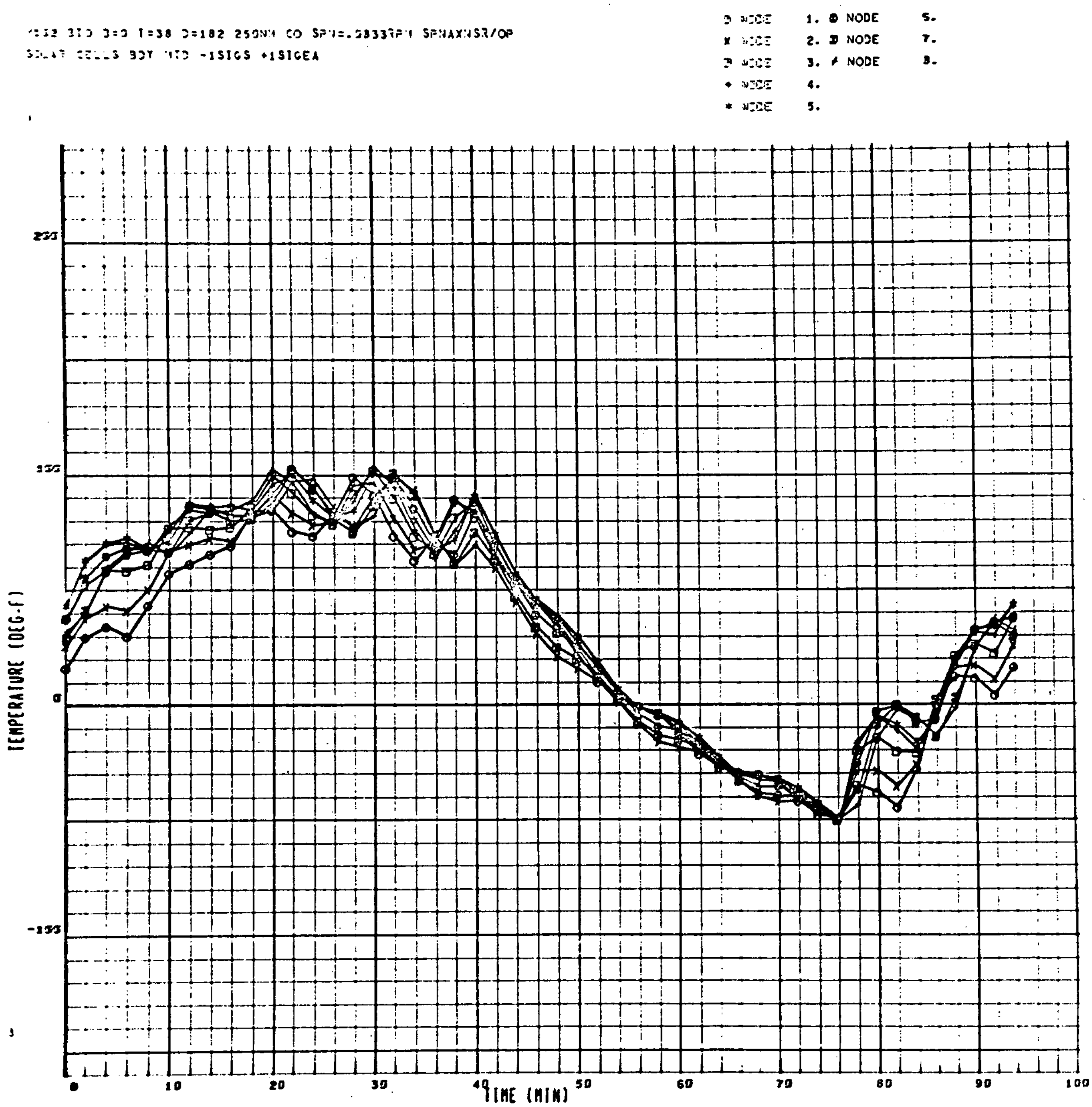

SOLAR CELLS BDY MTD -1SIGS +1SIGEA
O NODE 17.
X NODE 18.
NODE 19.
+ NODE 20.
* NODE 21.
NODE 22.
NODE 23.
NODE 24.
200
100
0
-100
TEMPERATURE (DEG-F)
10
20
30
40
50
70
80
90
100
TIME (MIN)

Figure IV-7



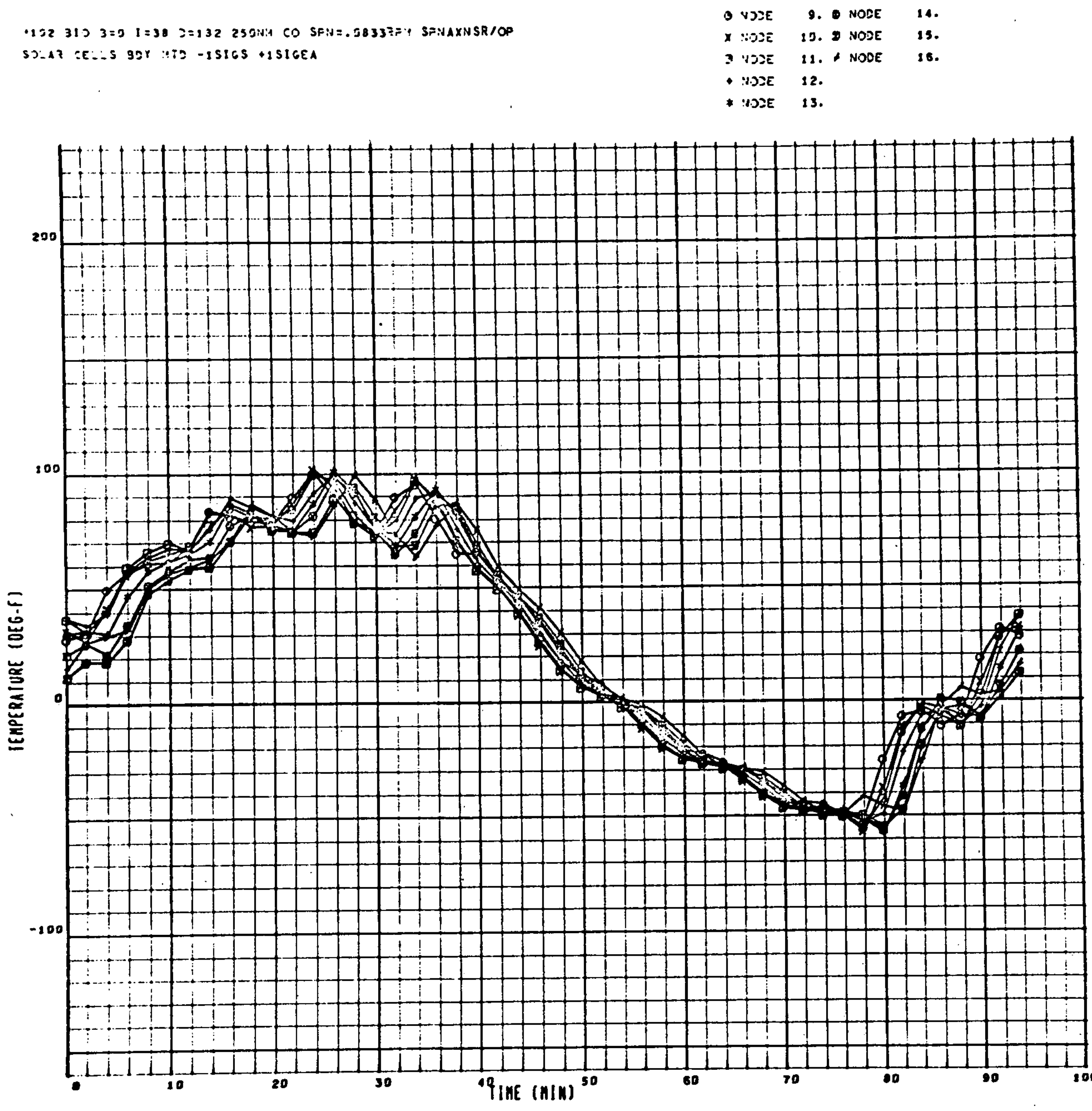
NODE 25.
NODE 26.
NODE 27.
NODE 28.
NODE 29.
NODE 30.
NODE 31.
NODE 32.
TEMPERATURE (DEG-F)
TIME (MIN)
200
100
0
-100
0
10
20
30
40
50
60
70
80
90
100

Figure IV-8



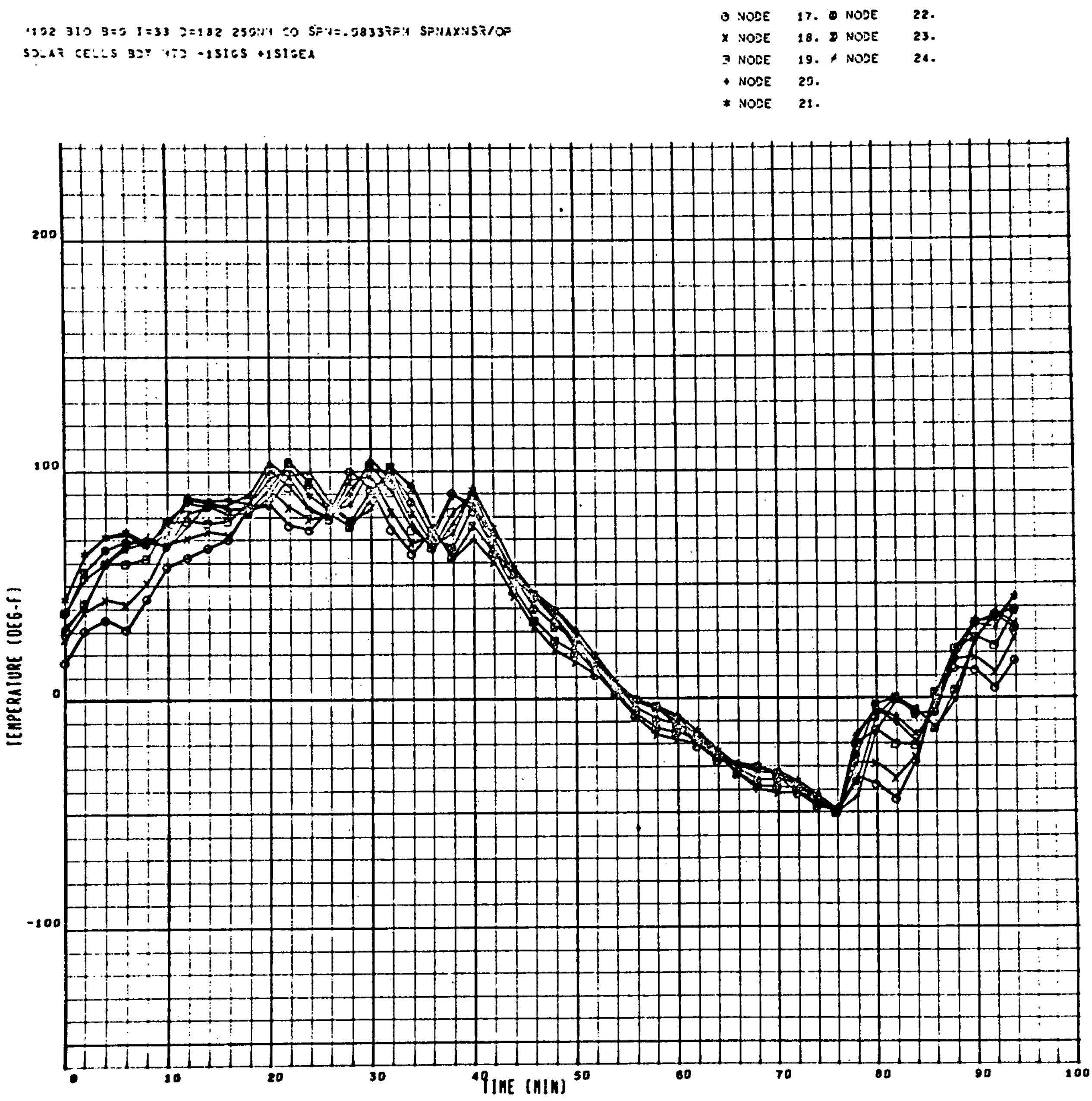
-1SIGS +1SIGEA
1. NODE
2. NODE
3. NODE
4.
5.
TEMPERATURE (DEG-F)
0
TIME (MIN)
10
20
30
50
60
70
80
90
100

Figure IV-9



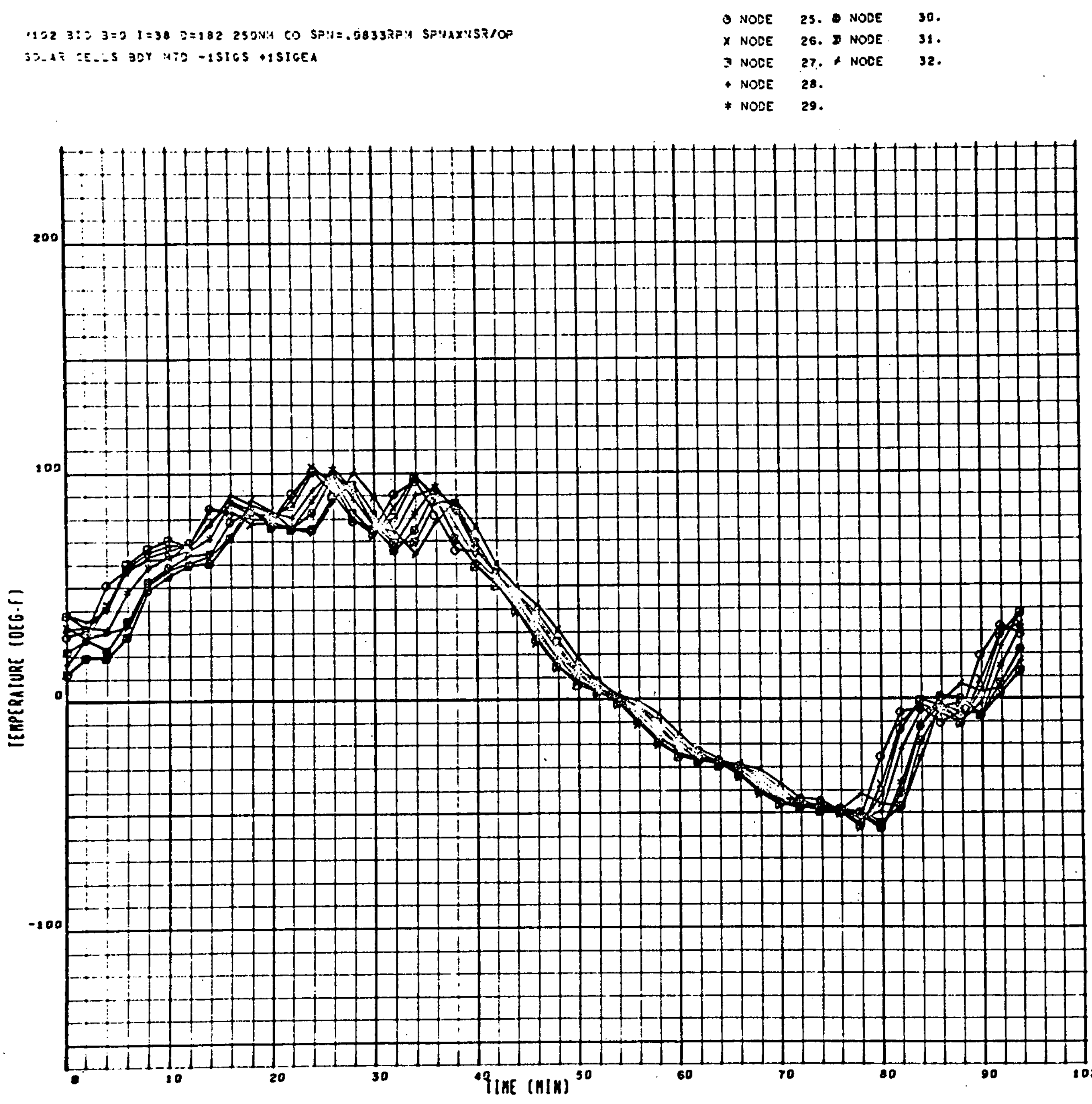
SOLAR CELLS BDY MTD -1SIGS +1SIGEA
NODE 9.
NODE 10.
NODE 11.
NODE 12.
NODE 13.
NODE 14.
NODE 15.
NODE 16.
200
100
0
-100
TEMPERATURE (DEG-F)
0
10
20
30
40
50
60
70
80
90
100
TIME (MIN)

Figure IV-10



4192 310 B=5 I=33 D=182 250NM CO SPN=.5833RPM SPNAXNSR/OP
SOLAR CELLS BDY MTD -1SIGS +1SIGEA
O NODE 17.
X NODE 18.
NODE 19.
+ NODE 20.
* NODE 21.
NODE 22.
NODE 23.
NODE 24.
200
100
0
-100
TEMPERATURE (DEG-F)
0
10
20
30
40
50
60
70
80
90
100
TIME (MIN)

Figure IV-11



'192 BID B=0 I=38 D=182 250NM CO SPN=.0833RPM SPNAXNSR/OP
SOLAR CELLS BDY MTD -1SIGS +1SIGEA
o NODE 25.
x NODE 26.
NODE 27.
+ NODE 28.
* NODE 29.
NODE 30.
NODE 31.
NODE 32.
200
100
0
-100
TEMPERATURE (DEG-F)
0
10
20
30
40
50
60
70
80
90
100
TIME (MIN)

Figure IV-12

(1) Utilization of the Nimbus IV (D) bare cell as a baseline (see Figure IV-13).

(2) Assume the following degradation and loss factors in obtaining beginning of life (BOL) and end of life (EOL) solar array output predictions.

a. α_g = 0.95 = Glassing Loss - Five percent glassing loss measured and reported repeatedly on Nimbus, Tiros and Classified programs.

b. α_r = 0.99 = Series Resistance Loss - Interconnection loss due to IR drop through the copper interconnects.

c. α_m = 0.99 = Cell Mismatch - The submodule I-V characteristics are identical, tungsten measurements typically show that the actual I out is 1% less than the average submodule. A series connected string is limited by the cell with the lowest short circuit current characteristics. A parallel connected string is limited by the cell having the lowest voltage.

d. α_i = 0.98 = Ionization Degradation - Bulk distortion of the silicon lattice typically 2% at lower earth orbits.

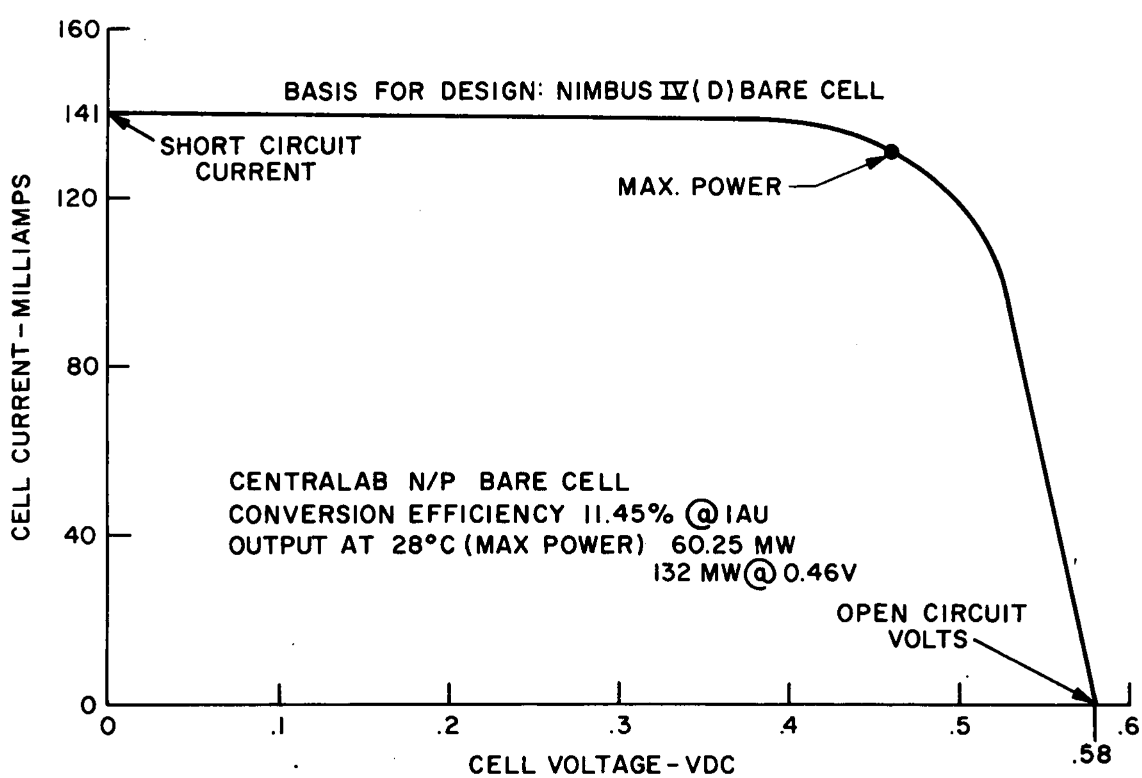


Figure IV-13. I-V Characteristics for the Bare Cell

e. $\underline{I}_v$ = 0.965 = Variation in Solar Intensity - The Earth orbit is elliptical varying from 1 AU. The solar constant, I_C, is calculated at 1 AU and air mass zero and must be corrected for the worst case earth orbit.

f. $\underline{\alpha}_u$ = 0.97 = Ultraviolet Degradation - UV damage seen as 0 to 8% loss in short circuit current (lab measurements). Loss is typically 2 to 3% and occurs during preflight, based on Nimbus experience.

g. $\underline{\alpha}_a$ = 0.99 = Alignment Coefficient - A function of the projected area of the incident photon.

h. $\underline{\alpha}_c$ = 0.98 = Thermal Cycling Allowance - Average was 4% thermal cycling loss for the old Nimbus design after 5000 orbits. New module fabrication tests indicate better stress relief and projects an approximate loss of 2%. Loss of operating cells due to failure of the interconnect caused by thermal cycling.

i. $\underline{\alpha}_t$ = 1.005 = Temperature Effect - Based on empirical data on similar base resistivity cells utilized on Nimbus and other programs. (Typically 0.45% loss/degree C above 28°C.) (Solar array temperature predictions are given in Section IV.4.)

j. $\underline{\alpha}_e$ = 0.96 = Engineering Uncertainties - Based on uncertainties in voltage and current measurement, air mass determination, sun spectrum uncertainty, water absorption of light, meter errors, etc.

Figure IV-14 shows the comparisons of the predicted solar array losses for Bioexplorer and the actual losses for Nimbus. This shows that the previous assumptions are slightly conservative and that the following design calculations can be backed up by flight experience.

To obtain an estimate of the beginning of life solar array output, the bare cell performance characteristics are modified by the applicable loss factors as follows:

(1) $E_{BOL} = (E_{\text{Bare Cell}}) \quad (\alpha_g \, \alpha_r \, \alpha_m \, \alpha_a \, \alpha_t \, \alpha_e \, I_v)$

(2) E_{BOL} = (60.25 Milliwatts) (0.862)

(3) E_{BOL} = 51.9 Milliwatts/cell

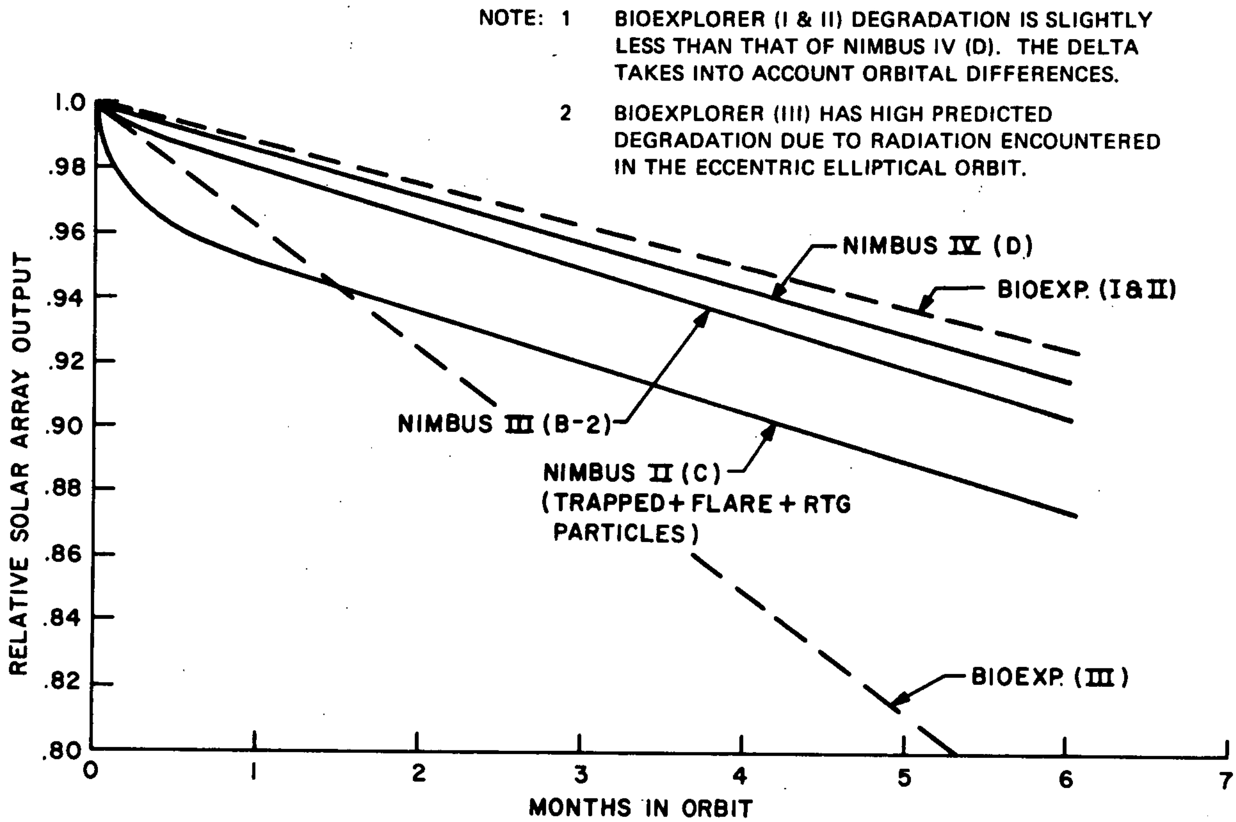


Figure IV-14. Actual Nimbus IV (D) Solar Array Degradation
Predicted Bioexplorer Degradation

A prediction of the end of life solar array performance can now be determined by assuming linear (with time) degradation of the array caused by ionization and ultraviolet degradation, and losses due to thermal cycling.

<u>For Missions I and II</u>

$$E_{EOL} = (E_{BOL}) \quad (\alpha_i \; \alpha_u \; \alpha_c)$$

$$E_{EOL} = (51.9 \text{ milliwatts}) \; (0.93)$$

$$E_{EOL} = 48 \text{ milliwatts/cell}$$

Assuming that the cells can be packaged with an 85% efficiency the output of the array will be 9.5 watts per square foot of array area perpendicular to the sun (projected area parallel to the roll axis). Therefore, in order to obtain the required array output for Mission I of 204.3W, the body mounted array must have a total projected area of 21.4 ft^2.

4.0 PADDLE ARRAY

The sizing of the paddle array was based upon reducing the Bioexplorer paddle configuration to the Bioresearch Module power profile. A simple ratio would have indicated a required paddle area of 26 $ft.^2$ However, since considerable layout work had been done on the eight panel Bioexplorer array for the BRM, it was decided to eliminate two of the panels, resulting in a six panel array with the same configuration as documented in the previous study. Further study could probably reduce the paddle sizes but does not significantly effect the results of this study.

5.0 BATTERY SIZING

The Bioexplorer requirements dictated the need for a 9.0 amp hour Ni-Cad. battery with a total weight of 31 lbs. The reduction in the battery storage requirements from 89.0 watts-hours to the 65.1 watt hours required for the Bioresearch Module would result in the need for a 5.8 amp hour or 167 watt-hour battery. The resultant depth of discharge would be 39%.

Figure IV-15 shows cycles to failure versus depth of discharge (DOD) at parametric temperatures. The Bioexplorer (Missions I and II) subjects the battery to 2800 cycles over the 6-month period. Since a 100% safety factor is usually applied to cycle life and the battery maximum temperature could be as high as 80°F, the maximum acceptable DOD for the battery would be about 39%.

The 5.8 amp hour battery would weigh 20.4 lbs for a resultant weight savings of 10.6 lbs.

6.0 BODY-MOUNTED ARRAY - ATTITUDE CONTROL SUBSYSTEM (ACS)

Among the studies conducted to determine the feasibility of a Bioresearch Module (BRM) design with a body-mounted solar array, was an investigation to determine attitude control subsystem requirements and to define a functional subsystem configuration. Of primary, initial, concern was the matter of orienting the BRM so that its roll axis (axis of symmetry) could be oriented normal to the ecliptic. Results of the attitude control studies show that it is possible to achieve the desired attitude and, in the cases of the Type II and Type III missions, to delete the requirement for a rate gyro package.

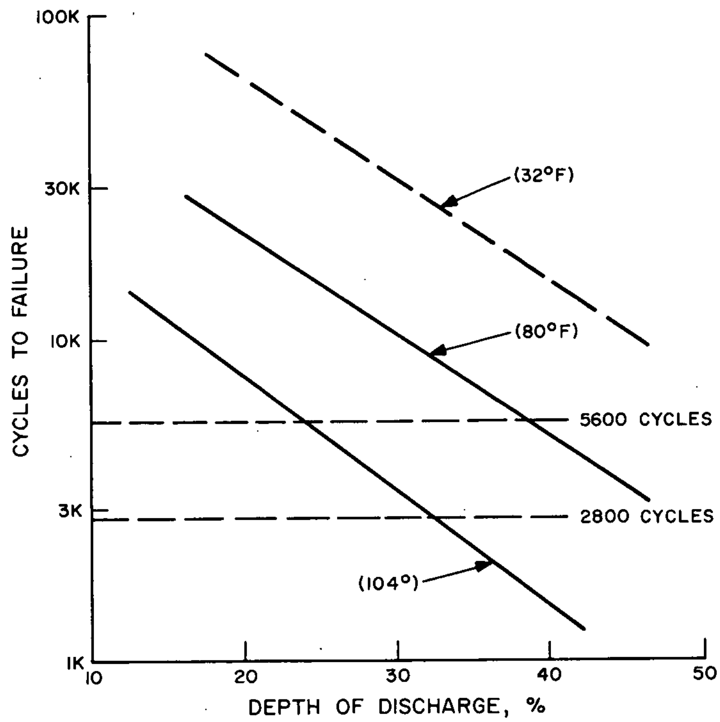


Figure IV-15. Cycles to Failure vs. Percent Depth of Discharge

6.1 SUN ACQUISITION

The change in BRM orientation in orbit resulted in a change in the sun sensor array configuration which can be implemented with the sensor system developed for Pioneer. Figure IV-16 illustrates how almost complete spherical coverage can be achieved with only two wide angle sensors mounted on a spinning spacecraft. As a result, sun acquisition can be accomplished without the necessity for a sun search mode used to offset the effects of launch and tip-off uncertainties. Following initial acquisition by a coarse sensor, the cold gas reaction control would be activated to precess the BRM so that the spin axis is normal to the sun vector. In achieving this orientation, the fine sun sensors, mounted diametrically opposite one another, as shown in Figure IV-16, would also be illuminated by the sun once per revolution and would provide the primary attitude reference for the remainder of the mission. Following this maneuver, the spin rate of the Type II and Type III BRM would be adjusted to the prescribed levels; thus, terminating the sun acquisition phase.

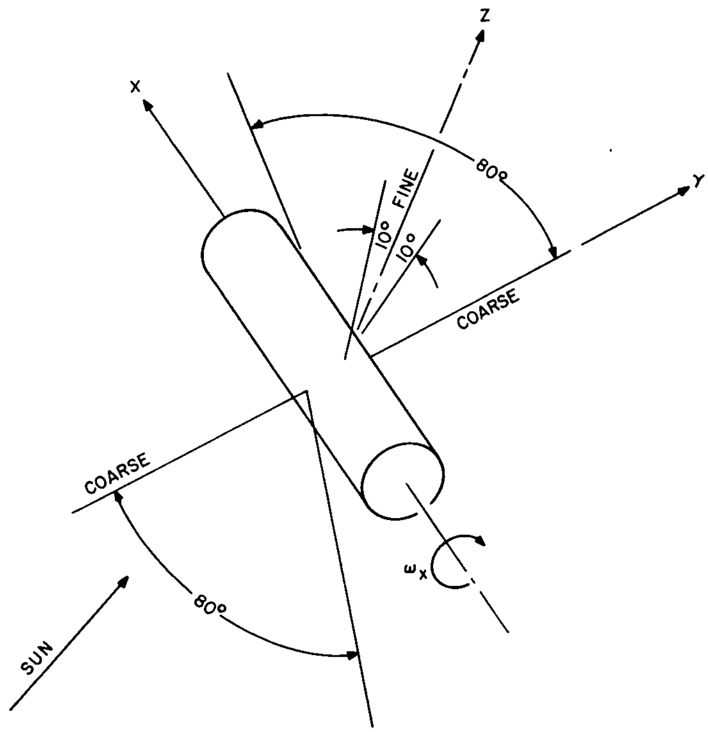


Figure IV-16. Sun Acquisition after Last Stage Burnout

On the other hand, the Type I BRM would be completely spun-down to zero rate in compliance with the near zero "g" requirement. Because spin rate of the Type I BRM will be limited to less than $|2|$ deg/sec, two additional sensors are required to provide 360° coverage as a contingency against a zero roll rate condition. Figure IV-17 shows how it is proposed to arrange the four sun sensors around the periphery of the vehicle.

