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STUDY OF APPLICATION OF REMOTE MANIPULATION TO SATELLITE MAINTENANCE

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FINAL REPORT   Volume II : Technical Report

GENERAL ELECTRIC

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JUNE 1969

A STUDY OF APPLICATION OF REMOTE MANIPULATION TO SATELLITE MAINTENANCE

FINAL REPORT

VOLUME II: TECHNICAL REPORT

CONTRACT No. NAS 2-5072
NASA REPORT No. R-73-339

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MISSION ANALYSIS DIVISION
OFFICE OF ADVANCED RESEARCH AND TECHNOLOGY
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FOREWORD

This study conducted under Contract NAS 2-5072 and entitled, "Application of Remote Manipulation to Satellite Maintenance," is documented in two volumes:

- Volume I is a summary containing the highlights, key results, conclusions and recommendations.
- Volume II is the technical report containing the details of the mission selection and analyses, laboratory simulations, system and subsystem designs, and system costs.

SECTION 1

INTRODUCTION

Scientists and engineers have devoted a great deal of effort in developing methods for achieving long life and high probability of success in satellite system programs. Design redundancy and extensive testing have been the most widely used techniques. Despite the relatively good success attained through these techniques, occasional catastrophic failures have continued to occur, most often in the more complex systems. Increased probability of success would be achieved if the failed satellites could be repaired on-orbit.

Another promising area of on-orbit operations is extension of the useful life of those satellites that are functioning successfully. These spacecraft offer two potential paths of future cost savings. First, the spacecraft whose experiment payload continues to provide useful data but whose housekeeping expendables such as fuel, cold gas, or batteries have been depleted could be refurbished on-orbit and have its mission life greatly extended. Second, those spacecraft with obsolete experiment payloads but with housekeeping subsystems that are fully operational could have both payload and expendables replaced on-orbit to provide a completely new mission. In fact, new experiments are being launched on existing or modified existing spacecraft in order to reduce the large design and development costs, thus demonstrating the compatibility of existing designs and new payloads.

While the potential of on-orbit maintenance is recognized, the use of man in an EVA mode to perform this function is limited in applications. The radiation environment which exists in some regions of space requires a substantial amount of shielding to protect man and consequently reduces his dexterity. The brevity of EVA periods reduces the amount of useful work that could be performed. In situations where the space station is a great distance from the worksite, propulsive requirements may be prohibitive and much valuable time would be spent traveling. The number of space stations to be orbited would be limited due to cost. Finally, the availability of astronauts to repair a random failure of a remote satellite is presumably low because of the tasks required of them in and around the space station. However, one thing is clear -- that man's intelligence and at least some part of his sensory and manipulatory capabilities are desired for on-orbit maintenance.

Remote manipulator systems allow man to be physically located in a safe environment, while extending his vision, feel, and motions to distant, hazardous locations. Today, manipulators have been built which possess both position correspondence as well as force reflection to provide the operator with a "feel" for his activities. In addition, a variety of terminal devices allow man to perform many tasks as well as he could manually and, in some cases, to perform tasks which he could not perform manually.

The foregoing suggests using manipulator systems in space to perform the following generic types of missions:

a. On-orbit repair.

b. On-orbit refurbishment.

c. Inspection and diagnosis of failed or degraded satellites. The purpose of this mission would be to obtain data on a failure otherwise unobtainable. These data would be of great value for redesign of follow-on spacecraft of the same family. These data could also be used for repair of the failed spacecraft.

d. Retrieval of scientific payloads or samples. Examples are retrieval of solar array sections or thermal coatings to examine radiation effects, retrieval of the detachable meteroid detection panels on Pegasus, retrieval of exposed photographic film, etc. Samples could either be deorbited or brought to a space station for analysis by astronaut scientists.

e. Other potential missions, such as erection of space structures, astronaut rescue, releasing fouled shrouds, hatches, or booms, and military missions.

These applications represent the direct extension to space of the hot lab manipulator technology already successfully applied to other areas on earth. The purpose of the study reported herein is to take a closer look at a specific remote manipulator spacecraft configuration to perform selected on-orbit repair and refurbishment missions. The remote manipulator spacecraft studied is a version configured for a single mission life. In operation it would be orbited separately to perform repair or refurbishments tasks on a selected satellite system. The study includes mission analysis and determination of system requirements. It also provides system design and system cost data, and a realistic evaluation of the system's ability to perform the missions. These data were derived in a manner which will allow both cost and technical comparisons of the scheme with alternate methods such as satellite replacement or man-attended maintenance.

1.1 REMOTE MANIPULATOR SPACECRAFT SYSTEMS

A remote manipulator spacecraft system consists of a spacecraft in orbit and men in a control station located on earth or in an orbiting space station. The operator's control station is equipped with master manipulators, visual displays, and controls. The spacecraft is equipped with slave manipulators, an operator-aimed camera, and the necessary housekeeping subsystems. Control of the spacecraft is through a wideband radio link.

The NASA established ground rules restricted the investigation to a system concept of the type depicted in Figure 1-1. This is a ground controlled single mission, single vehicle system. The rationale behind this is the belief that this is the lowest cost approach. By launching a new remote manipulator spacecraft for each mission, the spacecraft are produced in larger quantities and the recurring costs are low. Furthermore, the large propulsive requirements of orbit transferring resulting from a spacecraft with a multi-mission capability are obviated. Finally, a single mission system has a short operational life which alleviates the requirements for design redundancy and long life testing and reduces the costs. Other system concepts exist but are outside of the study scope and are not discussed.

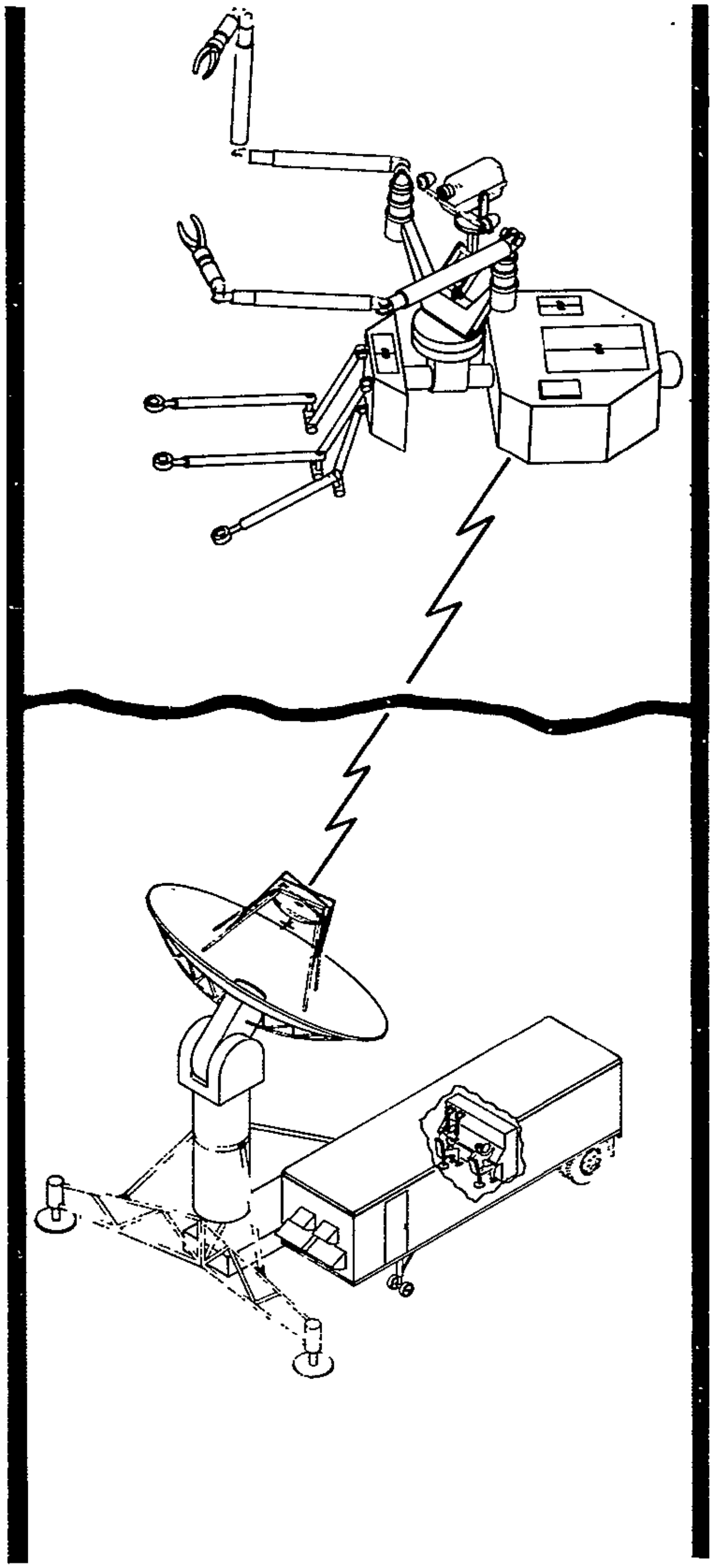


Figure 1-1. Remote Manipulator Spacecraft Systems

1.2 OBJECTIVES

This study is aimed at deriving fundamental data on the feasibility of using remote manipulator spacecraft systems to perform on-orbit satellite maintenance. The processes used to examine and establish feasibility included:

a. The selection of four satellite systems representing a broad cross section of designs and characteristics on which on-orbit repair or refurbishment missions could be performed. Analysis of the performance of these missions will yield a realistic set of requirements for which a remote manipulator spacecraft would be designed.

b. The recognition and identification of potential standard satellite design practices which could facilitate and simplify on-orbit maintenance. Although these standardized satellite design practices were selected specifically for enhancement of remote manipulator repair capability, these practices would aid an astronaut if he were called upon to perform EV maintenance.

c. The design of a remote manipulator spacecraft to meet as many of the system requirements as possible. The constraints on this spacecraft design were minimum cost, minimal complexity, ground control link only, and utilization of the spacecraft for a one-time mission.

d. A realistic reappraisal of the ability and limitations of the remote manipulator spacecraft design with regard to the total requirements of the selected four missions. Key design, technology, and operational problems were identified.

e. A cost estimate of an operational version of the selected remote manipulator spacecraft system. Costs were categorized as development, recurring, and sustaining costs.

The results of this study are intended to provide NASA with the basic information for realistically assessing the feasibility, and costs of developing and deploying a first generation remote manipulator system in space. The study furthermore identifies the areas for future analysis, design, and development required to provide a more complete understanding and more critical assessment of the missions which remote manipulators are capable of performing.

1.3 APPROACH

In order to meet the stated objectives of this study, the plan shown in Figure 1-2 was developed. The personnel participating in this study represented a mix of mission analysts, spacecraft designers, systems engineers, and manipulator design specialists from both the General Electric Company Space Systems Organization at Valley Forge, Pennsylvania and the General Electric Specialty Materials Handling Products Operation at Schenectady, New York. The laboratory facilities of the Research and Development Center were used to simulate portions of the maintenance missions. The setup included M-8 mechanical bilateral manipulators and a remote television display.

The first phase of the study dealt with the selection of two repair and two refurbishment missions from a complete listing of all NASA, unclassified military, and Comsat Corporation satellite programs. Included in the list were completed programs, programs in the hardware phase and conceptual spacecraft programs. Selections were made by assessing each satellite against a set of criteria established by the study team.

The second phase provided the design of a remote manipulator spacecraft system. The design was based on a set of requirements derived by analyzing the four selected missions. Determinations were made of characteristics such as manipulator force, torque, and reach requirements, mission duration, weight of the package containing the maintenance parts, thrusting requirements, special tool requirements, and docking equipment. Also derived from this phase were a set of satellite design practices which would facilitate future on-orbit maintenance missions by remote manipulator spacecraft.

The final phase consisted of estimating the cost of an operational, remote manipulator spacecraft system. The system included spacecraft, ground station, and factory test equipment. Development, recurring, and sustaining portions of the costs were specified. Specifically singled out was the cost of a space-qualified manipulator subsystem which represented a new major technology item.

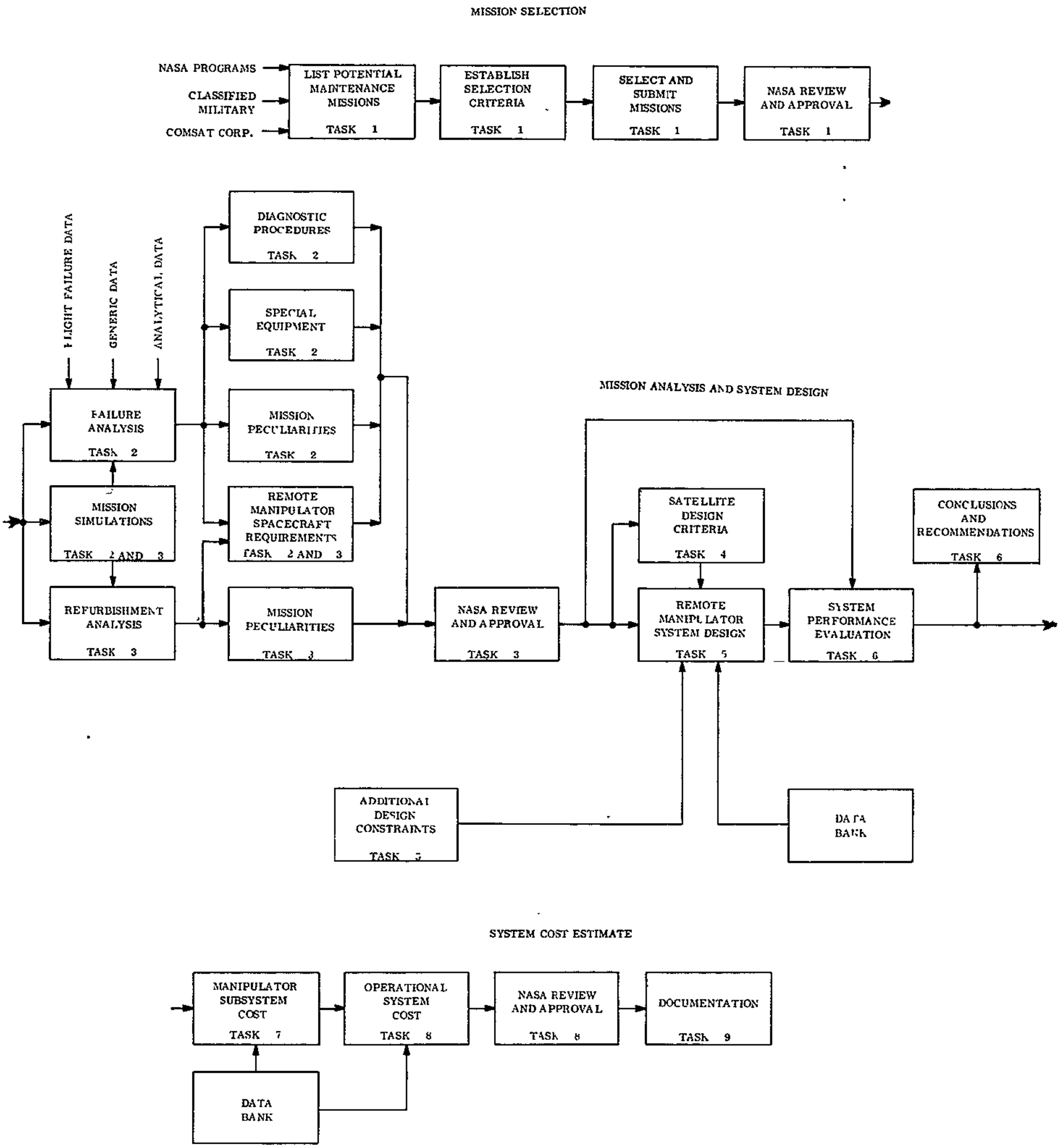


Figure 1-2. Study Plan

SECTION 2
MISSION SELECTION

Two on-orbit repair and two on-orbit refurbishment missions were selected for analysis (Table 2-1). The methodology used to arrive at these selections is discussed in this section. Where possible, actual flight failure data is used for defining repair missions. The refurbishment missions were derived by projecting the use of suitable existing spacecraft, with attendant modifications, into likely future mission applications. The methodology and selection were reviewed and approved by the NASA Mission Analysis Division.

2.1 CANDIDATE SATELLITE SYSTEMS

A complete listing of all unmanned NASA satellite programs, unclassified military programs, and other government commercial programs was extracted from References 1, 2, and 3 and is shown in Table A-1 of Appendix A. This listing includes programs that are in the early planning phase, those that are further along in development, and those already in hardware. Table 2-2 (identical to Table A-2 in Appendix A) is a reduced listing of candidate satellites.

2.2 GENERAL SELECTION CRITERIA

The satellite selection criteria used was intended to yield candidate satellite systems in low to medium orbits and synchronous orbits, as well as actively stabilized, spin stabilized, and unstabilized satellites. Each of these orbit and stabilization categories have unique problems associated with them and these problems had to be identified and examined so that an accurate set of design requirements could be defined for the remote manipulator spacecraft.

2.2.1 ORBIT

The problems of launching and rendezvousing a remote manipulator spacecraft with a satellite in a low to medium altitude orbit and a satellite in a synchronous orbit are different and this influences the design of many of the satellite housekeeping subsystems. For example, the propulsion subsystem design is affected because of the variations in tracking accuracies, injection accuracies, orbit period, and maneuvering time constraints.

Table 2-1. Satellite and Mission Selection

Satellite	Mission	Altitude (nm)	Dynamic State
FOR REPAIR			
Orbiting Astronomical Observatory (OAO)	Repair the flight failure that occurred on OAO-I	500	Tumbling at 0.5 RPM Tumbling at 34.0 RPM
Orbiting Solar Observatory (OSO)	Using generic failure rate data, apply a component failure and perform the repair	350	Spin stabilized at 26 RPM
FOR REFURBISHMENT			
Direct Broadcast Satellite (DBS) - Voice Broadcast Mission - UHF	Replace the DBS transponder with a transponder more suitable for the new mission	19,323	Actively stabilized
Nimbus	Replace the meteorological sensors with improved sensors for a new mission. Replenish the expendables.	500	Actively stabilized

Mission Analysis Summary

Mission	Duration (Minutes)	Maintenance Package Weight (Pounds)	Maximum Manipulation			Laboratory Task Simulations
			Reach (Inches)	Force (Pounds)	Torque (Inch-Lbs)	
OAO-A1 Repair	986	405	40	20	40	Yes
OSO-D Repair	265	31	40	15	40	Yes
DBS-VBM/UHF Refurbishment	494	110	40	15	40	Yes
Nimbus A-C Refurbishment	754	166	40	15	40	Yes
Nimbus D-E Refurbishment	287	1090	40	15	40	Yes

Results of Docking Analysis

Satellite	Results	Docking Procedure Comments
OAO	1. Docking would not be attempted at tumbling rates higher than 1.5 rpm.	1. Limited by operator control authority 2. Limited by potential danger due to motion of satellite spin vector in space
	2. Use of manipulator-held fluid jets to impinge and reduce satellite energy was found feasible.	1. Required large quantity of cold gas. Hot gas more attractive but may be contaminatory in some cases.
OSO	De-spinning would be accomplished with special manipulator-held and operated de-spinning devices	Figure 5-2 illustrates one of three such devices configured.
DBS	The satellite is actively stabilized and cooperative. Docking is straightforward.	
Nimbus	The satellite is actively stabilized and cooperative. Docking is straightforward.	

Table 2-2. Reduced Candidate Satellite Listing

Designation	Agency	Weight (lb)
NIMBUS	NASA	1,260
OAO	NASA	4,300
OGO	NASA	1,130
SERT II	NASA	1,080
ERTS	NASA	1,100
ASTRA	NASA	4,306
OSO	NASA	622
PEGASUS	NASA	23,100
ATS-F/G	NASA	1,843
DBS	NASA	1,000 to 3,300
DRSS	NASA	Undefined
USAM	NASA	Undefined
ATS-A-E	NASA	758
INTELSAT IV	CSC	1,200
NAV TRAFFIC CONTROL SAT	NASA	660
ATS-H-J	NASA	1,540 to 2,200
ATS-K-M	NASA	Undefined
ADVANCED SYNCH MET SAT	NASA	485 to 990

Continuous communications at low to medium altitude orbits require the use of synchronous relay satellites because of the short ground station passes (Appendix E). For a synchronous altitude mission, the remote manipulator spacecraft and the satellite to be maintained are geostationary and full time communications require only that the ground station be located within the beamwidth of the remote manipulator spacecraft antenna and, generally, a relay satellite is not necessary.

2.2.2 SATELLITE DYNAMIC STATE

Missions in which maintenance is performed on a stabilized and cooperative satellite, a spinning satellite whose spin axis is known but which has lost the ability to despin, and a randomly tumbling satellite whose dynamic state is totally unknown and which is uncooperative, place entirely different requirements on the docking process. The problem of docking with a stabilized spacecraft simply involves recognizing the points of the satellite to which the docking will be made, a simple translation maneuver until the satellite is within reach and there is nearly zero relative velocity, and finally the physical attachment. The point of docking can easily be selected so that no communications blockages occur.

The problem of docking with a spinning satellite is more complex because several measurements generally are made before any docking is attempted including determination of the satellite spin axis and spin rate. The approach to despin or dock is usually along the spin axis. This restriction may limit the times at which docking might take place because of communications outages. The process of despinning may require the use of special devices or providing an attitude control capability which would allow the manipulator spacecraft to be spun to match the spin rate of the satellite before translating into and attaching to it.

In the case of a randomly tumbling spacecraft, little is known about the dynamic state. The remote manipulator spacecraft must first determine the spin vector of the satellite and the motion of that vector in space. The docking hardpoints are next identified and then the determination that docking and stabilization are feasible without communications losses. The docking and stabilization are performed without the aid of special despinning devices because of the lack of prior knowledge of the axis about which the satellite is tumbling.

2.3 SELECTION CRITERIA FOR REPAIR MISSIONS

After arranging the selected satellites (Table 2-2) into groups by orbit altitude and dynamic state, criteria were established for selection of the most suitable repair and refurbishment missions. These criteria were:

1. Satellite cost
2. Availability of engineering data
3. Availability of reliability data
4. Satellite design life
5. Planned on-orbit or launch pad maintenance
6. Previous maintenance studies
7. Satellite program status.

These criteria were weighted so that a numerical value could be assigned to each satellite and mission selection could be made from those scoring highest. The philosophy behind the weighting system was that the satellite cost and design life were the most meaningful and measurable criteria relating to the desireability of attempting on-orbit repair. Availability of engineering and reliability data reflected the ability to adequately analyze that satellite mission. These criteria were allocated twice the weight of the others. Descriptions of these criteria and their weights are described in the following subparagraphs.

2.3.1 SATELLITE COST

The value of performing the repair mission of a satellite in orbit was measured directly by its cost versus the cost required to launch a replacement satellite. Satellite costs were obtained using a modified NASA/derived formula. The formula was an empirical fit to known program costs and used subsystem weights as the variables. The formula coefficient was modified by the General Electric Company to agree with the costs of later programs. The formula provided satellite recurring costs and it allowed the use of a common method to derive the cost data:

$$C = 0.0625* \quad \frac{W_T}{W_T - W_{P/L}} \quad (W_{TTC,D} + W_P) \qquad (2\text{-}1)$$

where

C = satellite recurring cost (in millions of dollars)

W_T = total spacecraft weight (lb)

$W_{P/L}$ = weight of payload (lb)

$W_{TTC,D}$ = weight of telemetry, command, and data processing subsystem (lb)

W_P = weight of power subsystem (lb)

The computations and results for the satellites listed in Table 2-2 are given in Appendix A.

Satellite Value	Weight (points)
> \$ 40M	10
> \$ 30M	8
> \$ 20M	6
> \$ 10M	4
< \$ 10M	2

2.3.2 AVAILABILITY OF ENGINEERING DATA

This criterion was used to define the quantity and type of data available on each of the Table 2-2 satellites and consequently the depth of analysis achievable on each. The data were gathered from sources at GE from the NASA/Mission Analysis Division, from NASA and Comsat Corporation Program Offices, and from Library searches.

Types of Data	Weight (points)
Hardware available for inspection	4
Drawings and photographs of the satellite	3
Related literature on the satellite	2
Conversations with personnel who work(ed) on the program.	1

Note that all four types of data could be available on a satellite allowing a total of 10 points.

*Original coefficient was 0.055.

2.3.3 AVAILABILITY OF RELIABILITY DATA

This criterion is similar to the availability of engineering data.

Types of Data	Weight (points)
Flight failure data	4
Failure mode and effects analysis	3
Reliability block diagrams	2
Generic failure rate data	1

2.3.4 SATELLITE DESIGN LIFE

The important criterion regarding satellite life for the repair mission is useful life after repair. However, because of the occurence of random failures this criterion is difficult to apply. The best measurement available is long design life. Repairs to random failures in systems with long design life are more likely to result in an operational system with long remaining useful life.

Satellite Design Life (Years)	Weight (points)
3.0	5
2.0	4
1.0	3
0.5	1

2.3.5 PLANNED ON-ORBIT OR LAUNCH PAD SATELLITE MAINTENANCE

This criterion was used to establish whether any effort went into the design of the satellite to facilitate on-orbit maintenance. Characteristics such as modularized design, accessibility of modules and quick connect/disconnect fittings would simplify on-orbit maintenance tasks. Launch pad maintainability requirements also have many of these ingredients. It was desirable for purposes of comparison to examine maintenance missions involving satellites designed with some degree of on-orbit or launch pad maintainability clearly specified against those without, but none were found.