6.2 ORIENTATION TO THE ECLIPTIC

The initial maneuver, after sun acquisition, merely orients the BRM roll axis perpendicular to the sun. However, in the absence of a third axis reference to help establish the desired orientation, it would only be by chance that the BRM roll axis would be perpendicular to the eliptic. Assume a condition, as depicted in Figure IV-18 where the roll axis is in the plane perpendicular to the eliptic, but displaced from the normal to the ecliptic by an angle θ. The inertially stabilized BRM would maintain

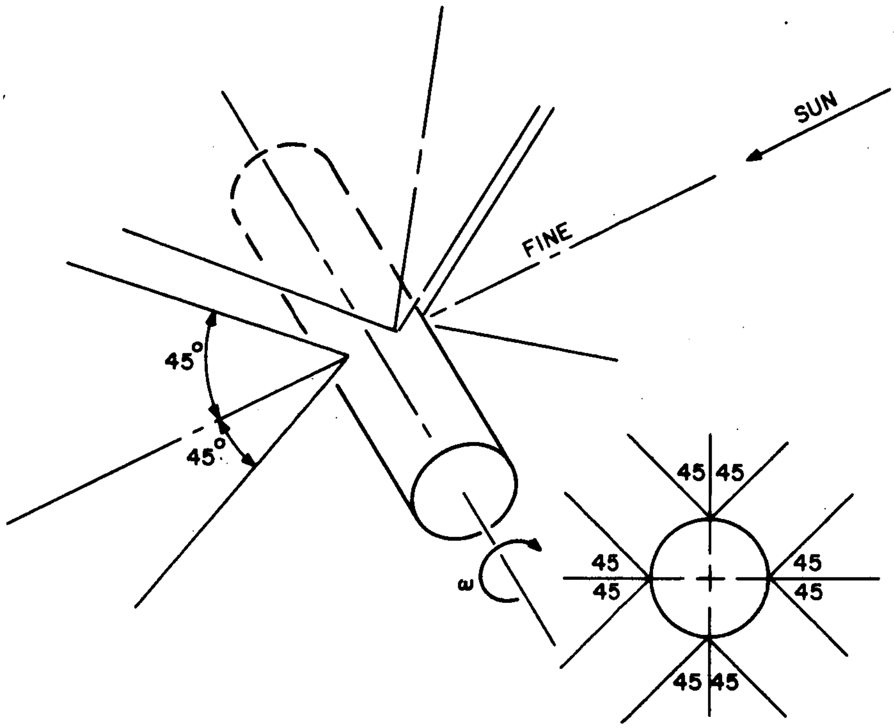


Figure IV-17. Fine Sun Sensor Acquisition and Stabilization, Type I Configuration

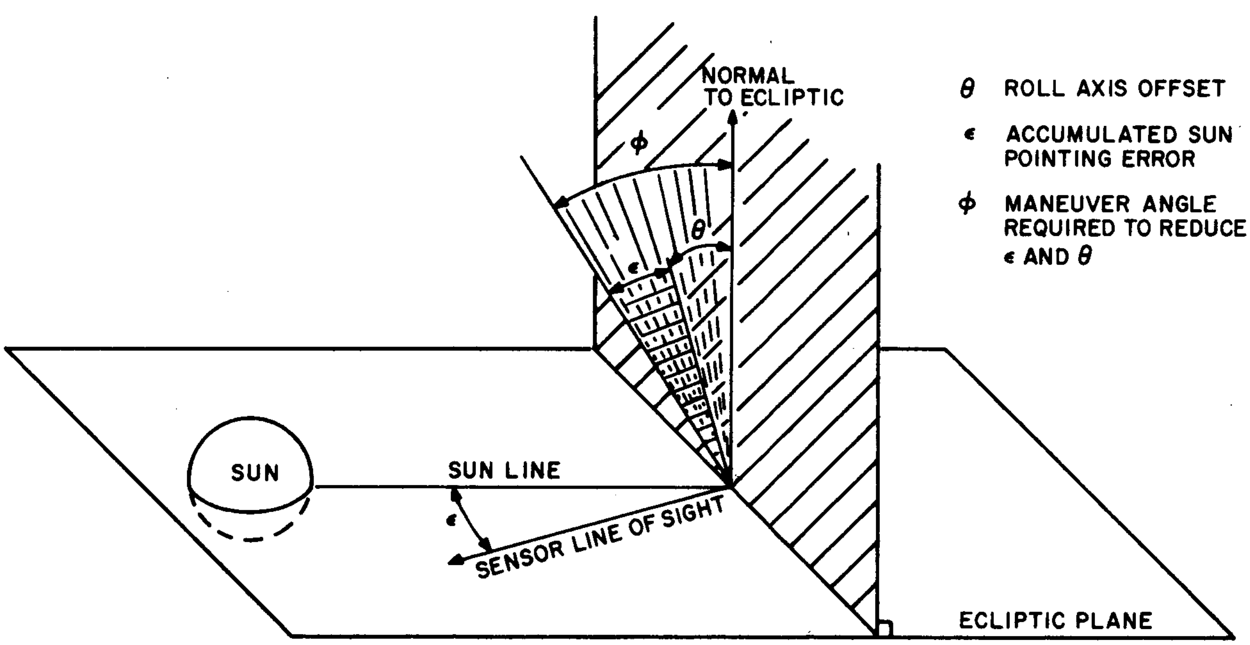


Figure IV-18.

2-3

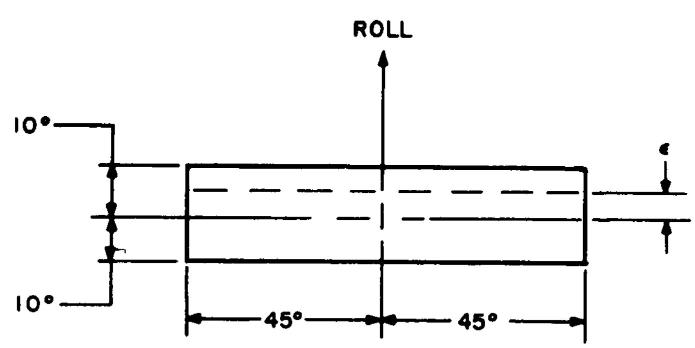
its attitude in the absence of perturbing forces. Since the Bioresearch Module is earth orbiting, its attitude, with respect to the sun, would change at a rate of 0.986 deg/day. This change would develop the angle ϵ as shown in Figure IV-18. This then can be considered an error function which would be detectable by the sun sensors as shown in Figure IV-19. With appropriately designed logic, an attitude maneuver can be determined to reduce ϵ and θ simultaneously. The maneuver (ϕ) would be accomplished about a known reference axis such as the sun sensor line of sight axis. Since ϵ is a measured quantity, the value of θ would have to be determined as follows:

$$\theta = \sin^{-1} (\epsilon/0.986 \, \Delta t)$$

where Δt is the total time since the initial orientation maneuver. The magnitude of the correction maneuver would then be based on the relationship:

$$\phi = \cos^{-1} (\cos \theta \cos \epsilon)$$

These calculations can be performed on the ground with telemetered sun sensor data, or on board the BRM, based on suitable time data obtained from the mission clock. Where these calculations are made depends largely on how frequently they must be performed and the penalties involved insofar as BRM weight is concerned.



$$\epsilon = \sin \theta \int_0^N 0.986 \, dt, \qquad N = \text{No. of Days in Orbit}$$

Figure IV-19. Sun Trace across Rotating Sensor Field of View

6.3 ACS IMPLEMENTATION

6.3.1 Type I BRM

A block diagram of the proposed ACS is shown in Figure IV-20. The subsystem shall be comprised of four fine sun sensors, two coarse acquisition sensors, a triaxial rate gyro package, a momentum wheel, control electronics and a cold gas reaction control. A 10 to 15% weight increase over the baseline ACS design weight for all three classes of BRM is anticipated to maintain stable orientation.

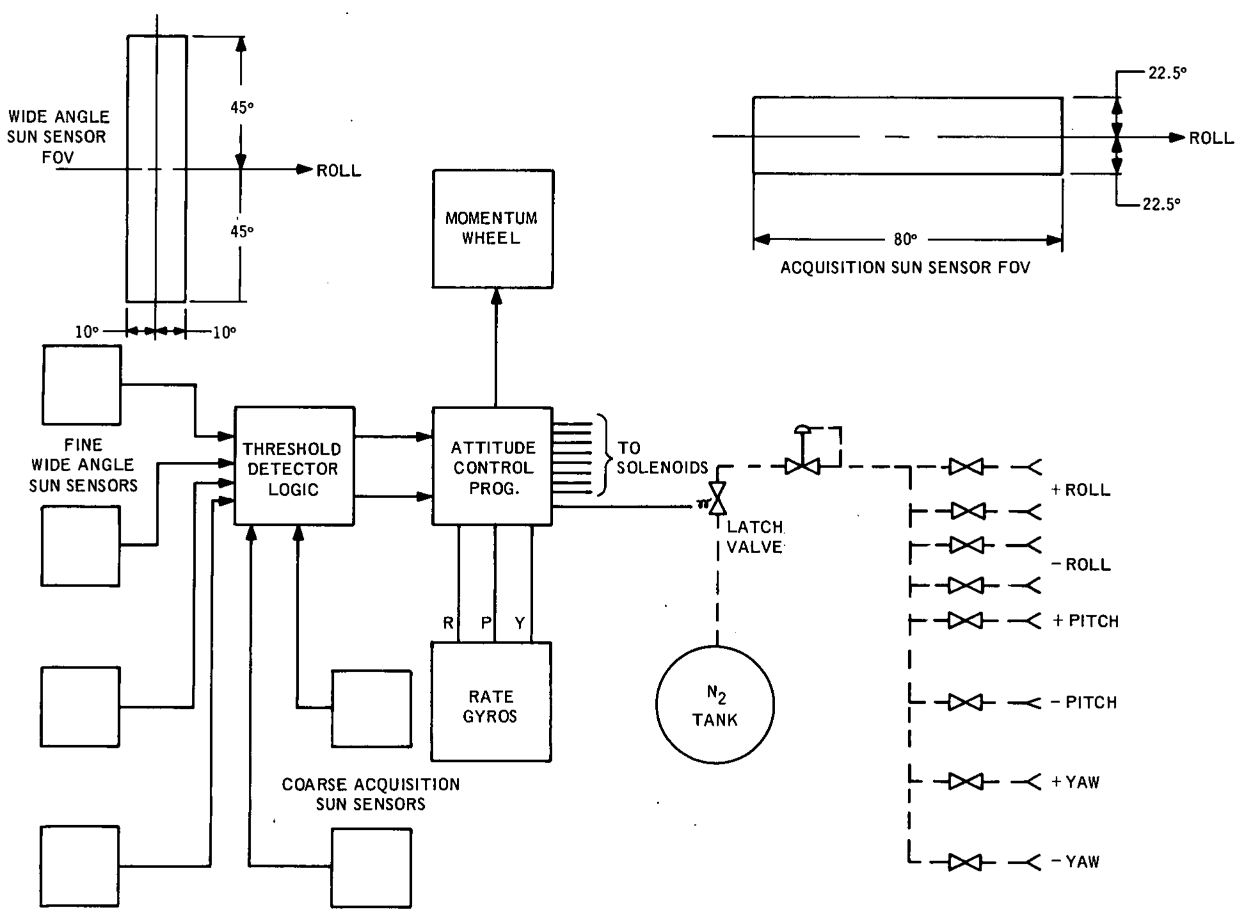


Figure IV-20. Type I. Attitude Control Block Diagram

6.3.2 Type II and Type III Implementation

Figure IV-21 is a block diagram of the proposed ACS solution for the Type II and Type III BRM. The Type II ACS will include, in addition to the sun sensors, control electronics and cold gas reaction control, four extendable booms to control the variable "g" spin rate. The use of the booms and overall BRM stability are discussed at length in Section IV. However, one point worth repeating is that stability can be built into the body mounted solar array BRM by extending booms of sufficient length to alter its inertia configuration to the extent that the roll moment of inertia is greater than the pitch/yaw moment of inertia. If this approach were pursued further, it is felt that a strong argument could be developed for separating from the spent rocket motor case based on weight and boom stability considerations. From the work completed to date, the minimum boom length required would be greater than the 22 ft required to control the baseline BRM spin rate.

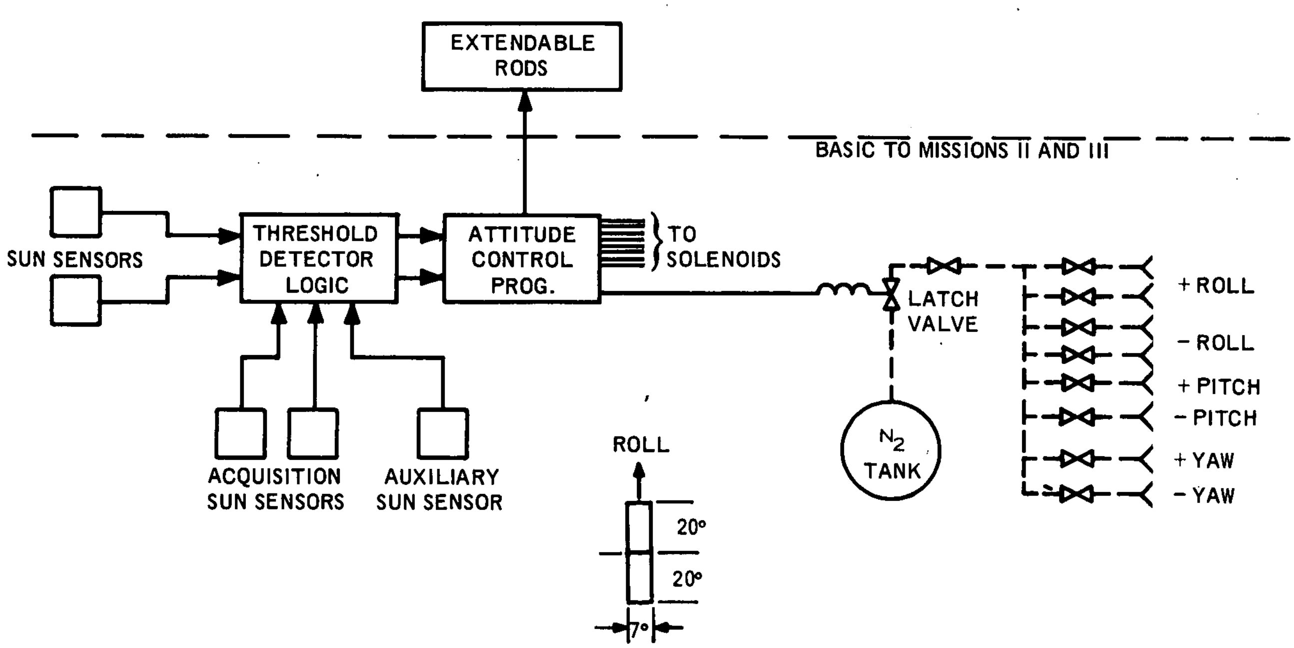


Figure IV-21. Type II Attitude Control Block Diagram

The Type III BRM could also be equipped with booms which could be used to provide the needed stability and to control spin rate to the prescribed prelaunch value.

Significant features of the body mounted solar array effecting the ACS are that periodic corrections of BRM attitude, which greatly influence control gas requirements, are not required. Secondly, an auxiliary sun sensor can effectively replace the rate gyro package in the Type II and Type III BRM.

7.0 BODY-MOUNTED BRM CONFIGURATION

The resultant BRM configuration required to meet the solar array projected area requirements is shown in Figure IV-22. The configuration was dictated by the Scout shroud limitations as shown in Figure IV-23. In order to meet the projected area requirement, it was necessary to utilize the total available Scout envelope, including the volume surrounding the fourth stage motor casing. It should also be noted that in the forward section, the array surrounds the current experiment module and extends 19.6 inches beyond the present coldplate. If the coldplate were to remain in its current location, it would be useless as a heat rejection surface since it would be radiating primarily to the array structure. As a result, the coldplate would have to be moved forward to where it would have a direct view factor to space. Since the coldplate was required to have direct contact with the experiment module it would also be necessary to extend the experiment volume as shown in Figure IV-23.

With the fourth stage of the Scout buried inside the BRM, the BRM-booster separation interface would become somewhat complex in order to assure the correct clearances while separating. It was, therefore, tentatively decided not to separate the fourth stage but to leave the expended motor as an integral part of the BRM. This, in addition to eliminating the separation problem, would save five lbs in the separation system weight.

The body mounted solar array BRM is configured to fit, with considered clearances, within the Scout shroud and present a maximum of projected area.

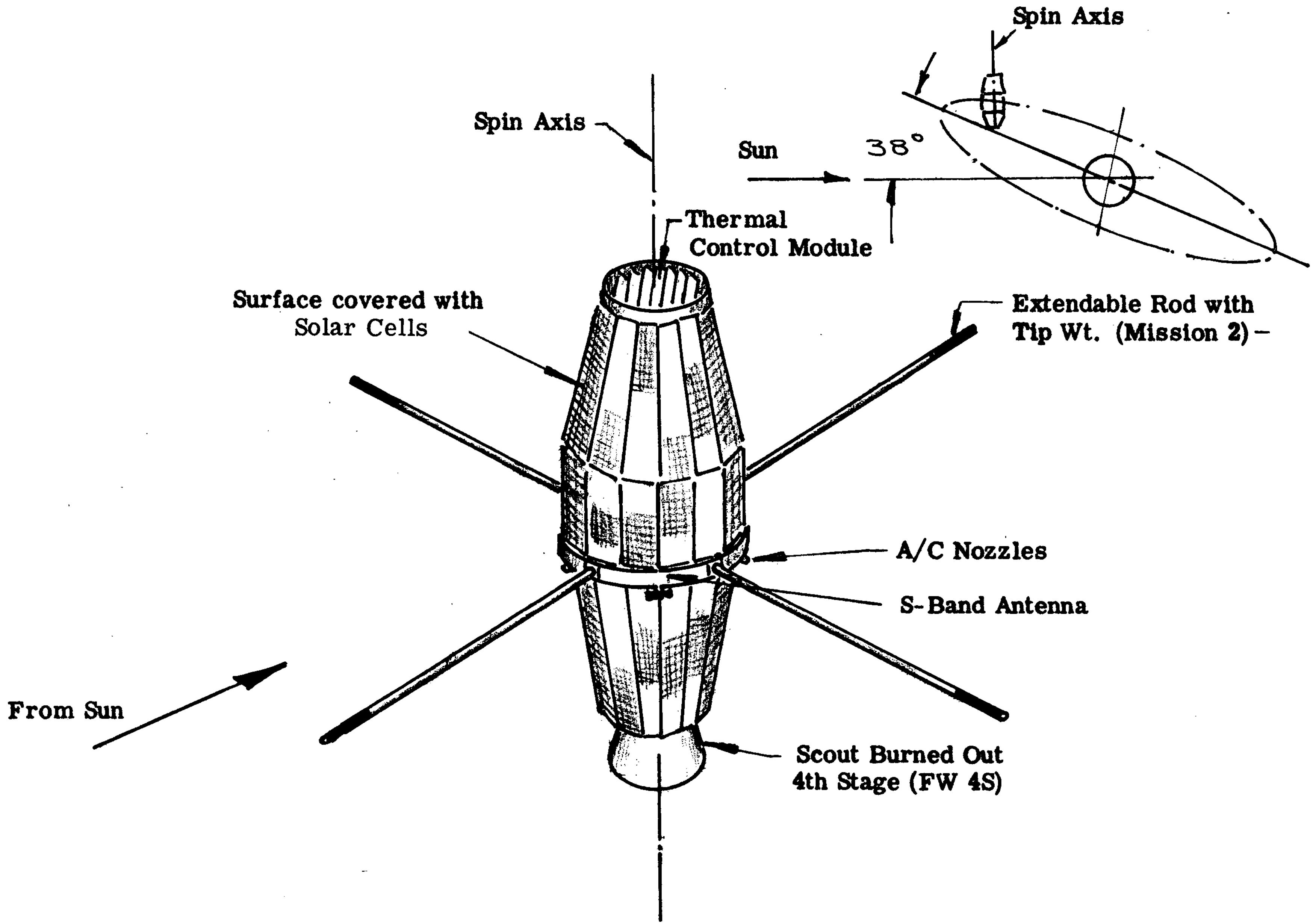


Figure IV-22. BRM Configuration

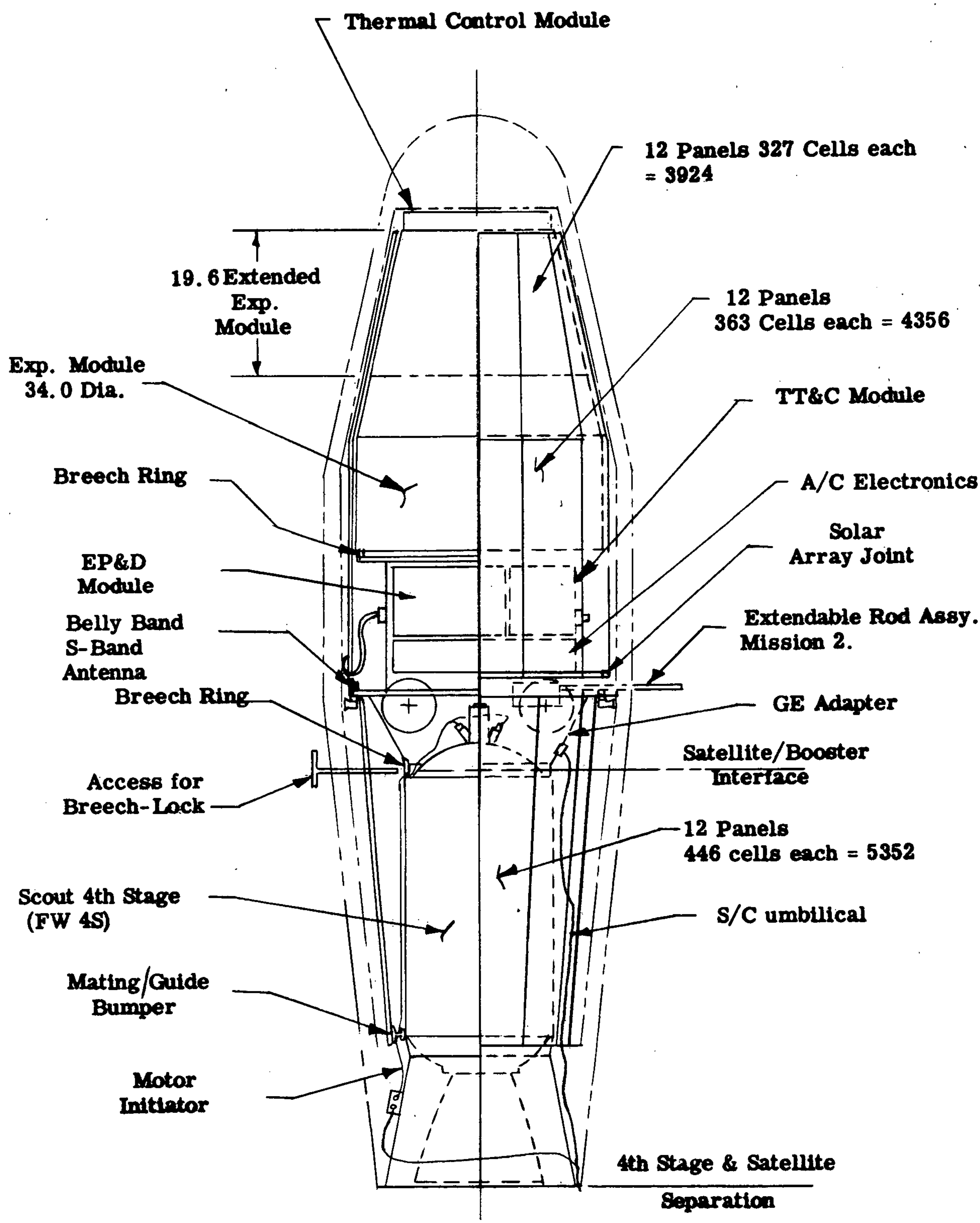


Figure IV-23. Body Mounted Array

The conclusion of this study was to choose a polyhedral cross-section rather than cylindrical due to packaging efficiency on the conical surfaces, keeping an even number of total strings on a side. Trade-offs, showed, that in the 8 to 16 side range, the power was quite insensitive to the number of sides. It was also established that 12 sides were the minimum that would fit at the aft end where the structure, including cells, had to fit into the anular limits defined by the shroud and the Stage IV Booster. Since complexity increases with the number of sides, 12 was chosen.

This configuration increases the available experiment volume, but at the same time compresses or buries the coldplate in the forward array fairing.

7.1 STRUCTURE

The BRM primary structure is essentially a shelf capable of reacting all boost, inertial and handling loads. The shelf is constructed of aluminum skinned honeycomb with appropriate adapter rings which mate with the forward and aft solar cell fairings. The forward side of the shelf supports the rack which provides support for the component modules and the experiment package. The shelf is bounded by a structural ring which contains the S-band antennas and has openings through which the extendable inertial rods pass. The ring also provides structure for ground handling and Scout shroud snubbers, if required. The aft side of the shelf provides reaction points for the thruster either through the V-band separation ring presently on the Stage IV thruster, or the proposed GE adapter. The attitude control components (tanks, lines, and nozzles) are also mounted on the aft side of the shelf.

The components are supported by a truss type rack which is permanently attached to the forward side of the shelf. The rack has tracks and locks which support the removable component modules. The mating half of the experiment breech ring is permanently mounted to the forward end of the truss.

The solar cells are supported by a light fairing type structure composed of either magnesium or aluminum honeycomb. The aft array fairing is fastened to the shelf through a bolted joint. The forward fairing connects to the shelf through a quick release arrangement for easy component access.

7.2 ACCESS AND HANDLING

The BRM has been designed so that access and removal and replacement capability of the experiment and component modules can be maintained on the pad.

Access to the modules and the component modules is acquired by the removal of the forward array. The array is unfastened by a quick release which necessitates the removal of only four screws. The removal of the array exposes the experiment and modules; either can be removed independently. The experiment is removed by first demating electrically, and then mechanically through the breech ring. The component modules are essentially drawers which are on tracks in the rack and held in place by latches. As the drawers are locked "home" by the latches, the electrical connections are made.

Access through the aft array is provided by four doors around the circumference just aft of the shelf.

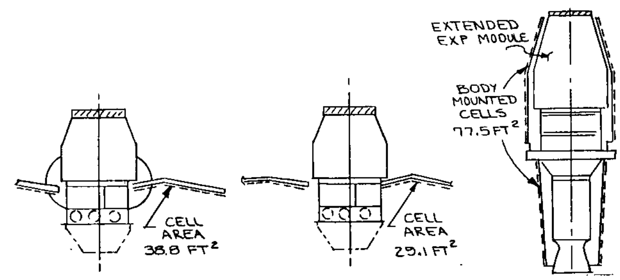
In addition to the performance improvements, the recommended Stage IV support change affects accessibility and handling procedures. The mate of the breech ring can either be accomplished through the access doors, or by lowering the array and attaching the breech ring.

One critical problem recognized at this time is the handling of the BRM and the arrays with the exposed solar cells. Procedures and equipment would have to be carefully designed to protect the solar cells.

7.3 WEIGHT TRADE-OFF

Weight statements for the three BRM configurations studies to date are shown in Figure IV-24 (S-band option is included in all three). If the number of paddles is reduced from eight to six, a net BRM weight saving of 22 lbs is achieved. The body mounted array vehicle is 48 lbs heavier than the six panel configuration. Although this margin can be reduced by 5 lbs due to the deletion of the separation system, the body mounted configuration is marginal with regard to the Scout boost capability and has zero growth capability.

CONFIGURATION



YEAR SOLAR CELL MOUNTING ELECTRICAL POWER	1970 PADDLES (8) 296 WATTS			1971 PADDLES (6) 222 WATTS			1971 BODY MOUNTED 211 WATTS		
MISSION	I	II	III	I	II	III	I	II	III
SYSTEM									
STRUCTURE	40.5	40.5	40.5	40.5	40.5	40.5	45.6	45.6	45.6
SOLAR ARRAY & DEPLOY MECH.	46.0	46.0	46.0	34.0	34.0	34.0	77.6	77.6	77.6
ELEC. POWER & DIST. (INCL. HARNESS)	45.0	48.0	76.0	35.0	38.0	56.0	35.0	38.0	56.0
TELEMETRY TRACK & COMM.	30.4	30.4	30.4	30.4	30.4	30.4	30.4	30.4	30.4
ATTITUDE CONTROL	57.3	49.8	49.8	57.3	49.8	49.8	64.9	57.4	57.4
THERMAL CONTROL	28.7	28.7	28.7	28.7	28.7	28.7	28.7	28.7	28.7
EXP. PACK. SHELL OR EQUIP SUPPTS	8.0	8.0	8.0	8.0	8.0	8.0	--	--	--
EXPERIMENT PACKAGE	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0
GROWTH & CONTINGENCIES	19.1	23.6	30.6	41.1	45.6	62.6	-7.7	-3.3	14.3
TOTAL SPACECRAFT	375.0	375.0	410.0	375.0	375.0	410.0	375.0	375.0	410.0
ADAPTOR	30.0	30.0	30.0	30.0	30.0	30.0	25.0	25.0	25.0
EMPTY MOTOR CASE (4TH STAGE)	58.5	58.5	58.5	58.5	58.5	58.5	58.5	58.5	58.5
TOTAL VEHICLE (INCL. ADAPTOR & MOTOR CASE)	463.5	463.5	498.5	463.5	463.5	498.5	458.5	458.5	493.5
"S" BAND TT WT. INCREASE	+8.6	+8.6	+8.6	+8.6	+8.6	+8.6	+8.6	+8.6	+8.6

Figure IV-24. Weight Estimates of Bioresearch Module

8.0 TRADE-OFF SUMMARY

(1) The body mounted array is 48 lbs heavier than the paddle configuration.

(2) Although the orbital fluxes have been reduced on the body mounted coldplate resulting in a lower sink temperature, the radiating area has also been reduced due to the moving forward of the coldplate (see Section III). Therefore, part of the conical surface must be used as a fin which results in the same problems which occur in the paddle design.

(3) The surrounding of the fourth stage with a section of the array will require modification of the stage to permit BRM-booster hookup.

(4) Although not separating the fourth stage from the BRM would solve the hangup problem, attitude control studies indicate that a serious stability problem would exist while deploying the rods.

(5) The body mounted attitude control system would eliminate the need for periodic updating of the sun vector alignment.

(6) Surrounding the entire BRM perimeter including the experiment module with the solar array will greatly complicate access and handling problems.

(7) The body mounted array can just meet the power profile requirements. No further growth in power can be accommodated since the full available volume for the Scout shroud is being utilized.

The above indicates that the body mounted array offers no significant advantages over the paddle configuration while being considerably heavier. In addition, the utilization of the full Scout volume should result in a more complicated booster interface. Thus, the results of the reduced power trade-off indicate that the paddle configuration is still the correct approach.

V. DATA STORAGE, TELEMETRY, TRACKING AND COMMAND SUBSYSTEM EVALUATION

1.0 OBJECTIVES

The objectives of this portion of the study were:

A. To prepare a preliminary design and to determine the impact on the baseline design of a real time or recorded playback television system to monitor biological activity in the experiment package for Missions I and II. Evaluate power, weight and available ground station support capabilities within the Bioresearch Module design restraints and develop an optimum design meeting the requirements listed below:

   (a) One minute continuous observations of biological specimens per orbit.

   (b) 7.5 inches by 10 inches frame size with 0.1 inch resolution.

   (c) Pre-flight selected frame rate of either 24 frames per second or one frame per minute, with frame rate to be experiment dependent.

B. To evaluate the design impact of using the Manned Space Flight Net (MSFN) either in addition to or in lieu of STADAN for ground station support. The MSFN shall be assumed available to support missions requiring television monitoring of the experiment.

C. To maximize command and data requirements using available off-the-shelf hardware with the Bioresearch Module weight and power requirements.

2.0 BASELINE DESIGN, BIOEXPLORER

As stated in the objectives, proposed specification changes were evaluated for impact on the baseline design. This baseline is the design determined in the Bioexplorer Spacecraft Preliminary Design Study and is summarized in the following sections.

2.1 BASELINE REQUIREMENTS

The general requirements were minimum cost, maximum use of off-the-shelf hardware, minimum weight, modularity, commonality between missions and a six month life.

2.1.1 The principal specific requirements for the baseline data storage and telemetry subsystems are as follows:

(1) A separate data "patchboard" shall be provided for experiment package hookup.

(2) The digital data signal from the experiments shall be a serial pulse train of 0-5 volts amplitude, maximum rate of 512 bps and 50% duty cycle during use. The required storage capacities are:

Missions I and II:	4,000 words based on one station pass per orbit
Mission III:	8,000 words based on one station pass per day.

(3) The analog experiment data signals shall consist of 100 signals of 0 to 5 volts amplitude with a maximum frequency of 0.01Hz and shall be transmitted with a resolution of 1.6% during each station contact in Missions I and II only.

(4) All data shall be transmitted as 6 data bits plus parity. Storage of parity of the digital experiment data is optional.

(5) The error rate for the telemetry subsystem shall be less than 10^{-3}.

(6) The following timing pulses shall be provided to the experiments:

a. 48 individually identifiable discrete hourly pulses.

b. A separate pulse every 10 minutes with a 1 to 5 millisecond rise time and 50 millisecond pulse width.

(7) A spacecraft clock shall be provided, compatible with STADAN requirements.

(8) The subsystems shall be compatible with STADAN (NASA Space Tracking and Data Acquisition Network).

2.1.2 The principal requirements for the command subsystem are:

(1) A "patchboard" shall be provided for experiment package commands.

(2) Twenty discrete commands are required for the experiments in Missions I and II; five are required in Mission III.

(3) The command signals shall consist of pulses with an amplitude of 27.5 volts and a width of 39 to 50 milliseconds.

(4) The subsystem shall be compatible with STADAN.

(5) Redundant receivers and decoders shall be provided.

(6) Commands shall be provided to allow complete control of the spacecraft operating modes by the flight controllers. As a result of the spacecraft design, the following command requirements were determined: Mission I, 26, Mission II, 31 and Mission III, 26.

2.1.3 Specific requirements for tracking are:

(1) At least one Minitrack beacon shall be provided for Missions I and II. Telemetry tracking may be used as a backup.

(2) A GR&RR transponder shall be used for Mission III.

2.2 BASELINE DESIGN SELECTION

The baseline design selected for Missions I and II is shown in Figure V-1. The transmitter phase modulates the 136 MHz rf carrier with the split phase signal from the data handling unit and has a 300 mw power output. The command receivers operate continuously in the 150 MHz command band.

The decoder is a tone command decoder that filters the audio output of the command receiver, detects the address and execute tones, and produces command output signals of 28 volt amplitude and a duration of 39 to 50 ms. The antenna is comprised of four semi-circular elements fed to form two crossed loops, fed 90° out of phase. For additional detail, refer to the final progress report, dated 29 November 1970, of the Bioexplorer Study.

3.0 MSFN COMPATIBILITY

3.1 STATION COVERAGE

The station pass coverage available from both the NASA Satellite Tracking and Data Acquisition Network (STADAN) and the Manned Space Flight Network (MSFN) for various

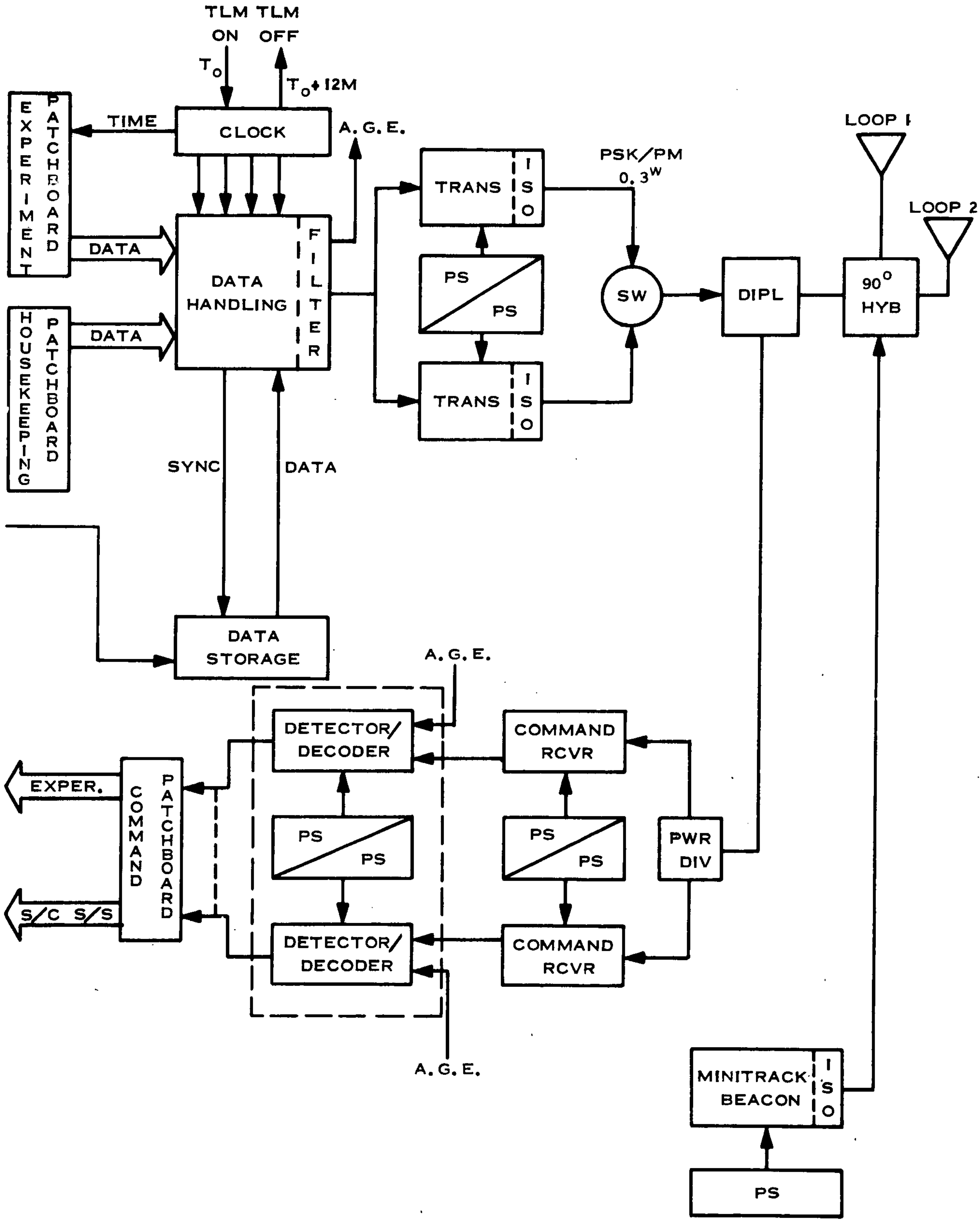


Figure V-1. Data Storage, Telemetry, Tracking and Command - Missions I and II inclinations, has been investigated. The inclinations used correspond to a due east Scout launch (38°) and space shuttle launches for the three basic missions; due east (28°), space station resupply (55°), and polar (90°). The study was made for an altitude of 200 nm and a 7° minimum antenna elevation at all stations. The results indicate that with MSFN, the BRM in any of the inclinations investigated could be out of contact with a station for two full orbits. With STADAN, the maximum time between contacts varies from 1.5 orbits to 2.3 orbits depending on inclination.

The stations used were:

STADAN (See Figure V-2)	MSFN (See Figure V-3)
Alaska	Hawaii
Rosman	Goldstone
Fort Myers	NTTF
Quito	Bermuda
Santiago	Madrid
Johannesburg	Canary
Tananarive	Ascension
Carnarvon	Carnarvon
Woomera	Honeysuckle Creek
Orroral	Guam

The following are the worst case conditions for each case:

1. STADAN, 28° Inclination

   A south to north (S → N) pass through the "hole" between Quito and Santiago is not picked up by Austrailia. As a result, the Bioresearch Module (BRM) can be out of contact for two full orbits. The next contact would be at Quito.

2. MSFN, 28° Inclination

   A S → N pass between Canary Island and Ascension will cause it not to be picked up until over Canary one orbit later. This results in two full orbits without a contact.

3. STADAN, 38° Inclination

   A S → N pass between Santiago and Quito is picked up by Orroral and Woomera. This results in 1-1/2 orbits without a station contact.

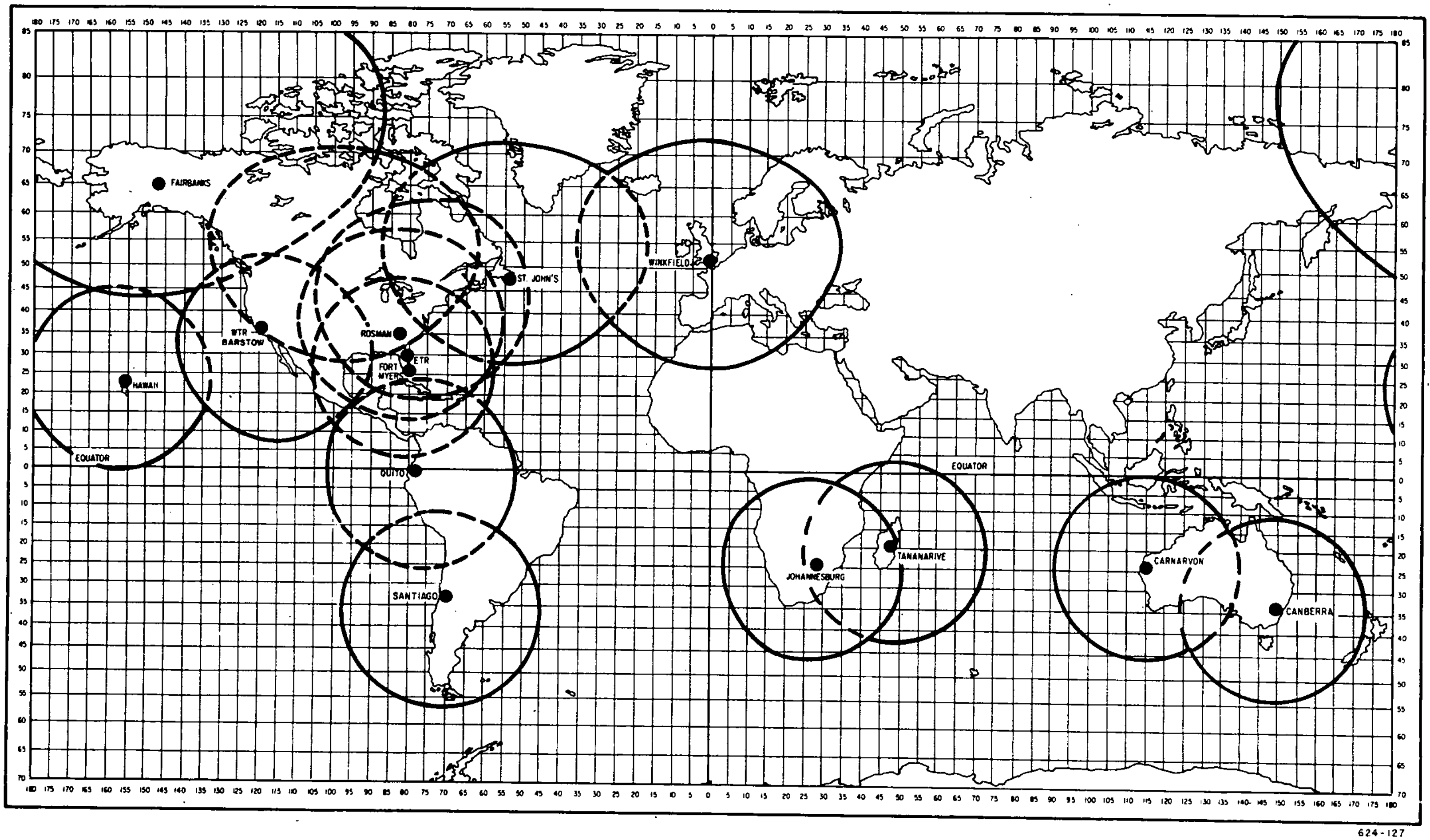


Figure V-2. STADAN Horizon-to-Horizon Acquisition and Command Range for 300-Statute Mile Altitude

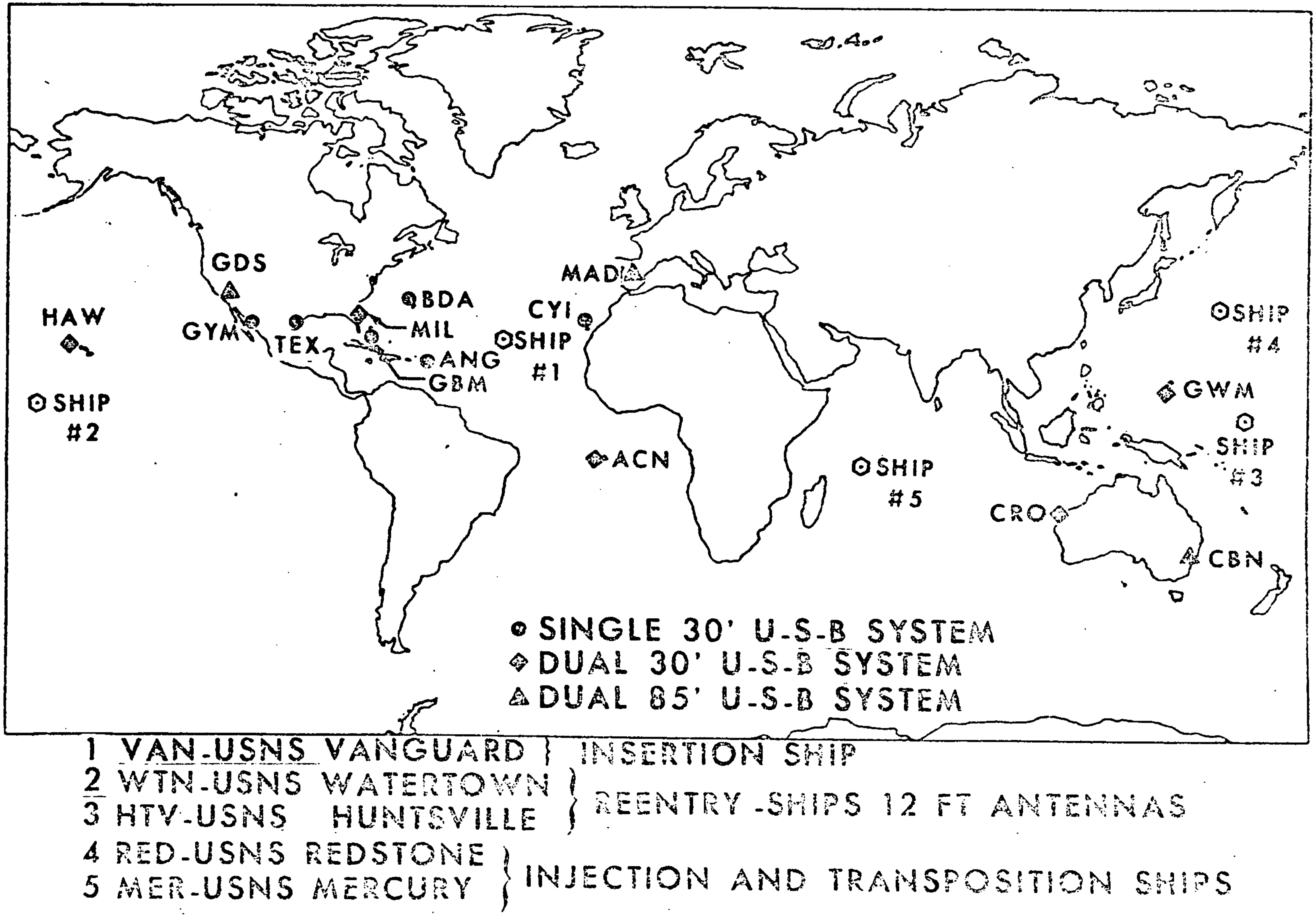


Figure V-3. MSFN Unified S-Band Stations

4. MSFN, 38° Inclination

   A pass just north of the Guam cone, then going through the hole between Hawaii and Goldstone results in two full orbits between contacts.

5. STADAN, 55° Inclination

   A pass just north of the Rosman cone is picked up by Johannesburg and Tananarive, resulting in 1-1/2 orbits between station contacts.

6. MSFN, 55° Inclination

   A N → S pass through the hole between Goldstone and Hawaii is not picked until over Hawaii one orbit later. This results in two full orbits between contacts.

7. STADAN, 90° Inclination

   A S → N pass, just west of the Johannesburg cone results in 2-1/3 orbits between contacts. In addition, misses between Alaska and Rosman and S → N passes just west of Alaska result in 2 and 1-3/4 orbits between contacts respectively.

8. MSFN, 90° Inclination

   Passes between Hawaii and Goldstone and passes between Bermuda and Canary both result in two full orbits between contacts.

   It can be seen that using MSFN makes very little difference in worst case time between station contacts.

3.2 MSFN COMMUNICATIONS

MSFN stations utilize the Unified S-band (USB) system, using S-band for command telemetry and tracking. The command code is PCM compared to the STADAN tone system. STADAN utilizes VHF for command and telemetry and either a minitrack beacon or a VHF range and range rate transponder (GRARR).

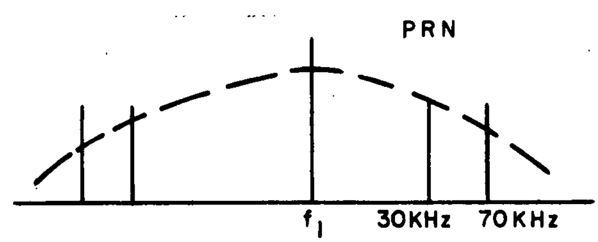
It should be noted that current NASA plans indicate that STADAN, MSFN and the Deep Space Net will be merged into one network in the next few years with similar capabilities at all the principal stations. MSFN, in addition to maintaining its present capabilities, will be implementing the STADAN tone sequential and tone digital command systems. STADAN will be implementing the USB system and will phase out the GRARR system. Currently the STADAN Alaska and Rosman stations are equipped for USB and the Santiago, Tananarive and Carnarvon stations have S-band range and range rate capability.

3.2.1 Unified S-Band

USB provides the capability for telemetry, tracking and command on a single carrier.

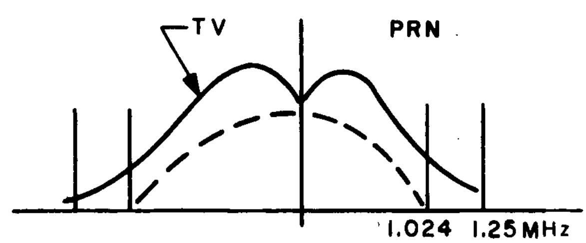
(1) Uplink

The uplink provides for PCM digital commands on a 70 kHz subcarrier and data on a 30 kHz subcarrier. It also can contain a pseudo random 1 MB/sec. range tracking signal on the carrier.

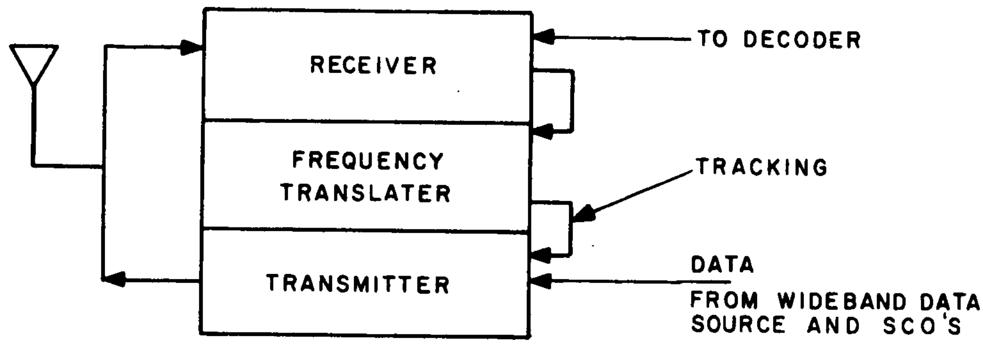


(2) Downlink

The downlink provides for range tracking by retransmission of the pseudo-random noise (PRN) signal on the carrier. PCM telemetry capability is provided on a 1.024 MHz subcarrier with another subcarrier, 1.25 MHz, also available for data. Wideband data, up to 0.5 MHz bandwidth (TV), is transmitted on the carrier.



The USB transponder, used for commands, telemetry and tracking is shown in block diagram form below:



3.2.2 Design Concepts

Essentially, compatibility with MSFN means replacing the VHF equipment proposed in the baseline study with S-band equipment. Five approaches have been examined, all of which meet the MSFN compatibility requirement, and are presented below.

Approach A — As shown in Figure V-4 and Table V-1, this approach uses the equipment that will be flown on the ERTS Program. It is a complete USB system, using digital command and has the PRN tracking capability. The average power is high and it is a heavy system (55 watts and 62 pounds). All equipment is flight qualified.

Approach B — As shown in Figure V-5 and Table V-1, this approach differs from the total USB concept in that there is no PRN tracking capability. The transponder is replaced with separate, redundant, receivers and transmitters. This system uses a digital command system as is currently in use at all the MSFN stations. The average power consumption of 75 watts is high, but the system weight of 12.4 pounds is low. The decoder would require qualification.

Appraoch C — As shown in Figure V-6 and Table V-1, this approach is very similar to Approach B. There is no PRN tracking capability and the system has separate,

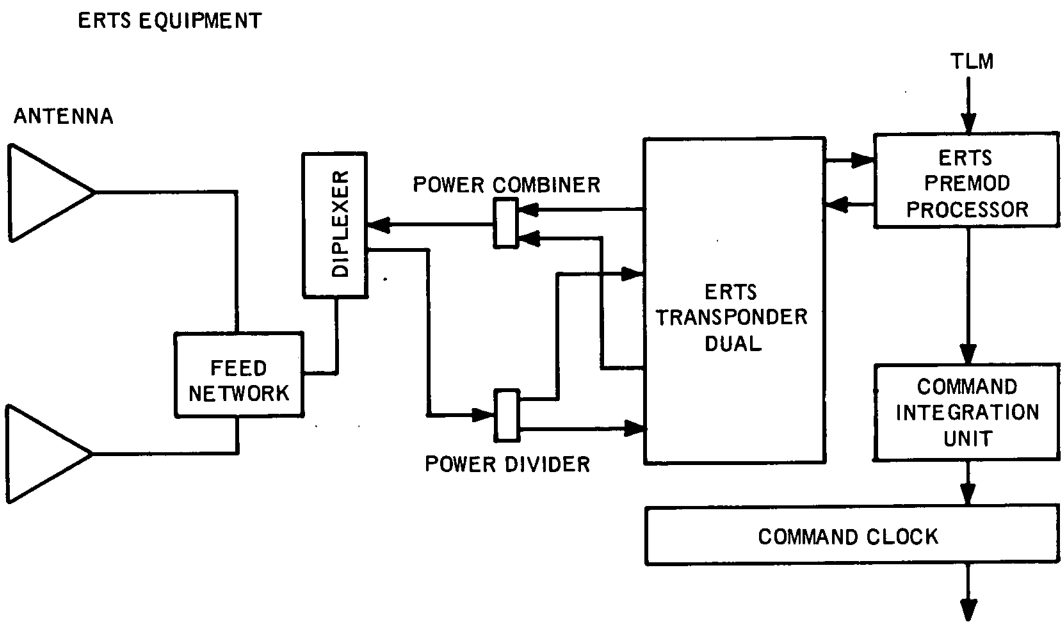


Figure V-4. Digital USB Command, Range & Range Rate (Coherent Tracking, Approach A)

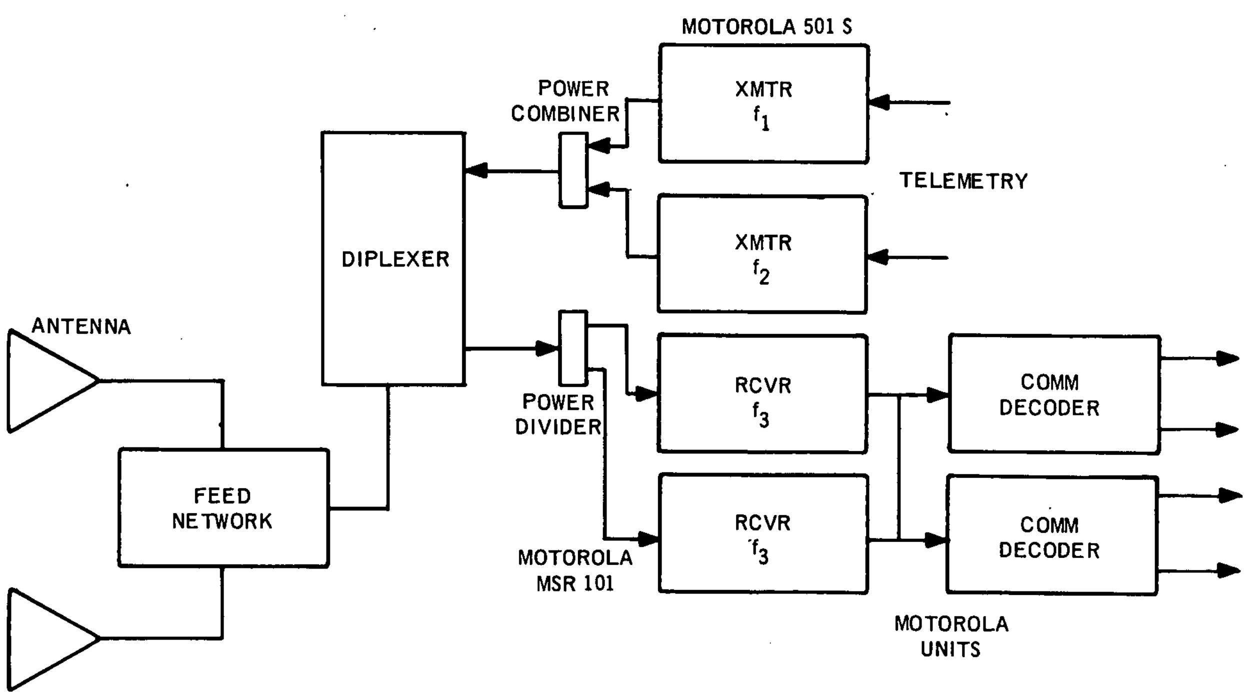


Figure V-5. Digital USB Command, Approach B

TABLE V-1. COMPARISON OF S-BAND APPROACHES

Totals	Transmitter	Receiver	Discriminator	Bit Detector	Command Decoder
55W 62# 1872 in^3 $350K	Motorola ERTS Transponder 28W (1 On) 24# 624 in^3 PM	($100K)	SCI ETS Premod Processor 4.4W ($75K) 10# 312 in^3	GE Command Integration Unit 2.5 W ($75K) 8# 312 in^3	Calcomp Command Clock 20 W ($100K) 20# 624 in^3
75W 12.4 # 220 in^3 $122K	2-Motorola MTT 501S ($10K ea) 70W (1) 4.6 # 64 in^3 FM	2-Motorola MSR-101 ($21K ea) 1.4 WSB 5.6 WOP 4.8 # 76 in^3	N. R.	N. R.	2-Motorola (Dev. Units) Based Upon Rcvr Operating with USB 4 W ($30K ea) 3 # 80 in^3
35W 14 # 197 in^3 $98K	2-Teledyne ($20K ea) TR 2400 25W (1) 5.6 # 72 in^3 FM or PM	2 Avco AED 303-2 ($10K ea) 5.6W OP 4.2 SB 2.4 # 48 in^3	N. R.	N. R.	1-AVCO AED 4-7 ($38K ea) 4 W Oper., 0.1 W SB 6 # 125 in^3
37 W 16.8 # 293 in^3 $108K	2 MSX 201 ($35K ea)* Transponders 37 W (1) 1.4 W SB 10.8 # 168 in^3 PM		N. R.	N. R.	1-Avco AED 407 4 W 6 # 125 in^3
	SB - STAND-BY OP - OPERATING	*PROVIDES INTERNAL SUBCARRIERS 1 - 128KBPS 1 - 10KBPS			

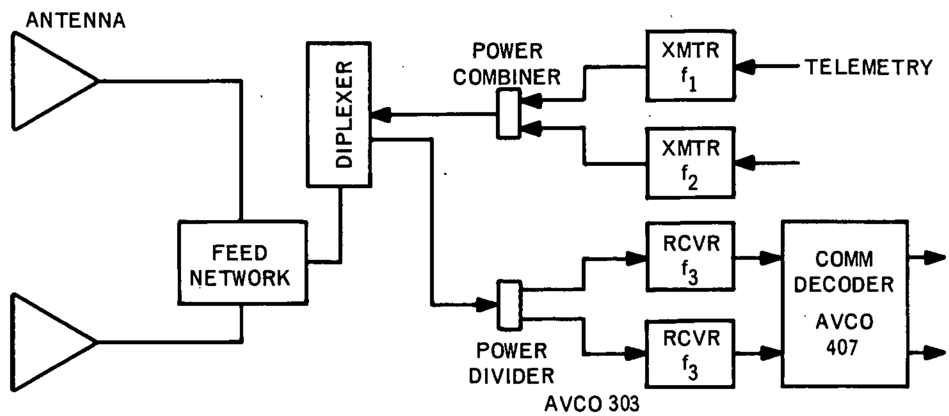


Figure V-6. Tone Command System, Approach C

redundant, receivers and transmitters. The difference between Approaches B and C is that Approach C utilizes a tone command system in place of the digital command system. Current NASA plans indicate that MSFN stations will have a tone command capability by 1973. This system uses considerably less power, 35 watts, than Approaches A or B, and the system weight is 14 lbs. All equipment except the receiver is qualified.

Approach D — Approach D as shown in Figure V-7 and Table V-1 uses a transponder to provide for range tracking in addition to commands and telemetry. This approach also uses a tone command system. The power consumption is 37 watts and the system weighs 16.8 lbs. All equipment proposed for use in this system is qualified.

Table V-1 indicates the equipment used in each approach along with estimates of weight, size, power consumption and cost.

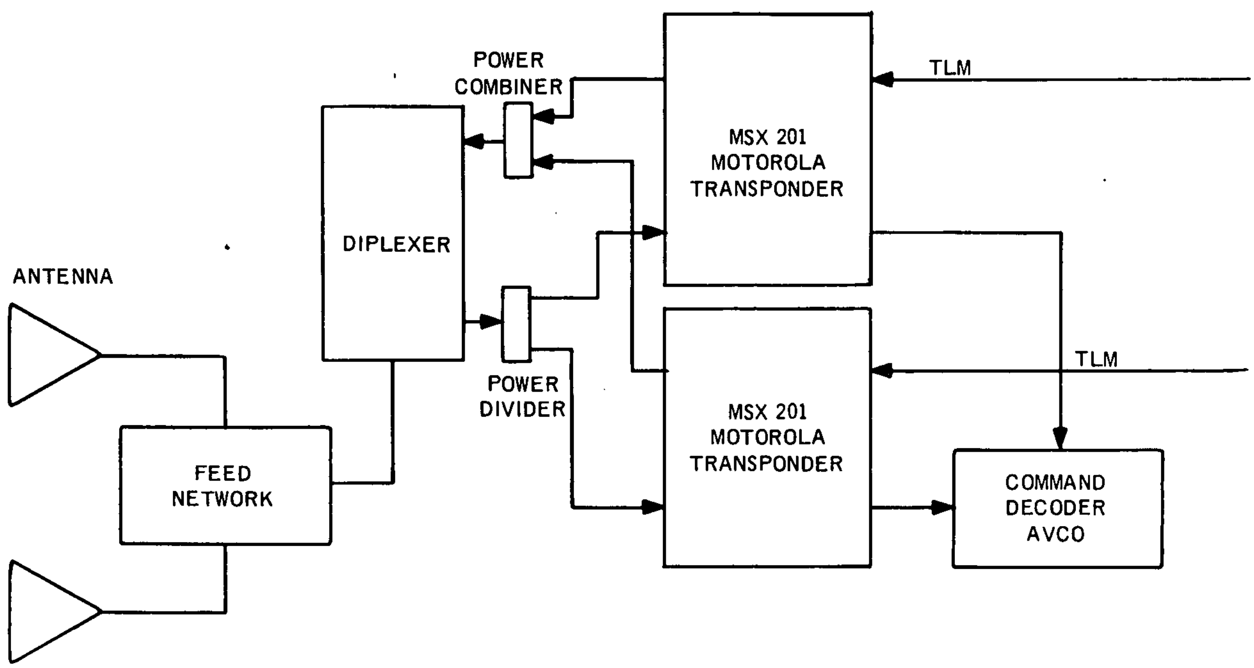


Figure V-7. Tone Command, Range Tracking, Approach D

In summary, Approach A has complete USB capability, but compared to the other approaches, has a very high cost in addition to high power consumption weight and volume. As a result, it is considered an unattrative approach. Approach B utilizes the digital S-band command system, but has no transponder for range tracking. Although the weight volume and cost are comparable to Approaches C and D, the very high power consumption of Approach B eliminates it from consideration. Approaches C and D both utilize the tone command system. C has no transponder for range tracking. To handle wide band data, the two transmitters in Approach C would operate simultaneously at different frequencies, one for wideband data (TV) and the other for the remaining spacecraft and experiment data. The system would have the capability of switching the inputs to each transmitter to preclude a loss of redundancy (see Figure V-8). Approach C, however, is somewhat lower in power, weight and cost and considerably lower in volume. Based on these comparisons, it is recommended that Approach C be used for the new baseline S-band design. This is shown in more detail in Figure V-8. No separate beacon or transponder is proposed for the S-band design for any of the missions. The beamwidth of the station antennas (1° for 30 ft disk, 0.35° for 85 ft disk) will permit accurate angle tracking. Doppler and skin tracking capabilities are available at the stations if needed. Doppler tracking on the telemetry carrier will provide range rate, and skin tracking by the MSFN stations will provide range data.

A transponder can be added if required. Qualified transponders for about $35K weighing about 6 lb, 100 cubic inches in volume and requiring 1 watt of standby power and 3 watts during operation are available.

3.2.3 S-Band Baseline Design Details

3.2.3.1 Antenna — The proposed VHF baseline loop antennas would have to be replaced for the S-band design. Two designs have been considered, slot arrays and stubs (see Figure V-9). In each case, two would be required because of the ground plane effect of the solar panels. The slot arrays would provide -4db gain over 4π steradians as compared to -12 db for the stub antennas.

3.2.3.2 Link Calculations — The S-band link calculations for uplink and downlink are shown in Tables V-2 and V-3.

3.2.4 Comparison of VHF and S-Band

The proposed baseline S-band system, as compared to the VHF baseline incurs very little penalty in terms of weight, power and volume. However, it provides an enormous increase in flexibility by providing for wide band data up to 0.5 MHz bandwidth. The 3.5 lb and 2.9 watt increases present no significant system design problems. Table V-4 presents a list of these comparisons.

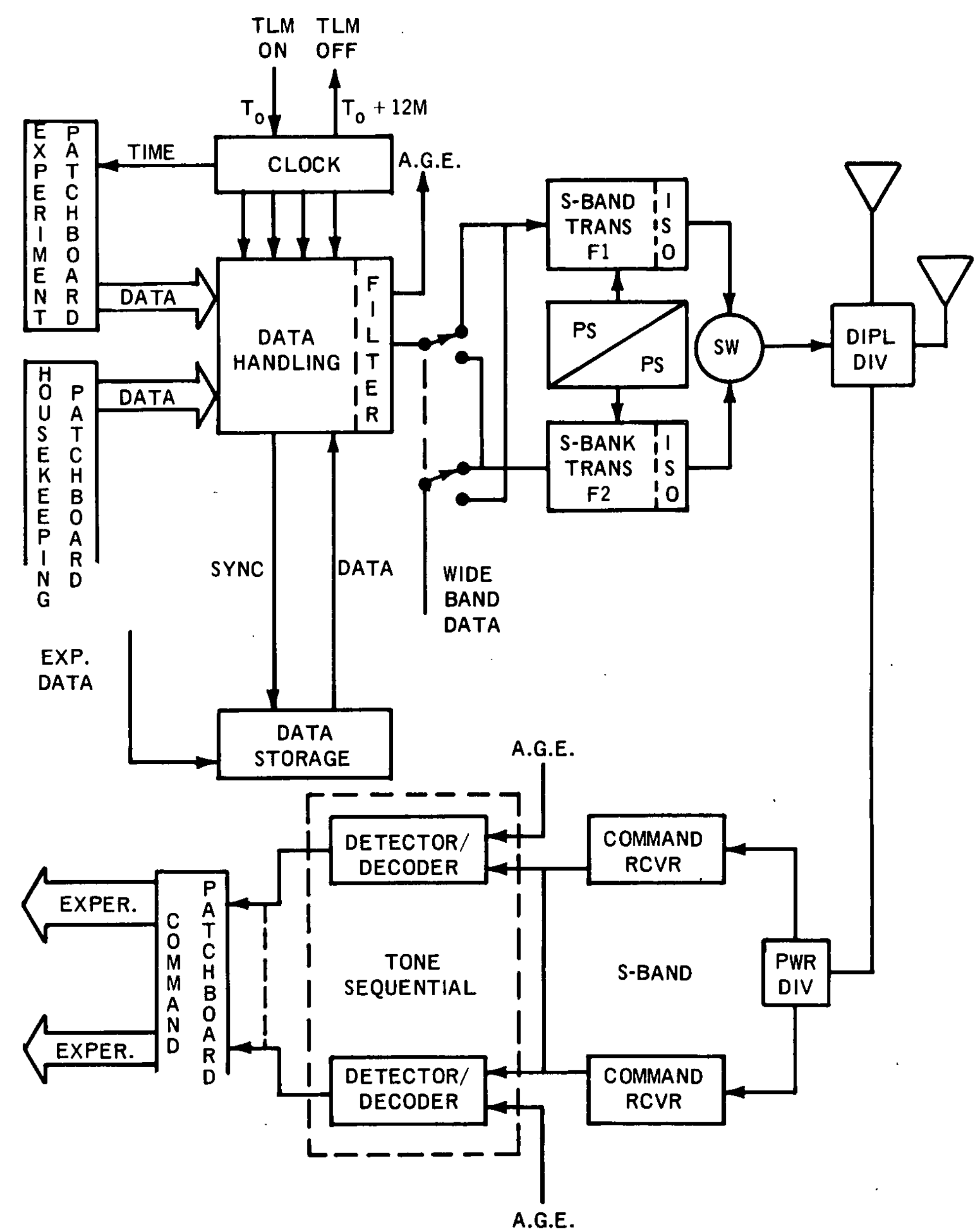


Figure V-8. Baseline S-Band Design - All Missions

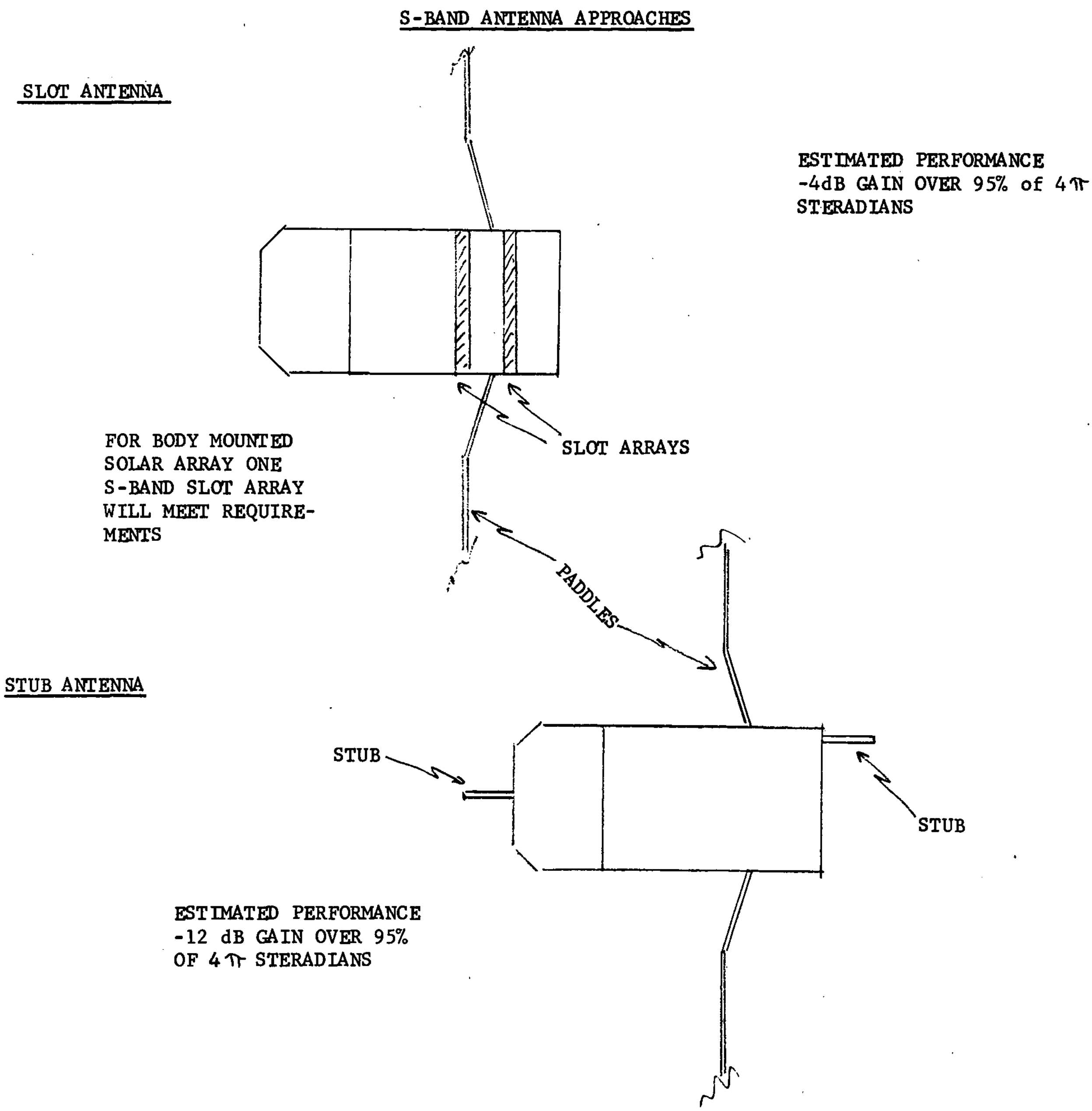


Figure V-9. S-Band Antenna Approaches

TABLE V-2. MISSIONS I AND II, S-BAND DATA LINK PERFORMANCE S/C TO MSFN GROUND STATION

	Transmitter power (5 watts)	4.8 dbW
	Transmit circuit loss	-3 db
	Transmit antenna gain	-4 db
	Path attenuation (F=2300, R=1250)	-166.6 db
	Polarization loss	-3 db
	Receiving antenna gain (30')	+42 db
	Receiving antenna pointing loss*	--
	Receiving circuit loss*	--
	Receiver power	-129.8 dbW
	Data Mod Loss	-0.8 db (-1.2 db WC)
	Data Power	-130.6
	KT (100° K)	-208 dbw/Hz
PM	C/No	77.4 dbHz
	Available with 8 db margin (-12 db antenna gain)	69.4 dbHz
	R with E/No = 9 db (10^{-3})	1.1×10^6 bps
	For 2 MHz - 13 db TV; C/No (500 KHz Baseband)	+76 dbHz
FM	Transmitter power	18 watts
	Noise power; 300 kHz BW	-153.2 dbW
	SNR (with 8 db margin)	+15.4 db

*Included in antenna gain

TABLE V-3. S-BAND COMMAND LINK PERFORMANCE, MSFN TO S/C

Transmitter power (20Kw)	+ 43 dbW
Transmitter ckt loss	- 1 dB
Transmit antenna gain (30 foot dish)	+ 42 db
Path attenuation (F=2200 MHz, R=1250 nm)	-166 db
Polarization loss	- 3 db
Receiving antenna gain	- 4 db
Receiving ckt loss	- 3 db
Receiver power	- 92 dbW
Receiver sensitivity (Avco 303-2)	-140 dbW
Margin	48 db

4.0 EXPERIMENT OPTIONS

4.1 TV REQUIREMENTS

The requirements for a TV system are:

(a) One minute per orbit
(b) 525 lines, 24 frames/second
(c) 0.1 inch resolution in a 7.50 inch by 10 inch format
(d) Black and White

4.1.1 TV Cameras

The following cameras were found to be best suited for this application. All are basically 525 lines, 30 frames/second cameras, but can be modified to 24 frames/second.