Status	Weight (points)
Planned	5
Not Planned	0

2.3.6 PREVIOUS MAINTENANCE STUDIES

This criterion was used to define whether any studies were previously performed involving on-orbit maintenance of each of the satellites by man in an EVA mode, or any other method. These studies would, of course, be useful to this effort.

Status	Weight (points)
Performed	5
Not Performed	0

2.3.7 SATELLITE PROGRAM STATUS

This criterion was used to define whether each satellite was in a flight hardware stage or a conceptual stage. Actual hardware is a more realistic case because design tradeoffs and optimizations are made which often result in more difficult accessibility and maintenance. Conceptual satellites reflect some of the future design practices which are tending towards ease of maintenance both on the launch pad and in-orbit, but caution must be used since many times the ultimate configurations differ significantly from the conceptual configuration after design, development, fabrication, and test is completed. Since no satellites were found which met the planned maintenance criterion, it was decided to rate conceptual programs highest and completed programs lowest.

Program Status	Weight (points)
Conceptual	5
Underway: Development	4
Underway: Hardware	3
Completed	1

2.4 SELECTION CRITERIA FOR REFURBISHMENT MISSIONS

There are two categories of refurbishment: refurbishment of expendables such as gas or batteries, and refurbishment of a satellite payload with a new payload while using the existing housekeeping subsystems. The latter refurbishment is the more general case, since in most successful missions the experiment is the portion of the satellite which finally becomes obsolete. Furthermore, studies made by GE show that the housekeeping subsystems represent approximately 75 percent of the satellite cost and represent a large investment which can be further extended through refurbishment.

Replacement payloads can be installed on spacecraft which were not initially designed for the type of payload to be installed. Consequently, the criterion of satellite cost begins to lose its meaning with regard to refurbishment and is replaced by two new criteria which reflect the ability of the satellite to accommodate new payloads. These are payload power available and original payload weight. Original payload volume is a good criterion but is a difficult figure to obtain and so weight is used in its stead. Payload weight impacts on the control capability of the attitude control subsystem and thus does indicate how much in new payloads might be added without upsetting the original weight and inertia balance.

The selection criteria in-orbit refurbishment missions were:

1. Availability of engineering data
2. Satellite design life
3. Planned on-orbit or launch pad maintenance
4. Previous maintenance studies
5. Original payload weight
6. Payload power available
7. Satellite program status.

2.5 SELECTION CRITERIA WEIGHTING FOR REFURBISHMENT

The weighting for those criteria not described in Paragraph 2.3 are:

Original Payload Weight (lb)	Weight (points)
< 100	1
100 to 1000	3
> 1000	5

Payload Power Available (watts)	Weight (points)
< 100	1
100 to 1000	3
> 1000	5

2.6 SATELLITE SYSTEM SELECTION

The satellite systems listed in Table 2-2 were measured against the criteria established in the previous sections and the selections made from those scoring highest. The details of the scoring and selection process are included in Appendix A.

2.6.1 SATELLITE SYSTEM SELECTION FOR REPAIR

The Systems selected were OAO and OSO. The factors which influenced selection of OAO were:

1. The availability of flight failure data from the OAO-I mission.
2. The interest in analyzing an actual mission.
3. The desire to examine an astronomical observatory mission because of the strong likelihood of more of these missions after the OAO Program.

The factors which influenced the selection of OSO were:

1. The desire to examine the docking problem with a problem with a spinning satellite.
2. The availability of previous maintenance study data (Reference 4).

2.6.2 SATELLITE SYSTEM SELECTION FOR REFURBISHMENT

The satellites selected were DBS and Nimbus. OAO and ASTRA were ruled out because OAO was already being examined under reapir and ASTRA was another astronomy satellite. DBS was selected because it was a synchronous satellite and early in the conceptual phase. In addition, GE had performed several studies of DBS and this data was readily available to the study team.

REFERENCES (Section 2)

TRW Space Log, Winter 1967-68

2. NASA/OSSA Prospectus 1966, Appendix B

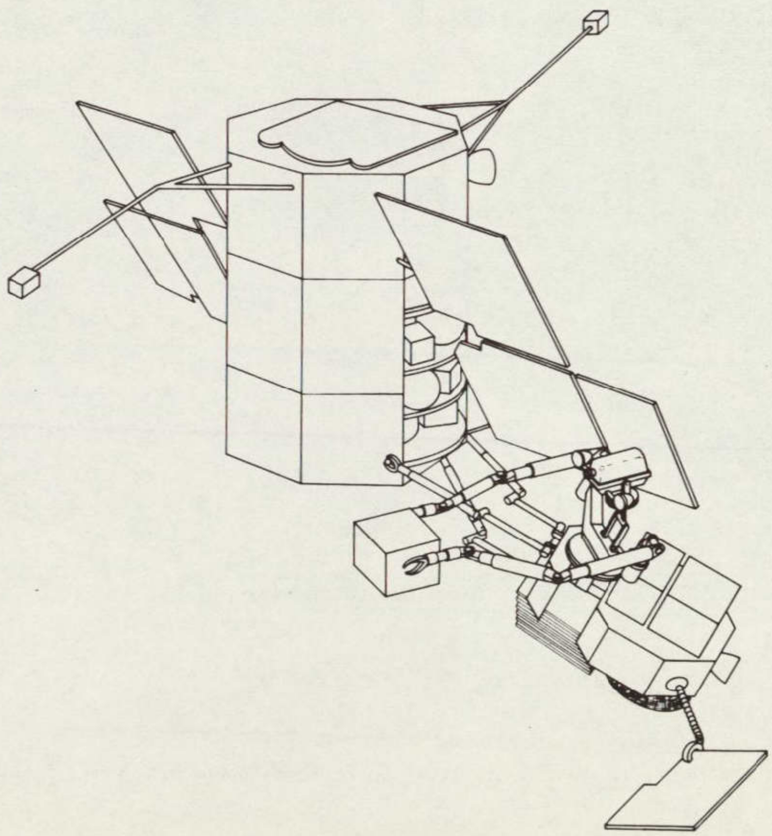
3. NASA/OSSA Prospectus 1967

4. ESMRO Study Program, Final Report, prepared by Ball Brothers Research Corporation for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center, Huntsville, Ala.

SECTION 3

MISSION ANALYSIS - SATELLITE REPAIR

. The use of a ground-controlled remote manipulator spacecraft to perform satellite repair missions selected in Section 2 was examined. In the case of OAO-A1 flight failure data was used to establish the repair tasks. This included replacement of the Battery Charger and Sequence Controller, the Spacecraft Data Handling Equipment, installing new batteries, and recharging the gas supply. For OSO, a failure mode and effects analysis was performed to define one of the higher probability of failure items. The analysis showed that the tape recorders were a weak item and therefore the replacement of a failed recorder and the recharging of the gas supply were selected as the tasks. The procedures required to correct each of these failures were defined. Laboratory simulations of key tasks were performed. Spare parts, special equipment and tools required to perform the mission were identified. The mission duration and manipulator requirements such as the number of arms, reach strength and speed were determined.



3.1 OAO SATELLITE REPAIR

The Orbiting Astronomical Observatory is illustrated in Figure 3-1. It is an accurately stabilized, unmanned platform for astronomical observations from well above the earth's atmosphere (500 n.m. apogee, 492 n.m. perigee, 100.9 minutes period, 35^{o} inclination). Of primary immediate interest is the observation of stellar radiation in the ultraviolet range which is severely limited even in balloon experiments because of absorption in the ozone layers.

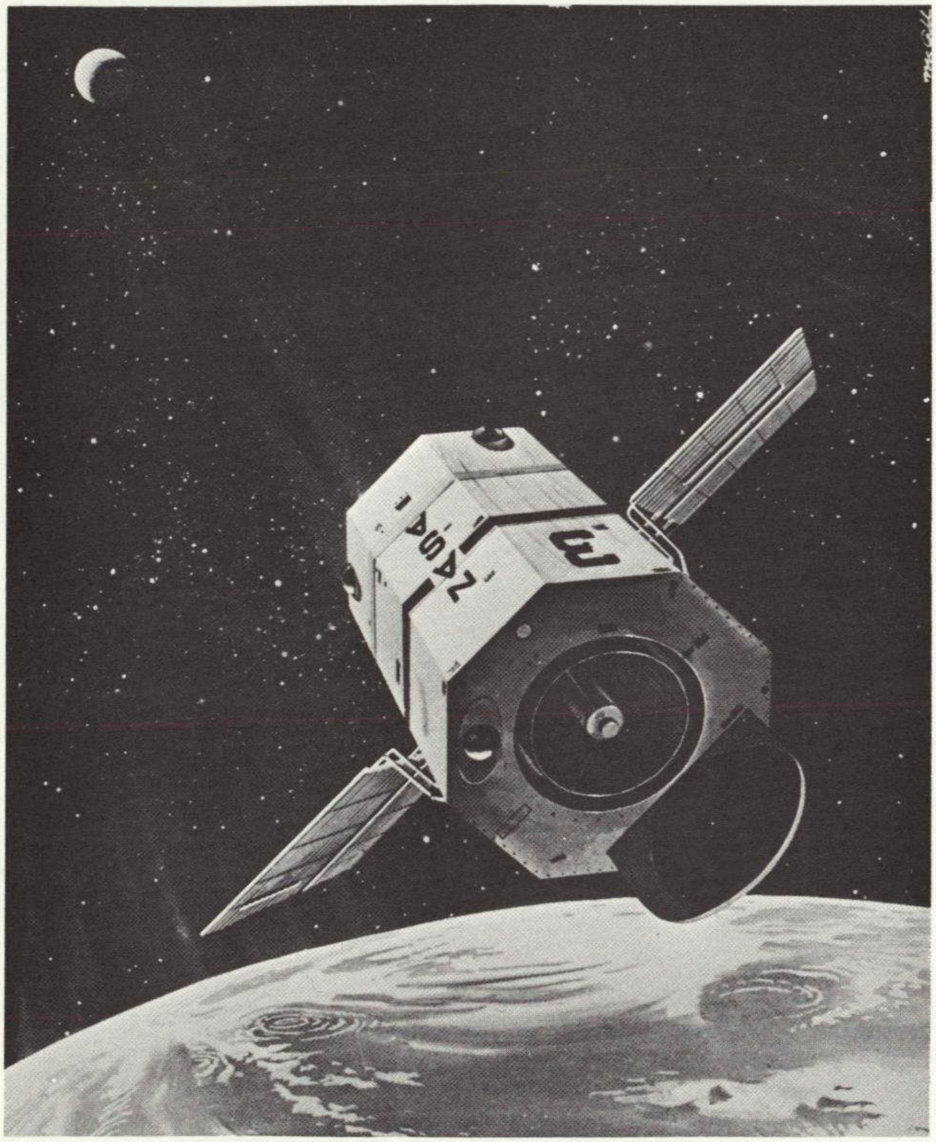


Figure 3-1. Orbiting Astronomical Observatory (OAO)

Details of the OAO-A1 mission were taken from Reference 8. Details of the satellite design, which also appear in Appendix B, were taken from References 9, 10, and 11. Reference 12 provided some data on cooperative docking concepts.

3.1.1 DESCRIPTION OF MISSION TERMINATING FAILURE

3.1.1.1 Orbital Operations

Information applicable to OAO A-1 orbital operations was extracted from Reference 1. The OAO A-1 spacecraft operated normally during the terminal countdown and launch phases. The solar paddles and booms were deployed and separation from the booster was achieved as planned. A summary of the anomalies observed at each orbital contact is given in Table 3-1. The first post-separation status data were relayed from Orroral, Australia at SET 7 (7 minutes after separation) and indicated that the OAO was performing normally in the roll search mode. At SET 8, star tracker No. 2 was commanded ON. The first indication of trouble was an apparent electrical transient which occurred at 18 seconds into SET 8. At this time, star tracker No. 3 inadvertently turned on and roll search was terminated. Subsequent tests by Grumman indicated that the transient was produced by arcing of star tracker No. 2 high voltage power supply. This, in turn caused squib firing and activation of star tracker No. 3. The current transient resulting from the squib firing caused termination of the roll search mode. The star tracker arcing also caused the Spacecraft Data Handling Equipment (SDHE) to change mode and resulted in loss of row No. 2 containing 20 spacecraft status data channels. After the transient, all equipments resumed normal operation with the only obvious permanent damage being loss of the SDHE channels.

During the contact with Rosman on the first orbit (Rosman 1), the data indicated that the spacecraft clock had been reset three times between Orroral and Rosman 1. The spacecraft was in the roll search mode, and rows No. 6 and No. 11 of the SDHE were lost presumably due to star tracker arcing. Battery No. 1 was on line and charging at a temperature of 88°]

Table 3-1. OAO A-1 Summary of Anomalies During Each Orbit

Contact	Anomalies Observed
Prelaunch	None
Orroral	At SET 8, Star Tracker 2 came on. At SET 8 minutes, 18 seconds, Star Tracker 3 came on, Roll Search Mode terminated, SDHE Mode changed, SCU went off, SDHE row 2 was lost. At 8 minutes, 40 seconds, the SCU came on.
Rosman-1	Three clock resets occurred by R-1. SDHE rows 2, 6, and 11 were lost. Star tracker 2 came on. Command memory out of synchronization with ground.
Rosman-2	Improper data storage. SDHE changed mode. Inverter switched. Tracker 6 was off; it should have been on. Battery temperature up to 103°F. SDHE not working properly. Tracker 5 was tracking earth. Tracker 2 was on; it should have been off.
Rosman-3	Tracker 5 wandering in track mode. SDHE changed mode. SDHE Row 1 was lost. RRG occurred causing clock reset. Tracker 2 came on. Battery 2 did not switch on in response to command. Battery at 111°F.
Rosman-4	Command memory stored. SDHE switched mode. Inverter switched to Backup unit. Tracker 4 came on. Battery switch command was not executed. Battery at 108°F Row 3 of SDHE lost.
Rosman-5	Tracker 4 went off. RRG occurred. SDHE row 10 was lost. Tracker 4 came on during Quito-5. Inverter switched during Santiago-5. Tracker 1 picked up a bit. Battery at 111°F.
Santiago-6	No data stored during back orbit. Tracker 4 came on. SDHE lost detail status. Battery at 119°F.
Santiago-7	SDHE mode changed. Tracker 2 was on. One gimbal on Tracker 5 had 14° error. Battery at 120°F.
Santiago-8	No data stored in Back orbit. Clock reset. Tracker 2 went off. Battery at 116°F.
Santiago-9	No data stored during orbit. Tracker 5 wandering in track mode. Battery at 133°F.
Santiago-10	No data stored during orbit. Tracker 4 off. Battery at 131°F.
Quito-11	No data stored during orbit. Trackers 2 and 4 came on. Battery at 142°F.
Rosman-13	No data stored during orbit. Battery transfer command not executed. Clock reset. Tracker 2 turned off. Tracker 5 tracking earth.
Rosman-14	No data stored during orbit. Battery transfer command not executed. Presence observed on BST under shade. Tracker's gimbal errors jumped. Battery at 158°F.
Rosman-15	Inverter switched. Battery transfer commands not executed. BST offset triggered. Clock reset. Roll search inhibited. Battery at 169°F.
Rosman-16	No data stored during orbit. Inverter switched. RRG occurred. SDHE in wrong mode. Battery at 164°F.
Rosman-17	Commands to switch Battery in dark not successful. Battery at 166°F. S/C spun up to tumbling mode.
Rosman-18	SDHE changed mode. Clock reset, secondary gas exhausted. Battery at 164°F.
Rosman-19	Battery switched to No. 2. Clock reset. Battery No. 1 dead. Battery temperature 171°F.
Santiago-20	BCSC motoring. Battery No. 2 and No. 3 voltage low. Battery temperature 174°F.
Santiago-21	No further contact.

Abbreviations

SET - Satellite Equivalent Time
SDHE - Spacecraft Data Handling Equipment
SCU - Signal Conditioning Unit
RRG - Restabilization Reset Generator Program
BST - Boresight Star Tracker
BCSC - Battery Charge and Sequence Controller

The overvoltage point for battery No. 1 was reached between Orroral-1 and Rosman-1, but the anticipated switch to battery No. 2 did not take place.

The battery temperature started to rise. It was 103°F at Rosman-2 and 111°F at Rosman-3. The Battery Charge and Sequence Control (BCSC) regulator was on at Rosman-3 due to the battery temperature exceeding 110°F. A command was issued to switch to battery No. 2, but it was not executed. During Rosman-3, Row No. 1 of the SDHE appeared lost.

At Rosman-4, the battery temperature was down to 108°F, apparently due to poor spacecraft attitude. A command was issued to switch to battery No. 3, but it was not executed. There was additional evidence of star tracker arcing and SDHE Row No. 3 was lost.

At Santiago-5, contact was in the dark. The spacecraft clock had been reset. Battery No. 1 was still on line and discharging; its temperature was 111°F. SDHE Row No. 10 was lost.

There was no further loss of SDHE channels, however, the battery temperature continued to rise and reached 169°F at Rosman-15. There were clock resets indicated at Santiago-8, Rosman-13, Rosman-15, Rosman-16, Rosman-18 and Santiago-19. Also, commands to switch off battery No. 1 were issued at Rosman-13, Rosman-14 and Rosman-15, but were not executed. During Rosman-16, commands were loaded in the memory to switch batteries in the dark, however, at Rosman-17 battery No. 1 was still on line.

During Rosman-17, real time pitch commands were issued to tumble the spacecraft in order to minimize power developed in the solar array. A tumbling rate of 5 revolutions per orbit was achieved. The plan was to alternately tumble and power up on successive contacts.

At Rosman-18, with the spacecraft still tumbling, the battery temperature held at 164°F and discharge current was high. An attempt was made to recover solar stabilization using the secondary gas supply, however, the secondary tank was exhausted and primary tank solenoid valve was shut off so recovery was unsuccessful.

At Quito/Santiago-19 battery No. 2 had come on line. Its temperature was 171°F and discharge current was high. Battery No. 1 was presumed to be dead.

At Santiago-20 both battery No. 2 and No. 3 showed on line, indicating possible motoring of the BCSC. Commands were issued and executed to turn all star trackers OFF. Voltages on batteries No. 2 and No. 3 were low and the voltage on battery No. 1 was zero. Recovery from the tumbling mode did not take place and no further data from the spacecraft was received after Santiago-20.

Throughout the mission there was occasional erratic ON/OFF operation of spacecraft components. Generally, these occurred out of line-of-sight contact with a ground station and were detectable as changes-of-state of some components (i.e., the main inverter would be "ON" at one contact and the standby inverter "ON" at the next). These erratic operations apparently produced no permanent damage except possibly the loss of the SDHE rows. Although the cause of these erratic operations was not confirmed until tests during post flight analysis, it was correctly presumed to be star tracker arcing.

There were also a number of anomalies affecting performance of the Stabilization and Control Subsystem (such as gimballed star trackers wandering in the track mode without star presences and indication of star presence while looking at earth). The effect of these problems was to reduce the control system's capability to function automatically. However, these difficulties could have been overcome through ground intervention and would not have been the cause of complete mission failure. These anomalies are summarized in Table 3-1.

3.1.1.2 Post Flight Failure Analysis

A complete failure analysis program was conducted by Grumman and cognizant subcontractors. (See Reference 1.) The program included performance of numerous tests to validate suspected failure causes. It consisted of hypothesizing causes for the anomalies observed during flight and then conducting tests to validate or disprove the therories suggested. The initial objective was to explore the OAO system for a single cause which would explain all the anomalies experienced in orbit. The analysis included reduction of A-1 flight data on a daily basis and review of previous component and spacecraft tests. A summary of the tests conducted during the post flight failure analysis program and the results of each test are given in Appendix C.

The failure analysis program resulted in the following conclusions:

a. The battery overheating was caused by the failure of a relay function in the BCSC which inhibited switching off battery No. 1.

b. Star tracker arcing did occur and was the cause of failure of the SDHE rows and other electrical disturbances.

c. Star tracker arcing was not the cause of the BCSC failure.

The power failure resulted in mission termination. The direct cause of mission termination was loss of battery power. However, this termination was a secondary failure. The primary failure was an open circuit in the K201 relay function in the BCSC which inhibited switching off battery No. 1 to one of the two other batteries. This caused overcharging and subsequent temperature rise in battery No. 1. When the temperature reached a dangerous level, the spacecraft was intentionally placed in a tumbling mode. It was anticipated that the off sun condition would reduce battery No. 1 charging and therefore lower the temperature. The temperature was to be controlled by tumbling and stabilizing the spacecraft on alternate orbits. While in the tumbling mode, battery No. 1 became heavily discharged. Battery switching eventually occurred but battery No. 1 had already become depleted. Batteries 2 and 3 came on but they were not fully charged and were at a high temperature. They soon were depleted. Recovery did not take place.

An investigation of the effects of star tracker high voltage supply arcing was also conducted. Arcing resulted in loss of half of the analog channels in the SDHE and caused a number of other spacecraft malfunctions. However, these effects were not mission terminating.

The criticality of each of the hardware items is indicated by the reliability block diagram for the OAO A-1 spacecraft functions given in Figure 3-2. The diagram indicates requirements for complete mission success, and shows redundancy at the component level only. Many of the components have internal redundancy at the piece part or subassembly level and there are many failure modes which would result in degraded performance rather than complete mission failure. However, the OAO system cannot operate without battery power and the loss of the three batteries is a mission terminating failure. The failure of the BCSC which controls battery charging must also be considered a critical failure. The loss of half of the SDHE channels, although not itself a mission terminating condition; does make the SDHE a candidate for replacement.

It is therefore concluded that restoration of the spacecraft to an operational condition requires at least the following maintenance actions:

a. Install new batteries

b. Replace the BCSC

c. Replace the SDHE

d. Recharge the gas supply

These are the general tasks that would be performed by a remote manipulator before undertaking the mission, however, alternative repair actions and potential secondary effects, particularly those due to complete loss of spacecraft power were examined.

3.1.2 DISCUSSION OF POTENTIAL PROBLEMS, SECONDARY EFFECTS, AND CASCADING FAILURES

3.1.2.1 Temperature

Previous studies performed by Grumman (Reference 2) indicated that the estimated equilibrium temperature of the OAO in orbit after a power failure was $-95^{\circ}F$. This assumed random tumbling with respect to the sun. The time required to reach the equilibrium temperature was estimated for typical OAO equipments based on an operating temperature of $100^{\circ}F$ at the time power shutdown occurred. The Grumman report also indicated a

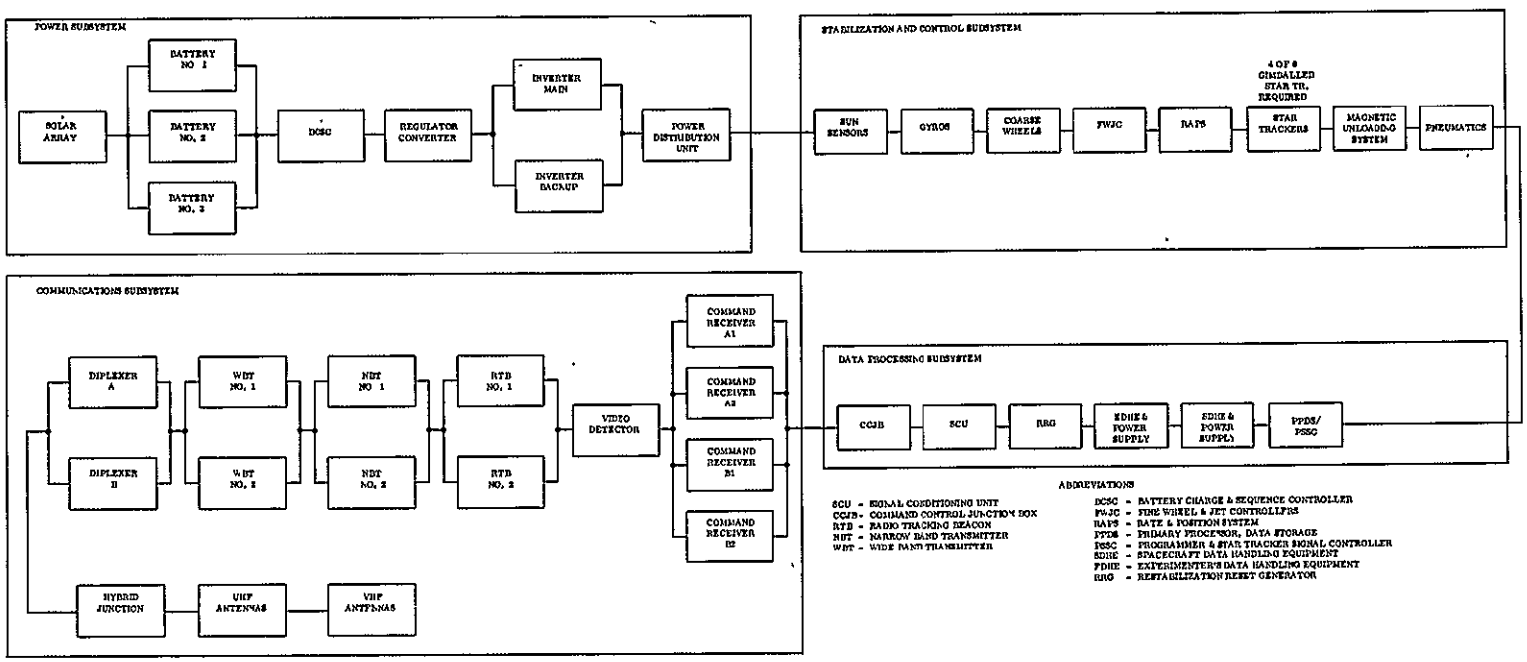


Figure 3-2. OAO Reliability Block Diagram

possible problem with operation of transistors below $-70^{\circ}F$ ($-57^{\circ}C$) and concluded that the OAO spacecraft must be reached for repair within two days to ensure the success of the repair mission.