TABLE V-4. COMPARISON ON OLD AND NEW BASELINE APPROACHES

	VHF (OLD BASELINE)	S-BAND (NEW BASELINE)
WEIGHT	26.9 lbs.	30.4 lbs.
Power	6.1 w. AVE	9.0 w. AVE*
Volume	562 cu. in.	573 cu. in.
TLM. margin (WC)	17.7 db (I&II) 8.7 db (III)	35.5 db**(I&II), 9.9 db***(III)
CMD. margin (WC)	47.7 db (I&II), 5.7 db (III)	48 db (I&II), 12.4 db (III)
MTL. cost	100K	120 K
Flexibility	very little	Wide band data, TV, variable PCM formats
Reliability	Redundant	Redundant
Compatibility	STADAN	STADAN/MSFN/DSIN
Other		Simultaneous data
		All missions identical
		Frequency diversity

* Can be reduced to 5.5 watts by modifying the S-band receivers

** Approximately 25 db margin is lost going to 0.5 MHz wideband data

*** Approximately 35 db margin is lost going to 0.5 MHz wideband data

	Size (Cu. In.)	Weight (lb)	Power (CW)	Qual.
RCA-Apollo	72	8.5	14	yes
Westinghouse - WTC	272	8	8	no
Westinghouse - Submin	11.3	0.6	6	no
Teledyne - Microeye	21	1.25	11	yes

The RCA camera is color but can be modified to black and white.

4.1.2 TV System

The resolution, lines per frame and frame rate requirements result in requiring a 3.5 MHz TV signal. There are no tape recorders on the market capable of recording the 3.5 MHz signal and meet the mission environment constraints. If recording is required, 1 MHz is the practical frequency limit. In addition to the constraint imposed by the recorder, ground station intercommunication links limit the signal to approximately 0.5 MHz (stations at Madrid and Goldstone have real time TV capability to Houston, Goddard can receive and display TV in a control-center and most other MSFN stations have monitor capability).

A 320 line, 10 frame per second Apollo camera format will result in a 0.5 MHz bandwidth and a record speed of 50 inches/second. All the cameras previously listed except the Teledyne can be modified to the 320 lines, 10 frames per second format. The proposed S-band baseline design has the bandwidth capability for this type of TV signal. The VHF baseline cannot handle TV.

Worst case conditions for Mission III will prevent real time transmission of TV data. The problem may be avoided by playing back the data near perigee or by replaying the data at a slower rate.

4.1.3 System Considerations

In addition to the TV camera, the TV system will also require a 60 watt (approximate) lamp inside the experiment package. At one minute per orbit, this adds an extra 0.7

watts average to the spacecraft power system. In addition, the recorder will require approximately 20 watts and weight about 15 lbs. The system power effect of the recorder is dependent on operation, but if one minute record and five minutes playback is assumed, the system power is increased by 4 watts average. Approximately ten extra commands would be required for control of the camera, lamp and recorder.

4.2 FROG OTOLITH EXPERIMENT

Figure V-10 presents concepts of how the frog otolith experiment could be handled by the VFH baseline system. The experiment data includes four otolith signals varying in frequency from 150 to 2000 Hz, 2 EKG signals of 6 to 100 Hz and one water pressure signal. Both options shown indicate the use of a separate SCO for each signal. Option one maintains the redundant transmitter baseline design concept. Considering a six minute station pass and the VHF bandwidth restrictions, the recorder playback speed ratio is limited to 10:1. This results in a one hour record capability. Further information on the nature of the experiment may prove that one hour record time between station passes is sufficient. This concept has minimum impact on the baseline design. Option 2 provides for a longer record time by utilizing both transmitters simultaneously. This allows a 20:1 playback, split between the transmitters but does not conform to the redundant transmitter design concept.

In addition to the options shown, many options for digital techniques are available, but at the cost of compromising experiment data or adding digital equipment. As examples:

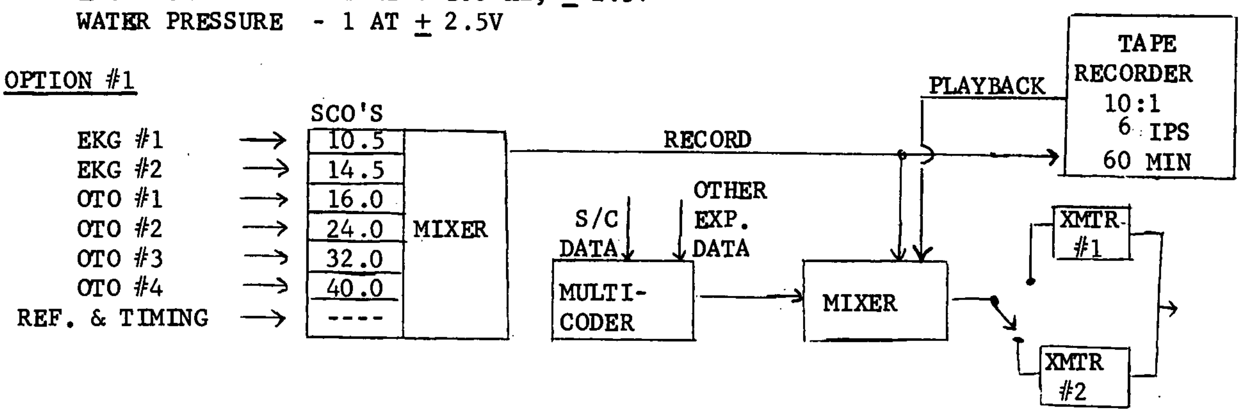
(a) Transmit the output of a pulse counter
(b) Convert N experiment pulses to one readout pulse
(c) Sequential sampling
(d) Data compression
(e) Adjust record rate and SCO frequencies according to expected otolith rates.

These techniques require lower record and playback rates resulting in more reliable recorder operation and/or more recording capacity. In general, these approaches are more expensive.

REQUIREMENTS:

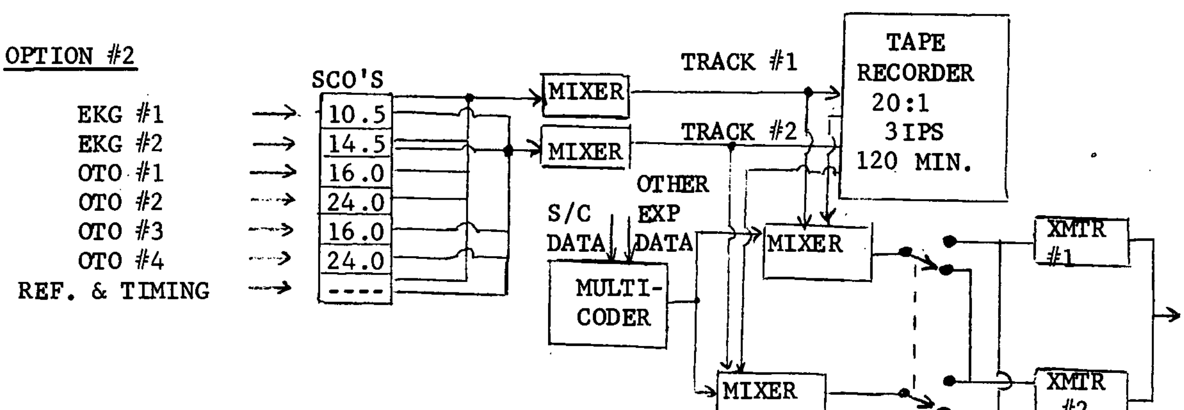
OTOLITH SIGNALS - 4 AT 150-2000 HZ, 5V P-P
EKG SIGNALS - 2 AT 6-100 HZ, ± 2.5V
WATER PRESSURE - 1 AT ± 2.5V

OPTION #1



ADVANTAGES - MINIMUM IMPACT ON BASELINE SYSTEM
- TRANSMITTER REDUNDANCY
- ALL DATA FROM ONE TRANSMITTER
DISADVANTAGES - 60 MINUTES RECORD TIME

OPTION #2



ADVANTAGE - LONGER RECORD TIME
DISADVANTAGE - TWO TRANSMITTERS ON FOR DATA
- LOSS OF COMPLETE TRANSMITTER REDUNDANCY

Figure V-10. Frog Otolith Experiment

5.0 COMPONENT DEVELOPMENT

The proposed baseline S-band design utilizes off-the-shelf transmitters, receivers and decoders (see Table V-1). The decoder would require qualification. The VHF baseline design utilizes off-the-shelf hardware for:

(a) Command Receiver - Avco AED301A
(b) Command Decoder - Avco AED404
(c) Transmitter - AACOM Model 101
(d) Beacon - AACOM Model 101
(e) Diplexer - Microlab FXR-KFA28
(f) Hybrid - ANZAC Model JH509
(g) Data Handling Unit - Same as used on RVTO-3B GE-RESD

Both baselines (S-band and VHF) could utilize the same data storage unit, a modified Electronics Memories model SEMS-5L, used on the OAO Program. The spacecraft clock will be designed and developed based on the timing devices developed by GE-RESD for the LAR and RVTO Programs.

VI. ATTITUDE CONTROL SUBSYSTEM BASE LINE OPTIMIZATION

1.0 PURPOSE

The purpose of the Bioresearch Module Attitude Control Studies was to perform a more detailed analysis of the variable "g" spin rate control system concept which entails the extension and retraction of booms to vary the Bioresearch Modules (BRM) roll moment of inertia properties. Specific objectives of these studies included the determination of overall BRM stability with booms in motion or at rest, boom stability, a discussion of suitable techniques for synchronizing boom deployment, the interaction of the cold gas reaction control system with deployed booms, and system reliability.

Results of these studies indicate that the proposed approach is definitely feasible by virtue of the fact that a boom configuration, that is one having adequate stiffness, is available and has a successful flight test history on the USAF sponsored SESP 71-2 Program. The booms are used to deploy a roll up solar array developed by Hughes Aircraft Co. In addition, these booms will also be used in a similar application on the Meteoroid Technology Satellite, a NASA program scheduled for launch in late 1972.

Attitude control studies also produced a subsystem approach which would align a vehicle, whose solar array was body mounted, normal to the ecliptic plane. Salient features of this proposed design include the deletion of the rate gyro package, use of Pioneer developed sun sensors and the ability to achieve and maintain the desired orientation with respect to the ecliptic.

2.0 ANALYSIS

Initial analytical studies established the complete equations of motion which defined the multi degree-of-freedom problem involving the spinning vehicle and the independent three-degree-of-freedom motion of each boom[1]. These equations were programmed for digital solution (see Appendix). However, because of the slow running time of the program (dictated by the dynamics of the problem), and the long turn-around time required to change problem parameters, it became necessary to modify this approach in order to meet program commitments. The only alternative then, was to turn to a simplified analog simulation of the problem.

This was accomplished by constraining spacecraft motion to only two-degrees-of-freedom (i.e., pure spin) and employing the expedient that all booms are deployed at a uniform rate. The resultant equations were analyzed and the desired stability criterion identified. As derived, the governing equations are:

Roll Torque Equation

$$I_x \dot{p} = -\sum_i^n K_i u_i r_i , \quad (i = 1, \ldots, 4) \text{ no. of booms} \tag{1}$$

where;

I_x = spacecraft roll moment of inertia, slug ft^2

$\dot{p}$ = spacecraft angular acceleration, rad/sec^2

K_i = boom spring constant at a given length r_i, lb/ft

u_i = lateral tip displacement of boom measured from a rotating coordinate frame fixed in the spacecraft, ft

r_i = deployed distance of a given boom, ft

Force Equations

$$F_i = m^* [\ddot{r}_i - r_i p^2 + u_i \dot{p} + 2 \dot{u}_i p] , \quad (i=1, .., 4) \tag{2}$$

(1) Cloutier, G.J., "Dynamics of Deployment of Extendable Booms from Spinning Space Vehicles," Journal of Spacecraft, Vol. 3, No. 5, May 1968.

where;

F_i = the total integrated force acting along the boom, lbs

m_i^* = effective mass of boom and tip mass, slugs

$\ddot{r}_i$ = linear acceleration of the boom in the direction of deployment, ft/sec^2

$r_i p^2$ = centrifugal acceleration ft/sec^2

$\dot{u}_i$ = rate of lateral tip displacement, ft/sec

$2\dot{u}_i p$ = Coriolis acceleration, ft/sec^2

and,

$$K_i u_i = m_i^* [2\dot{r}_i p + r_i \dot{p} - \ddot{u}_i + u_i p^2] \tag{3}$$

where;

$K_i u_i$ = the total integrated force acting to bend the boom, lbs

$\ddot{u}_i$ = linear transverse acceleration of the boom, ft/sec^2

Assuming that four booms will be used to control spacecraft spin rate and applying the assumptions used to facilitate this analysis, then the torque equation can be simply stated as

$$I_x \dot{p} = -4Kur \tag{4}$$

The equations defining boom motion in the spin plane are

$$\ddot{r} = rp^2 - 2\dot{u}p - u\dot{p} + F/m^* \tag{5}$$

$$\ddot{u} = r\dot{p} + 2\dot{r}p + up^2 - ku/m^* \tag{6}$$

It will be observed that if the deploying mechanisms are to have the capability of extending and retracting the booms at a uniform rate, then $\ddot{r}$ must equal zero. Therefore, equation (5) will be used to derive requirements for the deployment mechanism drive motor.

3.0 STABILITY CRITERION

The desired stability criterion is obtained from equation (6) in the following manner. If the motion of the boom is to be controlled, then it is desirable to restrict lateral acceleration so that $\ddot{u} = 0$. Next, by substituting the expression for $\dot{p}$, obtained from equation (4) into equation (6), the resultant equation can be manipulated to define the transfer function for lateral tip displacement as a function of deployment rate.

$$u/\dot{r} = 2p/[K/m^*(4r^2m^*/I_x + 1) - p^2]\text{sec} \tag{7}$$

In order for this transfer function to remain monotonic, it is necessary that the denominator on the right hand side always be positive. Hence, the stability criterion is defined. Examination of this expression reveals some interesting characteristics. For example, the term K/m^* is the undamped natural frequency, squared, of the boom at a given length, r, and the expression $(4r^2m^*/I_x + 1)$ is always greater than unity. The nature of this expression is to increase as the booms are deployed.

For the cases considered in this study, its values ranged from 1.0 to 3.8, which correspond to g levels of 1.5 to 0.1, respectively. To assure stability, that is, to limit the oscillations of the boom tip, it is necessary that the product K/m^* $[4r^2m^*/I + 1]$ always be greater than the spacecraft spin rate squared. This constraint can be satisfied by selecting a boom with the proper stiffness. The effective spring constant of the boom has been defined as

$$K = \frac{EI\ (2X^2 + 15X + 3)}{5r^3\ (8X + 3)^2} \tag{8}$$

where;

EI = boom stiffness, lb in^2

X = ratio of tip mass to mass of deployed boom

r = boom length, ft

4.0 IMPLEMENTATION STUDIES

During the Bioexplorer Study, when this variable "g" control concept was first presented, a tentative boom selection was made to determine the effect such a device would have on the BRM design. At the time, a 0.5 inch diameter configuration was selected and, it was upon its characteristics that deployment rates, maximum length, and other properties were selected. Quite logically then, this was the initial boom configuration evaluated in the stability studies. It was found that the 0.5 inch diameter boom with a stiffness of 10^4 lb in^2 tipped with a 0.25 lb mass did not satisfy the prescribed stability criterion. Figure VI-1 shows an analog trace of this boom's performance. The most significant characteristics to be noted in this run are, that once the boom starts to oscillate (the channel depicting the transverse displacement u), the oscillations develop and grow about a steady state displacement. Moreover, as these oscillations become more vigorous, their effect on the BRM spin rate appears as a small ripple on the p trace.

This constitutes an undesirable feedback condition that continues to exchange energy between the booms and spacecraft. Needless to say, this boom could not have endured the oscillations shown and would have surely broken off under other than simulated conditions. Figure VI-2 shows the variation of the spring constant of the 0.5 inch boom as a function of payload acceleration. Also, plotted on the lower part of Figure VI-2 is the characteristic of the stability criterion for the 0.5 inch boom. The point at which these two curves cross is the point at which the boom starts to oscillate.

The problem now was one of finding a suitable boom configuration before continuing with the remaining tasks of the attitude control studies. After contacting the vendor (Spar, Inc., Toronto, Canada) and discussing the boom requirements in more definite terms, a configuration was selected which had recent and successful flight test experience and which met the stiffness requirements derived from the most recent studies. The spring constant and stability criterion for this boom are shown in the upper portion of

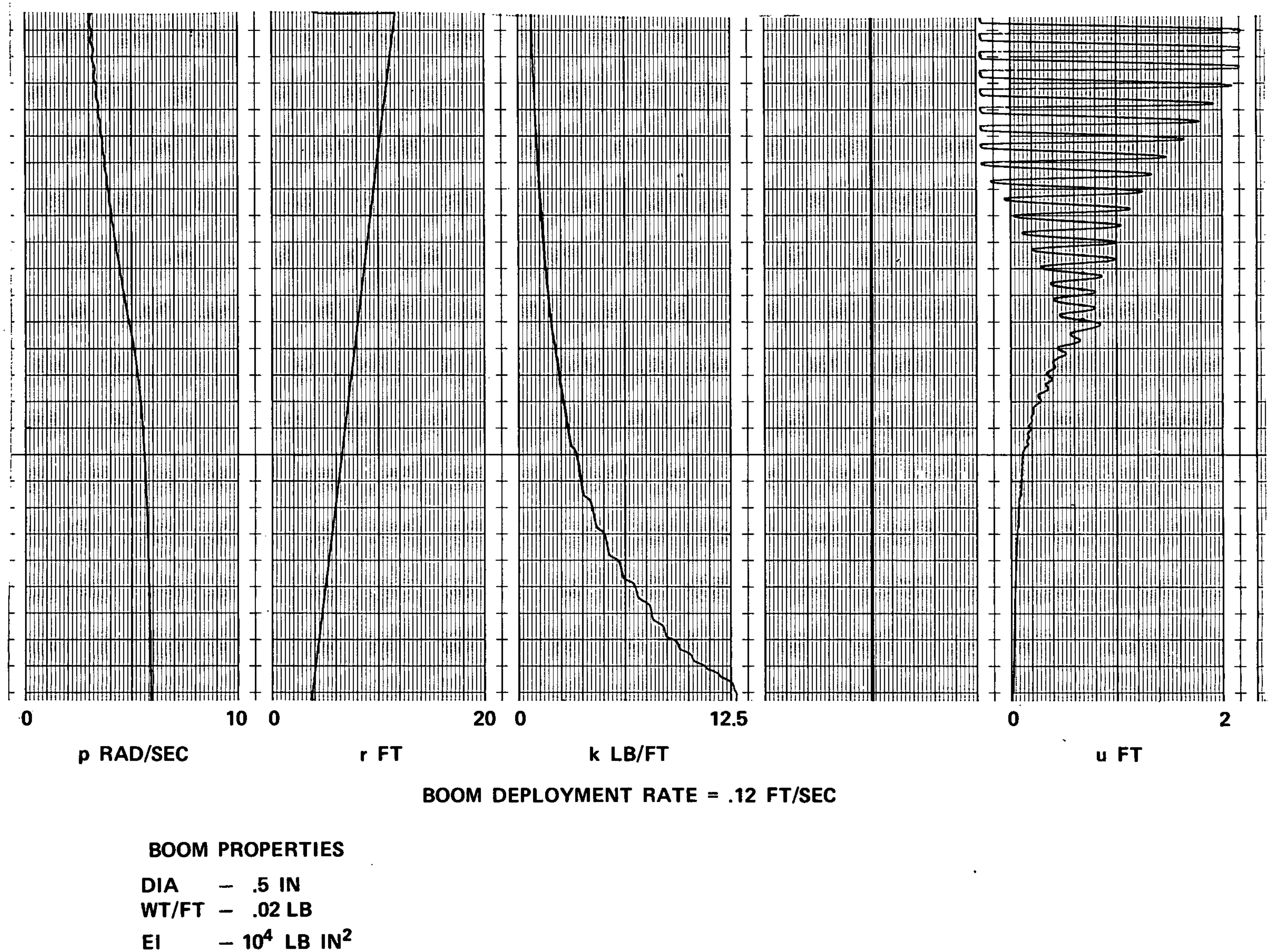


Figure VI-1. Attitude Control, Boom Deployment Rate = 0.12 ft/sec

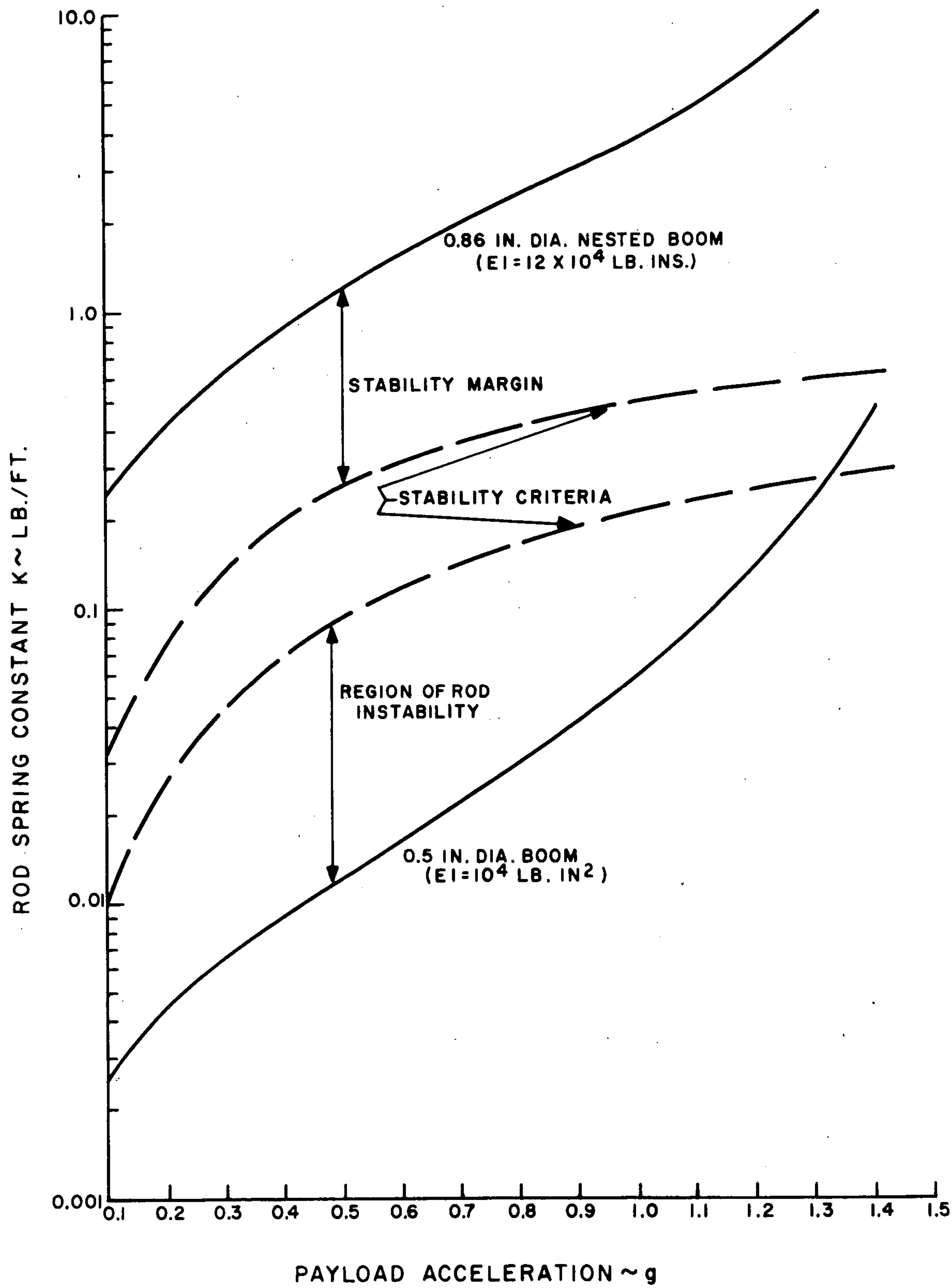


Figure VI-2. Attitude Control, Payload Acceleration (g) vs Rod Spring Constant (K) of Figure VI-2. The booms themselves consist of two elements, one nested inside the other, and are 0.86 inch in diameter. They are available as individual packages or as bidirectional packages. The latter configuration is the most attractive one to pursue from several points of view. Of prime interest, is the fact that it is the lighter of the two configurations. That is, a four boom design (comprised of two units) weighs 26 lbs, about 6 pounds less than four individual units. The total weight increase to the system due to the change from a 0.5 inch diameter boom to the 0.86 inch size is about 11 lbs. Another important feature of the bi-directional package is that a single motor can drive all four booms thus, assuring uniform deployment. An outline and installation drawing of the proposed bi-directional boom assembly is shown in Figure VI-3. Performance results with these booms are shown in Figures VI-4 and VI-5.

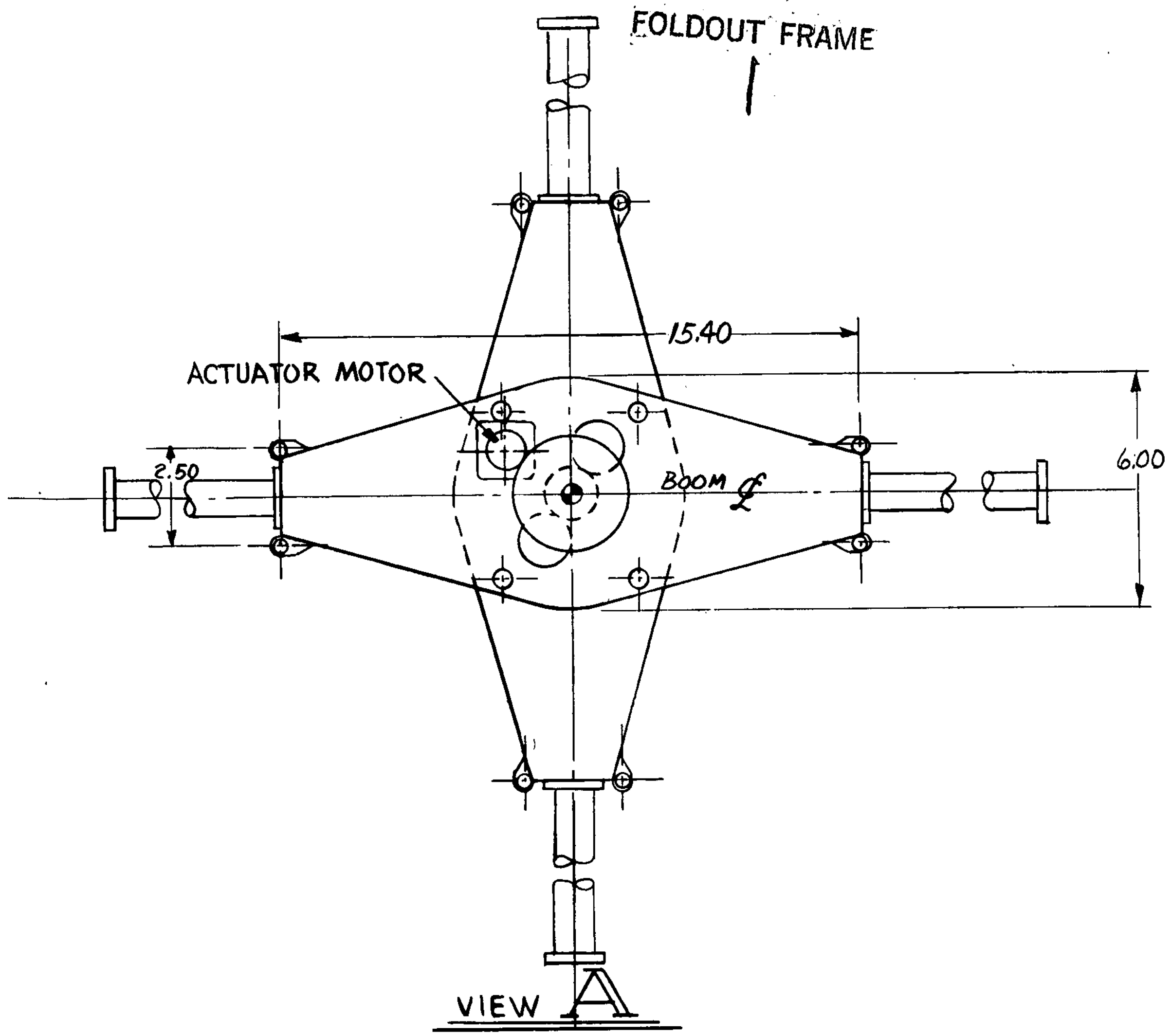
Figure VI-4 depicts the case where the booms are initially stowed and are subsequently deployed to the maximum 22 ft length which corresponds to a payload acceleration level of 0.1 g. Figure VI-5 shows the boom behavior and spacecraft response as they are retracted. One point to be made at this time is that the final boom selection has not been made and that more effort is required to further refine boom requirements. Boom characteristics described above show sufficient margin to justify further work in this area. Specific objectives of this activity would be to optimize boom size and weight. As a result of the studies reported on here, it is felt that a boom design can be developed to satisfy performance and to contribute a zero weight growth factor to the total spacecraft configuration.