In general, transistor specifications require satisfactory performance down to $-85^{\circ}F$ ($-65^{\circ}C$). GE has performed extensive testing of transistors down to $-102^{\circ}F$ ($-75^{\circ}C$) and found that no damage was incurred at this temperature (Reference 3). The devices tested included types used on the OAO Stabilization and Control Subsystem. In addition, recent tests were performed by GE on some power transistors at temperatures below $-200^{\circ}F$. It is therefore reasonable to expect that the equilibrium temperature of $-95^{\circ}F$ would not be a problem with respect to transistor operation. It is possible that some circuits will not perform satisfactorily immediately after application of power. However, normal operation should resume when the equipment warms up to within the design temperature tolerances.

Reference 4 also stated that several units in the OAO data processing system were designed for $0^{\circ}F$ to $-22^{\circ}F$. The limitation on these units is the ability of the magnetic storage device to withstand the extremely low temperature. This condition would impose a requirement that power be restored to the spacecraft in less than two days after mission termination.

The solder used for wiring connections was also investigated as a potential problem under the equilibrium low temperature conditions. The following quotation is excerpted from Reference 4:

> "Recent work has shown that the possibility exists of the allotropic transformation of tin (tin disease) taking place at low temperatures in solders containing from 40 to 70 percent tin. In order to prevent the transformation, solders should contain antimony or bismuth in quantities of 0.2 to 0.5 percent".

The solders used in the manufacture of spacecraft are generally procured to Federal Specification QQ-S-571 and are usually type Sn60 (60 percent tin). Appendix C shows the composition of solder-alloys as obtained from QQ-S-571.

The QQ-S-571 solders containing from 40 to 70 percent tin have 0.20 to 0.50 percent antimony and up to 0.25 percent bismuth. Therefore, no difficulty with tin disease is anticipated.

3.1.2.2 Cold Welding

Reference 5 indicated that cold welding may be a problem when bolts or screws are rotated for removal of equipment in the space environment. Cold welding can occur between metallic surfaces under high vacuum condition. The conditions conducive to the adhesion process are generated when the fasteners are rotated, causing the removal of the oxide films, and permitting metal to metal contact before these oxide films can reform. Experience has shown that cold welding of fasteners is not a problem under ground assembly conditions where there is an ample supply of oxygen to reform the oxides.

Reference 6 indicates that the expression for coefficient of adhesion σ is:

$$\sigma = \sigma_o t^n e \, (-Q/RT)$$

where

σ_o, Q, R, and n are constants for a particular configuration.

T = temperature (^oK)

t = time

The above equation shows that the tendency to cold weld decreases with decreasing time and temperature. In considering repair of OAO A-1 in the space environment, several factors act to reduce the likelihood of cold welding. First of all, the equilibrium temperature is low (-95^oF) which reduces the adhesion properties.

Secondly, the time during which pressure is applied can be kept low, which also reduces the cold welding effect.

3.1.2.2.1 Battery Recharging/Replacement

A study was made of the possibility of recharging rather than replacing the existing OAO A-1 batteries. Connections to the batteries could be obtained through one of the BCSC connectors, and recharging could be accomplished during replacement of the BCSC. The following limitations exist:

a. The battery temperature exceeded 170°F and it is likely that they experienced some permanent damage. A graph of battery temperature rise is given in Figure 3-3. Nickel cadmium batteries are normally designed to operate up to 100°F and the maximum temperature for satisfactory performance is 120°F.

b. The battery temperature should be no lower than 0°F charging. Generally, they will not accept a charge when at a temperature below -25°F. The 0°F criteria would required that recharging begin no later than two days after experiencing the power failure.

c. The charging voltage required is approximately 33 percent higher than the normal battery voltage. This would require carrying aloft a special battery, or providing special interconnections for the potential replacement battery in order to provide an adequate charging voltage.

e. Some means for disconnecting the spacecraft load from the battery would be required while recharging the existing batteries.

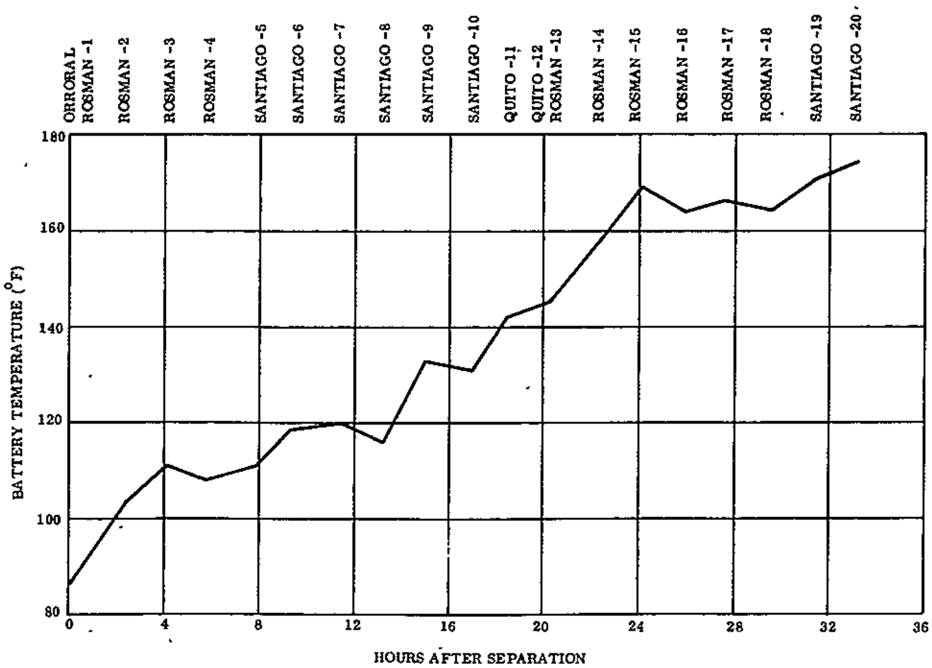


Figure 3-3. Battery Temperature Rise

The considerations imposed on recharging the existing batteries imply a high risk which could jeopardize the repair mission and the batteries should therefore be replaced. In this case, replacement does not imply physical removal of the existing batteries. The replacement batteries can be mounted at any suitable location and connected through the four connectors to be removed from the existing battery packs.

3.1.2.3 Star Tracker Exposure to Sunlight

The star tracker external sun shutter was designed to fail open in the event of a power failure. However, the photosensitive element is protected by a second shutter located behind the optics. This latter shutter is self-actuated when the field of view approaches the sun. The shutter is made of thin metallic material, however, and cannot withstand extensive exposure to the sun.

At the time of failure, it was assumed that the spacecraft was tumbling at a rate of 5 revolutions per orbit, and it is anticipated that his tumbling rate does not change significantly up to the time that repairs are started. Under these conditions, the one degree field of view of the star tracker will be exposed to direct solar flux for approximately 5 seconds if the field of view intersects the sun. Considering the spacecraft equilibrium temperature of $-95^{o}F$, a 5 second exposure should not damage the startracker shutter. The tumbling mode results in a random exposure of the six star trackers to direct sunlight, and the likelihood of the same tracker being exposed a sufficient number of times to burn through the self-actuated shutter is negligible.

In order to protect the star trackers from extensive exposure to the sun during stabilization of the spacecraft by the manipulator, the star trackers should be shielded from direct sunlight. Covers or tape can be applied by the manipulator as an initial task and for this purpose it is desirable that rendezvous take place in the dark.

If a single star tracker is found to be inoperative, the system can be considered degraded but still functional. In fact, basic system performance can be achieved with three of the six star trackers.

3.1.2.4 Exposure of Experiments to Sunlight

The experiment optics are protected from the sun by a shutter mounted on the top of the spacecraft. Grumman personnel have advised that this shutter was never opened during the OAO A-1 mission. There is therefore no danger of exposure to the sun during spacecraft tumbling.

3.1.2.5 Effect of Star Tracker Arcing

The effects of star tracker arcing were the loss of SDHE rows and spurious erratic operation of spacecraft components.

A total of six SDHE rows were lost. The last one to fail was detected at Santiago-5. There were no further rows lost or other permanent damage during the remaining 15 orbits even though there was further evidence of arcing throughout the mission. It is possible that arcing had abated with decreased degassing during extended time under high vacuum conditions. In any event, satisfactory control of the spacecraft could be accomplished through ground control.

3.1.3 MAXIMUM ALLOWABLE TIME TO BEGIN REPAIR

The repair actions which are being considered are predicatedon spacecraft conditions of no power, tumbling mode, and an equilibrium temperature of $-95^{\circ}F$. Under these condition, there is no limitation with respect to maximum allowable time to begin repair, with one exception. If it is necessary to keep the data processor temperature within the design limits of $0^{\circ}F$, then power must be restored in less than two days. Since this response time may not be practical, it would be necessary to also consider replacement of the data processor units.

3.1.3.1 General Comments

Many of the persons contacted during the investigation, including Grumman personnel at Goddard and technical specialists at GE, expressed various degrees of pessimism regarding the ability to reactivate the spacecraft. Most of the doubts centered around the effect of the low equilibrium temperature resulting from loss of power ($-95^{\circ}F$), and the possible existence of undetected secondary failures. Known problem areas have been investigated. However, detailed analyses and/or tests to validate equipment performance under extremely low temperature conditions, and the investigation of all possible secondary failures, is considered beyond the scope of this study. Since the spacecraft was not designed to survive a loss of power during orbit, there is some risk in assuming that it can be restored to operation by replacing the known failed times.

A review of the status data obtained during the flight indicates that a remote manipulator spacecraft repair mission launch would not have been practical within a few days after the OAO A-1 mission termination. The post flight analyses and tests required to identify and validate the causes of flight anomalies took approximately 18,000 engineering man hours and more than two months time by Grumman and cognizant subcontractors. The degree of confidence in the correctly presumed causes of flight anomalies increased as the post flight analysis progressed. However, the exact point at which the estimated risk would have been low enough to initiate the repair mission is highly subjective. It is estimated that this time would be a minimum of six weeks and possibly as much as twelve weeks after the OAO A-1 mission termination. This delay is due to the inability to directly isolate the failure causes from spacecraft data, necessitating extensive post flight analysis and validation through laboratory simulation.

The launch of a remote manipulator spacecraft to diagnose the causes of failure could have been made at mission termination. Data derived from this source could have provided earlier failure isolation and repair definition and a consequent earlier repair launch, but additional costs would be involved.

3.1.4 SATELLITE STATE

The state of OAO-A1 after the failures occurred was examined to determine the steps necessary to disable it and dock.

3.1.4.1 Disabling/Enabling

Before the remote manipulator spacecraft attempts to dock to the slowly tumbling OAO-A1, the OAO-A1 must be disabled either through subsystem failures, gas and power depletion, command instructions, etc. An analysis of Reference 1 indicates that at mission termination, the primary and secondary attitude control gas valves were closed. Furthermore, it is believed the secondary gas supply was depleted and the spacecraft gyros had slowed down appreciably due to the power failure. It was concluded that the OAO-A1 attitude control system would not react to the docking action of the remote manipulator spacecraft.

No action was taken to command off the spacecraft power subsystem, to interrupt some of it at the umbilical connector, or to disconnect the power either at the Battery Charge and Sequence Controller (BCSC) or at the batteries themselves because the batteries were in a weakened condition. Nevertheless, the array power and residual battery power could damage the remote manipulator spacecraft. Thus, the OAO-A1 overall repair procedure was:

a. Remove the BCSC unit first in order to disconnect the batteries and paddles from the spacecraft electrically. Connector caps carried in the Remote Manipulator Spacecraft supply-bin could be used to cap the array cable connectors.

b. Mount the new batteries externally.

c. Recharge the nitrogen gas supply.

d. Replace the Spacecraft Data Handling Equipment (SDHE) module.

e. Mount the replacement BCSC.

3.1.5 DOCKING TO TUMBLING OAO

The problems associated with docking a remote manipulator spacecraft with another orbiting vehicle were studied. The OAO-A1 spacecraft was selected as the candidate vehicle for the following reasons:

a. OAO-A1 tumbling rates are unknown.

b. OAO-A1 tumbling axis in body coordinates are unknown.

c. OAO-A1 tumbling axis in inertial coordinates are unknown.

d. OAO-A1 after-flight failure is an uncooperative vehicle.

e. OAO-A1 has externally mounted solar paddles and balance booms that are difficult to avoid and maneuver around.

f. OAO has only a limited number of hard points for docking.

g. Balanced moments of inertia (1300 slug $\cdot$ ft^2) reduce the chance of OAO spin stabilization about a "preferred" axis.

h. Because of the fragility of OAO skin, solar paddles, and booms caution and care are required in docking.

Thus, the failed OAO vehicle requires a docking maneuver that is both precise and delicate, while it swings a pair of lethal clubs (the balance booms) at the remote manipulator spacecraft.

Maintaining wideband communications is most important during the docking phase when the two vehicles are in close proximity. Therefore, this portion of the analysis assumes that the remote manipulator spacecraft is in constant communication with a synchronous-altitude data-relay satellite from the time docking maneuvers are initiated until physical contact and latching have taken place. To reduce the communications problem, the remote manipulator spacecraft may have to be configured to provide a stabilized antenna capable of pointing at the data relay satellite. If handover (from one data relay satellite to another during the docking phase) is discarded as to hazardous, then another constraint upon the docking maneuver is that it be completed in approximately 65 minutes; this is the time available for communication between a synchronous altitude satellite and the OAO at an altitude of 435 nautical miles and inclination of 28.5 degrees. The transmission path is shown in Figure 3-4. If the ascending or descending node of the OAO vehicle is not beneath the data relay satellite, the communications time will be slightly longer.

3.1.5.1 Preliminary Docking Maneuvers

When the remote manipulator spacecraft and OAO orbits are synchronized to within approximately 50 feet and the remote manipulator spacecraft is locked on to the data relay satellite, it will maneuver about the OAO to inspect the external condition and look for failures or changes in the vehicle configuration. This will include determining that the solar paddles and boom assemblies are properly locked in place, that the sun shade covering the telescope assembly is still latched, and that skin panels are all in place. The inspection maneuver will require that the remote manipulator spacecraft maintain a position for communication, while observing the tumbling OAO. The communication requirement will generally result in maintaining the remote manipulator spacecraft above and to one side of the OAO flight path. This position produces a viewing direction towards the earth's North or South Polar region and will prevent the remote manipulator spacecraft imaging system from accidently observing the sun, and it provides the operator with earth's horizon as an attitude cue. The remote manipulator spacecraft will maneuver regularly to maintain this position. Thrusting will most often be required in the crossplane direction, but maintaining the separation at less than 50 feet represents an insignificant portion of the mass expulsion system capacity.

The OAO tumbling axis and tumbling rate must be identified during the inspection period. The tumbling state of the OAO-A1 spacecraft was not determined after the initial positive pitch

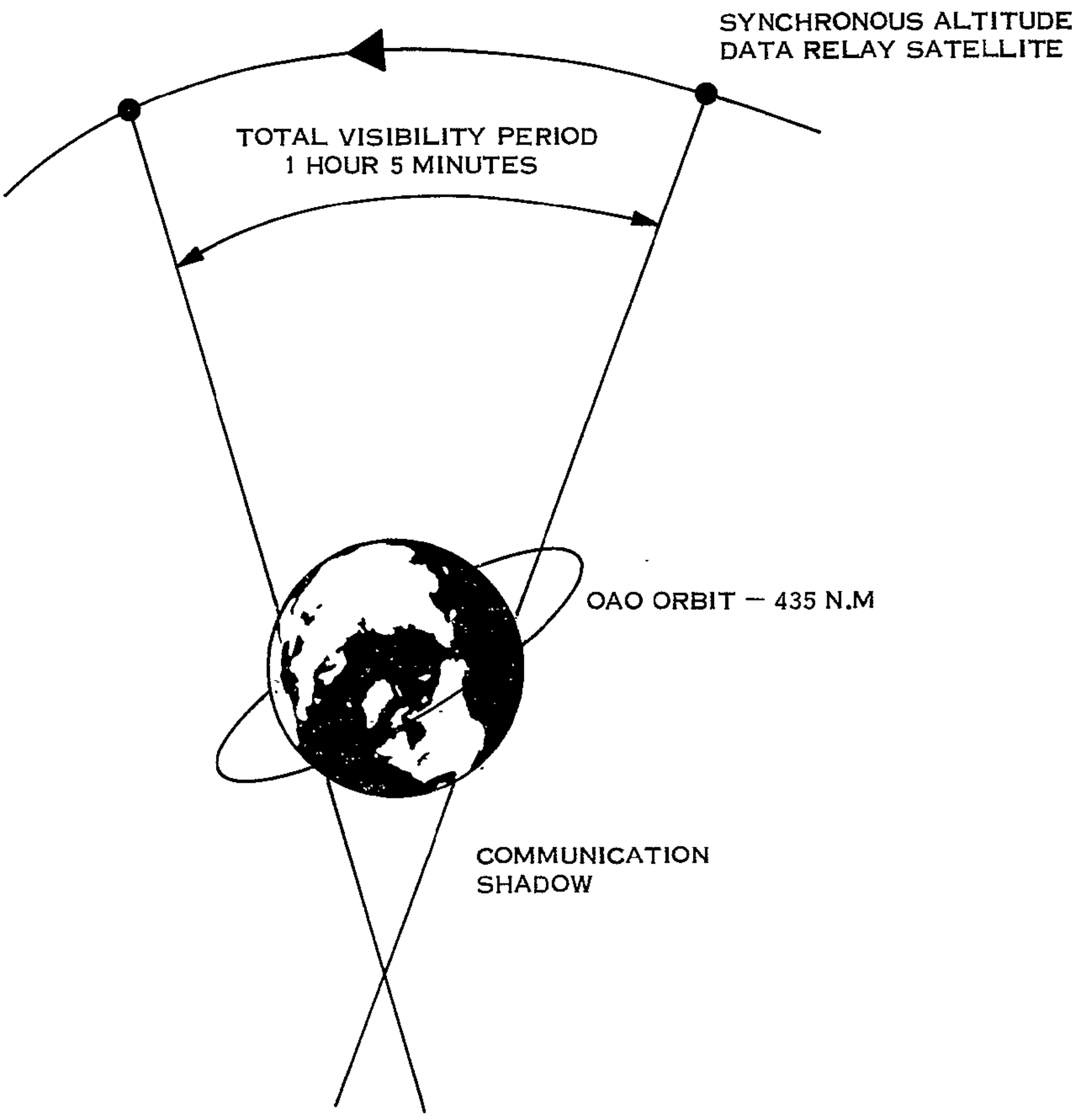


Figure 3-4. OAO, Synchronous Satellite Geometry

tumbling command was sent at Rosman-17, nor has it been positively determined from flight data. However, a nominal positive pitch tumbling rate of 0.30 degree per second appears reasonable after a review of the flight operations, as well as a potential roll rate of 0.25 degree per second. A review (Para. 3.1.1.1) of the last few active operational orbits of OAO-A1 reveals that:

a. At Rosman-17, the OAO-A1 was spun up about the pitch axis by eight manual pulses on the secondary high thrust jets. A Grumman report on OAO-A1 orbital operations states that the OAO was tumbled into a "random" orientation with respect to the sun. This does not necessarily mean random tumbling. It is assumed that the roll and yaw rates were zero because at Rosman-16 the OAO was still in the initial stabilization mode, with roll search inhibited, and the rates were controlled prior to the positive pitch spin-up of approximately five revolutions per orbit (equivalent to 0.30 degree per second). Nevertheless, very small roll or yaw rates could occur because of cross coupling from the pitch spin-up, and a failure could have commanded a full roll search rate of 0.25 degree per second. The roll search rate would be superimposed on the 0.30 degree per second pitch rate.

b. At Rosman/Quito-18, it was planned to recover solar stabilization by a computer progammed sequence; however, the program was not ready and manual entry was scheduled. The Rosman contact was short, but the realtime commands continued to inhibit roll search. The realtime commands were confirmed; solenoids were energized but no gas flowed because the primary solenoid valve was closed and the secondary high thrust tanks had been exhausted. It is assumed that OAO-A1 continued to tumble at approximately 0.30 degree per second when it left Rosman/Quito-18.

c. A few days after launch, visual reflections to a tracking station were timed and recorded, but the tumbling rate could not be determined because of the many highly-reflective surfaces (some of which are evenly spaced) on the OAO-A1. It was felt the same difficulties would be encountered with new visual data.

Tumbling of the OAO-A1 vehicle continued for the rest of the operational mission, which was terminated at Santiago-21.

Docking with OAO-A1 would be straightforward and unimpeded for either the pitch or pitch and roll axis tumbling conditions. The maximum tumbling rate of one revolution every 15.4 minutes means that the tip velocity of the balance booms is on the order of only 0.1 fps and that the centrifugal acceleration is only 2×10^{-5} g. The remote manipulator spacecraft would have response characteristics enabling it to maneuver around the rotating OAO and dock with it without fear of being struck a damaging blow by the balance masses or encountering centrifugal accelerations greater than could be produced by the thrusters. The docking problems that may occur become somewhat clearer if larger rotational rates are assumed.

A double failure that combines a failed-open high level thruster and a power or command failure could allow the entire supply of nitrogen to flow and spin up the vehicle. If the failures occurred on full tanks, the 24 pounds of nitrogen would produce an angular impulse of 4600 fps, and a 1300 slug-ft^2 OAO would develop an angular velocity of 34 rpm. With the OAO rotating once every two seconds, docking becomes a significant problem. The spatial relationships between the OAO, remote manipulator spacecraft and Data Relay Satellite must be set so that docking maneuvers can be started near the beginning of a communications cycle; this will provide most of the 65-minute communications period for the docking maneuver. Assuming that multiple data relay satellites are available, one will be chosen offering the best communication link geometry during the critical time that initial docking contact is made.

The video link will be used to determine the axis of rotation and the rotation rate. The rotational axis can be established using the earth as a visual reference with sufficient accuracy to plan the positioning of the remote manipulator spacecraft relative to OAO for best communications.

The location of the rotational axis relative to the OAO body axes will also be established to determine if a solar paddle or boom will obstruct the docking maneuver. For instance, if the OAO were rotating at a high rate and the rotational axis happened to lie in the same plane as the solar paddles, then the solar paddles would be in a way of any maneuver that attempted to produce docking at the rotational axis. Should this interference occur, the remote manipulator spacecraft has only to wait until nutational effects rotate the OAO around so that the solar array is no longer an obstruction.

The balanced inertias of the OAO prevent determination of a preferred spin axis prior to launch of the remote manipulator spacecraft, whereas a vehicle with unbalanced inertias should align the maximum moment of inertia axis with the angular momentum vector. The period of time required for alignment to occur is a function of the initial misalignment and of the damping caused by structural bending and fuel slosh. Vehicles with preferred spin axis may require special tools to permit docking, but the docking procedure is still simplified because the tumbling conditions can be anticipated.

After the OAO rotational axis is established, the rotational rate can be determined by using the video system's frame rate to determine the time required for one revolution. Because the video frame rate will will be 10 frames per second and because the OAO spacecraft vehicle's maximum rate is slightly in excess of one-half revolution per second, there is no incompatability between the two frequencies, and there is no chance of miscalculating the angular rate. There is an added safety factor produced because of movement of the OAO or any other vehicle must agree with the angular rate calculated by observing the motion from frame to frame.

3.1.5.2 Docking Procedures

There are two methods available for directly docking the remote manipulator spacecraft with another rotating vehicle. The first method is to approach the target vehicle along a path that is normal to the spin axis and then to synchronize the remote manipulator spacecraft to a convenient docking hard-point on the periphery of the target vehicle by continuously accelerating the remote manipulator spacecraft in a circular path until attachment can be accomplished. This method is straightforward and its advantage is that the operator uses the video link to determine errors in a familiar "chase" maneuver. This docking technique, herein referred to as "velocity matching", offers a good solution for slowly tumbling vheicles. Two disadvantages of this technique are:

1. The docking and stabilization must be accomplished in less than one-half of a revolution of the target vehicle in order to maintain communications.

2. The operator(s) controlling the remote manipulator spacecraft must command simultaneous rotation, translation, and gripping maneuvers.

The second docking method would allow the remote manipulator spacecraft to dock with vehicles rotating at higher rates. Instead of a chase maneuver and docking at the periphery of rotation, docking can be accomplished by spinning up to match the target vehicle and then docking at the axis of rotation. This method will be referred to as "rate synchronizing. The advantages of rate synchronized docking include the ability to dock with vehicles rotating at high rates, because a constant acceleration is not needed and the maneuver allows the operator(s) to separate the translational maneuvers from the rotational and gripping maneuvers. As in the velocity matching technique, the operator would first rendezvous with the target vehicle, roughly determine the vehicle's spin axis and rate, and then determine the proper docking geometry, taking into consideration the location of the Data Relay Satellites.

When the communications requirements are met, the operator would spin up the remote manipulator spacecraft to match the target vehicle's rotational axis, and rotational rate, and then translate to the target vehicle and dock with it by deploying manipulators and gripping mechanisms. The disadvantages of this method include problems associated with maintaining a communication link with a data relay satellite while spinning up and the alignment tolerances that must be maintained on the remote manipulator spacecraft among the thrusters, body axes, and moments of inertia to ensure a minimum of cross coupling so that the operator can produce pure rotational or pure translational motions. Finally, there is the problem of establishing the initial alignment so that the rotational axes of the two vehicles start out very closely aligned and have no significant drift rate.

Because multiple data relay satellites have been assumed, the communication problem can be solved by picking the best geometry between target vehicle spin axis and data relay location. Ideally, docking occurs when the remote manipulator spacecraft spin axis points at the relay satellite, but this is not likely to occur. Therefore, the remote manipulator spacecraft would have to be designed as a two-body spinner with the despun portion carrying an antenna that can be pointed at the relay satellite or with a design antenna. Figure 3-5 is a diagram of the relationships between the tumbling OAO, the remote manipulator spacecraft, the Data Relay Satellite. In the figure, the orbit is greatly exaggerated for clarity,

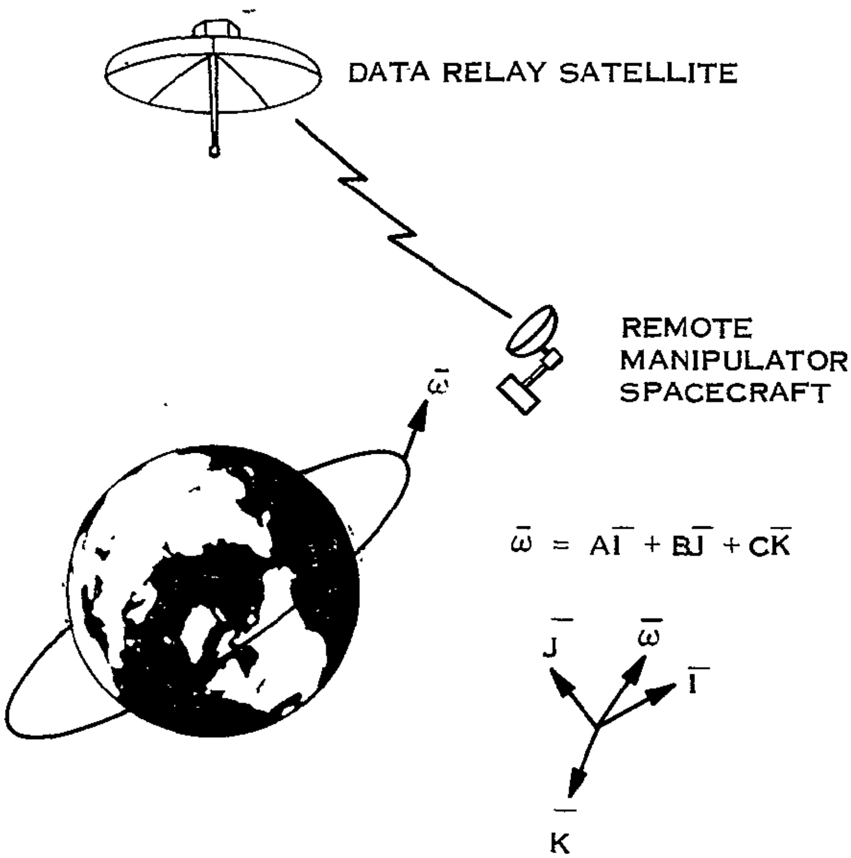


Figure 3-5. Relationship Between OAO, RMS, Data Relay and Angular Momentum Vector

and only one data relay satellite is shown even though there may be three or four. A representation of the spinning remote manipulator spacecraft is shown with a despun antenna pointed at the data relay.

A vector representing the angular momentum of the tumbling OAO can be described in the inertial coordinate set $\bar{i}$, $\bar{j}$, $\bar{k}$. The OAO spin vector can be represented by:

$$\bar{\omega} = A\,\bar{i} + B\,\bar{j} + C\,\bar{k}$$

From Figure 3-5, it can be seen that docking is simplest if A and C both equal zero. In this instance, despinning the antenna is unnecessary. However, if there is a large component of the spin axis directed along $\bar{k}$, docking will be conducted more simply if one of the other relay satellites is used. It can also be expected that C will have a non-zero value and the rotational axis will have a component in the $\bar{i}$ direction. The remote manipulator spacecraft must still be spun up with its spin axis aligned to the OAO spin axis, and the antenna is then directed towards the data relay. Regardless of the direction of $\bar{\omega}$, the angle between the spin axis and the relay can be made equal to or less than 90 degrees at the time of docking.

The difficulty of achieving an accurate alignment of the spin axes of the two vehicles depends upon the remote manipulator spacecraft configuration. Ideally the remote manipulator spacecraft spin axis will be aligned with the other vehicle's spin axis prior to spin-up. The operator must achieve alignment by observing the video image; it can be done most accurately if the optical axis is placed on the remote manipulator spacecraft spin axis and cross hairs are provided. Barring this, the alignment will have to be trimmed as the remote manipulator spacecraft is spun up. One other significant problem is the effect of the transmission time delay in the control loop. Delays greater than 1/2 second can occur; at 30 RPM 1/2 second is equivalent to 90 degrees of rotation. Once a steady dynamic state is achieved between the remote manipulator spacecraft and the other vehicle, this problem will be somewhat alleviated. Predictive displays may be useful for this condition. Although an operator can be trained to accurately operate a control loop with delays greater than 1/2 second, the ability of an operator to align the remote manipulator spacecraft at high RPM will have to be tested in a simulation to prove that with training an operator can master the required maneuvers.

3.1.5.3 Tumbling Rate Constraints

The maximum OAO tumbling rate that can be handled with the velocity matching technique will depend upon the remote manipulator spacecraft acceleration capability. Furthermore, the remote manipulator spacecraft acceleration capability must be based upon the control authority that can be used safely and satisfactorily by a remote operator attempting to gently dock with a large optical space systems. Handling studies for spacecraft docking that were conducted by Boeing (Reference 7) are represented in Figure 3-6. The analysis used this figure as a guide and assumed that a translational and rotational authority of 0.01 g and 3 deg/sec^2 represents an appropriate system repsonse. For the velocity matching maneuver, the OAO balance booms offer a good attachment point. For establishing angular rate limits, the inner and outer ends of the balance boom will be used. The inner or body end of the balance boom requires generation of less acceleration to overcome centrifugal force; therefore, represents the highest OAO angular rate that the remote manipulator spacecraft can dock to using the velocity matching technique. The angular rate for docking at the tip end must be lower, so that it represents a more conservative value.

The maximum tumbling rate for velocity matched docking is expressed by:

$$\omega_{MAX} = \frac{60}{2\pi} \cdot \sqrt{\frac{F}{r}} \text{ rpm}$$

where:

F = Control authority acceleration in ft/sec^2

r = Radial distance from OAO rotational axis to docking position.

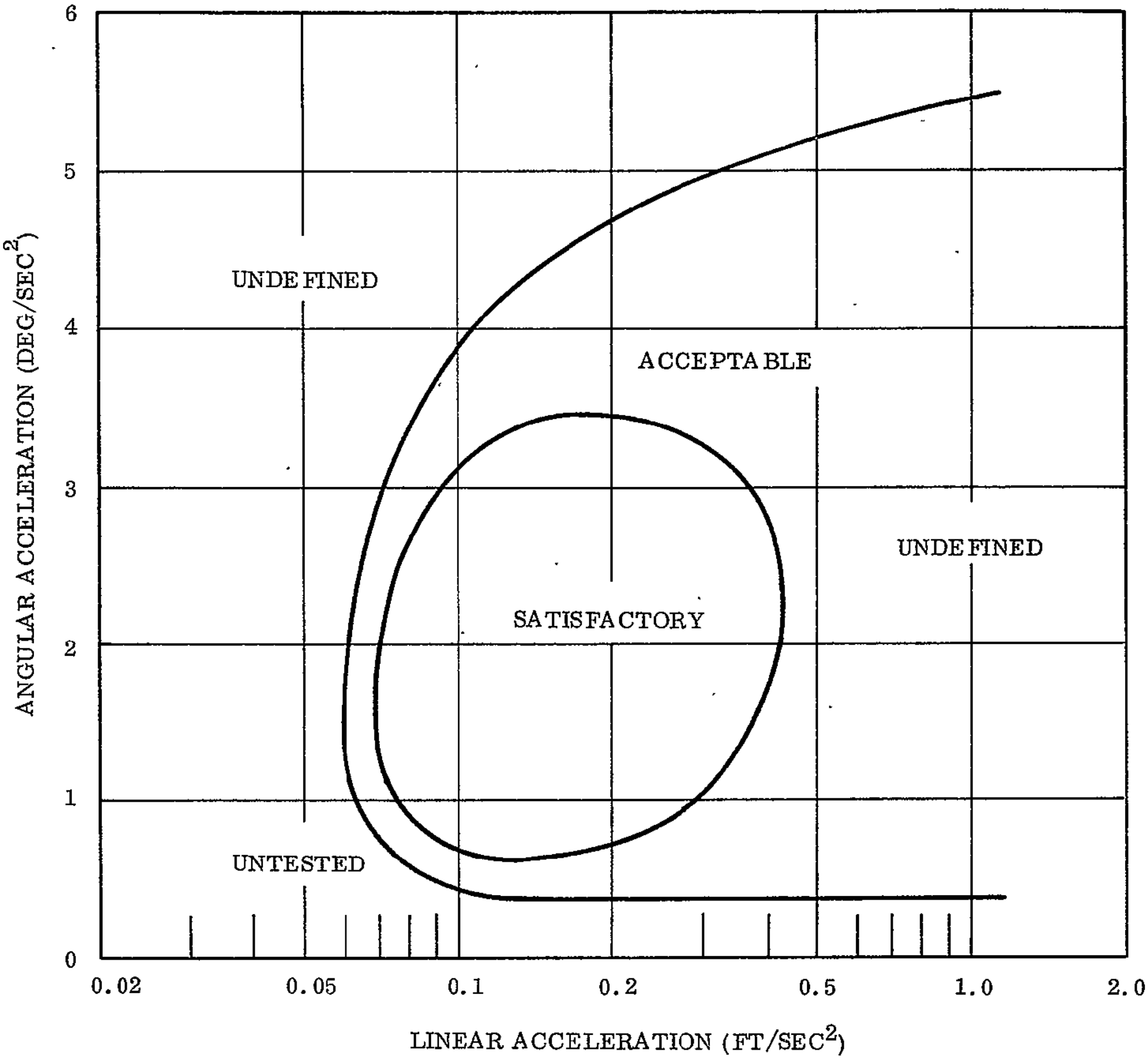


Figure 3-6. Required Docking Authority

With the indicated control authority acceleration of 0.322 ft/sec^2, the maximum tumbling rate permissible for docking at the base of the OAO balance boom (r = 6') is 2.3 rpm, and the maximum tumbling rate for docking at the tip of the OAO balance boom (r = 16') is 1.3 rpm. Both of these values are well above the probable tumbling rate of OAO-A1, but well below the OAO's 34-rpm capability.

The maximum tumbling rate at which the remote manipulator spacecraft can attempt a rate synchronized docking maneuver depends upon characteristics of both the remote manipulator spacecraft and the target vehicle. For the remote manipulator spacecraft limits will be set by the ability of the manipulators to function under centrifugal acceleration. For the target vehicle, limits will be set by the stability of the rotating or tumbling axis.

Present day bilateral electric manipulator technology is capable of producing a manipulator that can be extended in an outstretched position and support a weight one quarter of its own weight. (A manipulator that weighs 100 pounds can raise 25 pounds.) This means that the manipulator produces sufficient torque to just raise its own mass to an outstretched position in a gravity or acceleration field equivalent to 1.5 g. The value 1.5 g is based on a manipulator configuration where the center of mass is at the midpoint and the weight is held at the manipulator's extremity. Future manipulators are expected to be able to lift their own weight, meaning that they will be able to just function in a 3-g field. For this analysis, 1.5 g is considered to be the appropriate limiting condition.

The establishment of a rotational rate limit based upon the 1.5 g criteria requires the assumption of a manipulator configuration. The remote manipulator spacecraft and its manipulator are likened to a human torso with its arm attached at the shoulder. If the width from shoulder to shoulder is 3 feet and the axis about which spin-up occurs is 2 feet below the line connecting the shoulders, then the radius from center of rotation to the shoulder joint is 2.5 feet. The manipulators are assumed to be 4 feet long with a center of mass 2 feet from the shoulder. With this configuration, a rotating remote manipulator spacecraft will experience maximum torque on the manipulators, owing to centrifugal force, when the manipulators are fully extended and raised 53-1/2 degrees from an arm-at-the-sides position. The manipulator center of mass will be 3.2 feet from the axis of rotation and the angular rate which just produces a centrifugal acceleration equivalent to 1.5 g is 42 rpm. Limiting the centrifugal force to 66 percent of the manipulator capacity (or the equivalent of 1 g) reduces the maximum spin rate to 36 rpm. Under these conditions, the remote manipulator spacecraft is only expected to attach itself to the target vehicle and transmit the retarding force developed by thrusters to gradually remove the tumbling rate. As indicated previously, the OAO vehicle has a maximum rate of 34 rpm.

The other limit on tumbling rate for rate synchronized docking is the stability of the target vehicle itself. A vehicle tumbling or spinning around the axis of greatest moment of inertia axis will be stable if there are no external torques. Under these conditions, the vehicle will continue to rotate about the same body axis; this body axis is coincident with the angular momentum vector and remains fixed in inertial space.

The candidate OAO target vehicle is of special interest because the nominal centroidal inertias about all axes are equal and, therefore, the axis coincident with the angular momentum vector and fixed to the OAO should remain inertially fixed. Under these conditions, the remote manipulator spacecraft vehicle should be able to rendezvous with the OAO; orient itself relative to the spin axis; match spin rates; translate along the spin axis; and dock without any changes in the OAO spin axis. In reality, however, the OAO moments of inertia will not be equal (specification for OAO is $\pm$ 5%), and unless the OAO is coincidently spun up about the maximum or minimum moment of inertia axis, the spin axis will not remain fixed in the body. The OAO carries no liquids and is quite rigid except for balance booms and solar paddles, so the damping of this motion may be low.

The movement of the spin axis through the body of the OAO constitutes something of a hazard, since it means that the remote manipulator spacecraft runs the risk of being struck by solar paddles or balance booms. This hazard will be proportional to the rate at which the spin axis moves through the vehicle.

Analysis indicates that ω, the maximum rate of change of the spin axis direction as it moves through the body, is:

$$\omega = 1.25 \frac{\Delta}{I} \Omega$$

where:

Δ = Maximum value of incremental unbalance in moments of inertia

I = Nominal value of balanced moments of inertia

Ω = Tumbling rate of the vehicle

For the OAO, a 5 percent unbalance represents a potential rate of change of the spin axis equal to 2.1 rpm when the tumbling rate is 34 rpm. The analysis also indicates that the maximum rate occurs when the axis of maximum moment of inertia is 52 degrees from the angular momentum vector. The motion of the spin axis through the body, therefore, represents the most significant limitation on docking with a tumbling OAO.

All the hard points used for docking are associated with the bases of either the solar paddles or the balance booms. The maximum permissible motion of the spin axis during the docking maneuver is arbitrarily set at one-quarter of a revolution. One-quarter of a revolution represents a shift in the spin axis from top edge to bottom edge of the OAO. The actual time required for an operator to undertake the docking maneuver has not been tested, but if the one-quarter revolution shift in axis occurs over a period of 3 minutes this gives the operator 2 minutes to attempt to dock and 1 minute to retreat, if he is unsuccessful. One-quarter of a revolution in three minutes corresponds to one-twelfth of a revolution per minute and represents a required reduction in the maximum permissible tumbling rate from 36 rpm to 1.3 rpm, a value comparable to that required by velocity matching techniques.

If the OAO were a well damped and therefore stable system because of boom and solar paddle flexure, it would still be desirable to limit docking to low tumbling rates because of the unstabilizing nature of docking. As soon as the remote manipulator spacecraft docks with the OAO, an entirely new set of principal axis is formed, and the pair of vehicles will start to move about both a new center of mass as well as about a new axis of rotation. This not only offers potential danger to the remote manipulator spacecraft and OAO target vehicle, but also tends to break the communications link at a very critical point in the mission. The conclusion of the analysis can only be that docking with the OAO should not be attempted at tumbling rates in excess of 1.5 rpm. This low value does not appear to be incompatible with the OAO-A2 tumbling rate of 0.3 to 0.5 rpm. Another conclusion of this analysis is that docking in a rate synchronizing mode can indeed take place at rates as high as 36 rpm, if the target vehicle is stably spinning about its axis of maximum moment of inertia, and if the addition of the remote manipulator spacecraft mass and inertia to the spinning system does not cause it to go unstable.

3.1.5.4 Thruster Torquing

So far, the analysis has assumed tha the remote manipulator spacecraft must physically dock with the target vehicle and use the gripping capability of the manipulators to hold the two vehicles together while the remote manipulator spacecraft despins the target vehicle. The analysis also indicated that docking with a target vehicle may be feasible at only low spin rates unless the target vehicle was stablely spinning about its axis of maximum moment of inertia. However, the important function or need is to transmit a retarding torque to the tumbling vehicle rather than to physically dock with it. In a previous section, there was a discussion of devices configured to reach out from the remote manipulator spacecraft and by coming in contact with the target vehicle, apply a retarding force. This study has also investigated another torquing technique, the use of thrusters impinging upon the tumbling vehicle to impart retarding torques.

The remote manipulator spacecraft can use its manipulators to bring a small thruster near the tumbling vehicle and as surfaces such as solar paddles rotate by, the thruster can be fired and the gas from the thruster impinge upon the solar paddles or skin of the target vehicle to transmit a torque. The thruster can be made reactionless by providing two exhausts in opposite directions. This lessens the demands upon the remote manipulator spacecraft attitude control system and reduces the need for station keeping maneuvers.

An analysis was conducted of the force transmitted via plume impingement using a computer program recently developed to help analyze attitude control system response. The impingement mechanism was considered to be Newtonian and is therefore somewhat conservative as compared to free molecular flow. The analysis was conducted assuming a propellant of Freon 14 and a thruster developing 3 pounds of thrust from a nozzle with an expansion ratio of 60. The analysis considered both the force and torque reactions produced upon a 20 inch square plate at varying distances and angles with the thruster centerline through the center of the plate. Some of the results follow in Table 3-2.

Table 3-2. Force and Torque Transmission

Plate Size - 20 inches x 20 inches
Plate Area - 400 square inches
Propellant - Freon 14
Thruster Force - 3 pounds
Expansion Ratio - 60

Impingement Angle	Distance	10 in.	15 in.	20 in.
90°	F_x	2.77 lb	2.3 lb	1.92 lb
	M_y	0	0	0
45°	F_x	1.92 lb	1.69 lb	1.30 lb
	M_y	2.74×10^{-4} in.-lb	4.16×10^{-7} in.-lb	3.53×10^{-7} in.-lb

F_x is the force normal to the plate and along the x axis.

M_y is the torque about the axis which is in the plane of the plate and normal to and intersecting the thrust vector.

These results indicate that useable portions of the force from a thruster can be captured by impingement even at angles of 45 degrees.

A pair of thrusters rated at 3 pounds each and set in opposition to produce reactionless operation could be fired to impinge upon the OAO solar paddles without risk of damage to the remote manipulator spacecraft and only slight risk of paddle damage. Any damage to the solar cells is assumed to be an acceptable risk for this mission and would not result in catastrophic destruction of the entire array.

At 34 rpm, the outer most edge of the solar paddle will have a velocity of 39.2 feet per second. A thruster fired to impinge upon the solar paddle at an angle of 45 degrees can maintain a separation distance of 6 inches and still produce an average thrust of 1.53 pounds for 0.0425 second. This represents a total impulse of 0.715 foot-pound-seconds per impingement maneuver. If larger thrusters were used, larger impulses would result. After each impulse, the OAO angular rate would be lower and each succeeding impulse would therefore, be larger. The initial impulse producing the OAO's angular rate of 34 rpm amounted to 4600 foot-pound-seconds.

Producing a retarding torque by thruster impingement in a double thruster configuration, offers at best an efficiency of 50 percent. With a 45 degree impingement angle the efficiency is reduced to 1/3. Timing errors due to time delays and operator reaction time will reduce the efficiency even further. If the impingement geometry has an efficiency rating of 1/4 and half of the propellant is wasted due to timing errors, a total propellant load capable of producing 8 times the original disturbing impulse is required. For the OAO candidate target vehicle, tumbling at 34 rpm, 192 lb of nitrogen (or 256 lb of Freon-14) will be required.

3.1.5.5 Moment-Free Motion of the OAO Spacecraft

The moment-free motion of the OAO spacecraft is required to determine the maximum allowable tumbling rate for remote manipulator spacecraft docking maneuvers. This motion is analyzed below.

Nominally the centroidal inertias about all OAO axes are equal (all axes principal) and therefore there should be an axis fixed to the OAO which is both coincident with the angular momentum vector and inertially fixed. However, in reality OAO inertias will differ slightly for each axis and therefore only one orthogonal set of axes can be considered principal. Under these conditions, OAO fixed or body axes will remain inertially fixed only if the OAO angular momentum vector exactly coincides with the principal axis of minimum or maximum moment of inertia. This is an unlikely initial condition and in general it is appropriate to assume that any OAO axis will have an inertial angular velocity and the feasibility of the remote manipulator spacecraft docking maneuver will depend upon the maximum angular velocity and angular deviation of an OAO fixed reference axis.

The OAO is assumed to be a torque or moment-free rigid body in this analysis. Although spacecraft internal damping will tend to align the OAO's maximum moment of inertia axis with the momentum vector, this motion is assumed to be negligible when compared with the natural rigid body motion and the time of the attempted docking maneuver.

3.1.5.6 Analysis

The coordinate systems and angles describing OAO motion are shown in Figure 3-7. The general rotational equation of motion for a rigid body with external torques $\overline{M}$ and angular momentum $\overline{H}$ is:

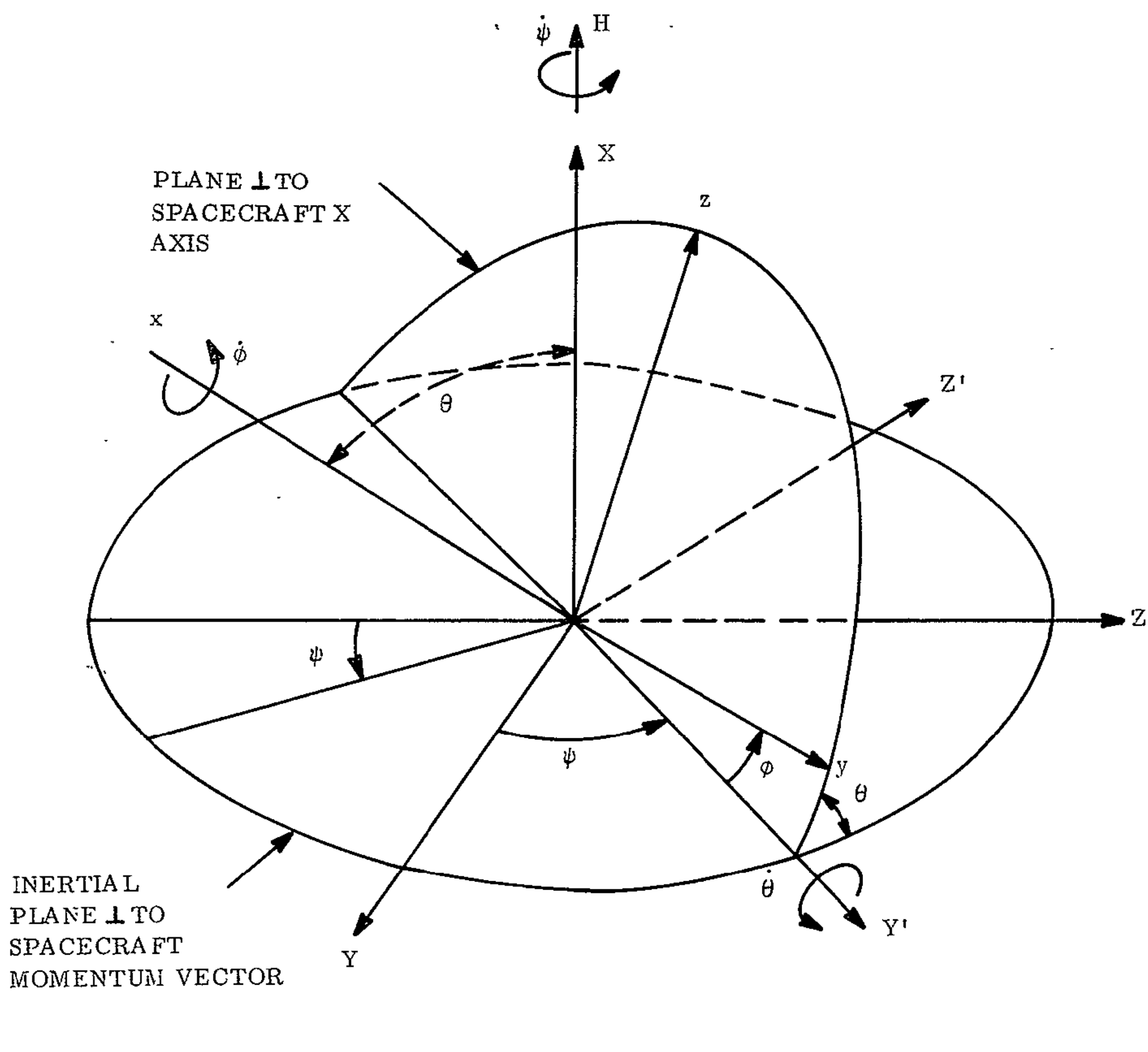
$$\overline{M} = \frac{d\overline{H}}{dt} \tag{3-1}$$

where the time derivative is taken with respect to inertial space. With zero external torques:

$$\frac{d\overline{H}}{dt} = 0, \ \overline{H} = \text{constant} \tag{3-2}$$

and the angular momentum vector is inertially fixed. Aligning the inertial X axis with the $\overline{H}$ vector:

$$\overline{H} = H\overline{X} \tag{3-3}$$



x, y, z - SPACECRAFT PRINCIPAL AXES
X, Y, Z - INERTIALLY FIXED AXES
Y', Z' - DISPLACEMENT FROM Y, Z, AFTER ROTATION ψ.