5.0 BIORESEARCH MODULE CONFIGURATION

5.1 BASELINE STUDIES

The established baseline spacecraft configuration used in these studies was one featuring deployable solar paddles. The paddles are symmetrically spaced around the periphery of the spacecraft so that the normal to the solar sensitive surface is essentially parallel to the spacecraft spin axis. In using such a configuration, it is necessary to periodically

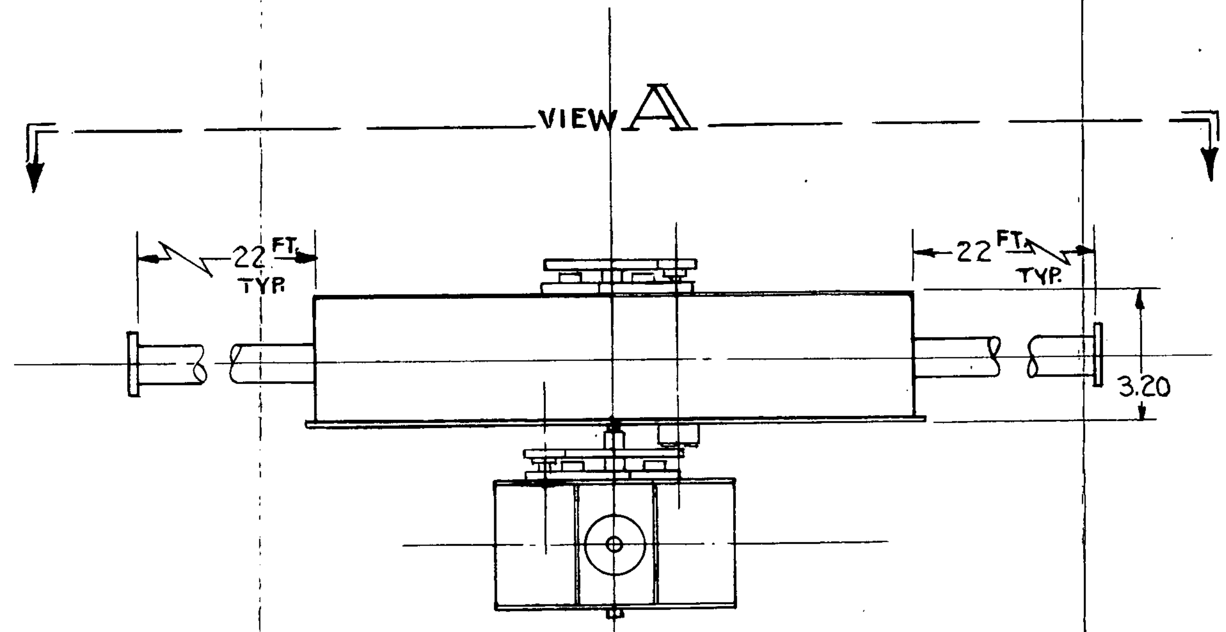
FOLDOUT FRAME 1



NOTES:

1. DIMENSIONS ARE FOR REFERENCE ONLY.
2. BOOM LENGTH - 22 FT. MEASURED AS SHOWN.
3. BI-STEM ELEMENT: NUMBER - DOUBLE NESTED, MATERIAL - 301 STAINLESS STEEL. DIAMETER - 0.86 INCH (NOMINAL), STRIP WIDTH - 2.375 TO 2.500 INCH. THICKNESS - 0.005 INCH.
4. ESTIMATED TOTAL ASSEMBLY WEIGHT - 26. LBS.
5. NOMINAL POWER REQUIREMENTS - 3.0 AMPS @ 24 VOLTS.
6. BOOM EXTENSION RATE - 1 TO 2 INCHES PER SECOND.
7. MOTOR - GLOBE - TYPE LL WITH PLANETARY GEAR HEAD.
8. MICROSWITCHES - HONEYWELL CONTROLS - 1HM1

FOLDOUT FRAME 2

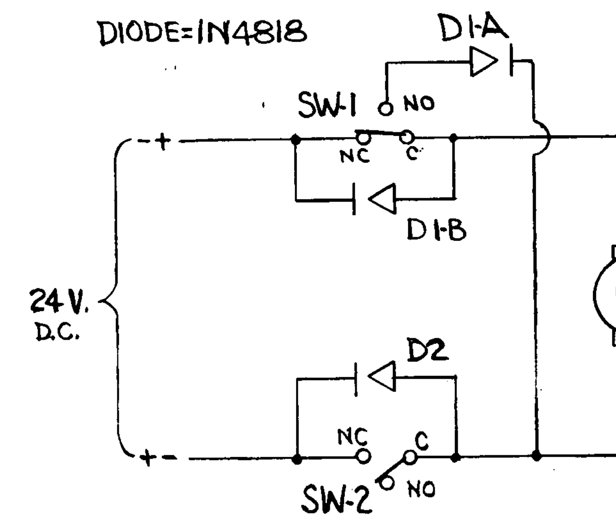


BOOM ACTUATOR ASSEMBLY

FIGURE VI-3

SW-1 EXTEND MICROSWITCH
SW-2 RETRACT MICROSWITCH

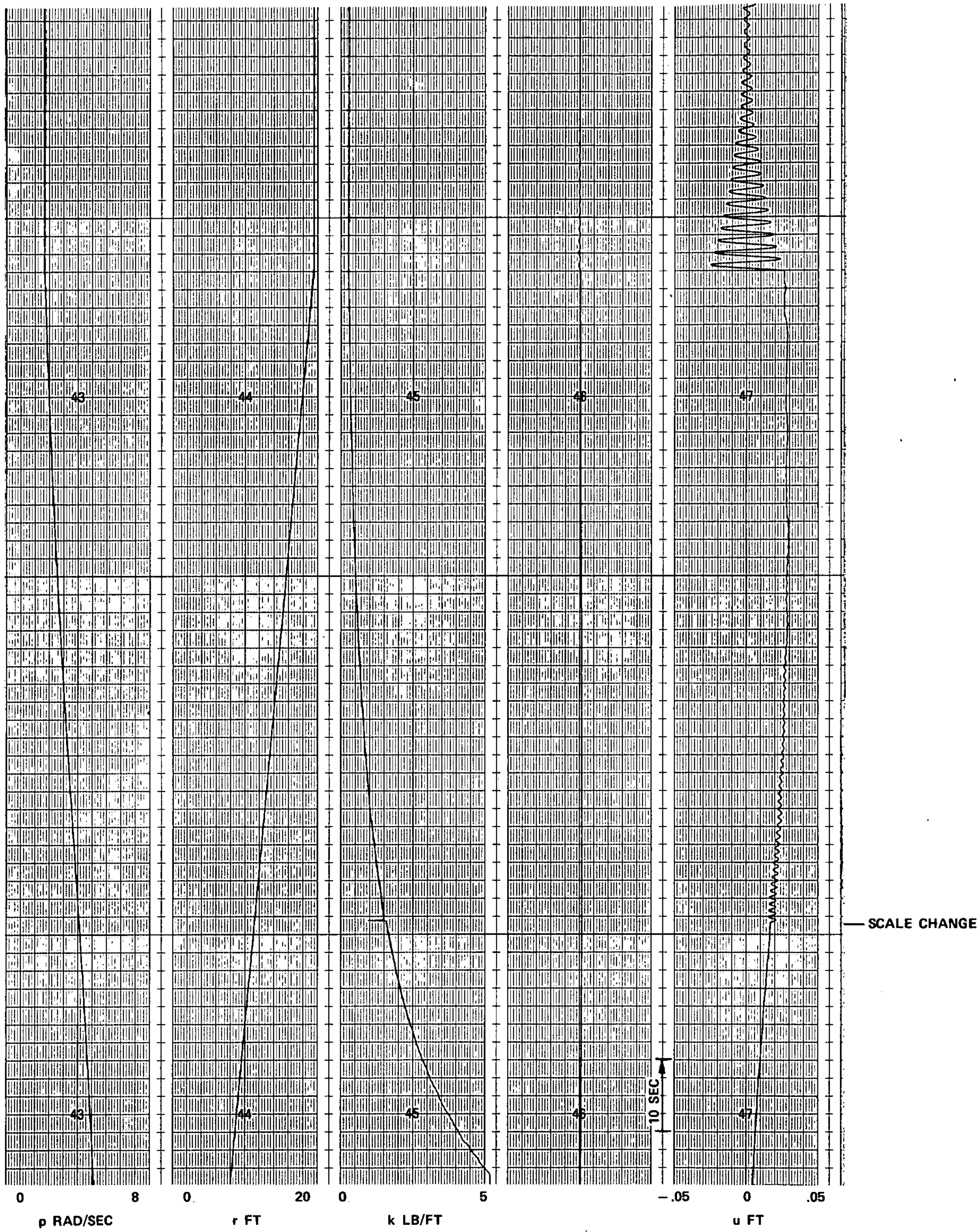
DIODE=IN4818



SCHEMATIC WIRING DIAGRAM

NOTE: CIRCUIT SHOWN IN FULLY RETRACTED POSITION.

Figure VI-3. Boom Actuator Assembly



BOOM PROPERTIES

DIA – .86 IN
WT/FT – .174 LB
EI – 12 x 10^4 LB IN^2

Figure VI-4. Deployment Rate = 1.2 ft./sec.

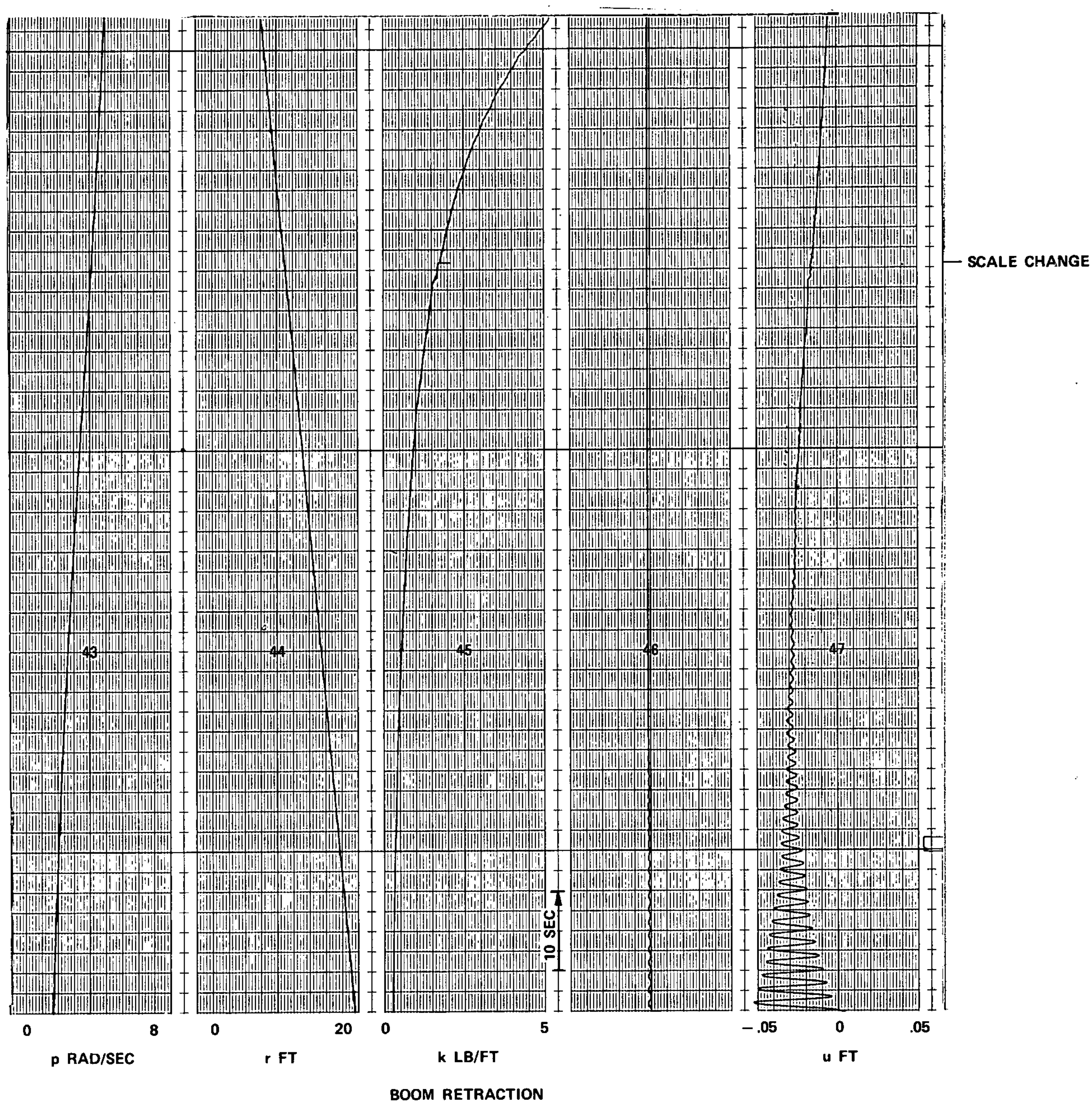


Figure VI-5. Boom Retraction

reorient the spin axis so that it is aligned to the sun within allowable limits. These orientation maneuvers would be accomplished through the use of the cold gas system mounted in the attitude control section of the spacecraft. Since the control torques exerted by the cold gas reaction jets would be normal to the spacecraft spin axis, they would also apply bending moments to the booms. The booms, being flexible members, would store some of the control energy and affect spacecraft motion at some later time depending on the natural frequency of the boom. Figure VI-6 shows the relative magnitudes of the spacecraft spin rate and boom natural frequency as a function of payload acceleration.

It will be noted that at the higher "g" levels the natural frequency of the boom is significantly higher than the spin rate indicating that there should be relatively little effect on spacecraft motion. Potential problems start to develop as the boom is deployed beyond 10 ft. At this distance, the boom natural frequency is greater than the spin rate by a factor of 2. This factor decreases asymptotically as the booms are extended further until at the minimum acceleration level, the ratio of boom natural frequency to spacecraft spin rate is only 1.4. Under these conditions, further investigation was warranted to determine the effect of these boom characteristics on spacecraft motion. This investigation was based largely on the stability studies documented by Flatley[2] which are related to partially flexible spinning satellites. Results of this investigation are shown in Figure VI-7. The technique employed in developing this data is described by Flatley in his paper. Of basic interest is whether or not the proposed design solution is stable. The two solid curves representing the upper and lower stability boundaries of Figure VI-7, identified as ΔI_{max} $(p - \bullet)$ and ΔI_{min} $(p - \infty)$, respectively, define the region wherein the motion of a spacecraft, whose principal moments of inertia are all equal, would be stable. The broken curve plotted between the two solid curves is representative of the required inertia configuration in order to

[2] Flatley, T.W., "Attitude Stability of a Class of Partially Flexible Spinning Satellites," NASA TND-5268, August 1969.

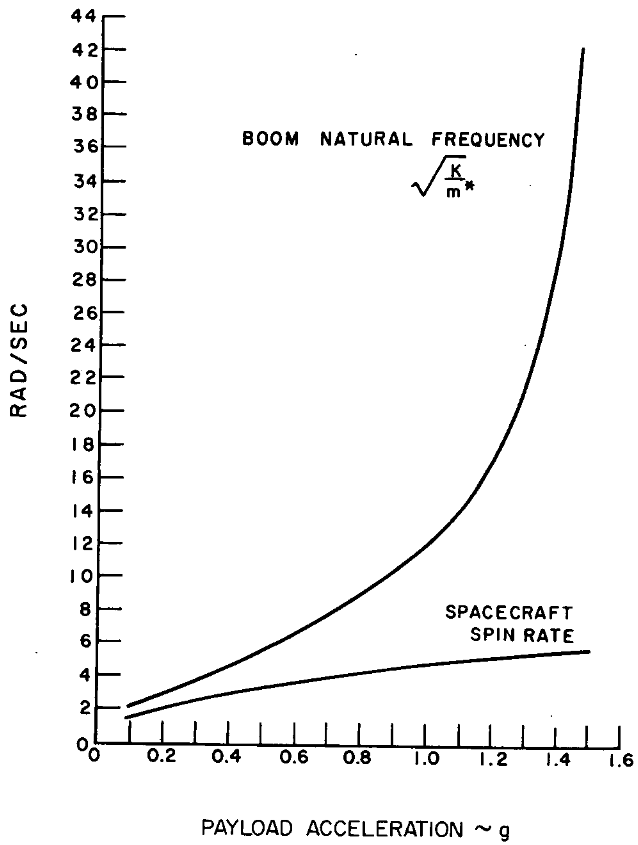


Figure VI-6. Payload Acceleration vs Rad/Sec

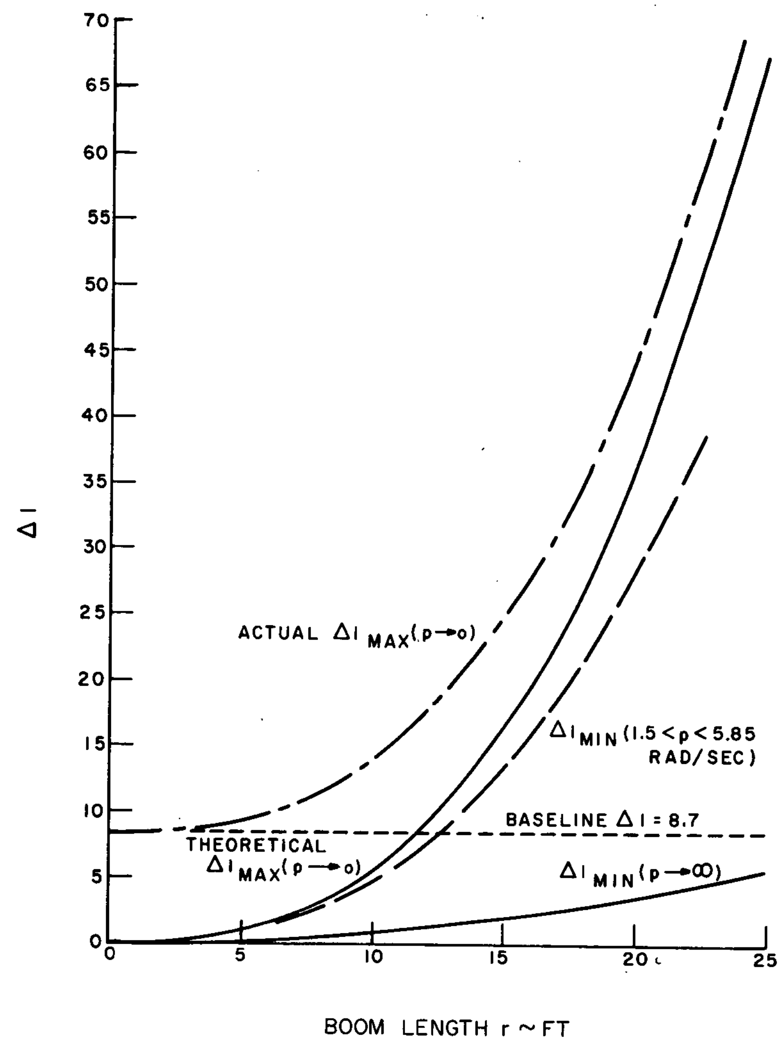


Figure VI-7. Boom Length vs ΔI

be compatible with Bioresearch Module spin rates as the booms are deployed. This performance, it will be noted, is quite close to the upper theoretical stability boundary. However, when actual baseline design inertia values are introduced it will be observed that the stability margin is increased as evidenced by the upper most curve in Figure VI-7. Since the baseline ΔI never falls below a fixed value, the spacecraft will be stable at any spin rate between 1.5 and 5.85 rad/sec.

Parameters which directly influence where within the stability region the actual performance curve will fall include, the lineal density of the boom, the spin rate, and the boom free length. Boom stiffness has an inverse influence on location of the performance curve. That is to say, for a given set of conditions if the boom stiffness were to be reduced, the performance curve would tend to move toward the lower boundary curve.

It is important that proper care be exercised in performing this particular trade-off because it effects the balance between spacecraft stability and boom stability. Suffice it to say, that the results of the current effort have successfully combined the properties of the boom, the spacecraft and the variable "g" spin rates to evolve a stable spacecraft configuration.

5.2 BODY MOUNTED ARRAY STUDIES

Two alternate spacecraft configurations were also investigated which featured a body mounted solar array. With this configuration it is possible to orient the spacecraft spin axis normal to the ecliptic plane and thus dispense with the chore of periodic attitude maneuvers to maintain sun orientation as is the case with the solar paddle configuration. Unfortunately, the moment of inertia properties of this configuration are such that spacecraft motion is only conditionally stable. That is, the spin rate required to develop the prescribed "g's" would be about the axis of minimum moment of inertia. Thus, any damping of transverse rates would have to be provided by the cold gas control system and could not be provided by deployed booms as in the case of the baseline configuration. Since the body mounted array configurations are not the primary design solutions, it was not deemed necessary to re-establish known truths which are available in any text book dealing with non-rigid body dynamics.

Applying Flatley's technique to this class of vehicles would require that a partially deployed boom configuration be found that is compliant with the established stability criterion. The booms would then be operated in the same manner as the baseline design to control payload acceleration levels. The resultant boom lengths would be substantially longer than the 22 feet specified for the baseline spacecraft.

In building stability into this class of spacecraft, the question of whether or not to separate the spent rocket motor case would have a profound influence on the design. Every indication is that it would be best to get rid of the motor case and derive the benefit of operating with the shortest booms possible.

6.0 DAMPING AND LONG TERM STABILITY

Damping is an all important characteristic in any dynamics problem. That is the case here also, because some damping is required to dissipate the energy in the booms as they are moved in and out. According to the manufacturer, there is some light damping inherent in the design of the boom which is ascribed to the overlapping configuration of the boom elements. In the studies just completed, only light damping (about 2%) was used. Evidence of this damping is shown in Figure VI-4. It appears after the boom travel has ceased and the Coriolis force is reduced to zero. At this time, the boom's steady-state equilibrium position shifts to its normal radius vector. The resultant motion shows the boom oscillating about this equilibrium point and the magnitude of these oscillations being slowly dissipated by the damping present. Since boom tip displacement is only a fraction of an inch, the problem of significant energy dissipation over a long time period is almost obviated.

A corollary consideration is one of the effect of thermal bending. This problem was treated by Etkin and Hughes[3] in their analysis of the anomalous spin behavior of the Alouette and Explorer XX satellites which experienced a decrease in spin rate of about 1 rpm/yr. Of particular significance is the fact that the plane of the flexible booms was parallel to the ecliptic; the same orientation proposed for the body mounted solar array spacecraft configuration.

The explanation presented by the authors is based on the hypothesis that the interaction of the solar radiation falling continuously on an unsymmetrical shape produces a torque on the body, which, when averaged for one revolution produces a non zero value. This effect would be of reduced magnitude in the case of the Bioresearch Module because of the amount of time spent in the earth's shadow and, would be no problem at all when considering the baseline spacecraft because the plane of the booms would be normal to the sun and all four booms would be equally illuminated. One other significant factor that has not really been exploited in these studies is the fact that there is an active control system aboard the spacecraft and sufficient sensing capability is provided to maintain the desired orientation. Consequently, when stability is discussed it must be with the knowledge that long term effects can be compensated for by expending modest amounts of stored energy. The significant findings of this study are related to the boom stability investigations which showed that care must be exercised in the selection of boom characteristics.

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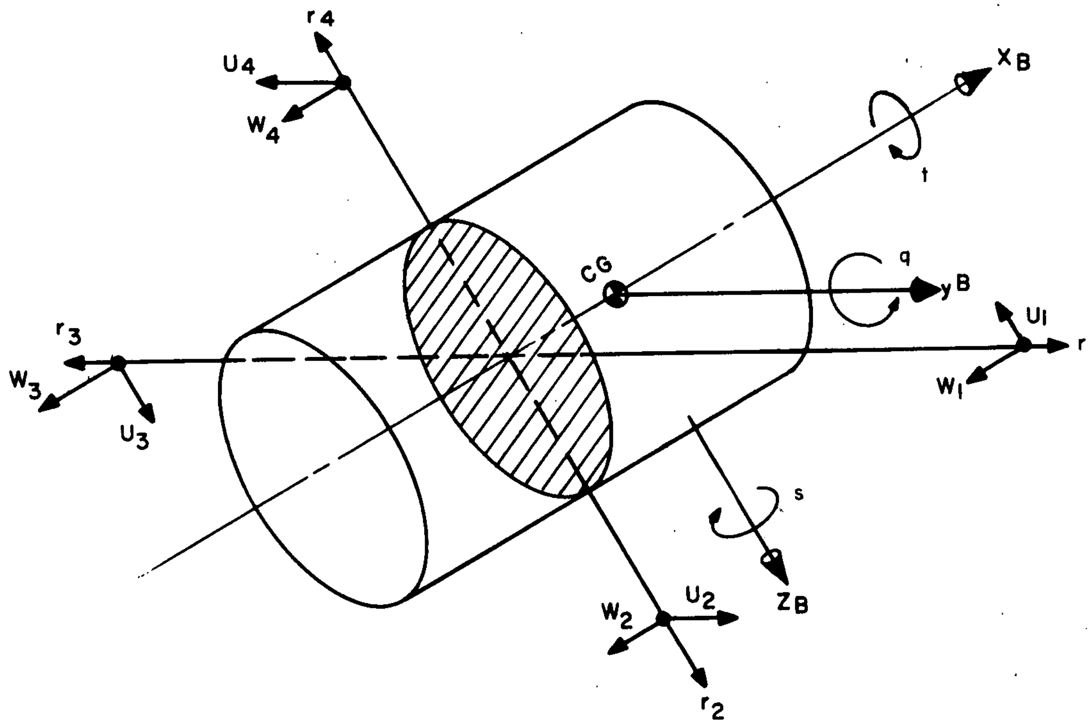
[3] Etkin, B. and Hughes, P. C., "Explanation of the Anomalous Spin behavior of Satellites with Long, Flexible Antennae, "Journal of Spacecraft, Vol. 4, No. 9 September 1967.

APPENDIX

DERIVATION OF GENERAL EQUATIONS OF MOTION

APPENDIX

DERIVATION OF GENERAL EQUATIONS OF MOTION



LIST OF TERMS & ASSUMPTIONS

p, q, s	-	Body spin rates
x_b, y_b, z_b	-	Body axes
u, w, r	-	Boom tip motion coordinates
m, m*	-	Boom tip mass and effective mass
U (u, w, r)	-	Boom strain energy equation
g (r)	-	Boom radial forcing function
I_x, I_y, I_z	-	Body moments of inertia

- Translational terms neglected
- Booms are co-planar, perpendicular to body spin axis and contain body cg
- Tip mass and boom mass can be expressed as simple equivalent mass (m^*)
- Radial boom velocity constant and equal ($\dot{r} = v_1 \; \dot{r}_1 = \dot{r}_2 = \ldots$)
- Boom strain energy can be expressed for u & w separate motion
- Boom tip displacements (u, w) are fairly small
- Body is axi-symmetric - Prod. of Inertia = 0

DYNAMICAL EQUATIONS

$$
\begin{aligned}
&\dot{p}\left[\frac{I_x}{m^*} + 4r^2 + \sum_1^4 u^2\right] \qquad (1) \\
&+ \dot{q}\left[r(w_1 - w_3) + u_2 w_2 - u_4 w_4\right] \\
&+ \dot{s}\left[r(w_2 - w_4) + u_3 w_3 - u_1 w_1\right] \\
&- r \sum_1^4 \ddot{u} \\
&+ p\left[8rv + 2\sum_1^4 u\dot{u}\right] \\
&+ q\left[r(\dot{w}_1 - \dot{w}_3) + v(w_1 - w_3) + u_2\dot{w}_2 + \dot{u}_2 w_2 - u_4\dot{w}_4 - \dot{u}_4 w_4\right] \\
&+ s\left[r(\dot{w}_2 - \dot{w}_4) + v(w_2 - w_4) - u_1\dot{w}_1 - \dot{u}_1 w_1 + u_3\dot{w}_3 + \dot{u}_3 w_3\right] = 0
\end{aligned}
$$

$$\dot{p}\left[r(w_1 - w_3) + u_2 w_2 - u_4 w_4\right] \tag{2}$$

$$+ \dot{q}\left[I_y/m^* + 2r^2 + u_1{}^2 + u_3{}^2 + \sum_1^4 w^2\right]$$

$$+ \dot{s}\left[r(u_1 - u_2 + u_3 - u_4)\right]$$

$$+ \ddot{u}_1 [-w_1] + \ddot{u}_3 [w_3]$$

$$+ \ddot{w}_1 [u_1] + \ddot{w}_2 [-r] + \ddot{w}_3 [-u_3] + \ddot{w}_4 [r]$$

$$+ p\left[r(\dot{w}_1 - \dot{w}_3) + v(w_1 - w_3) + \dot{u}_2 w_2 + u_2 \dot{w}_2 - \dot{u}_4 w_4 - u_4 \dot{w}_4\right]$$

$$+ q\left[2 \sum_1^4 w\dot{w} + 4rv + 2u_1\dot{u}_1 + 2u_3\dot{u}_3\right]$$

$$+ s\left[r(\dot{u}_1 - \dot{u}_2 + \dot{u}_3 - \dot{u}_4) + v(u_1 - u_2 + u_3 - u_4)\right] = 0$$

$$\dot{p}\left[r(w_2 - w_4) - u_1 w_1 + u_3 w_3\right] \tag{3}$$

$$+ \dot{q}\left[r(u_1 - u_2 + u_3 - u_4)\right]$$

$$+ \dot{s}\left[I_z/m^* + \sum_1^4 w^2 + 2r^2 + u_2{}^2 + u_4{}^2\right]$$

$$+ \ddot{u}_2 [-w_2] + \ddot{u}_4 [w_4]$$

$$+ \ddot{w}_1 [r] + \ddot{w}_2 [u_2] + \ddot{w}_3 [-r] + \ddot{w}_4 [-u_4]$$

$$+ p\left[r(\dot{w}_2 - \dot{w}_4) + v(w_2 - w_4) - w_1\dot{u}_1 - \dot{w}_1 u_1 + w_3\dot{u}_3 + \dot{w}_3 u_3\right]$$

$$+ q\left[r(\dot{u}_1 - \dot{u}_2 + \dot{u}_3 - \dot{u}_4) + v(u_1 - u_2 + u_3 - u_4)\right]$$

$$+ s\left[2 \sum_1^4 w\dot{w} + 4rv + 2u_2\dot{u}_2 + 2u_4\dot{u}_4\right] = 0$$

$$\dot{p}\,[-r] + \dot{q}\,[-w_1] + \ddot{u}_1\,[1] \tag{4}$$

$$+ p\,[-2v + w_1 s - u_1 t]$$

$$+ q\left[-2\dot{w}_1 - rs - u_1 q\right] + \frac{1}{m^*}\,\frac{\partial U_1}{\partial u_1} = 0$$

$$\dot{p}\,[-r] + \dot{s}[-w_2] + \ddot{u}_2[1] \tag{5}$$

$$+p[-2v - u_2 t - w_2 q]$$

$$+ s\left[-2\,w_2 + rq - u_2 s\right] + \frac{1}{m^*}\,\frac{\partial U_2}{\partial u_2} = 0$$

$$\dot{p}\,[-r] + \dot{q}[w_3] + \ddot{u}_3\,[1] \tag{6}$$

$$+ p\,[-2v - w_3 s - u_3 t]$$

$$+ q\,[2\dot{w}_3 - rs - u_3 q] + \frac{1}{m^*}\,\frac{\partial U_3}{\partial u_3} = 0$$

$$\dot{p}\,[-r] + \dot{s}[w_4] + \ddot{u}_4\,[1] \tag{7}$$

$$+ p\,[-2v + w_4 q - u_4 t]$$

$$+ s\,[2\dot{w}_4 + rq - u_4\,s] + \frac{1}{m^*}\,\frac{\partial U_4}{\partial u_4} = 0$$

$$\dot{p}\left[\sum_1^4 u\right] + \dot{q}[w_2 - w_4] + \dot{s}[-w_1 + w_3] \tag{8}$$

$$p\left[2\sum_1^4 \dot{u} - w_1 q - w_2 s + w_3 q + w_4 s - 4rt\right]$$

$$+ q\,[2\dot{w}_2 - 2\dot{w}_4 - 2rq + u_2 s + u_4 s]$$

$$+ s\,[-2\dot{w}_1 + 2\dot{w}_3 - 2rs - u_1 q - u_3 q]$$

$$+ \frac{1}{m^*} \sum_1^4 \frac{\partial U}{\partial r} = \frac{1}{m^*} \sum_1^4 g(r)$$

$$\dot{q}\,[u_1] + \dot{s}[r] + \dot{w}_1\,[1] \tag{9}$$

$$+ q\,[2\dot{u}_1 - w_1 q - rt]$$

$$+ s\,[2v - w_1 s + u_1 t] \quad + \frac{1}{m^*} \frac{\partial U_1}{\partial w_1} = 0$$

$$\dot{q}\,[-r] + \dot{s}[u_2] + \ddot{w}_2\,[1] \tag{10}$$

$$+ q\,[-2v - w_2 q - u_2 t]$$

$$+ s\,[2\dot{u}_2 - w_2 s - rt] \quad + \frac{1}{m^*} \frac{\partial U_2}{\partial w_2} = 0$$

$$\dot{q}\,[-u_3] + \dot{s}[-r] + \ddot{w}_3\,[1] \tag{11}$$

$$+ q\,[-2\dot{u}_3 - w_3 q + rt]$$

$$+ s\,[-2v - w_3 s - u_3 t] \quad + \frac{1}{m^*} \frac{\partial U_3}{\partial w_3} = 0$$

$$\dot{q}\,[r] + \dot{s}[-u_4] + \ddot{w}_4\,[1] \tag{12}$$

$$+ q\,[2v - w_4 q + u_4 t]$$

$$+ s\,[-2\dot{u}_4 - w_4 s + rt] \;\; + \frac{1}{m^*}\,\frac{\partial U_4}{\partial w_4} = 0$$

VII. SPACE SHUTTLE VEHICLE (SSV) INTEGRATION

1.0 OBJECTIVE

The purpose of this portion of the study is to investigate the impact of the availability of the SSV as a launch and retrieval vehicle on the baseline design of the Bioresearch Module. The SSV design and interface information were taken from the SSV Phase B study final reports by North American Rockwell and the McDonnell-Douglas Corporation in addition to the Orbiter/Payload Performance and Interface Requirements (NASA Controlled Document) dated April 2, 1971. The applicable portions of these reports were supplied to GE-RESD by NASA/ARC.

The primary tasks were to perform a mission analysis, define the required interfaces and then determine the baseline design changes required so that the Bioresearch Module will be a feasible and functional SSV payload. In addition, test and manufacturing changes caused by use of the SSV were evaluated.

2.0 MISSION ANALYSIS

2.1 Launch - The SSV is being designed to meet three basic mission requirements. These missions are:

ALTITUDE (NM)	INCLINATION	PAYLOAD (LB)	ΔV (FPS) REQUIRED
100	28.5°	65,000	900
270	55°	25,000	1500
100	90°	40,000	650

I

3-3

The two 100 nm (circular) missions are minimum altitude due east (from ETR) and the 270 nm altitude is the space station resupply mission. The SSV main propulsion system is being designed to put the orbiter in a 50 nm x 100 nm orbit. The additional ΔV required to achieve the orbit (as listed above) is supplied by the orbit maneuvering system (OMS).

The required 6-month lifetime in orbit for BRM can be achieved if the initial orbit altitude is 220 nm or higher for missions I and II. Accordingly, the SSV 270 nm mission presents no orbital lifetime problem (270 nm would provide a 2-year lifetime). However, if the maximum SSV altitude is only 100 nm, additional ΔV must be supplied by the BRM. It should be noted that the current SSV operational philosophy is that all payloads will be placed in the required orbit by the SSV. In fact, plans indicate that the SSV will insert a sun synchronous satellite in a 496 nm circular orbit. Table VII-1 indicates a breakdown of the traffic model used for the first ten years of shuttle operation. It can be seen that 80% of the planned missions are to 100 nm. The SSV OMS tanks are sized for 2000 fps ΔV, resulting in 1100 and 1350 fps excess ΔV for the due east and polar 100 nm missions respectively. For Missions I and II, 420 fps ΔV is required to change orbit from 100 nm circular to 220 nm circular (600 fps for 270 nm circular). However, considering the 'fly on missions of opportunity' concept suggested by NASA/ARC, and realizing that the SSV cargo bay may contain more than one payload, consideration should be given to developing a small ΔV package for Missions I and II. The concept and design will be modular so that if further SSV development precludes this requirement, there is no impact on the baseline design.

Mission III requires approximately 10,000 fps ΔV in excess of the 100 nm orbit and cannot be inserted by the SSV. Insertion from any of the planned mission altitudes appears to be feasible. Further study would be required to determine BRM launch window after release from the SSV.

Figures VII-1 and VII-2 indicate that a major portion of the SSV missions are dedicated to payload deployment.

TABLE VII-1. MISSION DISTRIBUTION BY ORBIT CHARACTERISTICS

ALT. / INCL.	100 NM	270 NM	OTHER
28.5° to 33°	59%	1%	1%
55° to 63°	6%	18%	-
90° to 100°	15%	-	-
TOTAL	80%	19%	1%

2.1.1 Mission I and II ΔV Requirements and Implementation

The ΔV required to change from a 100 x 100 nm orbit to a 220 by 220 nm orbit is 420 fps; 600 fps for 270 x 270 nm. Considering that the shuttle may also be slightly higher than 100 x 100 nm, the ΔV package capability should be approximately in the 300 to 600 fps range, and capable of being set within this range prior to launch. To achieve this orbit, the ΔV must be applied in two equal increments, 1/2 orbit apart. The thrust must be applied in the direction of the velocity vector.