Figure 3-7. Coordinate System for Describing OAO Motion

Referring to Figure 3-7, $\bar{X}$ can be written as:

$$\bar{X} = \cos\theta\ \bar{x} + \sin\phi \sin\theta\ \bar{y} + \cos\phi \sin\theta\ \bar{z} \tag{3-4}$$

Substituting Equation (3-4) into Equation (8-3)

$$\bar{H} = H\cos\theta\ \bar{x} + H\sin\phi\sin\theta\ \bar{y} + H\ \cos\phi\sin\theta\ \bar{z} \tag{3-5}$$

It is also possible to express the angular momentum in body fixed coordinates as:

$$\bar{H} = I_x\,\omega_x\,\bar{x} + I_y\,\omega_y\,\bar{y} + I_z\,\omega_z\,\bar{z} \tag{3-6}$$

where $\bar{x}$, $\bar{y}$, $\bar{z}$ are the principal axes, I_x, I_y, I_z are the corresponding inertias and ω_x, ω_y, ω_z, are the components of the angular velocity vector $\bar{\omega} = \omega_x\,\bar{x} + \omega_y\,\bar{y}, + \omega_z\,\bar{z}$.

Equating components of Equations (3-5) and (3-6) and solving for ω_x, ω_y, ω_z results in:

$$\begin{aligned} \omega_x &= \frac{H}{I_x}\cos\theta \\ \omega_y &= \frac{H}{I_y}\sin\phi\sin\theta \\ \omega_z &= \frac{H}{I_z}\cos\phi\sin\theta \end{aligned} \tag{3-7}$$

The angular velocity $\bar{\omega}$ can also be written as:

$$\begin{aligned} \bar{\omega} &= \dot{\psi}\ \bar{X} + \dot{\theta}\ \bar{Y}' + \dot{\phi}\ \bar{x} \\ &= (\dot{\psi}\ \cos\theta\ + \dot{\phi})\ \bar{x} + (\dot{\psi}\sin\phi\ \sin\theta + \dot{\theta}\ \cos\phi)\bar{y} + \ \dot{\psi}\cos\phi\ \sin\theta \\ &\quad - \dot{\theta}\ \sin\phi)\ \bar{z} \end{aligned} \tag{3-8}$$

Equating the components of Equation (3-8) with ω_x, ω_y, ω_z

$$
\begin{aligned}
\omega_x &= \dot{\psi} \cos\theta + \dot{\phi} \\
\omega_y &= \dot{\psi} \sin\phi \sin\theta + \dot{\theta} \cos\phi \\
\omega_z &= \dot{\psi} \cos\phi \sin\theta - \dot{\theta} \sin\phi
\end{aligned}
\tag{3-9}
$$

Substituting Equation (3-7) into Equation 3-9) and solving for $\dot{\psi}$, $\dot{\theta}$, and $\dot{\phi}$

$$
\begin{aligned}
\dot{\psi} &= H\left(\frac{1}{I_y} \sin^2\phi + \frac{1}{I_z} \cos^2\phi\right) \\
\dot{\theta} &= \frac{H}{2} \sin\theta \sin 2\phi \left(\frac{1}{I_y} - \frac{1}{I_z}\right) \\
\dot{\phi} &= H \cos\theta \left(\frac{1}{I_x} - \frac{1}{I_y} \sin^2\phi - \frac{1}{I_z} \cos^2\phi\right)
\end{aligned}
\tag{3-10}
$$

For nominal OAO $I_x = I_y = I_z = I$ and Equation (3-10) reduce to:

$$
\begin{aligned}
\dot{\psi} &= \frac{H}{I} = \omega \\
\dot{\theta} &= 0 \\
\dot{\phi} &= 0
\end{aligned}
\tag{3-11}
$$

For small variations in the OAO inertias ($I_x = I + \Delta_x$, $I_y = I + \Delta_y$, $I_z = I + \Delta_z$) Equation (3-10), after neglecting products of small terms and assuming that the total momentum remains constant will reduce to:

$$
\begin{aligned}
\dot{\psi} &= \omega - \omega \left(\frac{\Delta y}{I} \sin^2\phi + \frac{\Delta z}{I} \cos^2\phi \right) \\
\dot{\theta} &= \frac{\omega}{2} \sin\theta \sin 2\phi \left(\frac{\Delta z}{I} - \frac{\Delta y}{I}\right) \\
\dot{\phi} &= \omega \cos\theta \left(-\frac{\Delta x}{I} + \frac{\Delta y}{I} \sin^2\phi + \frac{\Delta z}{I} \cos^2\phi \right)
\end{aligned}
\tag{3-12}
$$

The velocity of a point at the OAO surface coincident with the momentum axis is:

$$\bar{V} = \bar{\omega} \times R \bar{X} \tag{3-13}$$

where R is the distance from the OAO mass center to the surface point and x is the cross product operator.

Substituting $\dot{\psi} \bar{X} + \dot{\theta} \bar{Y}' + \dot{\phi} \bar{x}$ for $\bar{\omega}$ and performing the cross product operation:

$$\bar{V} = -R \dot{\theta} \bar{Z}' - R \dot{\phi} \sin\theta \bar{Y}' \tag{3-14}$$

Substituting Equation (3-12) into Equation (3-14) and computing the magnitude of $\bar{V}$:

$$V = R \omega \sin\theta \left[\frac{\sin^2 2\phi}{4} \left(\frac{\Delta z}{I} - \frac{\Delta y}{I} \right)^2 + \cos^2\theta \left(-\frac{\Delta x}{I} + \frac{\Delta y}{I} \sin^2\phi + \frac{\Delta z}{I} \cos^2\phi \right)^2 \right]^{1/2} \tag{3-15}$$

Noting that:

$$V < \frac{\Delta}{I} R \omega \sin\theta \left[1 + 4 \cos^2\theta \right]^{1/2} \tag{3-16}$$

where Δ is the maximum value of Δ_x, Δ_y, Δ_z.

The maximum of Equation (3-16) occurs at $\theta = 52$ degrees and therefore:

$$V_{max} < 1.25 \frac{\Delta}{I} R \omega \tag{3-17}$$

The maximum angular velocity, ω_{max}, of the OAO axis coincident with the momentum vector is:

$$\omega_{max} = \frac{V_{max}}{R} < 1.25 \frac{\Delta}{I} \omega \tag{3-18}$$

It can be seen from Equation (3-12) that the worst case value of $\dot{\phi}$ is:

$$\dot{\phi} = 2 \frac{\Delta}{I} \omega \cos \theta \tag{3-19}$$

and therefore with time the coincident axis will cone, with an angle θ, around the OAO principal axis $\bar{x}$. Furthermore, this $\bar{x}$ axis will cone around the momentum vector with an angular velocity approximately equal to ω, and oscillates away from and towards the momentum vector with a maximum angular velocity equal to $\frac{\Delta}{I} \omega$.

3.1.6 SATELLITE GRIP HOLDS

There are a variety of grip-holds on the OAO-A1 available for the docking, locomotion, and repair phases of the mission. Table 3-3 describes them and Figure 3-8 shows their location. Figures 3-9A, B, and C show some of these grip-holds. The legend of abbreviations for Table 3-3 is as follows:

IB	- inertia boom (including damper)	PLP	- paddle latch on paddle
BC	- boom clevis	PC	- paddle clevis
BDC	- boom damper clevis	PDC	- paddle damper clevis
BL	- boom latch	PSP	- paddle spring pad
P	- paddle	SSC	- separation spring cup
PLB	- paddle latch on body	LF	- longeron fitting
BV	- bay vent	SS	- sun shade

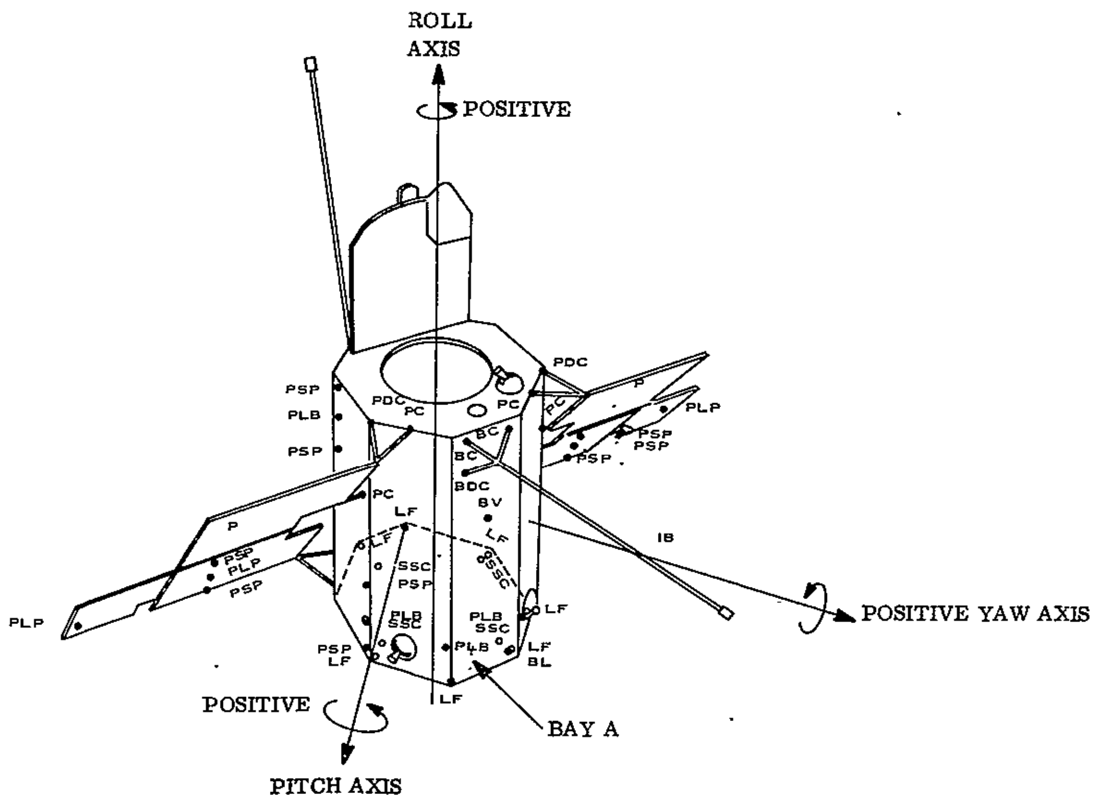


Figure 3-8. Grip Hold Locations

Table 3-3. OAO-A1 Grip-Holds

GRIP-HOLD (AND QUANTITY)	FIGURE NO.	DESCRIPTION APPROXIMATE PROTRUSION, E.G. BEYOND SKIN PANEL OR OPENED SNAP-COVER (INCHES)	COMMENTS	EXPOSED	ACCESS (STATIC) COVERED BY SPRING-LOADED SNAP-COVER	GRIPPABLE THRU CRUSHABLE SNAP-COVER, SKIN PANEL OR INSULATION	ADEQUATE ACCESS (TUMBLING AND STATIC), STIFFNESS, AND STRENGTH DOCKING PHASE HIGH TUMBLING RATE	NOMINAL TUMBLING PITCH RATE	LOCOMOTION & REPAIR PHASE
IB (2)	3-12	120	Alum Alloy, 1/8" wall	yes	N.A. (Not Applicable)	N.A.	No	Yes	Yes
BC (4)	3-13	2	Fiberglas.	yes	N.A.	N.A.	Yes	Yes	Yes
BDC (2)	3-13	2	Fiberglas.	yes	N.A.	N.A.	Yes	Yes	Yes
BL (2)	Not Shown	1	Steel, rugged for squibs.	no	yes	yes	No	Yes	Yes
P (4)	3-1	87	----	yes	N.A.	N.A.	No	Yes	Yes
PLB (6)	Not Shown	3	Steel, rugged for squibs.	yes (except two outer panel latches)	no (except two outer panel latches)	yes	Yes (Except two outer panel latches)	Yes	Yes
PLP (6)	3-9A	5	Steel, rugged for squibs.	yes	N.A.	N.A.	No	Yes	Yes
PC (8)	3-22	6	----	yes	N.A.	yes	Yes	Yes	Yes
PDC (4)	3-22	3	----	yes	N.A.	N.A.	Yes	Yes	Yes
PSP (16)	Not Shown	1/4	Rigid steel base for spring reaction.	yes	N.A.	yes	No	No	No
SSC (4)	3-9B	2	----	yes	N.A.	N.A.	No	Yes	Yes
LF (8)	3-9C	1	Steel, rugged for launch loads.	no	yes	yes	No	Yes	Yes
BV (approx. 36)	3-15	1	0.010" skin panels are non-structural	yes	no	yes	No	No	No
SS (2)	3-1	50	Open and close during mission	yes	N.A.	N.A.	No	No	No

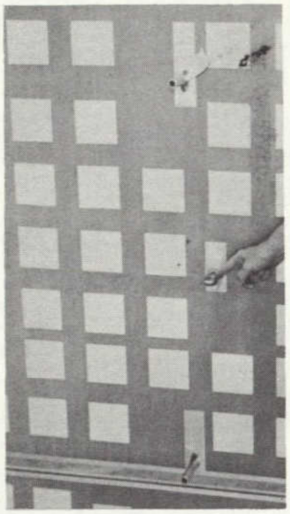


Figure 3-9A. PLP - Paddle Latch on Paddle

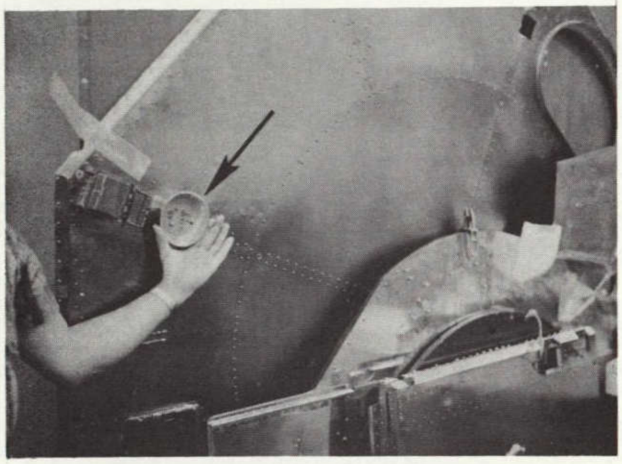


Figure 3-9B. SSC - Separation Spring Cup

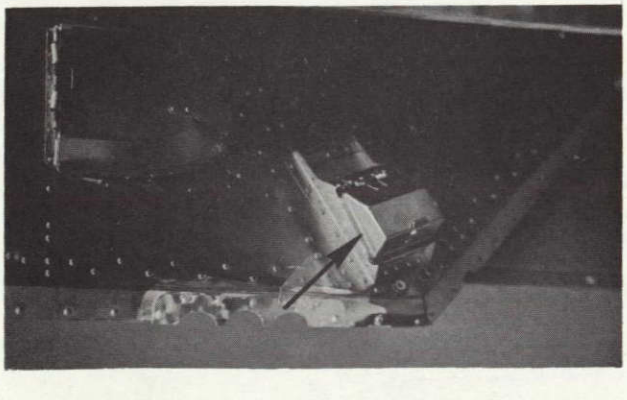


Figure 3-9C. LF - Longeron Filling with Snap-Cover Over It

Note in Table 3-3 that the boom clevises, paddle clevises, damper clevises, and main paddle latches are the only grip-holds which have adequate access, stiffness, and strength for all phases. Fortunately, the satellite tumbling rates do not require sacrifice of the booms and expensive paddles in order to dock to the OAO-A1 although they are relatively easy to replace.

Table 3-3 shows that almost all the grip-holds are adequate for gripping as the remote manipulator spacecraft travels on and repositions itself on the OAO-A1 after docking. This is fortunate because great care must be taken not to strike the many exposed, carefully-aligned components, namely:

- 6 star-trackers
- 18 pneumatic system nozzles
- 2 motorized sun shades
- Solar sensors
- Boresighted tracker

Furthermore, thermal insulation covers nearly all internal structural parts that could serve as grip-holds after skin panels are removed.

For the nominal tumbling rate of 0.5 rpm, the remote manipulator spacecraft is stationed in the tumbling plane and the manipulators grasp the slowing moving (0.25 fps) grip-holds such as clevises, separation spring cups, or snap covers over the longeron-fittings. After docking, it stabilizes the OAO-A1 to maintain a communications link to a relay satellite. The OAO-A1 has extremely thorough passive thermal control due to its super insulation, highly reflective skin panels, and thermal conduction isolation techniques. The spacecraft should be rotated periodically to limit component temperatures. The star-trackers should also be covered with shades to protect them from long exposure to direct sunlight.

3.1.7 OAO-A1 REPAIR TASKS

The tasks involved in repairing the failures described earlier are analyzed in the following sections.

3.1.7.1 Remove the Battery Charge and Sequence Controller (BCSC)

The BCSC unit is located in bay E-2 (Figures 3-10 and 3-11). Figure B-1 in Appendix B shows the OAO compartmentation. In order to obtain access to the BCSC, the bay E upper skin panel must be removed. However, the inertia boom at bay E-1 (Figure 3-12) is mounted on three clevises (Figure 3-13) which extend through three square holes near two edges of this skin panel. Therefore, it is necessary either to remove the boom or to cut away three small pieces of the skin panel with a special powered shears.

Because the latter method is the shorter of the two, it has been selected. Damage to the spacecraft, wiring, insulation, etc., is avoided by pulling the skin panel away from the OAO body structure as far as possible.

The following prelaunch preparations will simplify this portion of the repair mission by the remote manipulator spacecraft:

- Use semi-rigid tethers to store the skin panels. Unfortunately, the skin panels must be removed and installed according to a prescribed sequence because the skin panels overlap one another. The panels are removed in the following order:

  a. The upper (U) and lower (L) skin panels at bays A, C, E, and G

  b. Then the upper and lower skin panels of bays B, D, F, and H

  c. Then the middle skin panels of bays A, C, E, and G

  d. Then the middle skin panels of bays B, D, F, and H; however, the middle skin panels at B, D, and F are hinged about an axis parallel to the longitudinal axis of the OAO-A1.

- As a result, 6 skin panels must be removed in the following order: DM, FM, EM, DU, FU, EU. Skin panels DM and FM are hinged; the remainder are removed. Some of the skin panels are clamped and stored on tethers. The remainder are temporarily taped to the hardware protruding through them or are temporarily

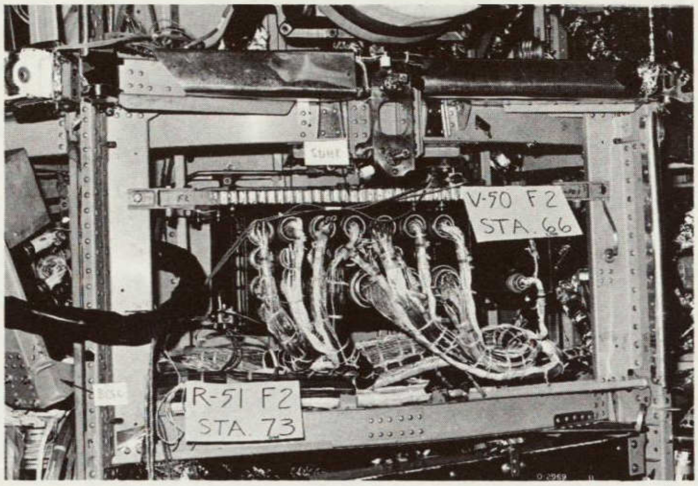


Figure 3-10. BCSC - Bay F-2

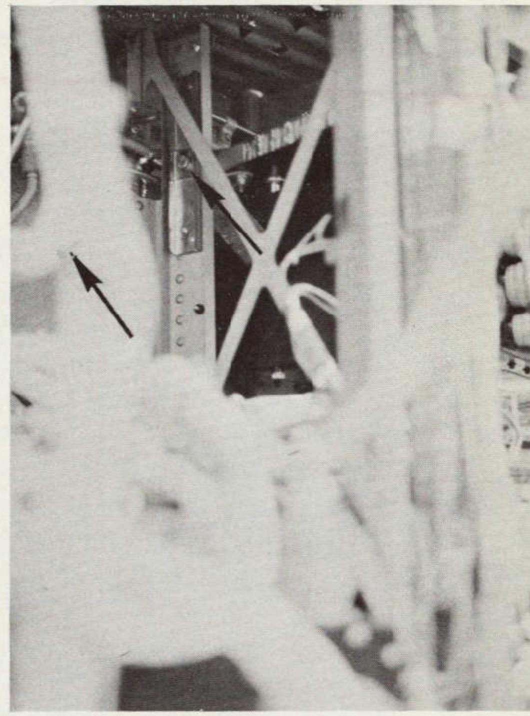


Figure 3-11. BCSC - Bay F-2

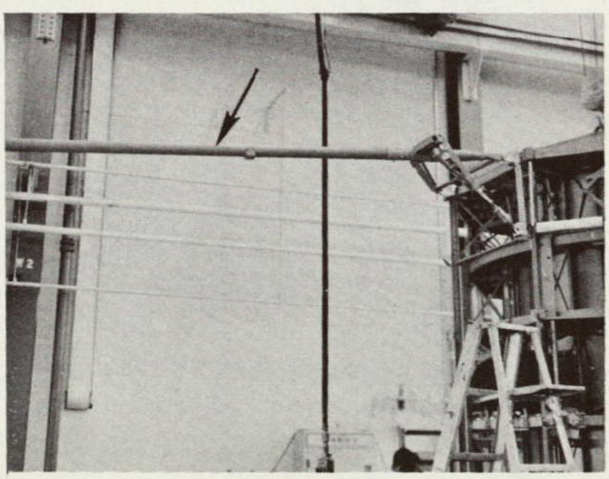


Figure 3-12. Erected Inertia Boom

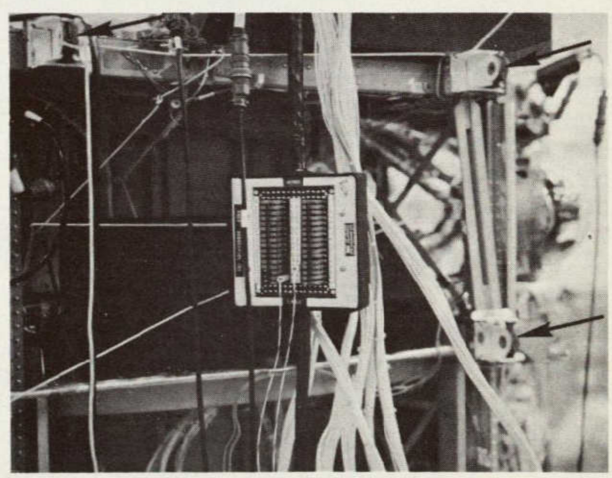


Figure 3-13. Three Mounting Clevises

re-installed over the bay according to the thermal protection requirements of the equipment. There are other hindrances to the easy, quick removal of the panel:

Some of the skin panels must be maneuvered around hardware that protrudes through the skin panels and around ledges on the thermal boxes over part of this hardware (Figure 3-14).

Tape and taped thermal insulation are used locally at gaps around the protruding hardware such as clevises and spring pads.

Fortunately, the skin panels do not have individual ground cables but do have springy fingers that ground the panels to each other. The skin panels are grounded by a cable at the paddle clevises which also ground the paddles.

Figure 3-14. Power Output Side of Battery Pack (TA-2 has three connectors as shown, A-1 has only two)

- The replaced insulation around the periphery of the replaced BCSC unit is attached to the new BCSC unit wherever possible. Super-insulation tape is attached to the insulation.

- The two replacement 0.25 inch Allen-head bolts for mounting the BCSC mounting plate have captive underhead washers, lead-in, and easily gripped or magnetic heads.

- The four replacement 0.25 Allen-head bolts and underhead washers for mounting the BCSC unit to the mounting plate have lead-in and are captive in the new BSCS unit.

- The BCSC unit has a handle on it to facilitate handling.

- Color coding, numbering, striping, etc. on connectors, equipment, etc.

- The new BCSC unit has a jack-harness attached to it in order to provide more slack; thereby, partial rotation of the BCSC mounting plate is avoided when the harness is connected.

The following steps are required to replace the BCSC:

	TASK	TIME (Minutes)
Step 1.	With the docking legs, grip the following to see and repair the BCSC unit:	3
	• The boom latch at bay E-6, the solar paddle clevises at bays D5 and D6, or the paddle damper clevis at bay D6 for a low position of the remote manipulator spacecraft, being careful not to strike the solar arrays when gripping these clevises.	
	• The damper boom and its three mounting clevises at bays E-1 and E-2 for a high position of the remote manipulator spacecraft when loosening the bay E upper skin panel.	
	• The solar paddle spring pads at bays C-1, C-2, G-1 and G-2 and the solar paddle latches at bays C-2 and G-2 for a high position of the remote manipulator spacecraft particularly when removing and re-installing the BCSC unit.	
Step 2:	Unclamp a stored, semi-rigid tether	2
Step 3:	Clamp the tether to the DM skin panel vent. Figure 3-15 shows the two vents on bay C upper skin panel. The upper two cut-outs are for the paddle clevis and paddle damper clevis at bay C-1 and the lower cut-out is for the other paddle clevis at bay C-2.	2
Step 4:	Reach into the tool bin and mate the special blade screw-driver bit with the power tool.	2
Step 5:	Unscrew the four No. 10, slotted-head, aluminum screws at each corner of the skin panel and eject and store them and their underhead washers on sticky surfaces in the supply bin.	6

TASK		TIME (Minutes)
Step 6.	Store the skin panel in a trailing position.	2
Step 7.	Repeat steps 3 through 6 to remove and store skin panels FM, EM, DU, FU, EU.	60
Step 8.	Reach into the tool bin for the special thermal insulation cutter (shears).	1
Step 9.	Remove insulation over the BCSC mounting plate and around the periphery of the hinged BCSC mounting plate and store it in the supply bin which has some sticky surfaces (Figure 3-16 shows typical peripheral insulation and overlying insulation at bays E-2 and F-2 on OAO-TA2).	8
Step 10.	Replace the insulation cutter in the tool bin.	1
Step 11.	Reach into the tool bin and mate the special hex head socket to the power tool.	2

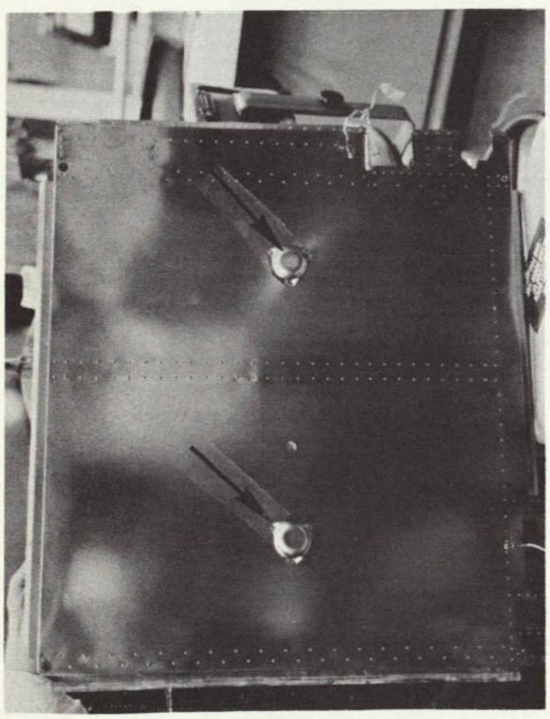


Figure 3-15. Typical Skin Panel Showing Vents

Figure 3-16. Typical Peripheral Insulation

TASK	TIME (Minutes)
Step 12. Reach into the tool bin for the special box wrench for the uncaptive nuts.	1
Step 13. Unscrew the two 0.25 inch hex-head bolts and their under-head washers and grounding strap that fasten the top of the BCSC mounting plate and eject and store them on sticky surfaces in the supply bin. A mirror may be required to see and hold the two uncaptive nuts.	8
Step 14. Demate and replace the special hex socket and power tool in the tool bin.	1
Step 15. Rotate the BCSC mounting plate approximately 45 degrees until motion is restricted by two cables attached to the two Deutsch connectors on the BCSC (Figure 3-10) shows the mounting plate rotated approximately 30 degrees.	1
Step 16. Reach into the tool bin for the special pliers and remove the CIR-CLIP on the small Deutsch connector.	4
Step 17. Replace the special pliers in the tool bin.	1
Step 18. Demate the small Deutsch connector and unscrew the larger multi-turn screw-type.	6
Step 19. Rotate the BCSC mounting plate approximately another 45^{o} degrees.	1
Step 20. Reach into the tool bin and mate the special hex head socket to the power tool. Also, reach for the special box-wrench to hold the nuts since they are not captive.	2
Step 21. Unscrew the four 0.25 inch hex-head bolts that attach the BCSC unit to the mounting plate and eject and store the bolts, washers (underhead and underwasher) and nuts on sticky surfaces in the supply bin.	6
Step 22. Demate and replace the special hex socket, power tool, and box wrench.	1

TASK	TIME (Minutes)
Step 23. Place the old BCSC unit in the supply bin for later deorbiting on the RMS.	2
	120 minutes to remove BCSC
	100 minutes total to replace BCSC (see Section 3.1.6.5)

3.1.7.2 Replace the Batteries

The batteries are located in compartment C-4 (Figure 3-17). There are two battery packs of six cells each which are connected in a leafed manner to form three batteries. The batteries, cases, and thermal heat sinks weigh 172 pounds.

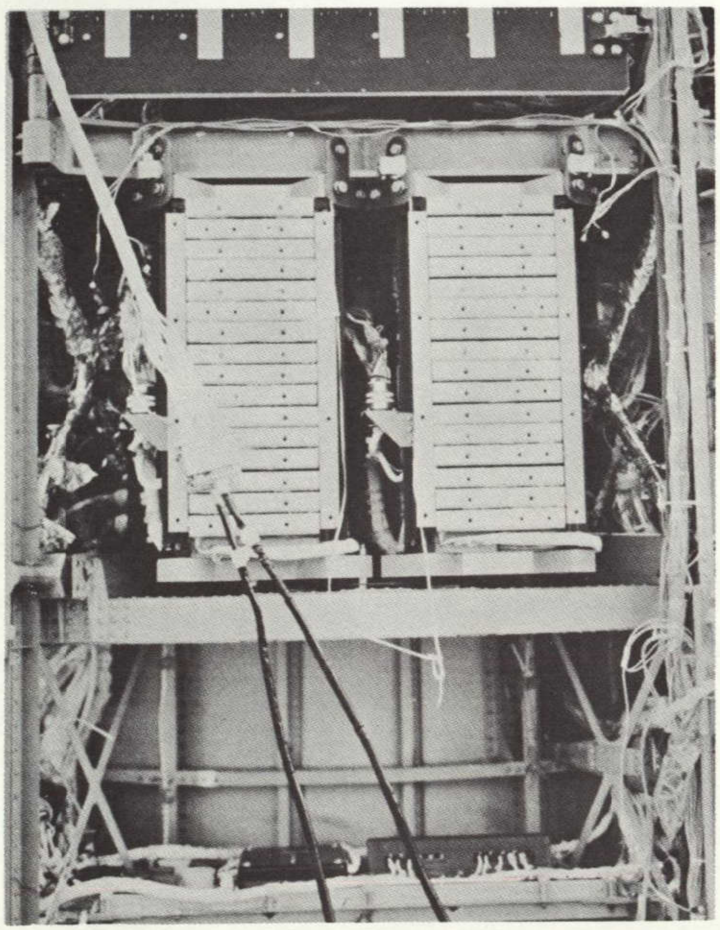


Figure 3-17. Two Battery Packs in Compartment C-4

a. The new battery unit will be mounted externally at bay C-6 instead of being exchanged for the two old battery packs. This location will not obscure the star trackers more than they are already obscured by the solar paddle. The old battery packs are not removed because too much time would be spent to remove and re-install the 17 Deutsch connectors with CIR-CLIP locking rings and 20 mounting bolts. Two of the bolts are fastened to the bottom of the battery packs (Figure 3-18); they could be reached only through nearly vacant bay C-5 (Figure 3-19) which necessitates removal of the bay C lower skin panel. Two other bolts are in blind spots and require the use of mirrors. However, since there are only four Deutsch connectors to the batteries themselves the new batteries are connected to those four power cables directly. Figures 3-14, and 3-20 show these connectors on the OAO-TA2 spacecraft which has six instead of four. The new battery unit is mounted externally at bay C-6 for several reasons:

1. The mass moment of inertia of the OAO-A1 spacecraft is approximately 1300 slug-feet about all axes. The clamping of a 175 pound additional battery unit (integral dual battery packs, case, heat-sink, super-insulation, power jack cable, and attachment clamps) externally onto bay C-6 will increase the mass moments of inertia as follows:

   Roll axis: 62 slug-ft^2

   Pitch or yaw axis: 168 slug-ft^2 (maximum)

   Although this maximum increase of 168 slug-ft^2 is a significant percentage of the original moment of inertia (13 percent), the attitude control performance (settling time after slewing) is not degraded prohibitively. However, the degradation would be less if further thermal analysis allows the battery unit to be mounted near the OAO-A1, center of gravity, and if diametral balance weights are effective. Ideally the battery unit should be divided into 4 masses mounted midway between the pitch and yaw axes, near the c.g., and on each side of the c.g. such as at vacant bays C-2, C-5, G-2, and G-5, but bay thermal balancing and the repair mission become too complex.

2. Installing the new battery unit on bays B, D, F, and H or on the top or bottom of the OAO-A1 would obstruct the view of the 6 star-trackers. Furthermore, the externally-mounted battery unit would degrade the performance of the attitude control system more if the unit were mounted on these bays since they are perpendicular to the pitch and yaw axes.

Figure 3-18. Bottom Lug Bolt of Battery Pack

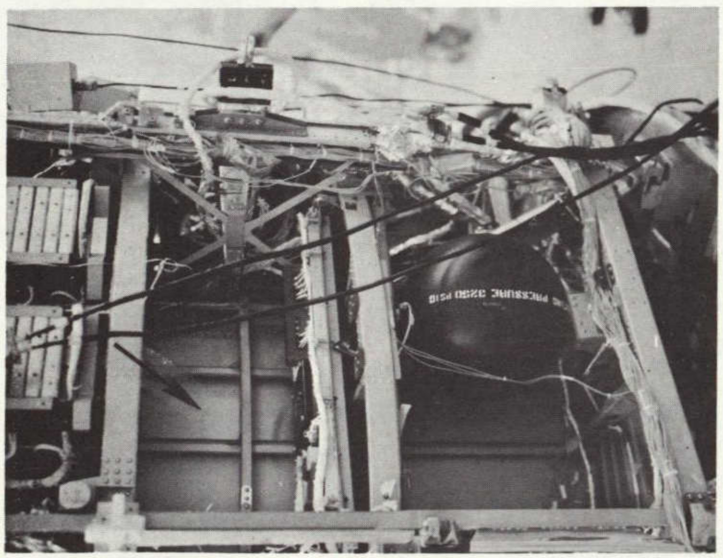


Figure 3-19. Nearly Vacant Bay (C-5)

3. The lower and upper solar paddle at bay C is removed in order to provide the needed access to the 4 battery connectors. Thus, there is good access to bay C-6 to mount the new battery unit. The other bays (bays A, E, and G) have either solar paddles or a damper boom on them.

4. In bay C-6 there is no gas container (Figure 3-19 shows one on OAO-TA2) which would not be affected by thermal outputs anyway from the new battery unit with a thermally designed backface. (Bays C-1 and G-6 are the only bays with only gas containers; however, the paddles near those bays also would have to be removed in order to avoid damage to the paddles during the repair mission. Bays A-3 and E-3 have not only gas containers but also other subsystem components.) Nor is bay C-6 affected by thermal blocking by the new battery unit as some bays would be (i.e., some bays are vacant for thermal balancing of the OAO-A1.

5. The clamps on the new battery unit can be attached as required to the nearby rigid and strong paddle erection hardware including paddle, latches, paddle clevises, and the paddle damper clevis (Figure 3-9).

6. The jack-cable from the new battery unit to the power cables is short and a hole can be easily cut for it in skin panel CM after the panel is removed.

7. The percentage of lower paddle area shadowed by an adjacent external battery unit is less than an upper paddle would be shadowed.

The following pre-launch preparations with simplify the battery repair mission by the remote manipulator spacecraft:

- Use rigidizeable tethers to store the solar paddle. Two of these tethers are illustrated in Figure 3-21 mounted on an astronaut belt. Use semi-rigid tethers to store the skin panels which are removed in this order: BM and CM.

- The new battery unit is as electrically, mechanically, and thermally similar to the old battery packs as possible in order to minimize the repair mission response time, but the heat sinks are pre-attached. The battery unit has integral clamps on it to facilitate clamping and grounding the unit to the OAO-A1. The jack-cable to mate with the power cables at the old batteries has adequate stifness and super-insulation on it. It also has a "plug" of super-insulation and tape for the hole in the skin panel through which the jack-cable and 4 paddle power cables pass.

- The two new paddle clevis bolts, paddle damper bolt, and self-locking nuts (to avoid cotter pints) have captive washers, lead-in, and easily gripped or magnetic heads.

Figure 3-20. Battery Bav

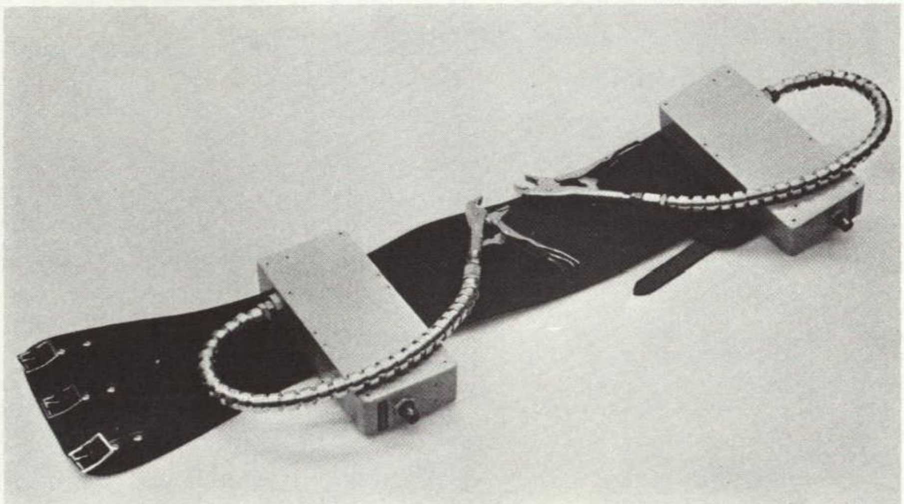


Figure 3-21. Rigidizeable Tether Concept

- The new skin-panel screws have lend-in, captive under-head washers and a large or magnetic heads for gripping during re-installation of the skin panels.
- Star-tracker covers automatically clamp onto the star-tracker domes.
- The new battery unit has handles on it to facilitate handling.
- Color coding, numbering, striping, etc. on connectors, units, etc.

After docking and stabilizing the OAO-A1, perform the following steps.

TASK		TIME (Minutes)
Step 1.	Grip with the docking legs the Bay C lower and upper main paddle latches and paddle/damper clevises as required to easily see and repair the batteries.	3
Step 2.	Reach into the tool-bin for the special thermal insulation cutter (shears).	1
Step 3.	Cut, pull-off, and fold-back enough insulation and insulation tape to uncover the 2 Cannon connectors (Figure 3-13 on OAO-TA2 shows 3)	
Step 4.	Replace the insulation shears into the tool-bin.	1
Step 5.	Reach into the tool-bin and mate the special screw-driver bit with the power tool.	1
Step 6.	Unscrew the 4 screws on the 2 Cannon connectors and demate them.	2
Step 7.	Demate and replace the special screw-driver bit and power tool into the tool bin.	1
Step 8.	Unclamp a stored rigidizeable tether from the remote manipulator spacecraft	1
Step 9.	Unrigidize the tether and clamp it to the latch post on the lower paddle.	2
Step 10.	Rigidize the tether.	1
Step 11.	Reach into the tool bin for the special pliers for removing cotter pins.	1
Step 12.	Remove the three cotter pins in the three castellated nuts at the two paddle clevises and the damper bolt (Figures 3-19 and 3-22).	4
Step 13.	Store the cotter pins in the supply bin on sticky surfaces.	2
Step 14.	Replace the special pliers in the tool bin.	1

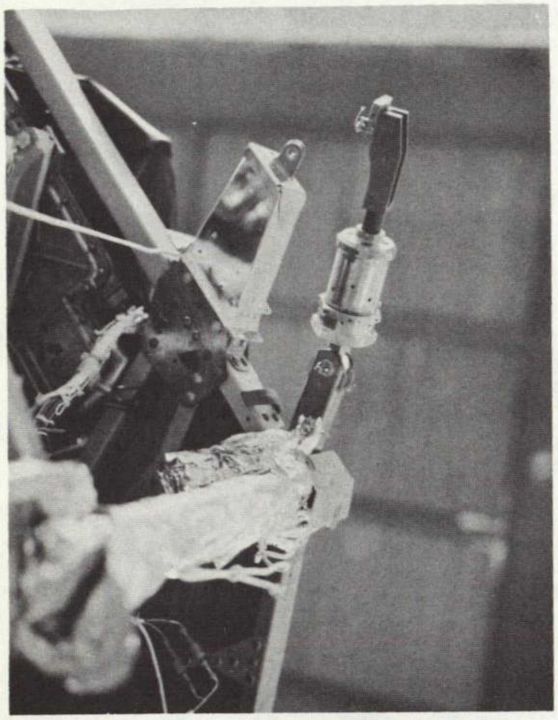


Figure 3 22. Upper Main Paddle Pivot and Damper

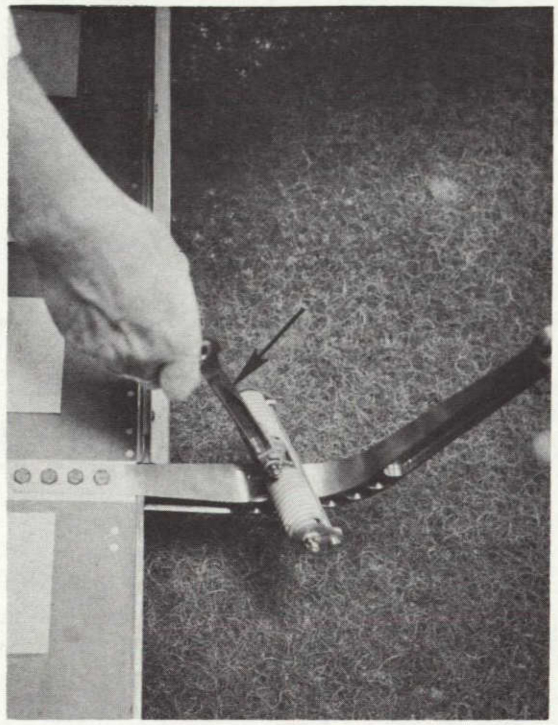


Figure 3-23. Paddle Damper Strut

Step 15.	Reach into the tool bin and mate the special hex socket to the power tool. Also reach for the special castellated-nut box-wrench.	1
Step 16.	Unscrew the three castellated-nuts.	3
Step 17.	Eject the nuts and washers from the special hex socket and box-wrench and store them in the supply bin on sticky surfaces. (Handling the separate underhead and undernut washers is tedious. Perhaps a dab of grease-like material on them will retain them sufficiently).	4
Step 18.	Demate the special hex-socket and power tool and replace them and the special box wrench into the tool bin.	1
Step 19.	Remove the two paddle clevis bolts, damper bolt, and underhead washers and store them in the supply bin. The torsion-spring-loaded damper strut (Figure 3-23) has to be repositioned slightly in order to withdraw the damper bolt.	3

Step 20.	Unrigidize the paddle tether sufficiently to store the paddle in a trailing position.	3
Step 21.	Repeat steps 1 through 22 to remove and store the upper paddle.	40
Step 22.	Unclamp a stored, semi-rigidized tether from the remote manipulator spacecraft.	1
Step 23.	Clamp the tether to the BM skin panel vent (Figure 3-15).	1
Step 24.	Reach into the tool-bin and mate the special blade screw-driver bit with the power tool.	2
Step 25.	Unscrew the four, No. 10, slotted-head, aluminum screws at each corner of the BM middle skin panel.	4
Step 26.	Eject and store the panel screws (and their underhead washers) in the supply bin.	4
Step 27.	Store the skin panel.	2
Step 28.	Repeat steps 24 thru 28 to remove the CM skin panel.	14
Step 29.	Reach into the tool bin for the special, thermal insulation shears.	1
Step 30.	Cut, pull-off, and fold-back enough insulation and insulation tape in order to thread the 4 paddle power cables through the holes in the skin panel.	8
Step 31.	Replace the insulation shears in the tool-bin.	1
Step 32.	Remove the skin panel, threading the four paddle power cables through the panel holes.	6
Step 33.	Reach into the tool-bin for the shears and enlarge the paddle power cable hole at the corner of the skin panel for the additional new jack-cable from the externally-mounted battery. Store the cut-out pieces in the supply-bin.	8
Step 34.	Store the skin panel.	2
Step 35.	Replace the shears in the tool bin.	1

Step 36.	Repeat steps 29 thru 31 to remove the thermal insulation around the periphery of the heat sinks (Figure 3-24).	10
Step 37.	Reach into the tool bin and mate the special Phillips-head screwdriver bit with the power tool.	2
Step 38.	Remove, eject and store the 94 Phillips-head screws and their under head washers in the supply bin. A dab of grease-like material on them will retain them sufficiently.	100
Step 39.	Store the heat sinks in the supply bin for later de-orbiting. Be careful not to transfer thermally conductive grease on the heat-sinks and batteries to other parts.	2
Step 40.	Reach into the tool bin for the special pliers and remove the 4 Deutsch connector CIR-CLIPS at the 2 battery packs.	6
Step 41.	Replace the special pliers into the tool bin, and get the special connector tongs (see Figure 6-2) for removal of Deutsch connectors in a narrow space.	1
Step 42.	Demate the 4 Deutsch connectors.	8
Step 43.	Replace the Special vise grip tongs into the tool bin.	
Step 44.	Reach into the supply bin and unclamp the new battery unit and its jack cable.	1
Step 45.	At bay C-6, clamp the new battery unit and its grounding strap to paddle erection hardware.	3

Step 46. Repeat steps 1 through 42, where required, in reverse fashion to reconnect the battery power cables and to re-install the BM and CM skin panels and solar paddles. For example, skip steps 41 and 42 because CIR-CLIPS are not necessary in orbit. Skip steps 35, 38, 39 because new heat sinks are supplied mounted on the new battery unit, and perform steps 2 through 7 preferably last in the OAO-A1 repair mission in order to interrupt array power as long as possible. The preflight preparations will simplify the replacement such as self-locking fasteners, captive fasteners, shaped thermal insulation and tape, pre-attached heat sink, etc. A special re-set tool may be required on occasion to reset the ball retaining band in the connector. (See Figure 6-3). It is not necessary to replace all skin panel screws. 170

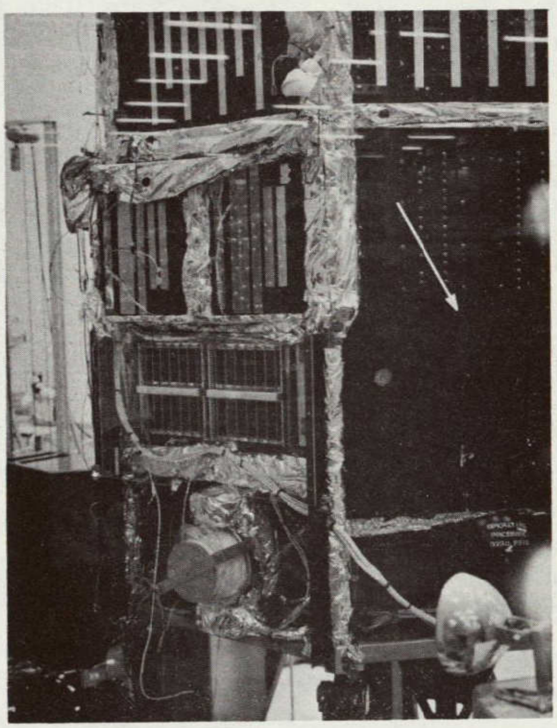


Figure 3-24. Battery Heat Shield

3.1.7.3 Recharge the Nitrogen Gas Supply

The exposed fill connections for both the 3500 psi primary and secondary pneumatic systems are at the interface of bays A-3 and A-4 on OAO-A1; Figure 3-25 shows the mounting block with only one filter connection at bay F-6 on OAO-TA2.