The most promising method of doing this is to use two solid rocket motors, one ignited by shuttle command, the second ignited 45 minutes later by a timer. This also requires attitude control to align the thrust vector.

Assuming a 400 lb. BRM, the 300 to 600 fps ΔV translates to 1860 to 3720 lb. sec. for each of the two rockets. Preliminary investigation shows that Thiokol makes two rocket motors in this category:

Engine	TE-M-542	TE-M-541
Capacity	2,000 lb. sec.	3,000 lb. sec.
Weight	10.2 lb	13.2 lb
Size	-- 6 1/4" dia x 9" long--	
Thrust	-- 660 lb. max. -------	
Qualified	-- Yes ---------------	
Off/over load	-- Yes ---------------	
Approximate cost	-- $6K to $7K ----------	

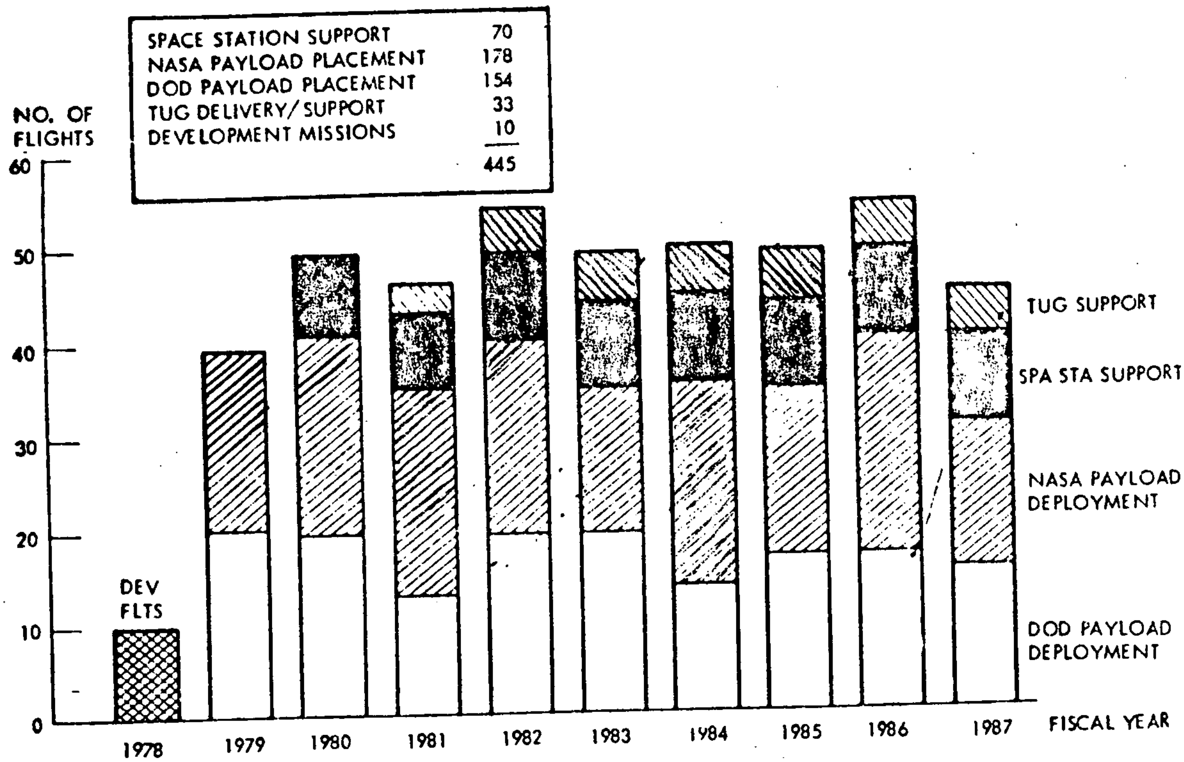


Figure VII-1. Model Missions

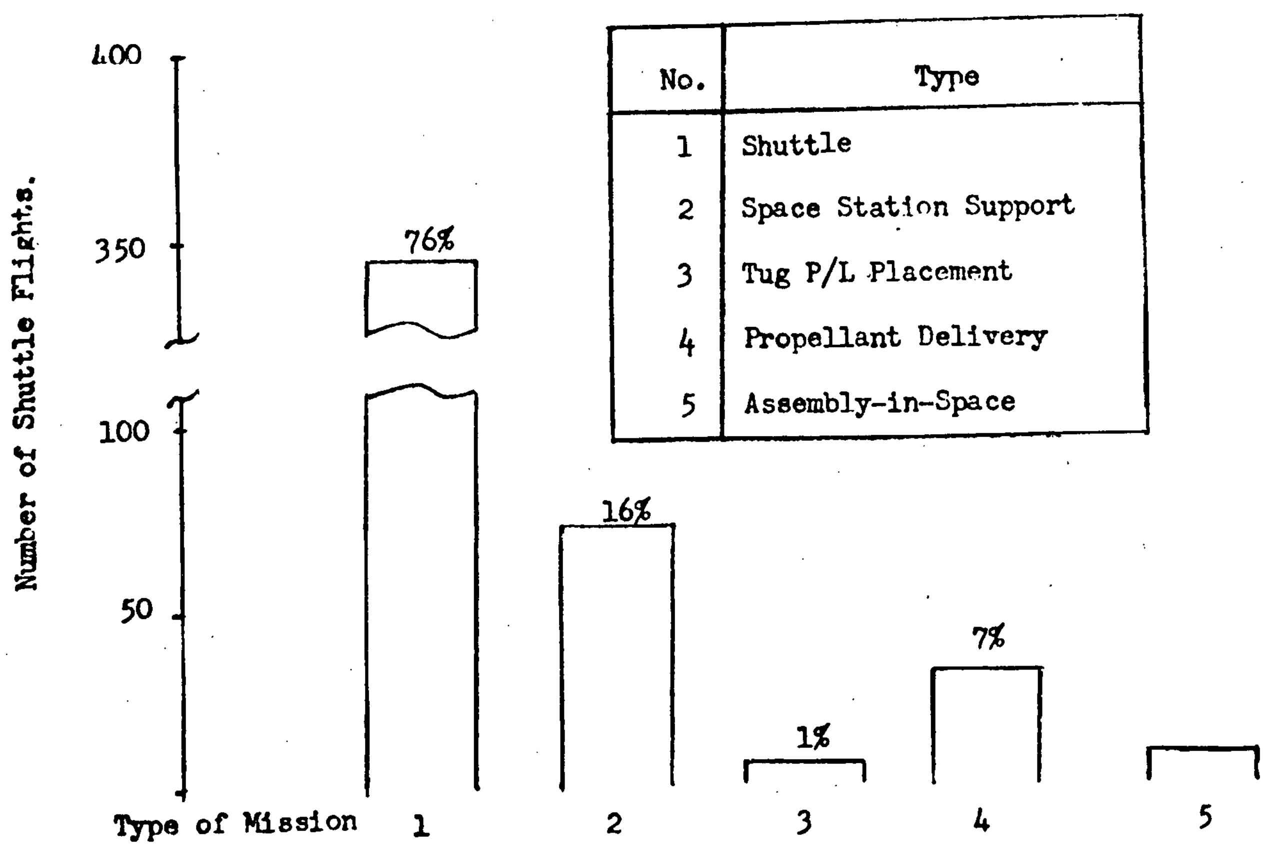


Figure VII-2. Mission Distribution

Thiokol also has 5,000 and 7,000 lb.-sec. units in the event that the BRM weight is increased.

Another method requiring only spin up and no maintenance of thrust vector parallel to velocity vector was investigated. This system proved not to be feasible. This maneuver definitely requires alignment of the thrust vector parallel to the velocity vector. Another possible implementation is to use the frame and guidance of an existing velocity package, such as Burner II, and attach the two small rocket engines to it.

2.1.2 Mission III ΔV Requirements and Implementation

For Mission Type III, an additional 10,000 fps (approximately) is required from any of the shuttle missions. This requires the use of one of the currently available velocity packages. Considering that the study may take advantage of the essential lack of weight restriction on the shuttle (e.g. to double up on experiment packages) performance characteristics are shown for spacecraft weights of 400, 800 and 1200 lb.

The chart below lists some of the planned and operational velocity packages along with performance characteristics.

Velocity Package	ΔV to 400 lb. Spacecraft	ΔV to 800 lb. Spacecraft	ΔV to 1200 lb. Spacecraft
OV1 (FW4) Operational	6100 fps	-	-
TE364 (1440) (Operational)	12,000 fps	8,800 fps	-
BII (1440) (Operational)	10,000 fps	-	-
BII (2300) (Proposed)	12,800 fps	10,000 fps	-
FIRE/FW4 (Proposed)	17,000 fps	13,000 fps	10,600 fps

The TE364 (1440) and the FIRE/FW4 are spin stabilized and must be considered in the payload design. The FIRE/FW4 velocity package is the only proposed or operational system that can provide 10,000 fps ΔV to a 1200 lb payload. The ΔV for Mission III can be applied in one increment, with no requirement for shutdown and restart of the engine. For all shuttle launched missions, initial attitude will be obtained from the shuttle deployment mechanism. Considering the operational velocity packages that can provide the required ΔV, an orbit error analysis was performed to assist in velocity package selection. The results are listed in Table VII-2.

TABLE VII-2. MISSION III ORBIT ERROR ANALYSIS

STABILIZATION	VERNIER ΔV	ΔV ERROR (FPS)	Δ ALTITUDE (NM)	APPROXIMATE COST ($)
SPIN, TABLE (BII)	YES	± 88	+ 44K - 26K	400K
SPIN, TABLE (BII or TE-364)	NO	± 110	+ 55K - 33K	350K
FREE SPIN (BII)	YES	± 200	+150K - 50K	400K
GUIDED (BII)	YES	± 6	+ 3K - 2K	550K
GUIDED (BII)	NO	± 74	+ 37K - 22K	500K

TE 364 has no vernier ΔV capability and no inertial guidance. BII has both vernier ΔV and inertial guidance. Table VII-2 indicates that a spin stabilized velocity package results in apogee errors of 44,000 and 55,000 nm (with and without vernier ΔV respectively). Using the approximation of 1.5 hr. period deviation for each thousand nm, this results in periods of 210 to 230 hr. These would probably be unacceptable for the experiments. However, it would also seem that the orbit stability would be a real problem with apogee out at 200,000 nm. A detailed analysis was not undertaken during this study. Because of these anticipated problems, it is recommended that a Burner II be used for the Mission III velocity package. In addition, the B II errors without vernier ΔV are also large, and the cost difference small. The B II used should include both inertial guidance and vernier control of the ΔV.

2.2 Rendezvous and Recovery

The NASA Controlled Interface Requirements Document (NCIRD) specifically states that all rendezvous maneuvers will be performed by the SSV. The only requirements on the BRM are that it carry an S-band transponder to allow the SSV to calculate the required orbit change maneuvers and that the BRM despin for recovery. The maximum rates have yet to be defined.

The maximum ΔV that the SSV would require for altitude change is 600 fps (100 nm to 270 nm). The ΔV required for phase change is time dependent. As an example, if the BRM and SSV are both at 270 nm, and 55° inclination, the time to rendezvous along with ΔV expended is:

Time to Rendezvous (days)	V (fps) catch up plus recircularize
1	450
2	250
3	175
4	145
5	120
6	100
7	95

The phase change and altitude maneuvers can be combined so that the total ΔV is not the sum of that required for each maneuver. If the BRM is at 270 nm (period = 5658 seconds) and the SSV at 100 nm (period = 5274 seconds), a 180° phase difference goes to 0° in 7 hours. In any event, the ΔV required is well within the SSV capability.

2.3 Station Coverage

Station coverage was examined for the three SSV design orbit inclinations along with the planned Scout launch inclination of 38°. Both STADAN and MSFN were considered. The maximum number of orbits between station contacts are:

Inclination	Orbits Between STADAN Contacts	Orbits Between MSFN Contacts
28°	2	2
38°	1.5	2
55°	1.5	2
90°	2.3	2

It can be seen that even at 90°, the time between contacts is not excessive. As a result of this and the excessive ΔV penalty (500 fps per degree), no plane change maneuvers are required.

3.0 INTERFACE DEFINITION

The SSV-payload interfaces at present are not completely defined. In addition, at this stage of SSV design and development, it is also recognized that the defined interfaces are subject to change. The conceptual design approach for the BRM is to minimize the effect on the BRM design caused by changes in the SSV interface by maintaining the Scout launched baseline design and by interfacing with the SSV through buffer equipment. As a result, the baseline BRM design will also fly on the SSV, with maximum flexibility.

In general, the SSV payloads will be installed in the VAB with the orbiter in a horizontal position. The concept is to complete payload installation 72 hours prior to launch. Pad access is planned, but current designs indicate that the cargo bay door will not be opened on the pad. Access is planned through two or three small (approximately 20 by 20 inches) doors. Unless plans are changed, the experiment will have to be inserted at about T-80 hours. The effects of this early insertion on the experiments must be evaluated.

3.1 Structural interfaces

The SSV cargo bay dimensions are planned to be 15 ft. in diameter by 60 ft. long. There is very little detail on mounting payloads in the cargo bay. However, the NCIRD specifies that all payloads will be pallet mounted. In the two phase B studies, North American Rockwell (NR) proposes use of a boom manipulator (Figure VII-3.) for mounting and extraction, with no pallet; McDonnel Douglas (MD) uses a deployable rack (Figure VII-4) with a pallet mount. In accordance with direction received from NASA/ARC, this study considers only the boom manipulator. The separation from and post recovery re-attachment to the SSV is to be done automatically, with no reliance on intra-vehicular activity (IVA). All deployable appendages must be stowed during launch and prior to recovery. As a result of this requirement, a design change to the baseline BRM is required. Either a retraction mechanism for the solar panels or replacement of the panels with a fixed ring. Figure VII-5 indicates the concept of the fixed solar array. As can be seen, the overall BRM diameter is reduced by about 3 ft.

3.2 Power Interfaces

The SSV will provide power to the payloads from payload installation up to release from the cargo bay. Connections will be by junction boxes. The NCIRD, NR and MD studies differ on the type and amount of power to be supplied. In summary the values are:

	NCIRD	NR	MD
TOTAL (kwhr)	50	20	20
AVG (watts)	1,000	500	500
PEAK (watts)	1,500	800	800
Voltage	NOT DEFINED	*28 VDC ± (TBD)	120 VDC ± 10%

*LOWER THAN BRM BUS

If power is required post recovery, reconnection must be automatic. Further definition is required, not only of total power available to the cargo bay, but also of the power specifically available to the BRM.

3.3 Commands and Data Interfaces

The SSV will have command capability by hardline connection to the cargo bay and by RF to the payload, after payload deployment. Data transmission capability will be provided. The NCIRD and the two studies indicate that from 20 to 50 kbps data downlink bandwidth will be allotted to the payloads in the cargo bay. Further definition is still required.

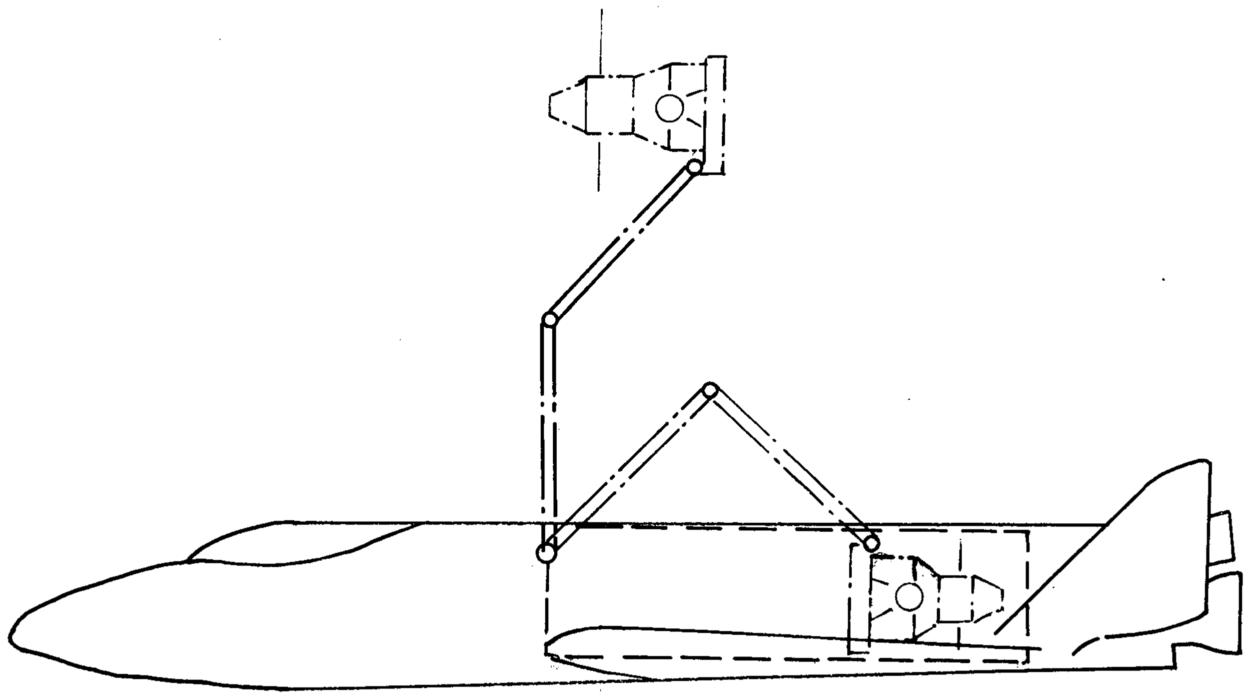


Figure VII-3. Boom Manipulator

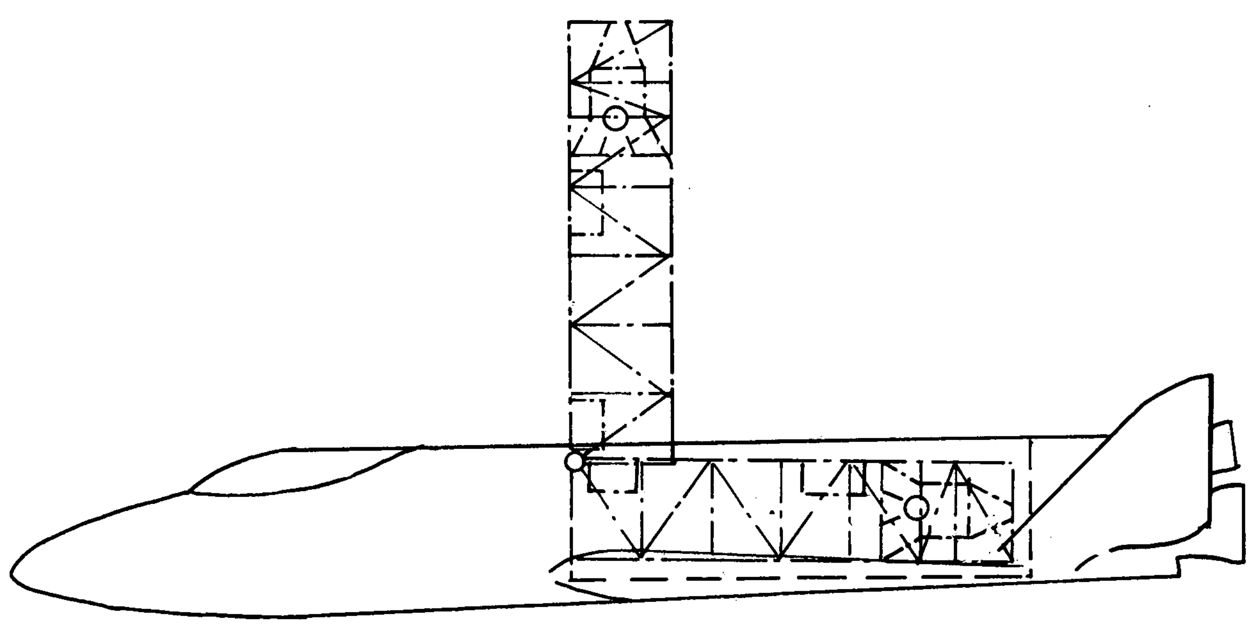


Figure VII-4. Rack Deployment

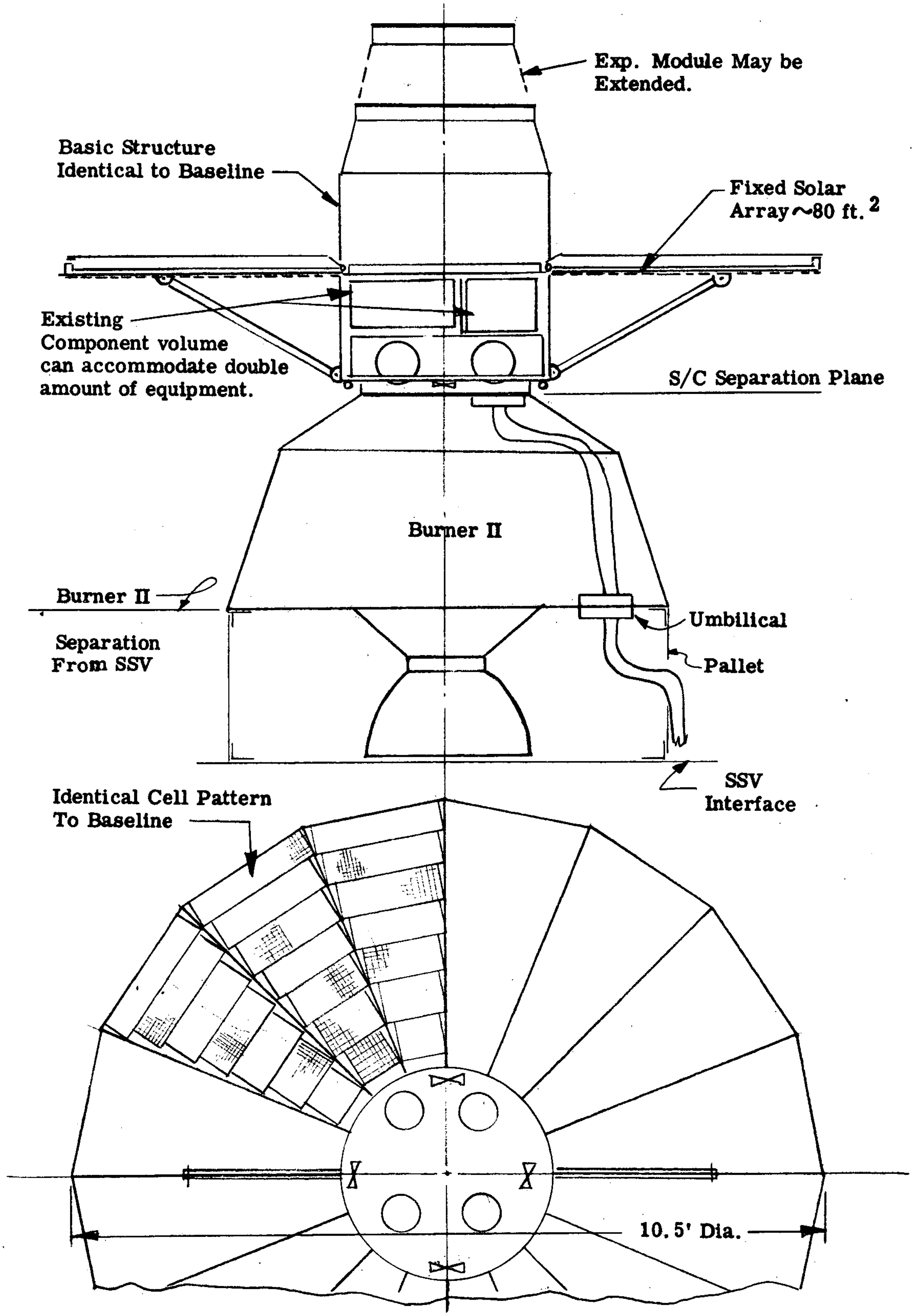


Figure VII-5. Fixed Solar Panel

The SSV will use USB and the BRM requires a transponder for recovery. The shuttle will have capability to monitor payload data. It will provide go-no-go console monitors and/or computer analysis of the payload data. Again, this interface is not fully determined.

3.4 ENVIRONMENTS

The vibration and steady state load factors are much less severe than the Scout requirements. A summary of the requirements defined at this time in the NCIRD and NR study is:

- Steady-State Load Factors, NASA/NR, (g)

	X	Y	Z
Launch	*3/3.3	± 1.0	± 1.0
Entry	.25/.5	± .5/1.0	- 2.0
Flyback	.25/.5	± .5/1.0	-2.5 to 1.0
Landing	-1.0 to .8/1.3	±.5	-2.5/2.7
Emergency Landing	-10	+1.5	-5

*ENGINES THROTTLED

- Payload Bay Thermal Environment, NASA/NR, (°F)

	MIN.	MAX.
Prelaunch	-100/300*	+ 120
Launch	-100/300*	+ 150
On-Orbit-Door Closed	-100/300*	+ 150
-Door Open	TBD	TBD
Entry and Post Landing	-100/300*	+ 150/350

*NASA/ARC has directed that the -100°F value be used.

- NR also defines:
  - Max Vibration — $2 \times 10^{-2}\ G^2/Hz$
  - Depressurization — 90% in 80 Seconds
  - Repressurization — 90% in 1400 Seconds

3.5 THERMAL INTERFACE

The SSV will provide a coolant loop to interface with the payload support equipment (not the payload itself). This coolant will be supplied from payload installation to payload deployment using ground cooling prior to launch, cryogenic H_2 during launch, and a radiator in orbit. The only definition of the coolant is that the coolant capacity will be equal to the power supplied. At this time, no information on type of coolant, temperature or flow rate is available. It can also be inferred that additional cooling may be available by fluid venting because the NCIRD states that overboard vents will be made available to the payloads.

4.0 DESIGN CONCEPTS AND EFFECTS ON BASELINE DESIGN

To determine the feasibility of using the SSV to launch and recover BRM missions, four tasks were undertaken. The first was essentially mapping out a sequence of events to determine the functions associated with an SSV launch and from them to determine if any design changes are required. The second task was to come up with approaches for providing power to the BRM and maintaining the command and telemetry function. Because of the interface uncertainty at this time, a number of approaches are suggested, final selection of which will be dependent on the interface definition. However, due to the diversity of these approaches, one or more will be applicable to the final SSV definition. The third task was to develop methods for maintenance of the cold plate temperature while in the cargo bay. As with power and telemetry and command, several approaches are suggested. Finally, a set of configuration concepts were developed. In each of the subsystem tasks, two concept extremes were examined, along with others exhibiting a mix of these extremes. One extreme is for minimum shuttle interface - no hardware connections, no coolant connections. An adapter service module will provide all needed support for launch and recovery. The other extreme was to minimize the support equipment flying with the BRM-hardware connections to the SSV with a buffer console for an electrical interface and to provide coolant to the BRM.

All approaches were developed to have minimum effect on the baseline design. The adapter/service module will carry additional equipment and for automatic mechanical, electrical and coolant connections.

Many of the proposed approaches are also applicable to the SSV sortie missions. As an example, the adapter designed to meet the minimum interface concept could easily be modified to support the BRM for 30 days in the cargo bay. In addition, due to the modular nature of the BRM design, the sortie BRM power system would not include solar panels (obviously not required), the prime power would be supplied by batteries in the adapter, and no attitude control module would be included. The data, command and thermal control functions would be as designed for SSV launch and recovery.

4.1 SEQUENCE OF EVENTS

This sequence of events assumes command control of the BRM while in the cargo bay. The functions listed are all those that could be required. For example, it may not be required to go to a BRM low power mode between SSV orbit insertion and BRM deployment, or buffer recorders may not be used, but the functions are still listed in the sequence.

4.1.1 ΔV Package on Missions I and II

If the SSV deploys the BRM at an altitude below that required for the specified mission life, a ΔV package (described previously) will be included with the BRM. This will require an RTC to ignite the first rocket after separation from the SSV; a timer to ignite the second rocket approximately 45 minutes later, and to initiate sun orientation, activate the momentum wheel or spin up as required.

4.1.2 Mission III Sequence

For Mission III, after separation of the BRM from the SSV an RTC is required for Burner II ignition. Burner II is capable of being programmed for sun search and spin up and will be used to perform these functions. Burner II will provide the signal for solar panel deployment and will initiate separation.

4.1.3 Summary

As is evident from the proceeding sequences of events, no major changes to the BRM baseline design are required to accomplish Missions I, II and III using the SSV as a launch vehicle. Several new commands are needed and the transponder and sequencer will be 'add-on' equipment if required. The other change of any magnitude is the re-design of the solar panel attachments to allow retraction if required or the possible replacement of the panels with a fixed ring as discussed previously.

4.2 TELEMETRY, TRACKING, COMMAND AND POWER

At this point, the SSV electrical interface definition is not detailed well enough to determine a unique BRM design that will be compatible with the SSV. A number of approaches were examined that encompass both the minimum SSV interface concept and the concept of minimum adapter/service module capability that will fly with the BRM. These approaches are summarized in Table VII-3 and discussed in more detail below.

SEQUENCE OF EVENTS FOR MISSION I AND II

Approximate Time	Event	Source	Notes/Effect on Baseline
T-80:00	Install BRM in Orbiter		Orbiter horizontal, all interfaces connected
T-74:00	Power On	Ground Crew	
	Switch to internal power	RTC	No change to baseline
	Interface verification	RTC	No change to baseline
			Verification by ground and/or shuttle. Parameters: BRM voltages Shuttle to buffer voltages & current Buffer to BRM voltages & current Continuity Loops Temperatures Gyro and Momentum wheel operation N_2 and other tank pressures Valve positions On-off indicators Experiment clock Mission time Command accept Experiment parameters
T-72:00	Complete verification and confidence check, close cargo bay door, commence orbiter rollout.		Assume no further physical access, determine impact on experiment.
T-50:00	Oribter/booster mate		BRM parameters can be monitored.
T-00:10	Configure BRM for launch: Gyros on, Momentum wheel on, Activate data storage, Telemetry on, timer bypass	RTC's	No change to baseline
	Buffer recorders on	RTC or hardline	Command line to buffer
T=00:00	Launch		
T+00:08	Insertion		50 x 100 nm orbit
T+00:28	Open cargo bay door, Deploy orbiter radiator		
T+00:51	First Hohmann burn		50 x 220 nm orbit (210 fps ΔV over 100 x 100 nm orbit)
T+01:38	Second Hohmann burn		220 x 220 nm orbit (210 fps ΔV)
T+01:40	Prepare for post insertion coast;		
	Gyros off	RTC	No change to baseline
	Momentum wheel off	RTC	No change to baseline
	Telemetry off	RTC	No change to baseline
	Unlock thermal shutters	RTC	No change to baseline
	Activate refrigerator evaporator/heaters	RTC or hardline	Command line to buffer

I

MISSION I AND II (Continued)

Approximate Time	Event	Source	Notes/Effect on Baseline
T+(TBD)	Pre-release preparation:		
	Switch to internal power	RTC	No change to baseline
	Telemetry on	RTC	No change to baseline
	Remove BRM	Hardline	Mechanical, electrical and coolant disconnects.
	Orient BRM		
	Activate BRM:		
	Gyros on	RTC	No change to baseline
	Momentum wheel on	RTC	No change to baseline
	Disable evaporator/heater	RTC	New command
	Deploy solar panels (Release BRM)	RTC or sequencer	New command, possible addition of sequencer
	Attitude control on	RTC or sequencer	No change
	Spin up (Mission II)	RTC or sequencer or spin table	New command
T+6 months	Rendezvous		
	BRM transponder on	RTC	New command
	Retract solar panels (if required)	RTC	New command and function
	Despin (Mission II)	RTC	New command (requires boom retraction and lock)
	Dock with manipulator		
	Attitude control off	RTC	No change
	Connect and lock BRM to SSV	Hardline	Mechanical, electrical, coolant
	Switch to external power	RTC	No change
	Configure for post recovery coast.		Same as post insertion coast
	Configure for re-entry and landing		Same as launch

The following options represent the most practical options available to provide compatibility of the Bioresearch Module Electrical System with a space shuttle launch and recovery capability. These options provide minimum or no impact on either the Baseline Bioresearch Module design or the present space shuttle/payload interface concepts.

TABLE VII-3. SUMMARY OF APPROACHES

	RF Link	Command	Telemetry	Tracking	Data Display	Power
(1)	None	Direct xmission to BRM	Direct to ground from BRM	Shuttle radar or visual	None	Shuttle power
(2)	Payload antenna on STS, BRM radiates in cargo bay to parasitic antenna	Tone sequential thru shuttle transponder and cable to BRM decoder or to console	Xmit thru shuttle transponder	S-band xponder in BRM	Safety only thru shuttle payload display panel	BRM furnished battery supply
(3)	Payload antenna on STS, coax to BRM	PCM thru shuttle transponder - cable to buffer console - discretes or tones to BRM	Store all data-playback later		As above with panel in crew compartment	Combination of 1) and 2)
(4)	Payload antenna on STS, hardwire from BRM to console, coax from console to antenna	Manual discretes from payload crewman	Shuttle computer processing - computer readout via telemetry or voice		As above with non safety data in cargo compartment and TV scan	
(5)	Any of above with coupling off shuttle main antenna	Programmed from console or adapter	Data read by payload crewman from display console			
(6)	Complete shuttle RF capability					

4.2.1 RF Link

(1) No Real Time Command or Telemetry Data Link with BRM in Payload Compartment

This option represents the highest risk and the least flexible configuration but eliminates, to the greatest extent possible, any operational or electrical interface with the space shuttle. The BRM is not checked out until withdrawn from the payload compartment by the cargo arm. The check-out and the pre-release sequence is carried out by direct ground station to BRM data link exactly as in the orbital mode. Release from the orbiter is carried out after ground station approval. If experiment or spacecraft data is needed from the launch phase, a tape recorder in the adapter section with readout during the pre-release or orbital phase could be employed. The same recorder could also be utilized during the recovery phase. This option would not necessarily be constrained by the BRM design but experiment requirements could make this option unsatisfactory. Disadvantage of this option involves the inability to monitor possible hazardous conditions associated with the BRM while in the cargo compartment. (Requirement in the interface concept study).

(2) Payload Dedicated Antenna Mounted on Shuttle, Parasitic Antenna Radiating In Cargo Compartment to BRM

This option has the advantage over (1) in that checkout and experiment monitoring can take place in real time while the BRM is stowed in the cargo compartment. No provisions in the present shuttle/payload interface concept allow for a payload dedicated antenna. This option allows checkout of the BRM without depending on equipment utilized by the shuttle in its normal mission functions or equipment shared by other payloads. This option, as does (1), allows communications without the necessity for electrical disconnect/connect capability. Problems may result from radiating free energy (5W telemetry output) in the cargo compartment as far as safety or EMI is concerned which, if allowed, would probably require additional analysis, compatibility testing or restricted usage due to RF quiet period requirements. Shielding around the BRM except in the path to the parasitic antenna probe would improve the desirability of this option. A cooling blanket, if needed for thermal control, could be designed to be RF absorbant. Either an RF probe to pick-up BRM telemetry or crew contact with the ground stations will be necessary to monitor BRM safety related parameters in the shuttle.