The following preparations will simplify the gas resupply repair mission by the remote manipulator spacecraft:

- The fill nozzle wing-nut has a lead-in into the fill connection on the OAO-A1 that also orients the nozzle attachment nut perpendicularly to the fill connection.
- The new fill connection cover has captive Allen-hand screws with lead-in and captive underhead washers.

The following tasks are necessary to recharge the secondary nitrogen gas supply:

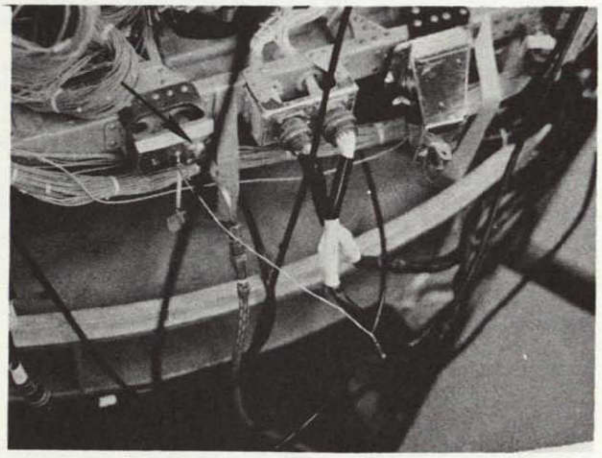


Figure 3-25. Nitrogen Supply Fill Vent

TASK		TIME (Minutes)
Step 1.	Grip with the docking legs the separation-spring cups, the paddle mounting clevices at bays F5 and F6, and the boom latch at bay E6.	
Step 2.	Reach into the tool bin for the special Phillips-head screwdriver bit and make it to the power tool.	2
Step 3.	Unscrew the 9 Phillips-head screws that attach the fill connection cover to the skin panel (has 9 captive nuts) and store them, their underhead washers (not captive) and the cover on sticky surfaces on the supply-bin.	13
Step 4.	Reach into the tool bin for special filler cap box-wrench (3/4 inch).	1
Step 5.	Unscrew the secondary gas supply filler cap.	1
Step 6.	Unclamp the fill nozzle from the supply bin and mate it with OAO-A1.	2
Step 7.	Tighten the fill nozzle wing-nut directly using the manipulator jaws.	1
Step 8.	Open the hand-valve on the fill nozzle to recharge one of the OAO-A1 nitrogen gas supplies.	1
Step 9.	Repeat steps 2 through 6, where required, in reverse fashion to cap filler connection.	14
Step 10.	Repeat steps 2 through 9 where required, in reverse fashion to recharge the primary nitrogen gas supply (See Figure 3-25). A review of the flight operations indicates that the secondary system was depleted and the primary system may be approximately 1/3 full.	35*
		70

*70 minutes total for both pneumatic systems

3.1.6.4 Replace the Spacecraft Data Handling Equipment (SDHE)

The SDHE unit is located in bay F-2 (Figure 3-10). Ground preparations similar to those for replacing the BCSC would simplify the SDHE repair. The following tasks are necessary to replace the SDHE:

	TASK	TIME (Minutes)
Step 1.	With the docking legs, grip the following as required to easily see and repair the SDHE unit (Figure 3-10). • The solar paddle spring pads at bays G-1, G-2, G-5, and G-6 for both a high position and a low position of the remote manipulator spacecraft. • Three mounting clevises for the damper boom at bays E-1 and E-2. • The solar paddle clevises at bays H-1 and H-2 and the paddle damper clevis at bay H-1 (but do not strike the solar arrays when gripping these clevises).	
Step 2.	Unclamp a stored rigidizeable tether from the manipulator spacecraft.	1
Step 3.	Clamp the tether to the bay HM skin panel vent (Figure 3-20 shows the two vents on skin panel CU).	2
Step 4.	Reach into the tool bin and mate the special blade screwdriver bit with the power tool.	2
Step 5.	Unscrew the No. 10, slotted-head aluminum screws and underhead washers at the corners of the upper skin panel. Eject and store them in the supply bin.	6
Step 6.	Store the skin panel.	2
Step 7.	Repeat steps 3 through 6 to remove and store skin panels GM, HU, and GU.	40
Step 8.	Reach into the tool bin for the special thermal insulation	1
Step 9.	Remove the insulation over bay E-2 (Figure 3-9 shows this insulation on OAO-TA2; note the vent hole) and where required in bay G-2, and store it in the supply bin which has some sticky surfaces.	8

TASK	TIME (Minutes)
Step 10. Replace the insulation cutter in the tool bin.	1
Step 11. Reach into the tool bin for the special pliers to remove the CIR-CLIPS on the Deutsch connectors and remove the 22 Deutsch connectors.	24
Step 12. Replace the special CIR-CLIPS removal pliers in the tool bin.	1
Step 13. Demate the 22 Deutsch connectors.	
Step 14. Attach slightly spring-tensioned retractors or tape (from the supply bin) on SDHE cables where required to keep them out of the way.	7
Step 15. Reach into the tool bin for the special thermal insulation shears.	1
Step 16. Remove additional insulation over and around the SDHE unit and store it on sticky surfaces in the supply bin.	8
Step 17. Replace the insulation cutter in the tool bin.	1
Step 18. Reach into the tool bin and clamp the special mirrors (and possibly adjustable lights) into bays E-2, F-2, and G-2, in order to see the four 0.25 inch nuts and four 0.25 inch bolts that fasten the SDHE mounting plate to the main structure (Figures 3-10 and 3-11); the two innermost nuts are captive.	4
Step 19. Reach into the tool bin and mate the special hex head socket to the power tool.	2
Step 20. Reach into the tool bin for the special box wrench.	1
Step 21. Unscrew the four 0.25 inch hex head bolts and their underhead washers, nut, undernut washers, and grounding strap and eject and store them on sticky surfaces in the supply bin.	20
Step 22. Place the old SDHE unit in the supply bin for later deorbiting on the remote manipulator spacecraft.	2

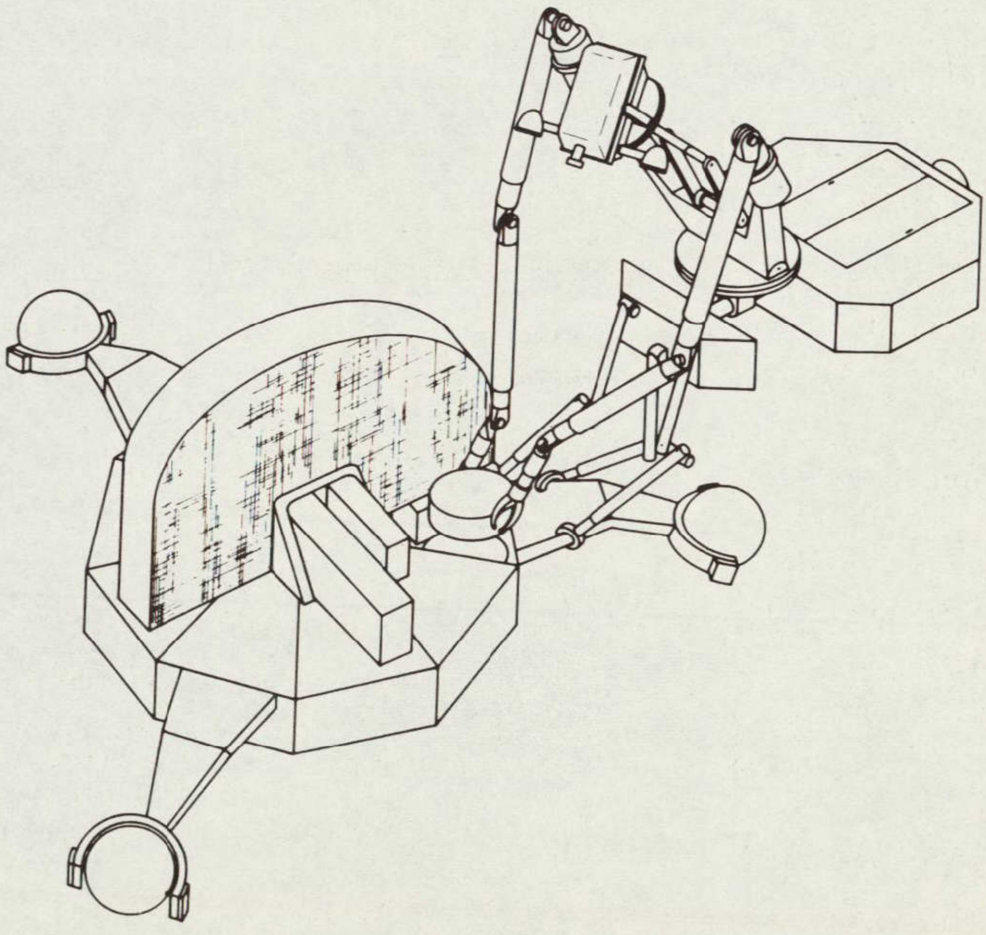
TASK	TIME (Minutes)
Step 23. Unclamp the new BCSC unit from the supply bin.	2
Step 24. Repeat steps 2 through 20 of Section 3.1.6.3. (Replace the BCSC), where required, in reverse fashion to replace the new BCSC unit. Skip steps 14 and 17 because the CIR-CLIPS are not necessary in-orbit. The special reset tool may be required on occasion to reset the ball-retaining band in the connector. The preflight preparations will simplify the replacement such as Allen-head screws/bolts, captive fasteners, pre-attached thermal insulation. The BCSC unit and the bay E upper skin panel and then the boom at bay E-1 are not replaced until the SDHE unit is replaced in bay F-2 by viewing one of the inboard mounting pins on the SDHE mounting plate. Fortunately, bays G-2 and G-1 do not have modules in them that would have to be removed in order to view the other inboard mounting pin; however, thermal insulation has to be uptaped and folded back locally. It is not necessary to replace all skin panels screws.	100
Step 25. Unclamp the new SDHE unit from the supply bin.	2
Step 26. Repeat steps 2 through 21 above, where required, in reverse fashion to replace the SDHE unit. Skip steps 11 and 12 because CIR-CLIPS are not necessary in-orbit. The special reset tool may be required on occasion to reset the ball-retaining band in the connector. It is not necessary to replace all skin panel screws. The preflight preparations will simplify the replacement such as allen-head screws/bolts captive fasteners, pre-attached thermal insulation, and new SDHE mounting plate attachment fasteners.	120

256 minutes are required to remove and replace SDHE.

986 minutes total for the four OAO A1 repair tasks.

3.2 OSO-D SATELLITE REPAIR

The Orbiting Solar Observatory (OSO) is a satellite platform for experiments intended primarily to observe the sun and to study its influence on the interplanetary space near the earth (see Figure 3-26). The orbit has a 354 nm apogee, 336 nm perigee, 32.9 degree inclination, and 95.9 minute period. The primary objective of OSO is to obtain high-resolution spectral data within the 1 Å to 1350 Å range. However, the experiments measure solar x-rays, gamma rays (with energy levels in excess of 100 mev), UV, and other phases of solar activity such as the intensity of the nuclear components of primary cosmic radiation and the long-term radiation effects on selected surfaces. Details of the OSO-D mission were taken from Reference 16, 17 and 18. Details of the satellite design, which also appear in Appendix B, were taken from References 15, 19, and 21. Reference 20 provided some design data on docking to a spinning OSO.



OSO-D Satellite Repair

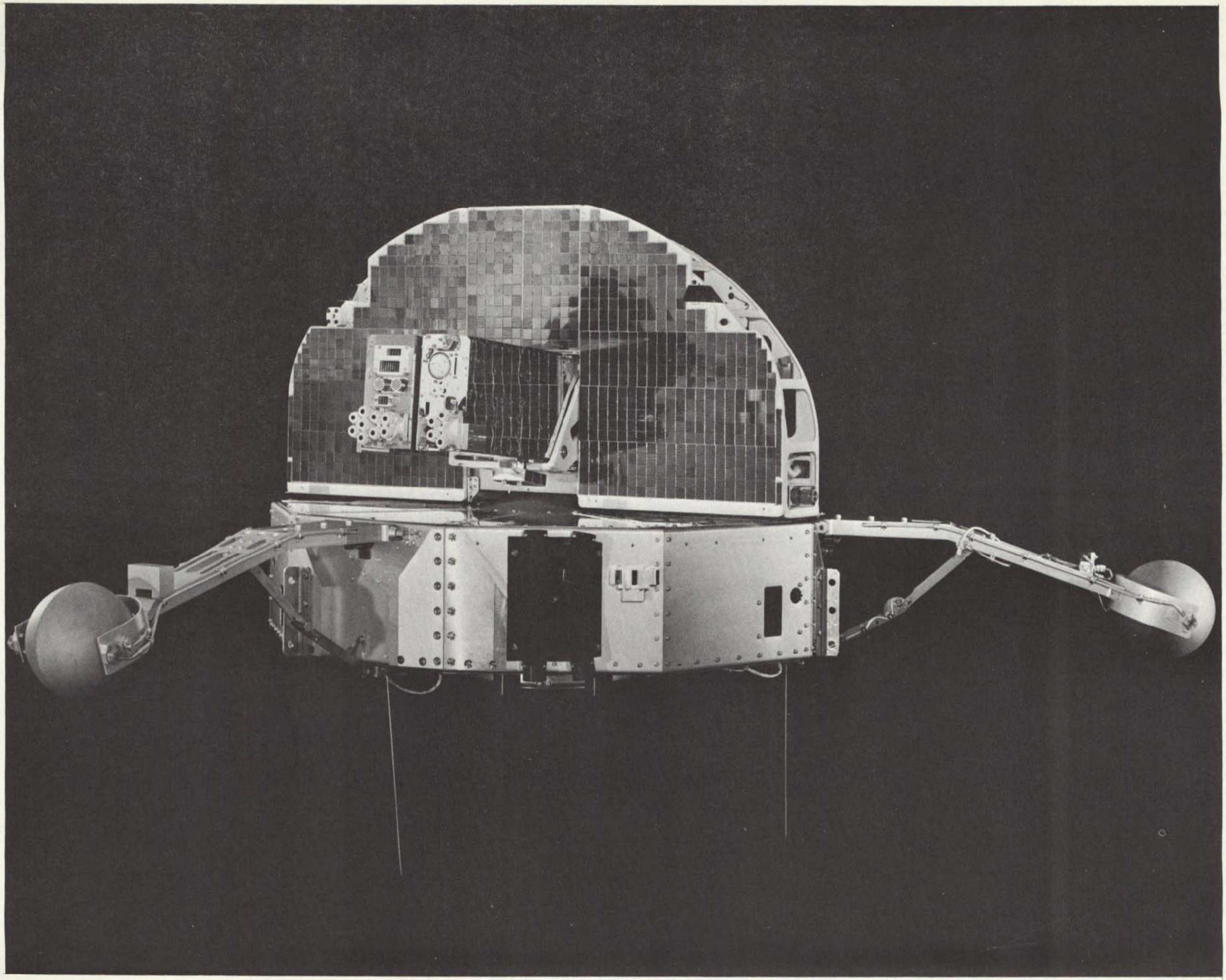


Figure 3-26 Orbiting Solar Observatory

3.2.1 FAILURE EFFECTS ANALYSIS

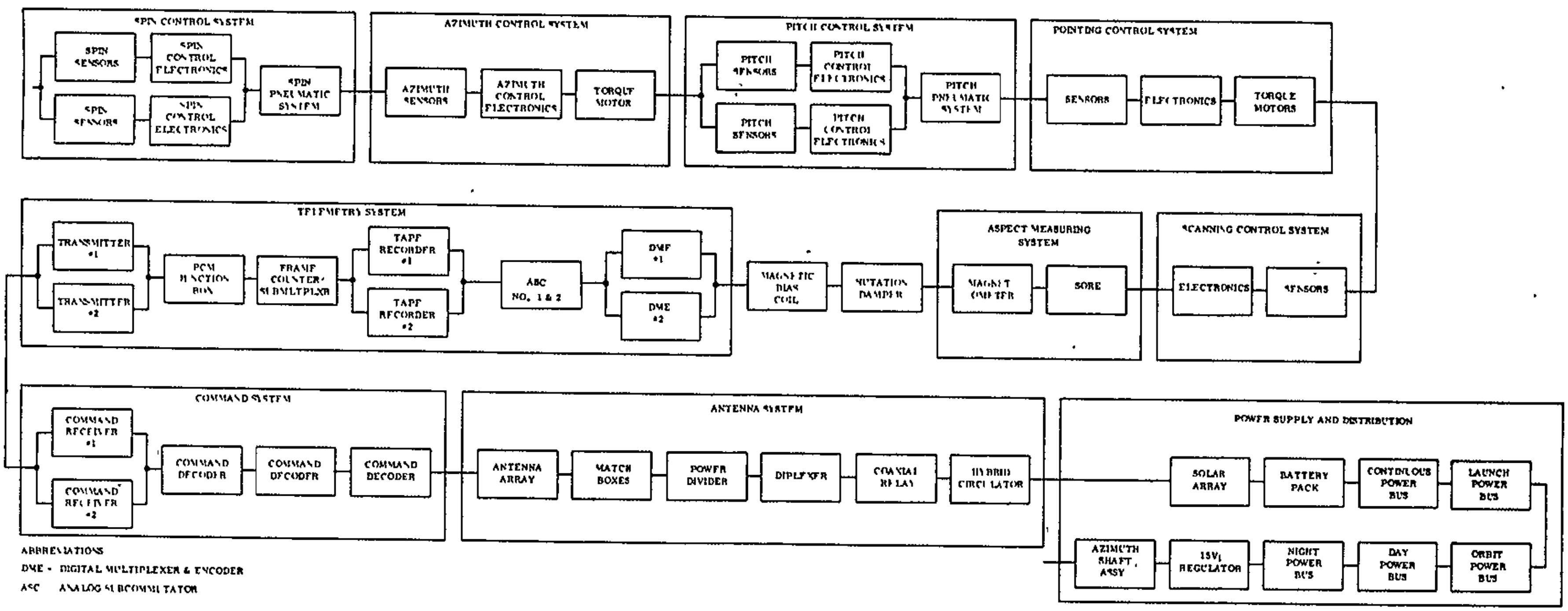
In order to define a repair candidate for the OSO spacecraft, a failure effects analysis was performed on each subsystem to identify the possible functional failure modes. Each failure mode was then ranked for criticality and probability of occurrence. The product of these two rank values for each item resulted in a total ranking which was used for selection of the repair candidate.

The OSO failure effects analysis is given in Appendix D. In this analysis, the indicated failure effect reflects the impact of the failure on system performance. There are, of course, many failure modes within each item which would result in some level of degraded performance rather than complete failure. We are, however, concerned with identifying the most critical failures, and, therefore, no consideration was given to degraded modes within each item.

The criticality ranking for each item was established by reference to the OSO reliability block diagram supplemented by the estimated impact of each failure on overall system performance. The reliability block diagram for the OSO-D spacecraft functions is shown in Figure 3-27. The diagram indicates requirements for complete mission success, and shows redundancy at the component level only. Many of the components have internal redundancy at the piece part or subassembly level, which is reflected in the probability of failure ranking of Table D-1.

The criticality ranking was assigned as follows:

Rank Value	Estimated Effect on System Performance
1	Practically None
2	Loss of up to 25% of data
3	Loss of 25% to 50% of data
4	Loss of 50% to 75% of data
5	Loss of more than 75% of data.



ABBREVIATIONS
DME - DIGITAL MULTIPLEXER & ENCODER
ASC ANALOG SUBCOMMUTATOR

Figure 3-27. OSO-D Reliability Block Diagram

The assignment of a probability of failure ranking for each item in the failure effects analysis was based on generic failure data for each item. For this purpose, the failure rate data in references 13 and 14 were reviewed for both components and functions. The estimated failure rate for any component will depend on many factors such as: application conditions, number of failure experiences, and total number of component operating hours accumulated. It is not the intent here to establish accurate failure rates for each component or function, but rather to rank each item on the failure effects analysis for the purpose of selecting a repair candidate. These generic ranking values were, therefore, established as follows:

Rank Value	Estimated Probability of Failure
1	Low
2	Medium
3	High

The probability of failure rank values shown on the failure effects analysis pertain to the indicated function. Design redundancy was considered wherever applicable. For example, the Digital Multiplexer and Encoder component would have a probability of failure rank of 2; however, since two components are used in a block redundant configuration, the resulting rank is 1. Similarly, a single tape recorder has a rank of 3; however, the system contains a redundant recorder and the rank for the tape recorder function is therefore 2. In general, where redundancy is used, the probability of failure rank for the item is one lower than the value for the function without redundancy.

3.2.2 FAILURE SELECTION

The items with the highest total ranking were the Azimuth Shaft Assembly, the Pneumatic System, and the Tape Recorders. The tape recorder was selected for the OSO repair mission because they are a known limited-life item, and also because both tape recorders had failed during the OSO-D mission.

3.2.2.1 Tape Recorder Failure Modes

Most tape recorder failures are caused by failures of the mechanical components or devices. The elements which fail most frequently are motors, bearings, clutches, belts, and the magnetic tapes.

Specific information on the OSO tape recorder flight failures was not readily available, but analyses of the Nimbus recorder failures indicated that they were most likely caused by broken tapes and jammed motors.

The basic reliability problems of tape recorders are as follows:

a. The recorders have limited life; this is especially true of the tape itself, and the condition is aggrevated when an endless loop tape is used. Much of the available operating life can easily be used up during ground testing resulting in early mission failures.

b. Tolerances are very critical and proper adjustment is difficult to maintain over long periods of time due to tape wear and collection of debris from the tape.

c. Failures can be induced by the launch environment. For this reason, tape recorders are often operated during launch to reduce the risk of failure.

3.2.2.2 Additional Considerations

OSO tape recorder failures are not completely mission terminating. There are two recorders used in a block redundant configuration. The tape recorders record information during the entire 95 minute orbit and on command play back the recorded information from the previous orbit in 5 minutes. Real time data transmission also occurs simultaneously with data recording. Assuming that real time data is collected once each orbit for a period of 14 minutes, a total of 3.73 hours of data is collected per day. This can be interpreted as a mission effectiveness of approximately 16 percent.

3.2.3 MAXIMUM ALLOWABLE TIME TO BEGIN REPAIR

There is no limitation on time to initiate on-orbit repair action. The remote manipulator spacecraft can be launched at any time provided enough useful satellite life remains after repair.

3.2.4 SATELLITE STATE

3.2.4.1 Disabling/Enabling

The OSO-D is assumed to be uncooperative. The spin-gas of the wheel spin-system is depleted and will be recharged as part of the recorder repair mission; therefore, the spinning OSO-D cannot be despun by either the automatic or the command manual spin-despin systems.

The following command actions are taken to further disable the satellite prior to docking:

- Wheel and sail attitude control are commanded off.
- The experiments are commanded off.
- Power to the recorder is commanded off.

As a result of these commands, the OSO-D will not react to any of the docking actions of the remote manipulator spacecraft. The freely rotating sail spins up to the wheel spin-rate (resulting in a combined spin rate of 26 rpm) when the power is commanded off or during satellite night. After docking, the remote manipulator spacecraft clamps the freely rotating sail to the wheel to assure that neither the sail nor the two, long transverse experiments do not strike the spacecraft nor block access to the recorder compartment and the wheel spin-gas charging valve.

The attitude of the OSO-D is quite stable before, during and after docking, the 26 rpm spin rate about the yaw axis gyrostabilizes the spacecraft in roll. Instrumentation on OSO-B indicated a spin-rate decay of approximately 2 percent per month or 80 percent of nominal after a year. The satellite rotates about the roll axis approximately one degree per day. Furthermore, the OSO-D pitch attitude and rate, controlled by the cold-gas pitch system on the sail, are quite stable as shown by the few pitch corrections required in orbit; the pitch damper in the wheel that utilizes the earth's magnetic field is effective. The wobble is only 10 minutes of arc due to the balanced state of the spacecraft. The nutation damper at the top of the sail limits the complex motion of combined wobble and nutation to one degree of arc until the spacecraft settles down to the small wobble motion.

After the OSO-D recorder is repaired and the wheel spin-gas recharged, the OSO-D (including the unclamped sail) is spun up by its own spin-gas to the nominal 30 rpm initially by ground command and then by the automatic spin-system.

3.2.4.2 Docking and Locomotion

Docking the remote manipulator to the spinning uncooperative OSO-D is achieved by utilizing a synchronized, attachment head on a fixture held by the manipulator (Figures 3-28, 3-29, and 3-30). Figures 3-28 and 3-29 show rotating heads that are synchronized to the satellite spin rate then centered visually on the adapter interface flange on the OSO-D and finally clamped or engaged to the satellite. Three toggle clamps (Figure 3-28) are automatically triggered individually to snap close on the flange upon toggle release. After clearing the three whip antennas, (Figure 3-29) three arms engage the satellite de-spin gas arms. The nesting surfaces on the heads assist in the centering. The battery powered electric motor spins up the heads before engagement, and despins the satellite. Figure 3-30 shows an alternate docking head that is not electrically powered but despins the OSO-D through viscous or eddy current devices after the initial brake shoe initially grasps the OSO-D.