(3) Payload Dedicated Antenna Mounted on Shuttle with Coax to BRM

This eliminates the undesirable radiating features of (2), but requires a physical connection to the BRM. A single coax line however, is much easier to connect by remote means than one or more multipin connectors. If communications after recovery is not a prime consideration, this option has no significant advantage.

(4) Same as (3) but Coax to Checkout Console with Hardwire to BRM

This option provides the greatest flexibility with no dependence on the space shuttle electrical system. In addition to preserving the direct RF link between ground station and BRM, this method also allows functioning and monitoring of the checkout console from the ground. This capability would add to the complexity of the checkout console and would require a means of reconnecting the console to BRM cables if operation after recovery is required.

(5) Any of Above with Coupling Off the Shuttle Main Antenna

This option has the advantages and disadvantages of the previous corresponding options except that it requires dependence on the Shuttle antenna and multicoupler. This would presumably be no problem as far as dedication is concerned if the multicoupler output to the BRM operates at a carrier frequency different from the shuttle frequencies. No provisions of this nature are presented in the study reports. Although allowance is made for antenna sharing in the interface concept studies, provisions for dedication to a particular payload are not.

(6) Use Complete Shuttle RF Capability

This option depends on the utilization of the shuttle unified S-band system thus resulting in maximum dependence on the shuttle for RF capability. A separate unified S-Band system dedicated to the payloads, not now planned, would help alleviate the problem.

4.2.2 Command

(1) Transmission Directly to BRM

Transmitting tone sequential commands directly to the BRM minimizes the electrical interface with the shuttle and is incompatible with planned shuttle decoding equipment. This makes complete command verification by the crew impossible.

(2) XMIT Tone Sequential Through Shuttle Transponder

- Cable to BRM
- Cable to Buffer Console

The tone sequential commands (BRM baseline command system on shuttle opposed to the PCM unified S-Band command system utilized by the shuttle) can be modulated directly on one of the unified S-Band subcarriers. Direct modulation of the carrier is also very likely an acceptable mode. In any case, the detected tones can then be hardwired to the BRM decoders or can be used to initiate functions through the checkout console. The disadvantage of this option is that the crew will not be capable of directly monitoring or generating the command inputs unless buffering equipment is supplied.

(3) XMIT PCM Code Through Shuttle Transponder

- Cable to Buffer Console
- Discretes to Buffer Console from Computer

The disadvantage of (2) is overcome if unified S-Band PCM command codes are utilized. However, conversion to the tone sequential codes or to console discretes would have to be accomplished in the checkout console. Particular discretes could be generated by the shuttle computer from the PCM codes but this would result in a more complex interface. This approach could enable the payload crewman to initiate PCM commands himself assuming that the shuttle will be provided with this equipment.

(4) Manual Discretes from Payload Crewman

Manual discretes, based on instructions from the ground or from a BRM prelaunch operations manual, could be carried out by the payload crewman either with a payload furnished switch panel or through IVA operation of the checkout console. IVA operations are the least desirable due to safety and operational complexity considerations.

(5) Programmed Commands

Programmed commands could be generated by a sequence generater, housed either in the console or in the BRM adapter, and initiated by a timer. This option provides no flexibility for contingencies or for experiment control but does minimize dependency on shuttle.

4.2.3 Telemetry

(1) Transmission Direct to Ground from BRM

Transmitting telemetry directly to the ground station minimizes the electrical system interface with the shuttle and imposes no data or bandwidth restrictions.

(2) Transmit through Shuttle Transponder

Both the narrow (PCM-housekeeping and low rate experiment data) and wide band data could be transmitted by conventional unified S-Band (i.e., wide band on carrier simultaneous with narrow band on sub-carrier). This option again depends on availability of shuttle capability. Transmitting only wide band or only narrow band would probably affect the allowable transmission time. Present restrictions of 25 kbps down link data (to be shared by all payloads) will not permit transmission of wide band data without slowing it down.

(3) Store All Data for Later Playback

Intervals of data could be stored on tape for playback. Launch data could be played back after BRM release and recovery data played back on the ground. This option is particularly attractive with option 4.2.1. (1) with launch data playback occuring during check-out, prior to release. The tape recorder would be located in the adapter unless provisions could be made to utilize a recorder that is needed for orbital experiment data storage.

(4) Shuttle Computer Processing with Computer Readout via Telemetry or Voice

The shuttle computer could be used for telemetry limit checking and/or for data conversion. The payload crewman could read the data back to the ground station. Although eliminating dependency on the unified S-Band system for direct read-out, this option may have greater time restrictions associated with it in terms of computer availability.

(5) Data Read-out by Payload Crewman from Payload Furnished Display Panel

The telemetry data could be processed by the checkout console with critical out-of-limit parameters displayed to the payload crewman in a manner specified in 4.2.5-(3).

4.2.4 Tracking

(1) Utilize Shuttle Radar or Visual Tracking

This option would require no additional hardware. The present S-Band baseline design contains no provisions for active tracking other than on the telemetry carrier. Telemetry carrier tracking is not adequate for BRM tracking of the shuttle, particularly not for determining closing rates. This option is not presently feasible due to the interface requirement for a PRN tracking capability on payloads to be recovered.

(2) Provide S-Band Transponder in BRM

All present requirements can be met utilizing a PRN transponder located in the adapter section with its own individual antenna.

4.2.5 Data Display

(1) None

Having no display would require buyoff that there are no conditions which might prove hazardous to the shuttle mission due to malfunctions of the BRM. Present requirements are that these parameters must be constantly available for display to the crew on an shuttle warning display panel.

(2) Safety Related Data only Through Shuttle Payload Display Panel

Providing only parameters for display which could indicate hazardous conditions results in considerably more complexity than (1), but relatively little less complexity than (3) or (4).

(3) Same as (2) but with Display Panel in Crew Compartment

The non-safety related data display is to be payload furnished. At the present time only data of the go/no-go type can be considered for crew involvement according to the study reports.

(4) Same as (3) but with the Non-Safety Related Data Display in the Payload Compartment

Readout of non-safety related data display in the cargo compartment can be accomplished by remotely controlled TV scanning the checkout console display or through IVA. TV in the payload compartment is not now planned but is feasible for future consideration. Planned IVA is not presently considered for routine payload checkout.

4.2.6 Power

(1) Shuttle Supplied Power

The power to be supplied by shuttle, according to the interface study results, amounts to 20 kwh, 500 w. average and 800 w. peak, to be shared by the various payloads and their associated equipment. Although the BRM can be run in a low power mode (with the possible exception of the experiment module), some shuttle launch and recovery mission opportunities would be lost due to other payload power priorities. One study indicated payload power could be supplied only during shuttle orbit. The power would have to be furnished by the BRM battery during the remaining time.

(2) Payload Furnished Battery Supply

A battery could be supplied to power the BRM, checkout console and cooling system. The battery could be housed either in the console or in the adapter section or both. The principal portion of this power would be most likely be required by the cooling system and might likely make this approach impractical.

(3) Combination of (1) and (2)

The two most practical implementions of this option are the entire load shared between the two power sources by switching or diode blocking or the use of shuttle power to supply power only to the console and coolant system with all BRM power supplied by a battery located in the adapter section. This alternative is attractive from an EMI standpoint since the BRM power subsystem is isolated from the shuttle, and the other payloads and their support equipment. The latter minimizes direct electrical connection between the shuttle and BRM but could require an additional smaller battery in the console if shuttle power is available only during the orbital phase.

4.2.7 Configuration Selection

Three configurations have been chosen and their implementation is described below. The first configuration represents the simplest, and therefore, less costly, interface with the shuttle. There is no direct electrical connection between the BRM and the shuttle necessarily either during launch or after recovery. This design will minimize such costly factors as interface documentation and meetings, compatibility testing, crew training, designing to more stringent man-rated requirements (e.g. EMI safety, etc.) and payload prelaunch checkout. Although an adapter section is necessary with this configuration the elimination of the checkout console should result in an overall cost saving and, in addition, this adapter design can provide augumented orbital capability. The disadvantage of this concept is less flexibility and contingency capability.

The second configuration represents a more costly approach but has the advantage of being more compliant with present shuttle requirements and offers greater flexibility and thoroughness for BRM checkout and monitoring and also allows more complex pre-release and recovery experiment operations. This configuration requires a check-out console but no adapter section (transponder would be mounted within BRM if needed).

The third configuration represents complete independence from the SSV systems while affording complete command control of the BRM by the ground controllers along with complete telemetry contact. All BRM support equipment would be located in the adapter. All advantages of the second configuration also apply to this configuration. However, it is yet to be determined if the parasitic antenna concept will be permitted in the SSV.

The basic design of the three configurations will change as shuttle requirements evolve but the philosophy indicated above will probably require continuing consideration of three design approaches until late in the program concept phase.

4.2.8 Implementation

4.2.8.1 Configuration No. 1 — Configuration No. 1 includes options 4.2.1-(1), 4.2.2-(1), 4.2.3-(1) 4.2.4-(2), 4.2.5-(1) and 4.2.6-(2). (BRM on battery only). The only significant problem with this configuration, as far as present shuttle requirements are concerned, involves safety related parameter monitoring, which will probably not be a factor unless the delta V pack is included. There is no shuttle checkout console in this configuration. The Scout console, with added capability for the adapter section, will be used for checkout prior to last access to the payload. The adapter section contains a sequencer for programmed experiment operation with the BRM stowed, a battery for BRM operation during launch and after recovery, and solar cells, a battery charge regulator to recharge this battery in orbit and the required cooling equipment. A

Beacon transponder with an independent stub antenna for those BRM's that are to be recovered is also included. The adapter battery could be used for contingency orbital operations and for providing BRM orbital power until first sun, thus minimizing the depth of discharge on the BRM battery and, therefore, eliminating any launch window constraints imposed by the time to first sun. A tape recorder could be included in the adapter section, for playback through telemetry or when recovered, if the experimenters require data during the stowed phase.

4.2.8.2 Configuration No. 2 — Configuration No. 2 includes options 4.2.1-(6) (with coax to the checkout console and hardwire to the BRM), 4.2.2-(2), 4.2.3-(2), 4.2.4-(2), 4.2.5-(2) and 4.2.6-(3) (with load sharing). This configuration represents an optimum tradeoff between economy, flexibility, and thoroughness of pre-release checkout. This configuration requires dependence on shuttle operations and requires a test console but no support equipment in the adapter section. The checkout console consists of the console utilized for the scout launches with an equipment rack for shuttle peculiar BRM checkout facilities.

4.2.8.3 Configuration No. 3 — This configuration includes options 4.2.1-(2), 4.2.2-(1), 4.2.3-(1), 4.2.4-(2), 4.2.5-(1) and 4.2.6-(2). The adapter section would carry a sequencer (if needed), a battery for BRM operation during launch and after recovery, solar cells and a battery charge regulator to charge up the battery during the orbital phase of the mission, the required cooling equipment, and the S-band transponder and antenna as required. This option configuration provides minimum risk and maximum control while the BRM is in the cargo bay.

4.2.9 Summary

As is evident in the foregoing sections, the BRM design concept is flexible and can accommodate the SSV interface. Before final selection, as stated previously, the SSV side of the interface, along with operating restraints must be defined. In addition, the experiment operational requirements and design will also have a major impact on the configuration design selection.

4.3 THERMAL CONTROL

The experiment package coldplate must be maintained at 40 ±5 °F while the BRM is in the SSV cargo bay for launch and recovery. The thermal loads during these periods are assumed to be the same as during the orbital phase of the mission:

(a) Peak; 350 BTU/hr for 10 minutes each hour

(b) Continuous; 180 BTU/hr to 270 BTU/hour

The SSV wall temperature extremes, as defined in the NCIRD are -100 °F and +140 °F (the NR and MD studies indicate a low value of -300 °F).

To maintain the coldplate temperature a thermal control system must be provided. The relationship:

$$Q = A\sigma\epsilon\ (T^4_{ssv} - T^4_{brm})$$

where:

A = heat transfer area in square feet,

σ = Boltzmann's Constant, 0.1713×10^{-8} BTU/hr ft^2 °R^4

ϵ = Emissivity

T_{ssv} = Wall temperature of SSV in °R

T_{brm} = Required coldplate temperature in °R,

was used to determine the quantity of heat to be removed or added to the coldplate to maintain it at 40 °F. This analysis considered that the coldplate is insulated from other sections of the vehicle and that the thermal control of the coldplate can be evaluated independently. In addition, it was assumed that for all sections of the BRM, except the coldplate, the SSV environment will impose no severe design constraints.

The results of the analysis conducted for a heat load of 270 BTU/hr and a coldplate radiating area of 4.63 ft^2, presented in Figure VII-6, indicate that if the shutters are locked in a closed position, heat removal is required for the entire range of SSV wall temperature. If the low range is determined to be -100° F, and the shutters are locked open, heat removal is required over the range of wall temperatures; however, if the low limit is -300° F, and the shutters open, heat must be added from -300° F to approximately -116° F. Still considering the -300° F low, if the shutters are allowed to operate in the cargo bay, no heat need be added. Shutter blade angle variations from 40° to 90° (full open) will control to the required coldplate temperature and heat removal will be required for the remainder of the wall temperature range.

Three methods of heat removal can be considered, again, dependent on the final SSV design; refrigeration, heat exchange and evaporation.

Figure VII-7 is a block diagram of the refrigeration concept that could be used. It is a single stage vapor compression cycle. Figure VII-6 indicates the power that would be required to run a refrigeration unit of this type for both open and closed shutters on the BRM. The compressor power is a realistic number in that standard refrigeration efficiency and coefficient of performance were used in its derivation. Practical selection of a system will be dependent, not only on wall temperature extremes, but on estimates of expected temperature variations (duty cycle).

Two other methods have been considered for heat removal in the cargo bay. Figure VII-8 indicates a heat exchanger using SSV supplied coolant. Coldplate temperature is maintained by varying the coolant flow through the coldplate. This method is practical, consumes little power, but requires a coolant interface with the SSV. The NCIRD specifies that this interface must be with support equipment and not with the payload. Further pursuance of this concept must await the definition of SSV coolant type, temperature and flow rate. Figure VII-9 shows another method for removing heat from the coldplate. A sublimator type of water evaporator would be used for the heat exchange. Coldplate temperature is maintained by varying the coolant temperature into the coldplate by controlling the rate of evaporation. The NCIRD indicates that

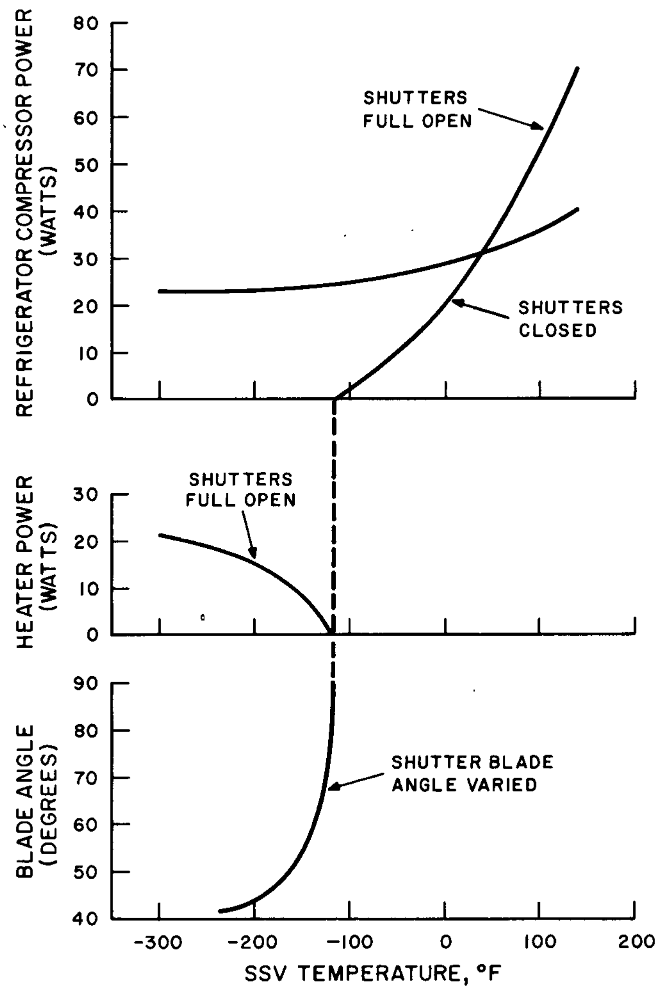


Figure VII-6. Power and Shutter Blade Angle Requirements to Maintain the Coldplate at 40°F On Board SSV

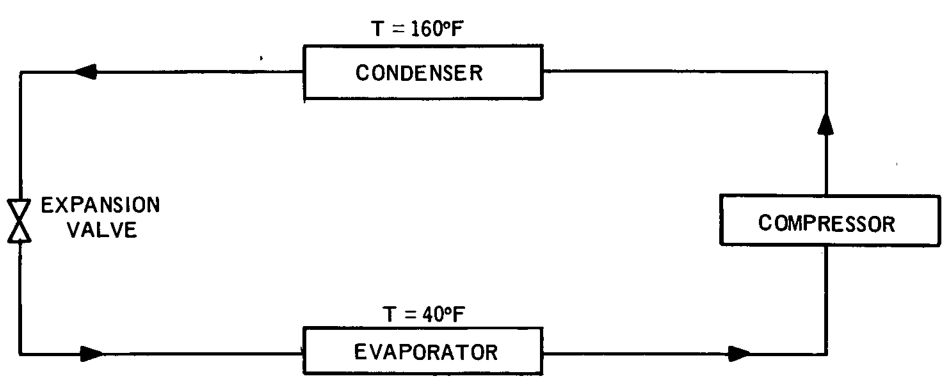


Figure VII-7. Refrigeration Concept

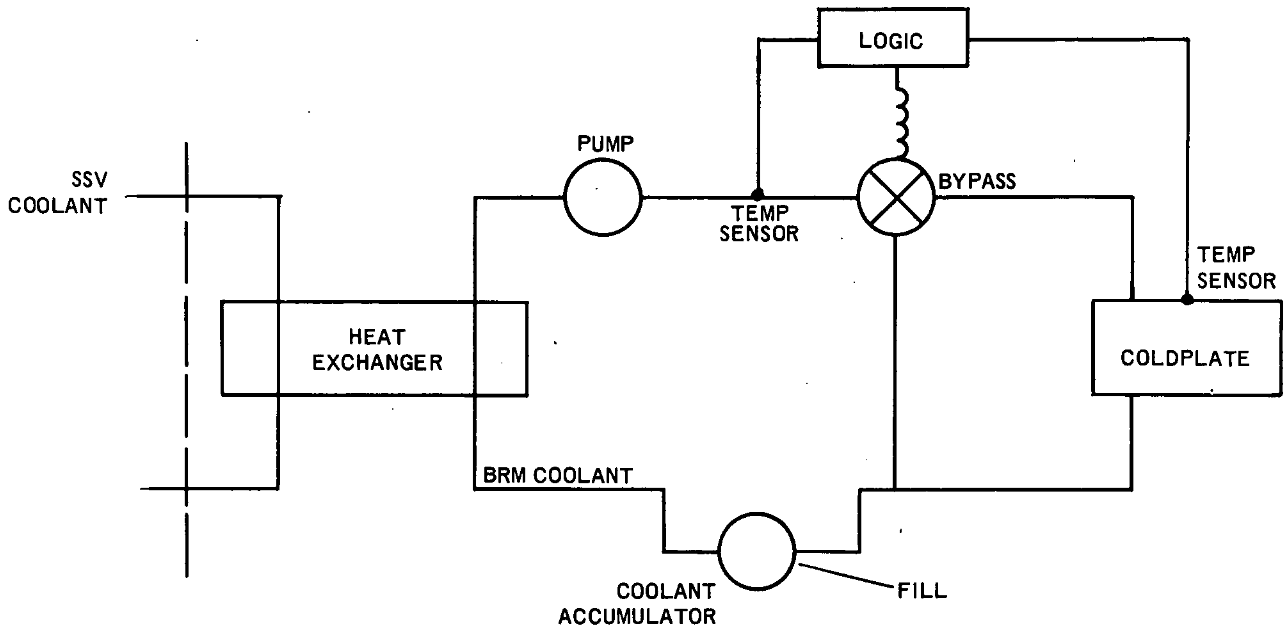


Figure VII-8. Heat Exchanger

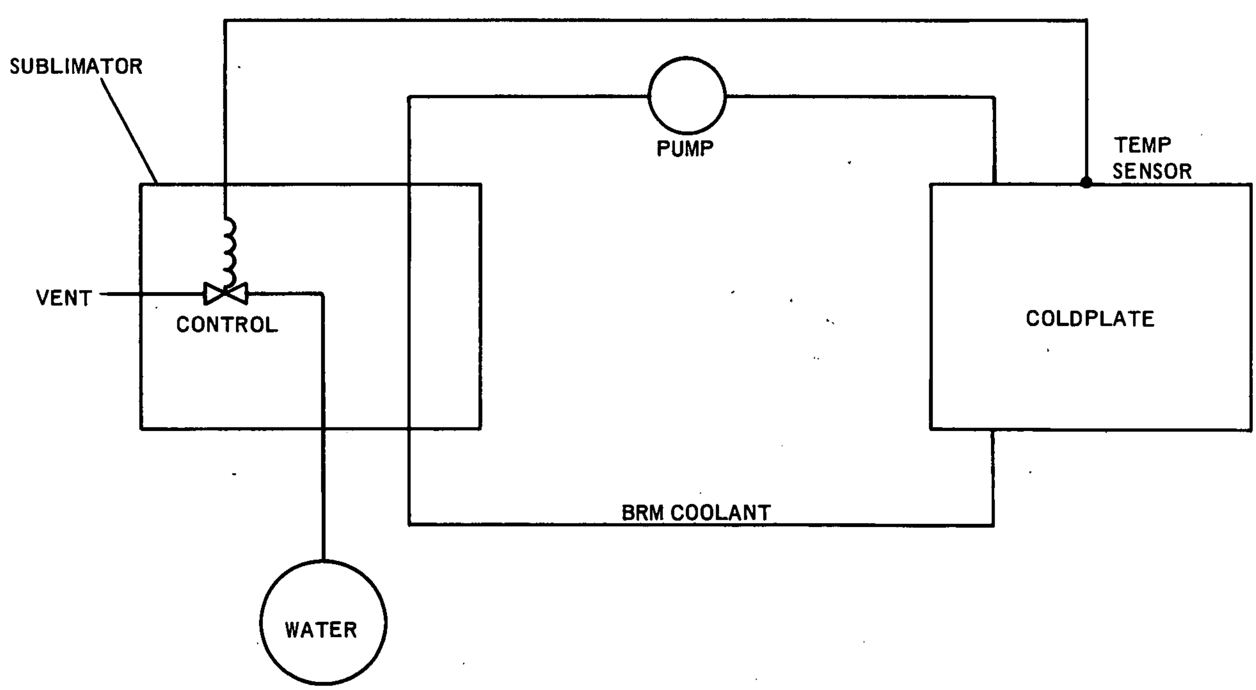


Figure VII-9. Evaporator

overboard vents will be available to payloads. This type of system can be completely contained in the adapter service module, however, it is dependent on being capable of venting while in the cargo bay.

4.4 CONFIGURATION CONCEPTS

This section is a compilation of various configurations that could fit in the SSV cargo bay. The means of attachment in the SSV plus the articulation of the manipulator arm have yet to be established. However, several concepts are shown indicating the flexibility and adaptability of the BRM to the SSV as a launch vehicle.

4.4.1 Burner II Configuration

Figure VII-10 shows the Burner II attached to the Scout baseline design BRM by means of a new adapter. Support equipment for this concept would be shuttle mounted. The solar panels are the same as the Scout design (no retraction required because there is no recovery of Mission III) except that a new tie down to, and release from, Burner II is required.

4.4.2 Stowage

Figure VII-11 indicates the BRM stowed in the SSV cargo bay in various positions, with the solar panels deployed and retracted. The position with BRM roll axis parallel to the cargo bay long axis provides for lift-off loads in the same direction as Scout. However, because the SSV loads are significantly lower than Scout, the lateral position should not cause any problems. The sketch also indicates that the BRM with solar panels deployed will also fit longitudinally or laterally. Stowage with panels deployed could increase the possibility of damage in handling.

4.4.3 Canisters

The concept of stowing the BRM in a canister in the cargo bay appears attractive. It affords maximum protection against handling damage not only in the cargo bay but

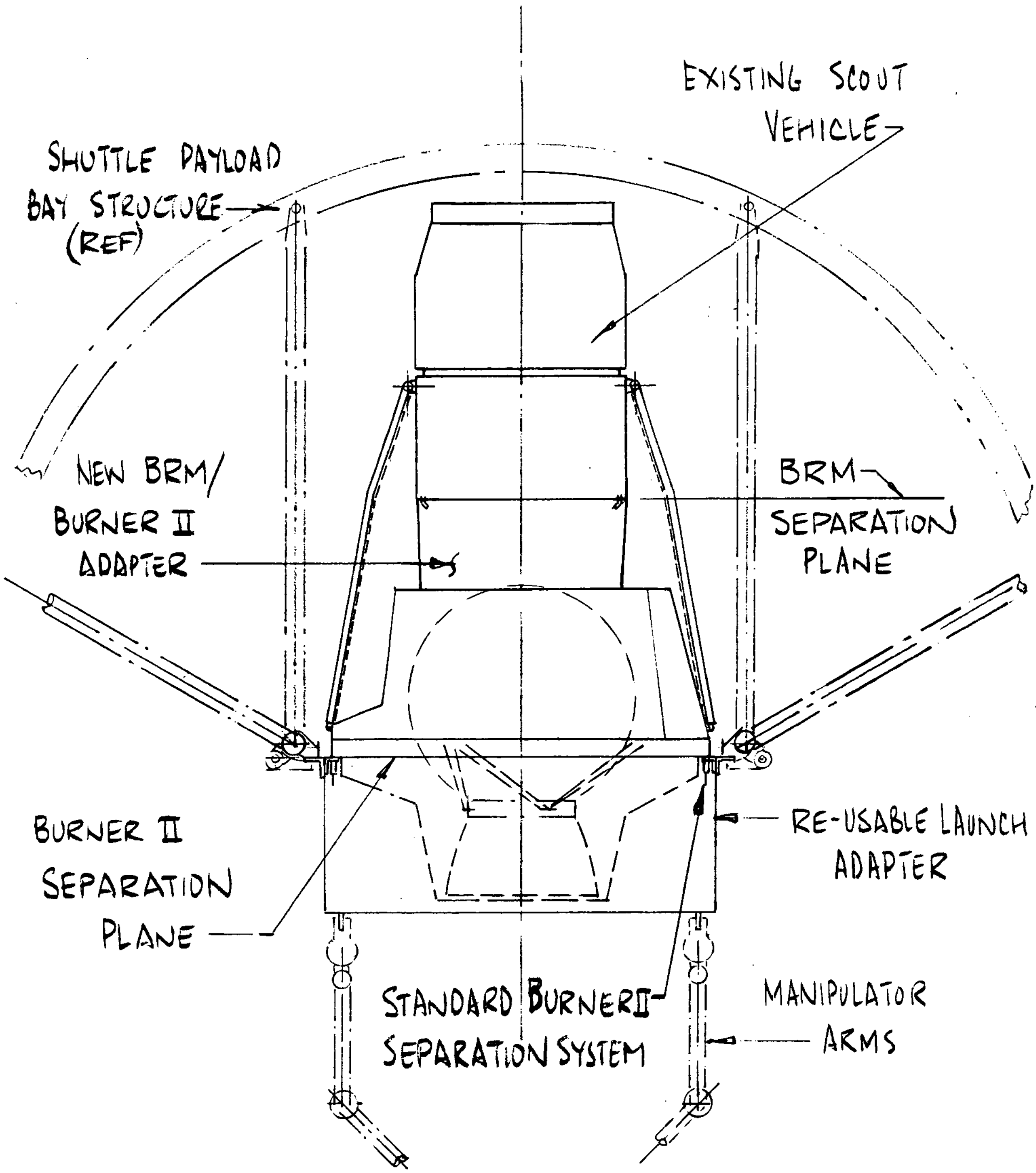


Figure VII-10. Burner II Configuration

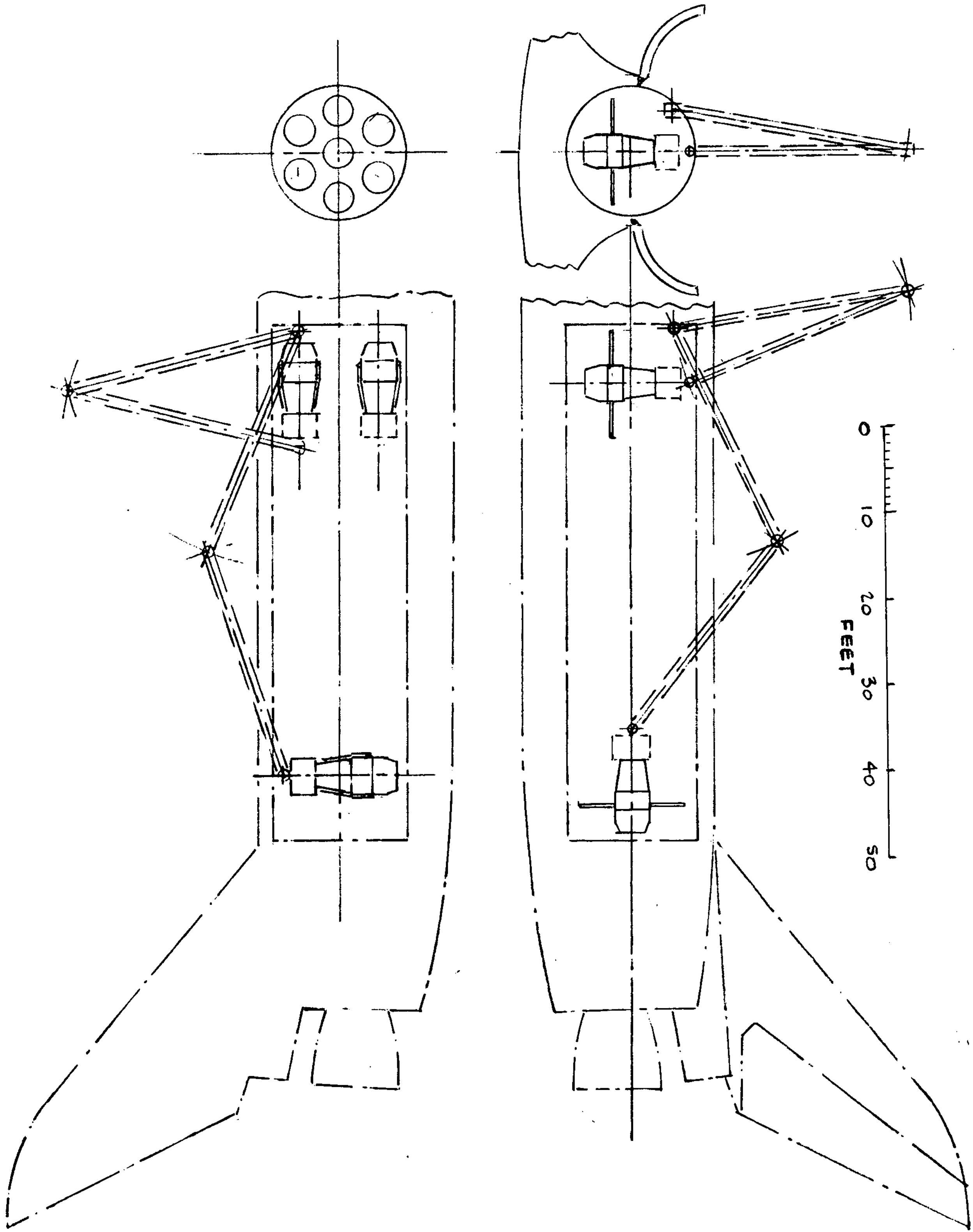


Figure VII-11. Shuttle Stowage Options

from factory to pad, flexibility for the manipulator, and a degree of environmental control. Its largest disadvantage is the additional weight, which should be no problem for the SSV. Figure VII-12 shows several canisters mounted in the cargo bay while Figure VII-13 indicates some of the details of the BRM in a canister.

4.4.4 Shuttle Interface

Figure VII-14 shows a possible method of manipulator interface with the BRM. It is patterned after the Apollo docking adapter design. This concept is capable of providing push off ΔV and umbilical disconnect on release. Axial alignment is required for docking with radial alignment provided by ramps and rotational by the manipulator arms. Figures VII-15 and -16 indicate possible adapter designs for the BRM. Both adapters are the same size so that the manipulator interface is clear of the retracted solar panels. The first is for the concept of using external support equipment on the SSV. The second is for the minimum interface concept in which the adapter would carry all the support equipment required for the BRM. In both configurations, the attitude control nozzles are moved to attain more torque and SSV attachment is by an add-on structure around the experiment. Figure VII-17 is a layout of the support equipment in the adapter that would be used in the minimum interface concept.

4.4.5 Launch and Retrieval

Figures VII-18 and -19 indicate various methods of launch and recovery of the BRM by the SSV. At this writing it appears that the manipulator will be the method selected for payload launch and retrieval.

5.0 SERVICE, MAINTENANCE AND TESTING

5.1 SERVICE AND MAINTENANCE

A preliminary evaluation of various service and maintenance concepts indicates that on-orbit service and maintenance for the SSV launched BRM missions should be limited to recovery of the BRM for return to earth. This was primarily based on achieving a

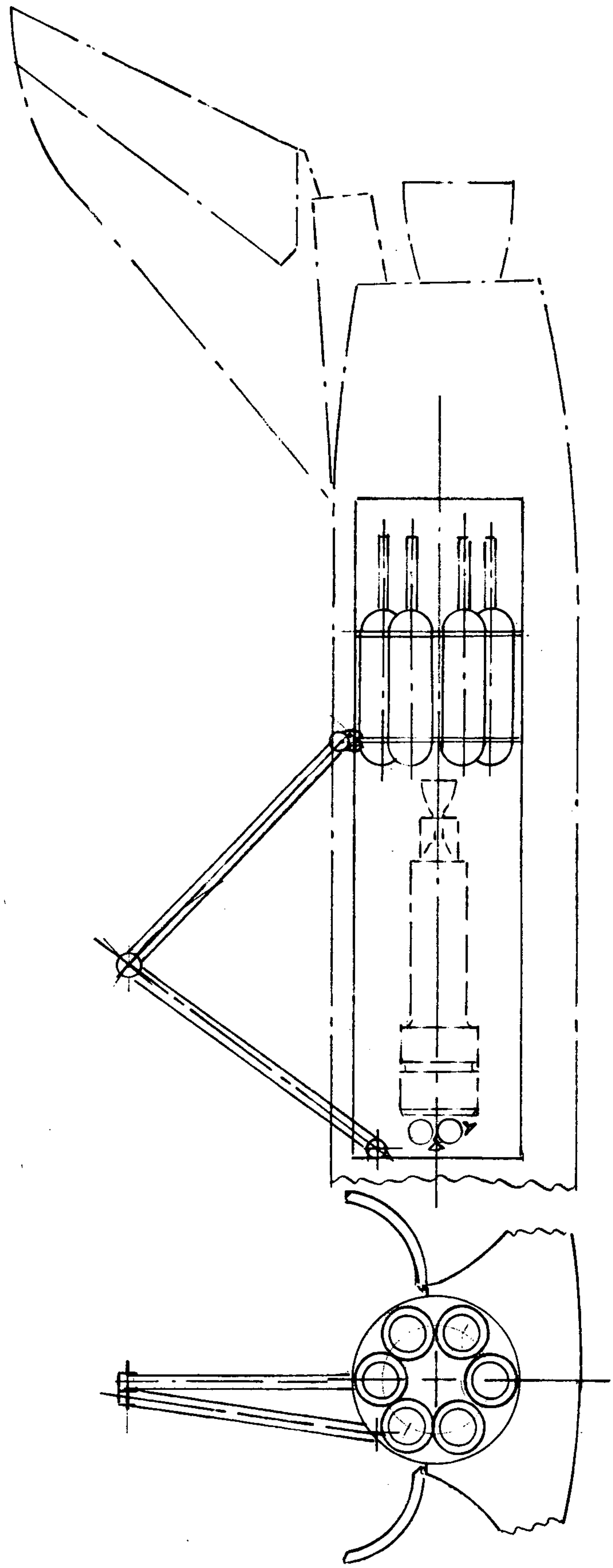


Figure VII-12. Stowage of Containerised BRM's

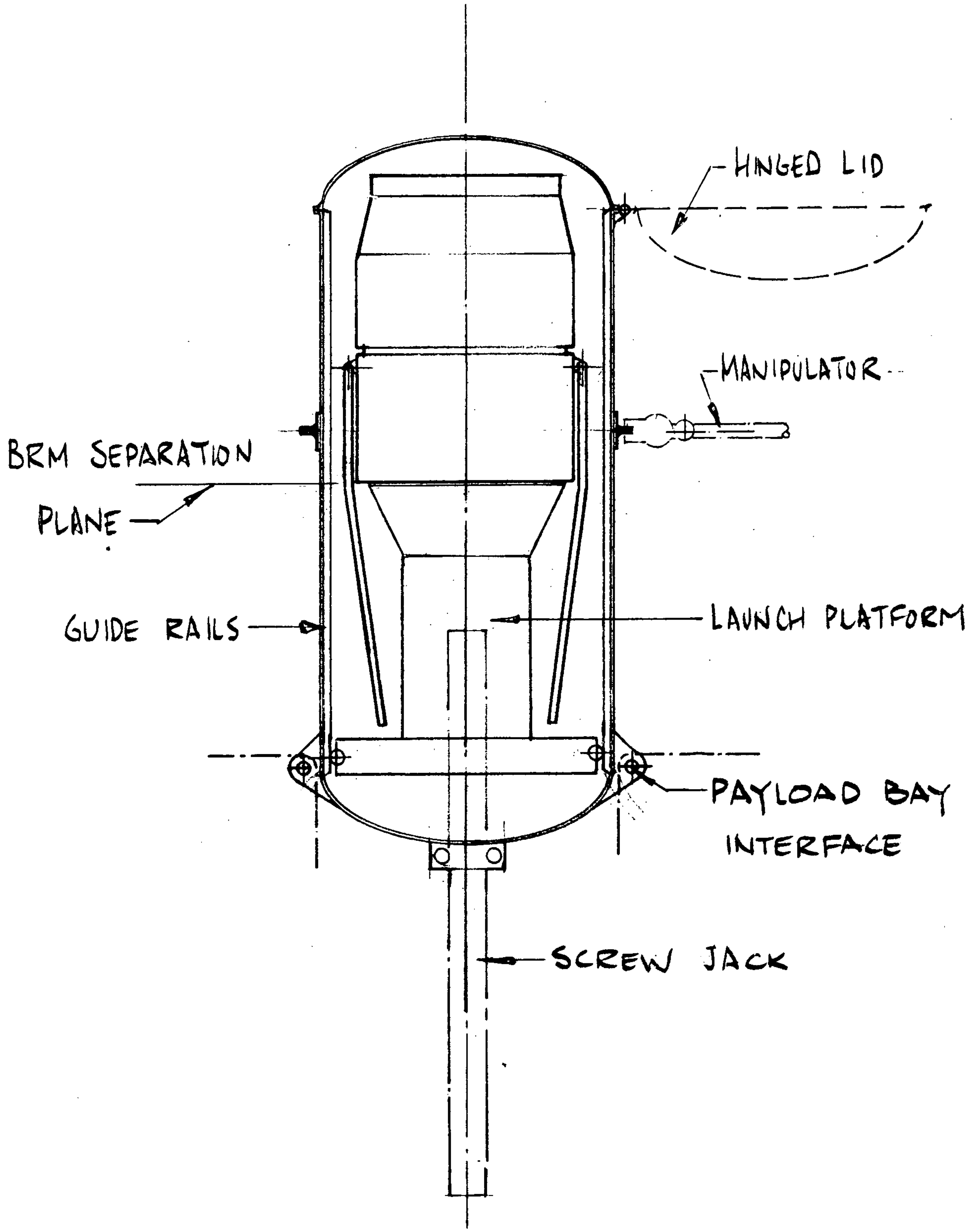


Figure VII-13. Containerized Spacecraft

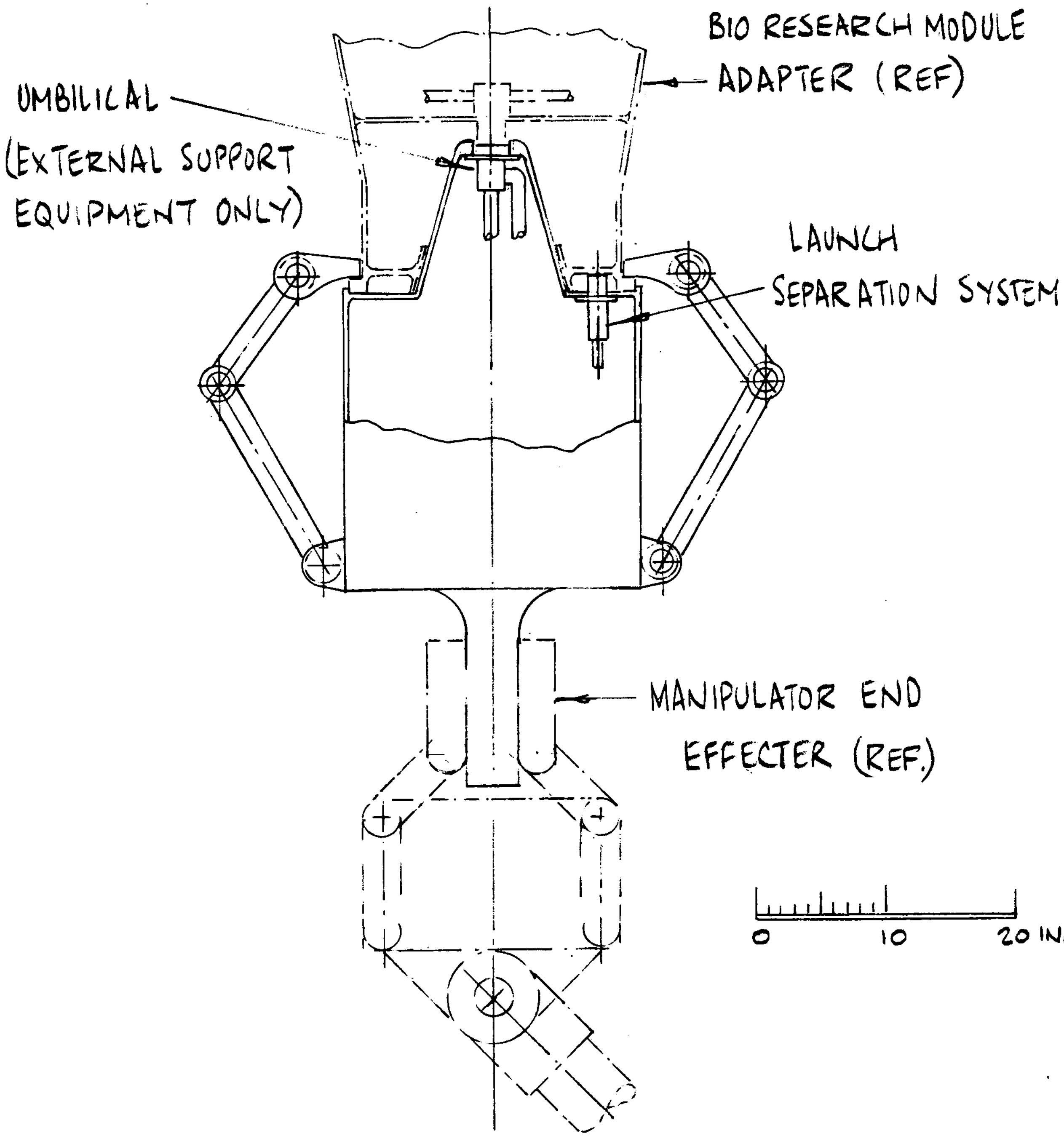


Figure VII-14. Manipulator Attachment

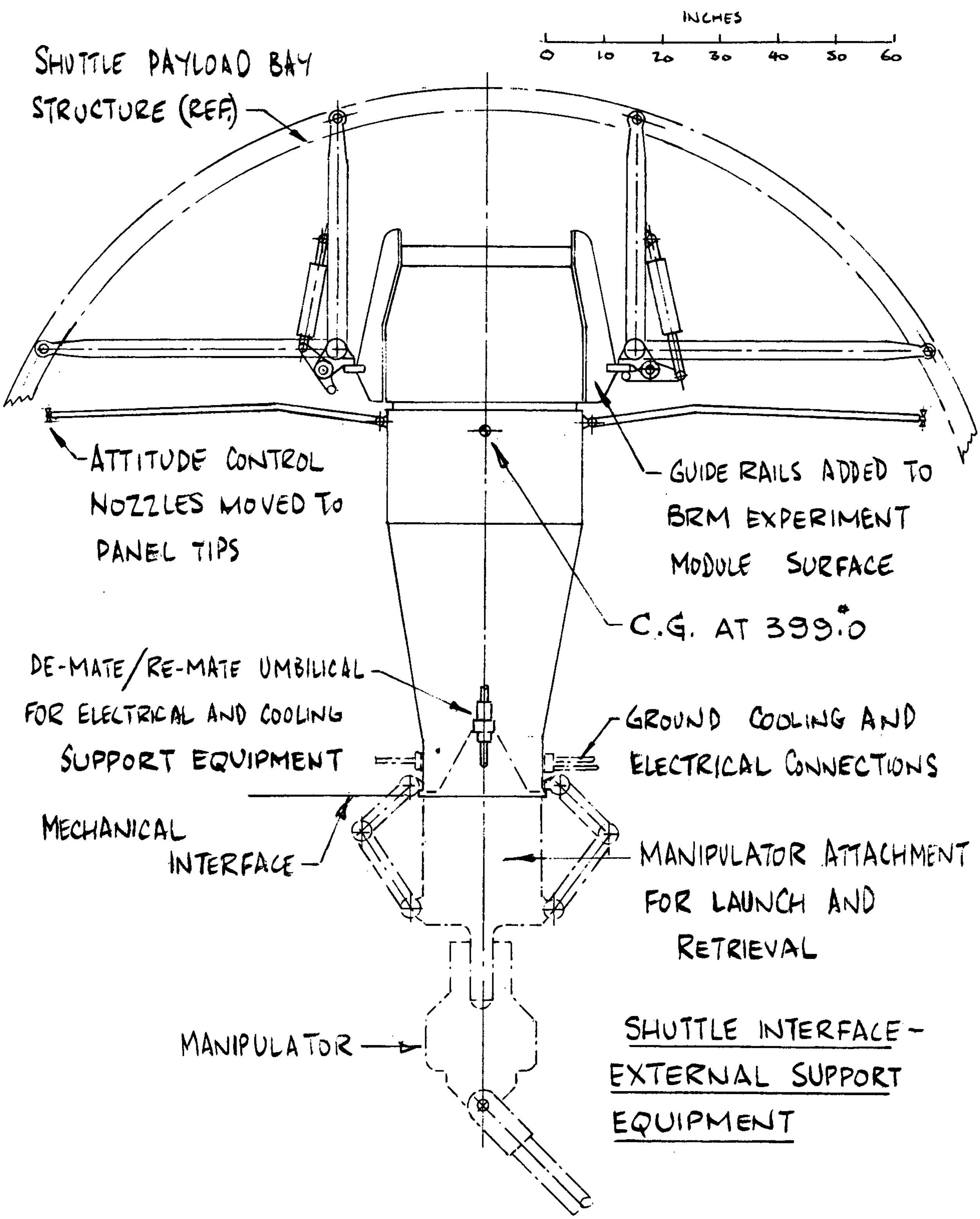


Figure VII-15. Shuttle Interface - External Support Equipment

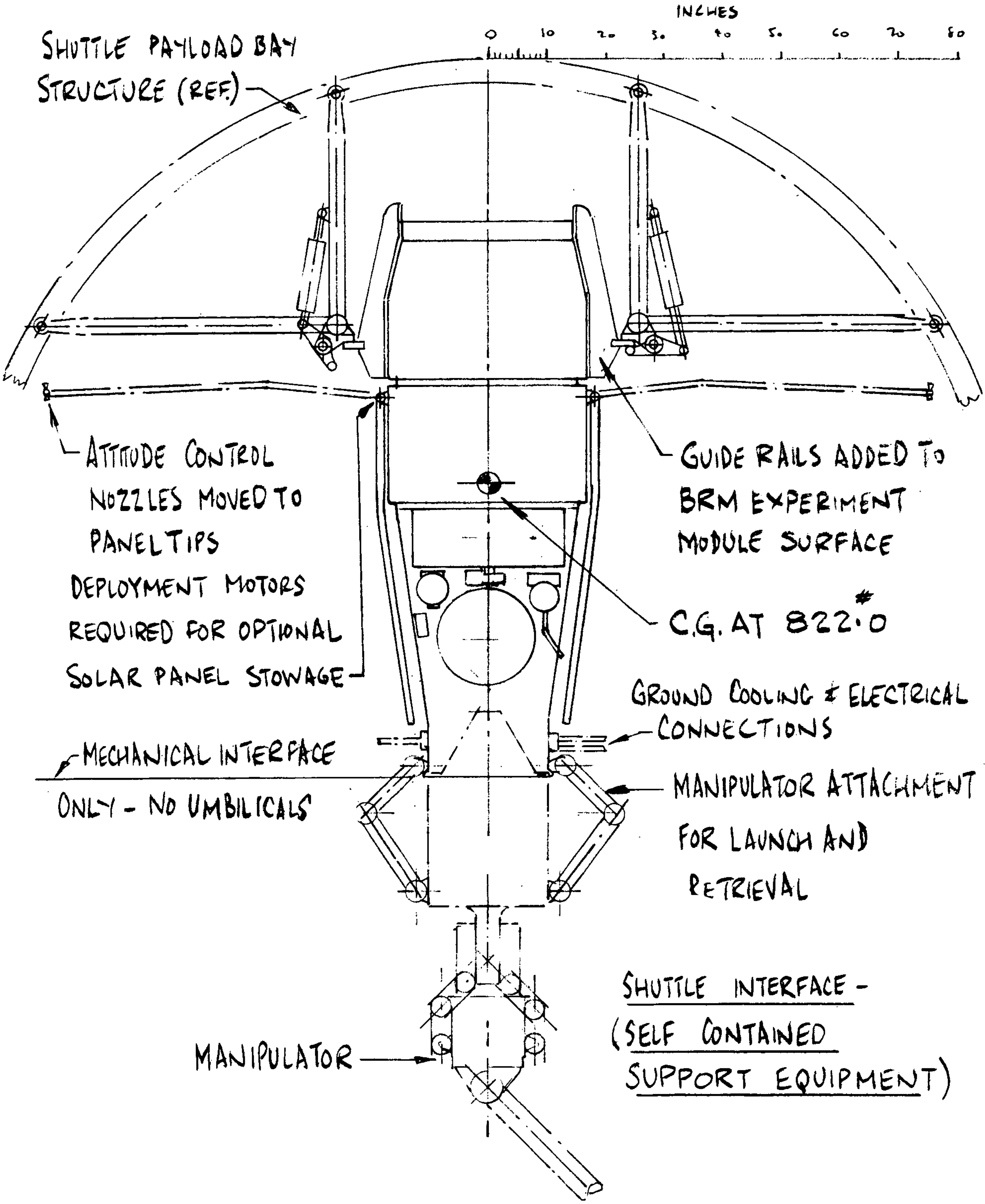


Figure VII-16. Shuttle Interface - (Self Contained Support Equipment)

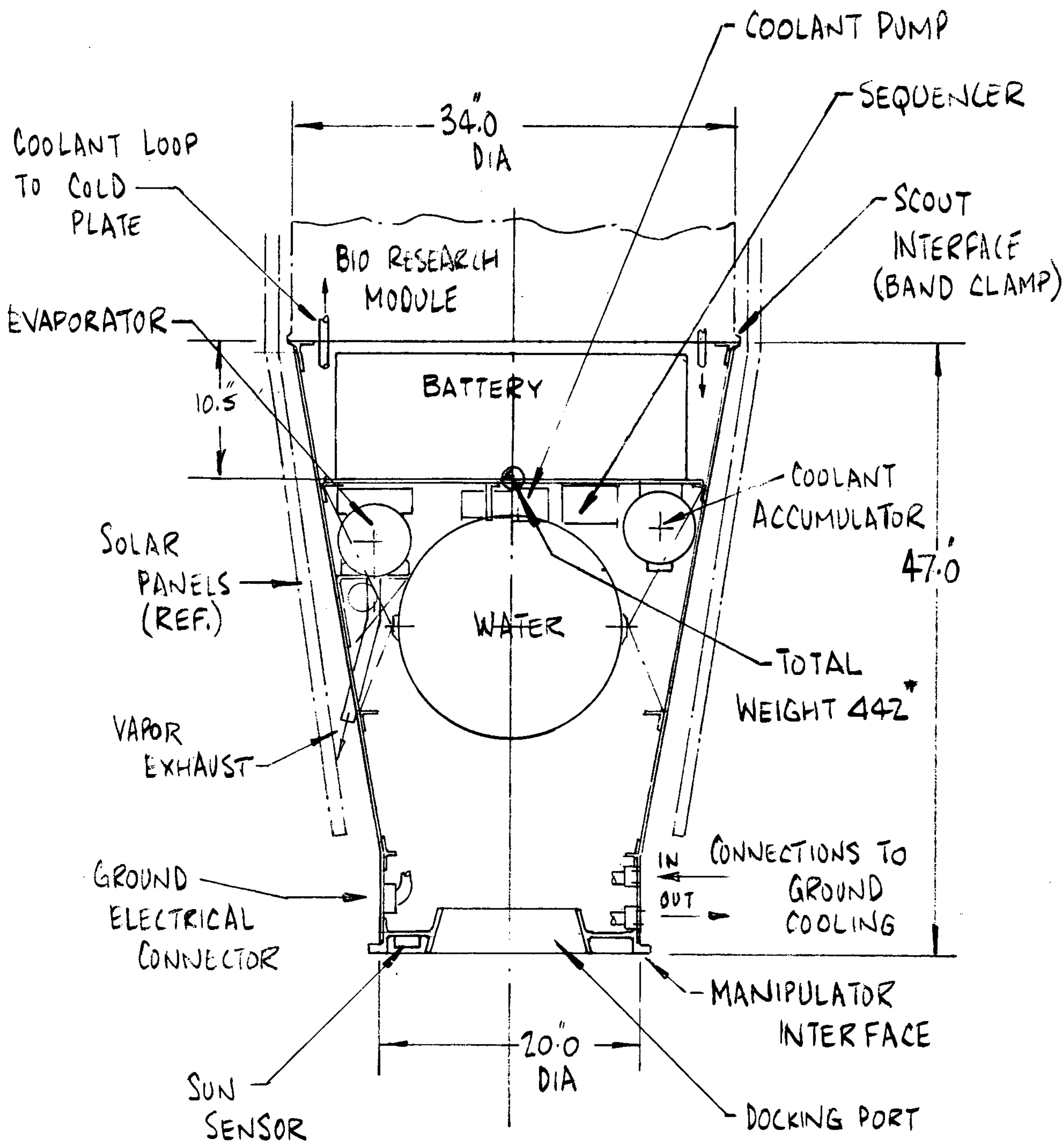


Figure VII-17. Adapter Arrangement for Self Contained Support Equipment

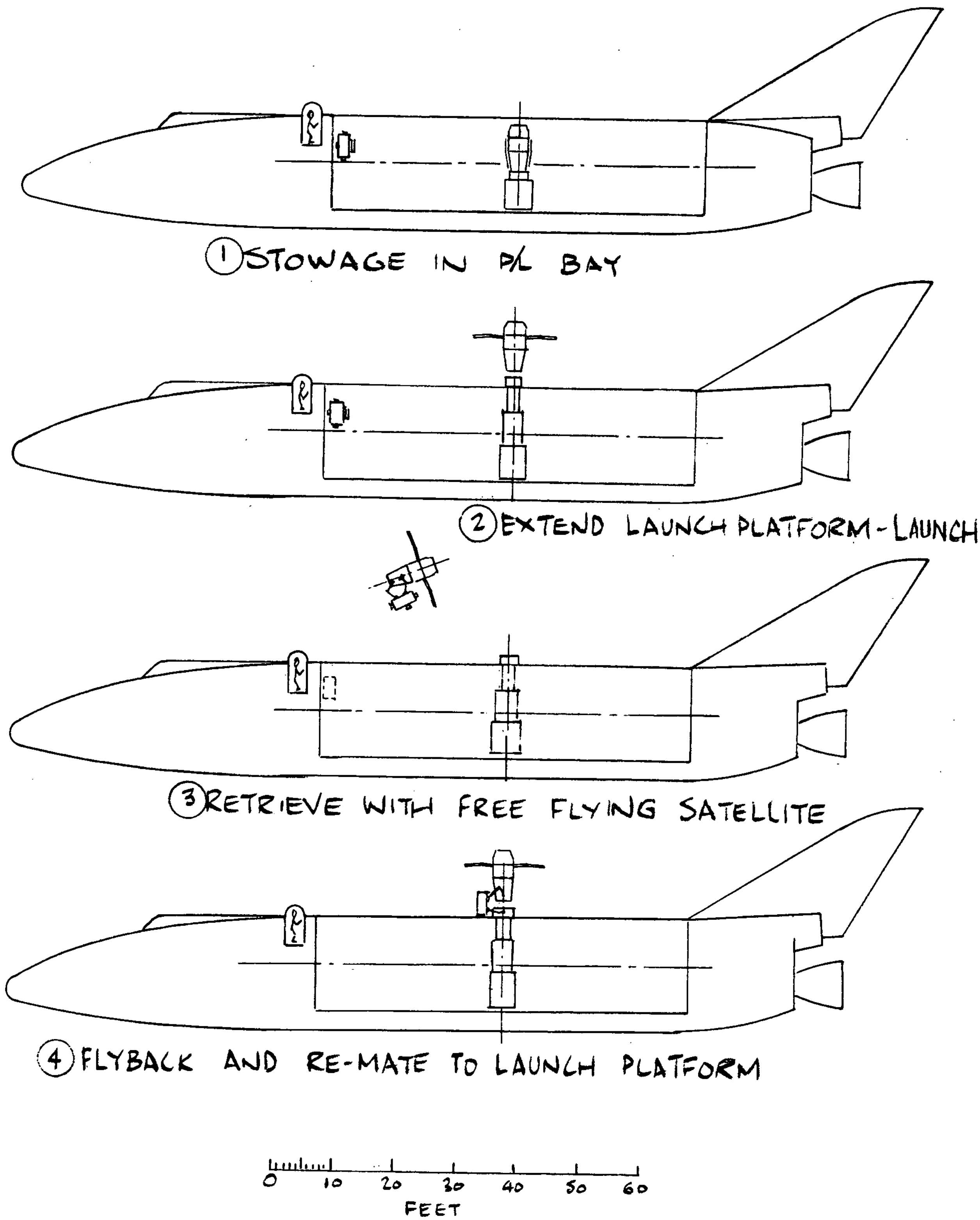


Figure VII-18. Launch and Retrieval - "Free Flying" Satellite

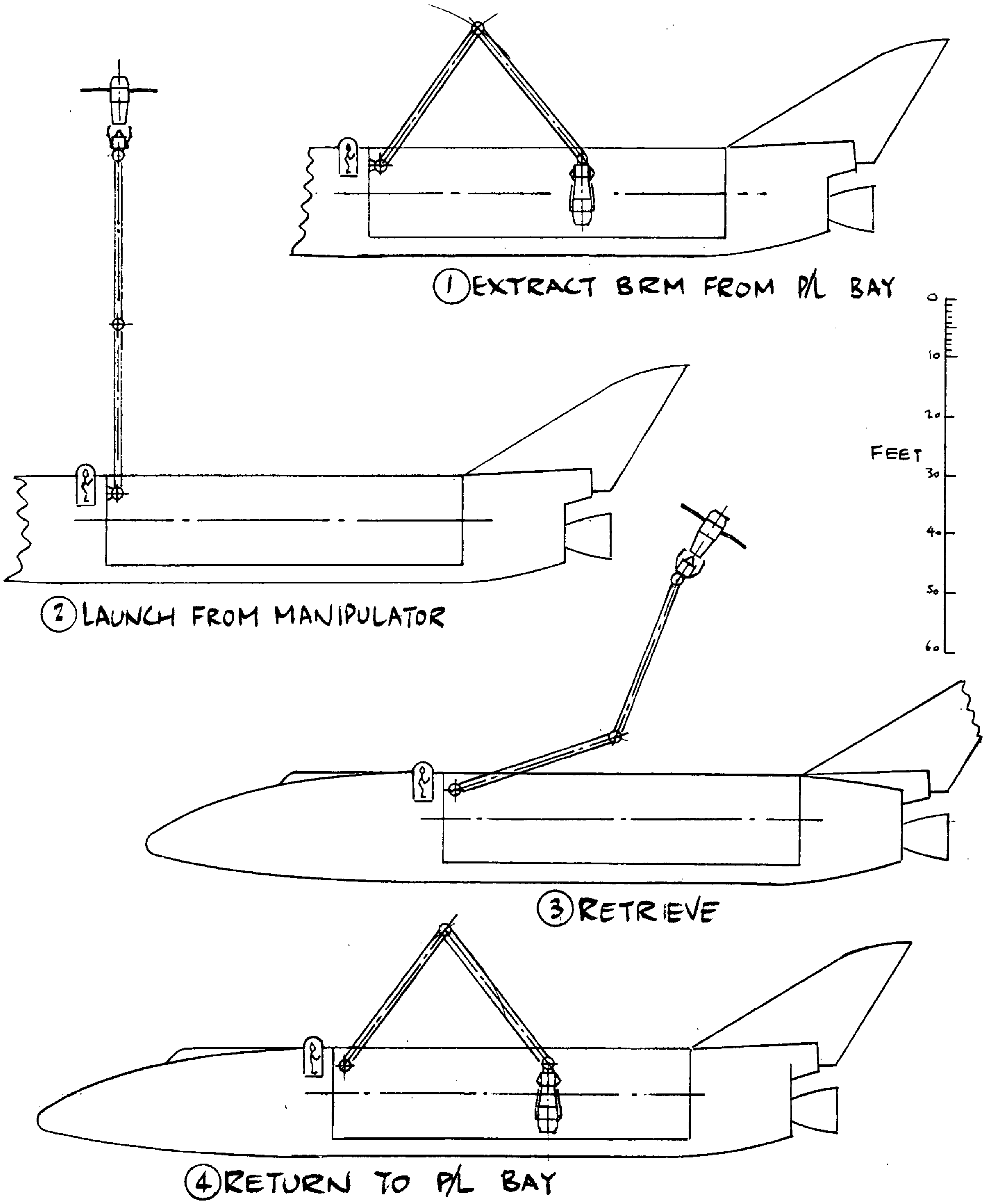


Figure VII-19. Launch and Retrieval - Manipulator cost effective service and maintenance concept with a planned mission schedule of one 6-month flight per year. This will result in minimizing annual costs for hardware, service and maintenance. There will be no significant impact on the design and operations of the SSV and minimal impact on the BRM design. Table VII-4 summarizes the concepts considered and the respective program implications of each concept. Table VII-5 indicates the SSV configuration and capabilities applicable to evaluation of the on-orbit service and maintenance concept. The information, not yet finalized, was taken from the NCIRD and the Phase B study reports.

5.1.1 Concept Impact

While all concepts considered appear to be technically feasible, there does not appear to be a significant improvement in benefits by the selection of only one concept. Table VII-6 indicates the overall impact of the general service and maintenance concepts on the design and operations of the BRM and the SSV. Although no unique benefit is accrued, when one considers the significant impact on the design and operations of the BRM and SSV of the other concepts this preliminary study indicates that concepts other than recovery and return to earth are not justified. As a result, the recommended approach is recovery and return to earth of the BRM at mission completion or in the event of a malfunction. However, continued evaluation of this approach is required as the SSV definition becomes more detailed or if the basic BRM mission constraints are altered.

5.2 SSV OPERATIONAL EFFECTS

It is recognized that the BRM may be one of several payloads on a particular SSV mission, resulting in a loss of the dedicated launch vehicle preparations to accomodate the payload. It can also result in a much longer duration between launch and the required orbital environment (zero "g" or variable "g"). Table VII-7 summarizes the experiment constraints imposed by the SSV and compares them to Scout.

TABLE VII-4. SERVICE & MAINTENANCE CONCEPTS

Concepts	Implication
1. Recover Complete System (Return to Earth)	Normal Recovery Procedure Increased Experiment Data Reuse of Hardware
2. Replace Experiment (Completed or Malfunctional) on Functioning Spacecraft	Normal Rendezvous & Docking Manipulations to Remove Completed Experiment and Install New Experiment. Normal Pre-Deployment Checkout Deploy SSV AGE to Support Experiment(s) (In Lieu of Normal S/C)
3. Replace Spacecraft (Malfunctioning or Expended) on Functioning Spacecraft	Normal Rendezvous & Docking Manipulations to Exchange Spacecraft Modified Pre-Deployment Check Deploy SSV AGE to Support Experiment During Exchange
4. Replenish Consumables Replace (Malfunctioning) Components/Subsystems	Normal Rendezvous & Docking Manipulations for Access to and Replenishment of Consumables Manipulations for Access to and Removal/Replacement of Malfunctioning Components or Subsystems Modified Pre-Deployment Checkout Deploy SSV AGE to Support Consumables Supply

TABLE VII-5. SSV CONFIGURATION/CAPABILITY BASELINE FOR ON-ORBIT SERVICE & MAINTENANCE

Function	Baseline
SSV Payload Structure and Interface	Limited Access. "Standard" SSV Interface. Support AGE. Complete AGE is Handled by Payload Deployment Device.
SSV Rendezvous and Docking	Match Orbits. Establish Physical Contact. Mate with AGE and Complete the Docking. Return of Payload to Cargo Compartment.
SSV Payload Monitoring and Data Pass Through	Hardware Data Paths. AGE Will Determine GO/NO-GO Status. Experiment Data Will "Pass Thru" SSV System. Separate SSV AGE Monitor Performance After Separation of S/C from AGE. Visual Observation.
SSV Manipulation of Payload	Only Basic Manipulation Needed to Deploy, Dock, Stow are Currently Assumed.
SSV Crew	Equivalent Service of 1 Man Day (24 Hours) Dedicated for Payload Support and Operations.

TABLE VII-6. IMPACT OF SERVICE AND MAINTENANCE CONCEPTS

Concept	On BRM Design and Operation	On SSV Design and Operation	On Additional Parameters
Recovery and Return to Earth	None	None	None
Replace Exp. or Replace S/C	Exp/S/C Interface Accessibility. Design Compatible with SSV Operational Constraints, EVA or Manipulator. Effects on S/C Reliability of S&M Manipulations.	EVA, IVA, or Special Service Sta. Equip. Crew Training Mission Time Consumption Manipulator Capabilities Availability for Service and Maintenance (Scheduled vs. Non-Scheduled)	2 S/C + Spares Effect on Experiment of S&M Manipulations
Replenish Consumables and/or Replace Subsystems	Lower Reliability Accessibility/ Maintainability AGE Trouble Shooting Less Space on Board for Consumables Design Compatible with SSV Maintenance Capabilities	Trouble Shooting (Remote) Spare Parts Stowage EVA, IVA or Special Service Station Equip. Crew Training Mission Time Consumption	Effect of Repeated SSV Launches on Spares Carried Effect on Experiment of Service and Maintenance Manipulations Requires Spares Pod

TABLE VII-7. EXPERIMENT CONSTRAINTS BY LAUNCH VEHICLE

Event	SSV	Scout
Mate S/C to Launch Vehicle	T-80 Hours	T-9 Days
Mate Payload to S/C	T-72+ Hours (Complete)	T-8 Hours (Start)
Access to Payload	Limited by Cargo Bay Configuration Crew Tunnel, Access Doors	360° Around S/C and Full Height
Experimenter Walkaway	T-72	T-6
Deployment into Final Experiment Environment	Up to T+72 Hours (or More)	T+17 Minutes (Mission I)
Launch Environment	3.3G Axial (Remainder TBD)	Static: Axial 13.5G Lateral 3.0 Spin 180 rpm/max. 60 rpm Oper. Spin Accel. 12.56 rad/sec Dynamic: Axial ±6.0G
Telemetry	1 Channel 25 KBPS 2 Way Simplex Via SSV (If No Payload Antenna)	Self Contained RF Link

5.3 TEST PHILOSOPHY

A preliminary evaluation of the potential impact on the test program indicates that the baseline test program for the Scout launched BRM can, with relatively minor modifications, establish design confidence and qualification status for both Scout and SSV missions.

Three cases were considered, the impacts of which is listed in Table VII-8. However, the current design philosophy is such that case number 2, minimal baseline change, is the most probable program plan.

It should also be noted that the acceptance test cycle for refurbished equipment from SSV missions may be significantly reduced although repeated use of hardware may result in additional test requirements. This direction can only be determined when a specific program plan along with specific SSV capabilities are determined. Other influential factors are that the SSV launch environment is not as severe as the Scout launch environment and that the anticipated on-board checkout capability of the SSV, after the launch environment, will increase confidence of mission success.

6.0 COST

It is felt that the preceding sections have justified a design approach which uses the Scout baseline design with minimal modifications for the SSV launched missions. The additional cost for design, development and manufacturing of the BRM itself will be quite small. Added cost will be for design, development, manufacture and test of the adapter/service module, any SSV mounted support equipment and vehicle integration and support. At this time, the selection of this equipment is not possible so rather than confuse the options listed in this report, cost impact estimates will not be made.

TABLE VII-8. SUMMARY MATRIX OF IMPACT ON TEST PROGRAM

Assumed Configuration	Potential Impact on Baseline Test Program
1. Baseline Bioexplorer Configuration Virtually Unchanged for SSV Operations.	No Significant Changes Required to Basic Test Program. Additional Compatibility Tests will be Required. Can be Incorporated into Basic Test Program.
2. Minimal Modifications to Baseline Configuration (Most Probable).	No Significant Changes Required to Basic Test Program. Additional Tests Required to: Achieve Qual. Status of SSV Peculiar Hardware Demonstrate Compatibility
3. Major Modifications to Baseline Configuration (Not Likely).	
3.1 Mods Incorporated into Baseline Configuration.	No Significant Changes Required to Basic Test Program. Minimal Changes Required to Achieve Qual. Status for Both Scout and SSV Missions.
3.2 Mods Incorporated after Scout Missions.	May Require Additional Tests and Test Hardware to Achieve Qual. Status with Two (2) Different Configurations. Possibility of Performing Additional Tests on a Component or Module Level.