Inadvertent, off-center bumping of the adapter interface flange by the heads during both the synchronization centering phase and the clamping phase will not disturb the satellite spin axis appreciably because of the high degree of OSO gyro stabilization. The docking heads can wobble about the pitch and yaw axes slightly (3 degrees limit stops) to compensate

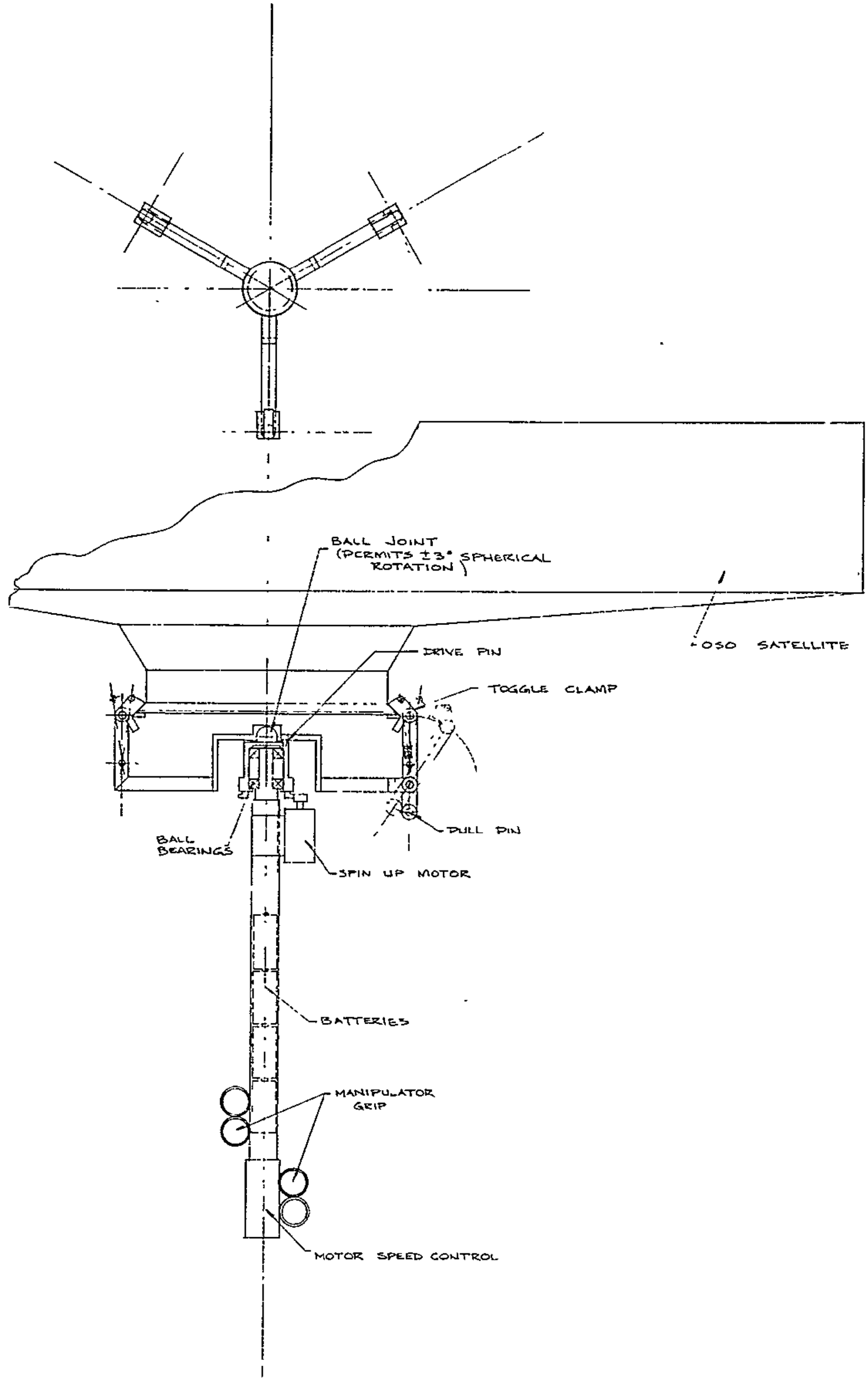


Figure 3-28. OSO-D Despin Fixture Rotating Attachment Head

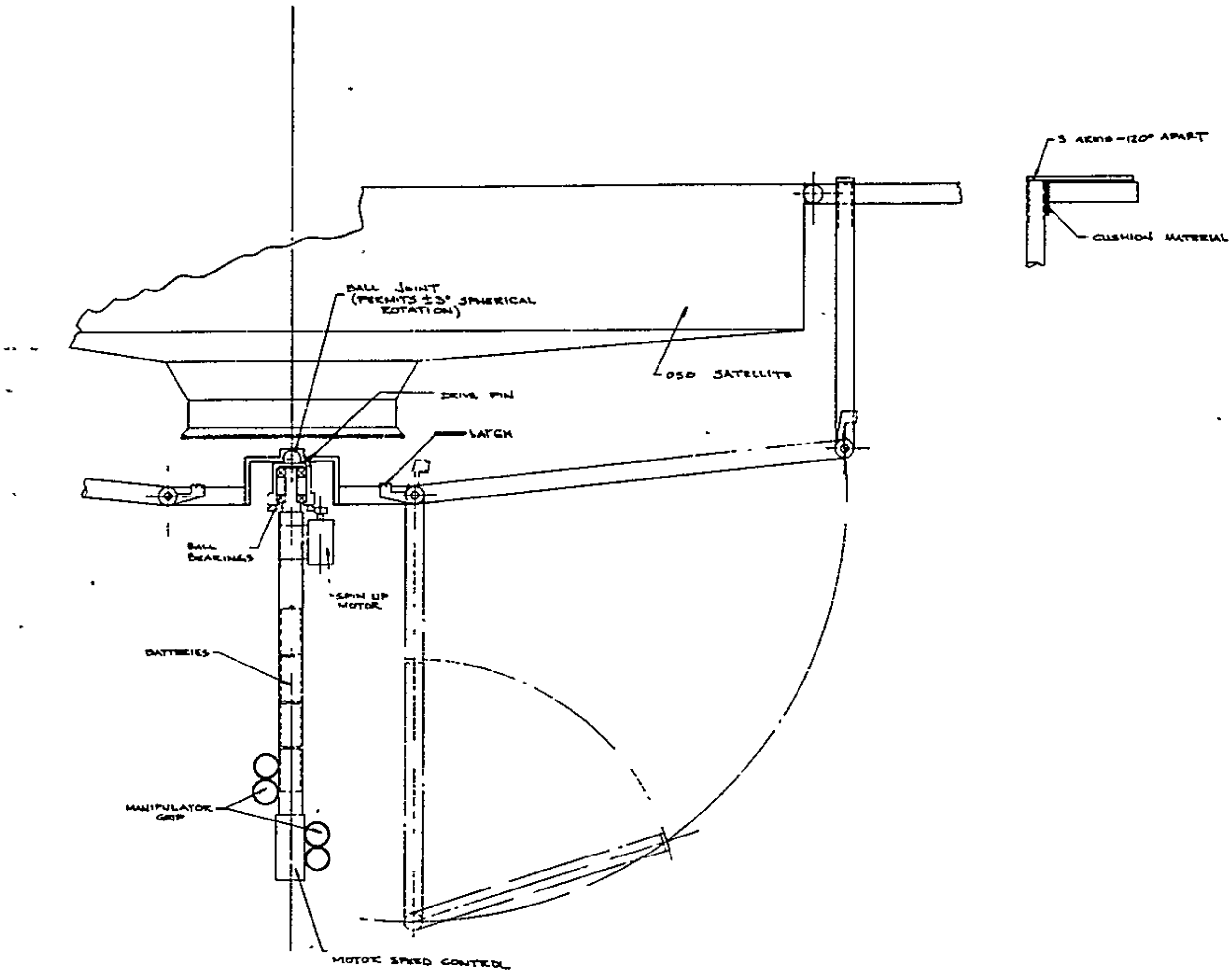


Figure 3-29. OSO-D Despin Fixture Rotating Hook Arms

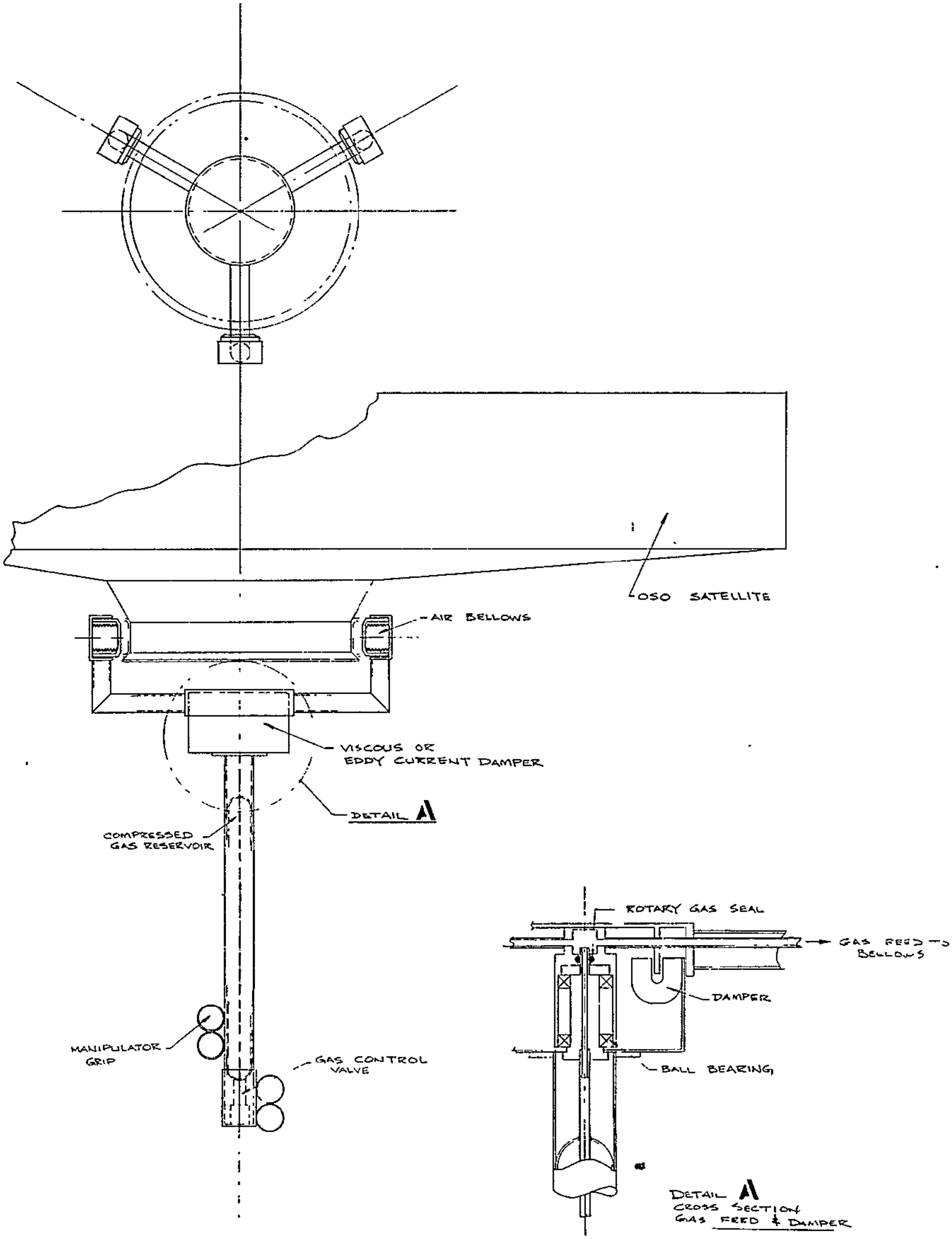


Figure 3-30. OSO-D Fixture Nonrotating Air Bellows Attachment Head

for mismatching due to centering tolerances, nutation and wobble of the OSO-D spin axis that would feed back to the remote manipulator. Wobble occurs especially when the sail is free to rotate because of the relatively loosely-controlled mass balance of the sail. If the satellite spin gas systems have been turned off long enough, the satellite will achieve a steady state condition in which the spin momentum vector is aligned with the axis of maximum moment of inertia. In this steady state condition, a pure wobble of not more than 10 arc minutes will be present. If the satellite is not in the steady state condition, then a transient condition may exist in which a combination of wobble and nutation of several times the 10 arc minutes is possible. This transient condition is characterized by a beat frequency between the spin and nutation frequencies. This beat frequency occurs because mass unbalances have the effect of applying a moment (rotating at the spin frequency) to the satellite. Fortunately the estimated combined motion of wobble and rotation is only 1 degree and easily can be accommodated by the wobble device in the attachment heads.

Docking to the adapter interface (Figure 3-28) is preferred to engaging the three spin-gas arms (Figure 3-28) for several reasons:

- The docking attachment is localized.
- There is less chance of damage to the OSO experiments, the 3 axial whip antennas (at the spin-gas arm hinges), and particularly the spin gas pneumatic lines on the arms.
- The de-spin fixtures are centered visually on the adapter interface.
- The adapter interface flange offers 360 degrees of attachment surface; thus, indexing about the yaw axis between the protruding satellite components during docking is not required. Also, the flange has no functional use after separation; thus, possible brinelling damage to it during docking is acceptable.
- The docking process can be aborted more quickly and attempted again if the occasion arises.
- The adapter interface structure on the OSO is adequate for the docking and de-spinning loads since it is designed for the OSO launch loads.

An advantage of the fixture in Figure 3-29 is that it can despin the OSO without exerting any forces along the spin axis except for possibly a minor pulling force to assure that the hooks on the fixture arms remain engaged with the OSO-D spin gas arms.

Reference 10 includes investigations of other docking techniques such as nets, bolar tape, and a synchronized adhesive pad; synchronized mechanical attachment techniques also were preferred in that study. A synchronized three arm fixture that hooks onto the three spin gas arms as shown in Figure 3-29 and a rotating interface attachment head as shown in Figure 3-27 were described there.

After docking, remote manipulator locomotion on the OSO-D is easy because:

- The OSO-D size is approximately the same as the remote manipulator size and all of it is within reach of the remote manipulator docking legs and manipulator arms. As a result, the need to travel on the OSO is small.
- The grip holds such as the spin gas arms (Figure 3-31) and adapter interface flange are adequate in strength, rigidity, number, and distribution.
- The remote manipulator can clamp the rotatable sail where desired in order to reposition itself.

3.2.4.3 Time Required to Dock

The analysis described in Reference 20 was used. The time required to dock is based on the following remote manipulator spacecraft docking tasks.

	TASK	TIME (Minutes)
Step 1.	Remote manipulator is initially at a position 300 feet from the spacecraft.	
Step 2.	Remote manipulator spacecraft travels to a position of close proximity, to the spacecraft during daytime of first orbit, approximately 10 feet.	10

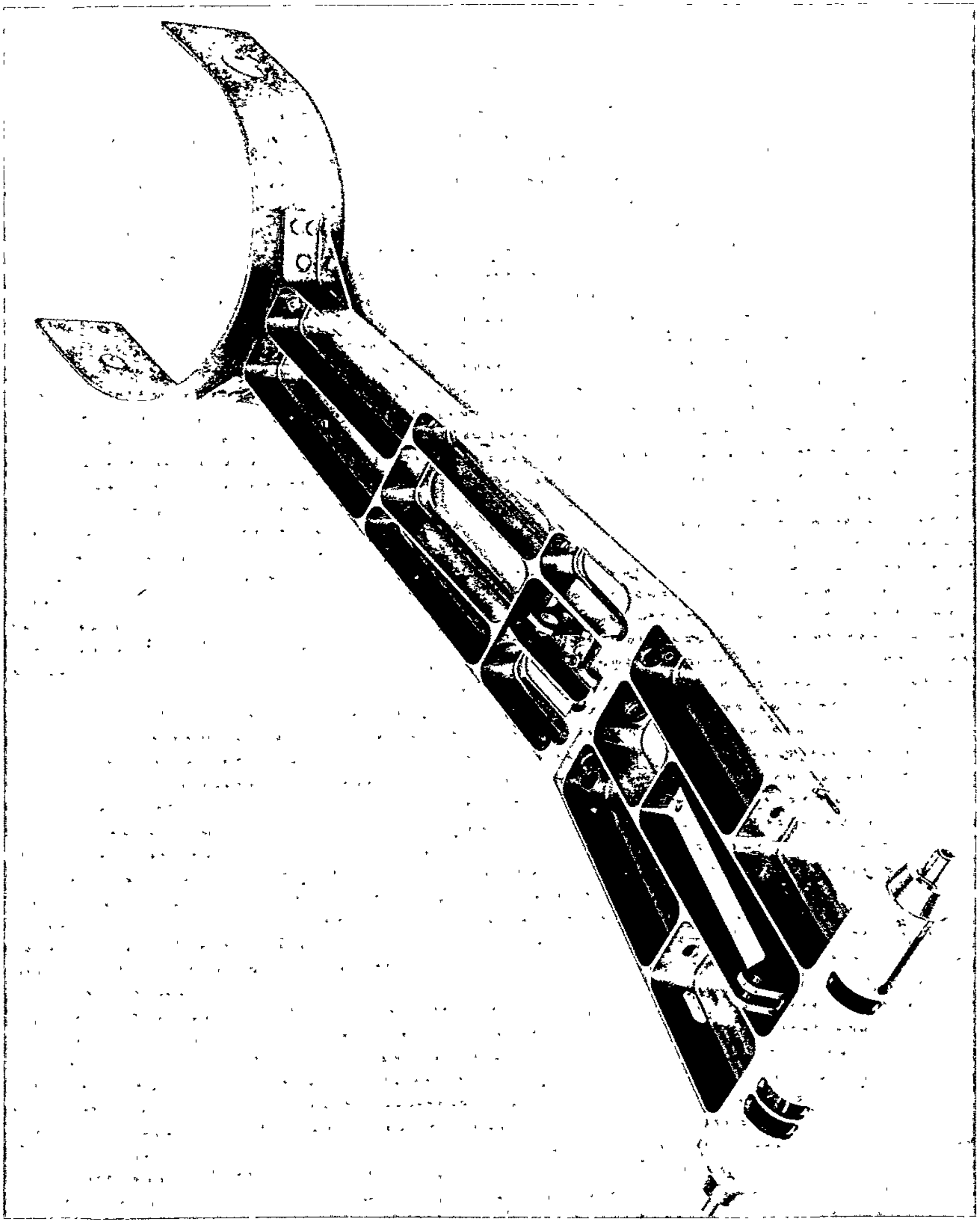


Figure 3-31. Spin-Gas Arm

	TASK	TIME (Minutes)
Step 3.	Remote manipulator spacecraft performs stationkeeping and inspection functions.	15
Step 4.	Remote manipulator spacecraft continues to circumnavigate the spacecraft and conducts predocking functions such as:	15
	a. Observing and determining the attitude rates of the spacecraft. For the high yaw rate the remote manipulator spacecraft positions itself along the yaw axis in order to accurately determine the yaw rate.	
	b. Observing the condition of the spacecraft for unusual colorations, loose or damaged appendages, erection state of the appendages, temperatures, etc.	
Step 5.	The remote manipulator travels to the satellite and aligns itself with the docking axis. (Probably the yaw axis.)	5
Step 6.	The remote manipulator engages the despinning fixture and despins the satellite.	20
Step 7.	The remote manipulator re-orients the satellite for appropriate thermal control and communications with the relay satellites and ground station.	5

Total time if satellite need not be despun: 50 minutes
Total time if satellite must be despun: 70 minutes

3.2.5 ORIENTATION OF OSO DURING MAINTENANCE MISSION

If the remote manipulator spacecraft holds the de-spun OSO-D in a fixed attitude throughout the repair phases of the maintenance mission, thermal problems will arise. The OSO balances the thermal input from the sun on the wheel rim panels through the action of the spin rate and by transferring heat to the top and bottom of the wheel where it is radiated to deep space. When the wheel is despun, the side of the wheel facing the sun increases in temperature, and the opposite side decreases in temperature. In direct sunlight, the temperature gradient reaches levels beyond the design limit within a few minutes after stopping the wheel. Therefore, to maintain the temperature in the OSO within acceptable limits, the remote manipulator spacecraft must either periodically rotate the spacecraft with respect to the sun or control the OSO-D temperature with thermal shields, shades, etc.

3.2.6 OSO REPAIR MISSION

3.2.6.1 Replace the Upper Recorder

Only the upper tape recorder of the two stacked recorders (Figure 3-32) in compartment No. 8 (Figure 3-33) is replaced in orbit for several reasons:

- The bottom recorder is very inaccessible and requires 8 to 10 hours of technician time to remove on the ground. Two compartment panels and the telemetry transmitter must be removed before the lower recorder becomes accessible (Figure 3-33).
- A second new recorder could not be clamped to the outside of the wheel although wheel unbalance and increased wheel spin/despin gas consumption would be tolerable, because of very low accessibility of the two cable connectors of the lower recorder.
- The recorders are redundant for the real time transmission of OSO data, and replacement of the upper recorder restores the satellite to full operational status.

The new recorder unit with integral recorder clamps and jack cables will be installed above the lower recorder in compartment No. 8. There are fewer thermal insulation blankets and no severe alignment and thermal deflection problems as an OAO-A1 mainly because the pointing accuracy of almost all OSO experiments is 3 degrees. In OSO-G and OSO-H the three stanchions mounted to a base plate (Figure 3-31) are no longer a two-piece design but are a one-piece design which allows both the upper and lower recorders to be easily and quickly replaced through the top of the wheel. A service loop for cable slack is also provided for the bottom recorder. These are good examples of minor design changes that greatly improve in-orbit maintenance.

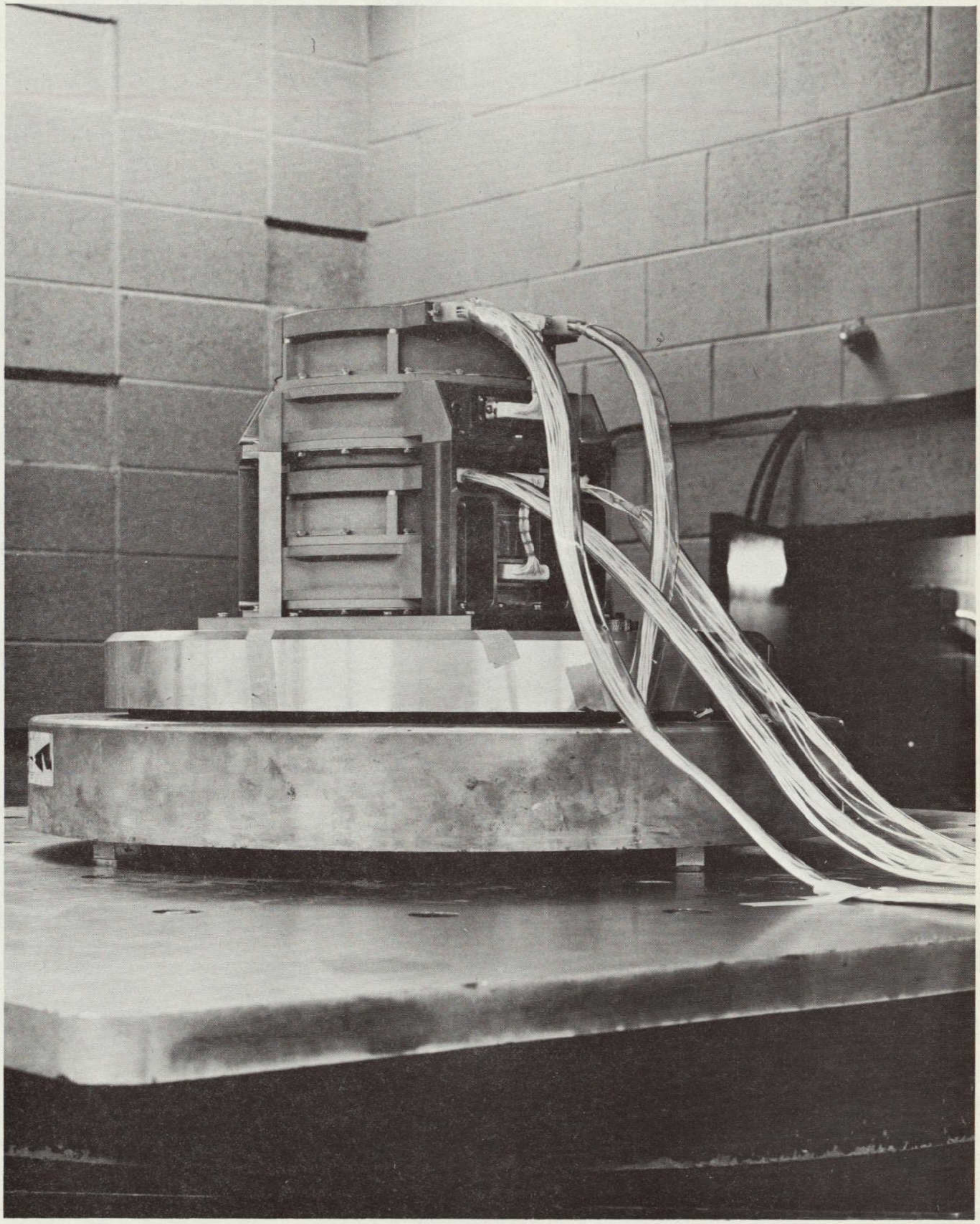


Figure 3-32. Tape Recorders

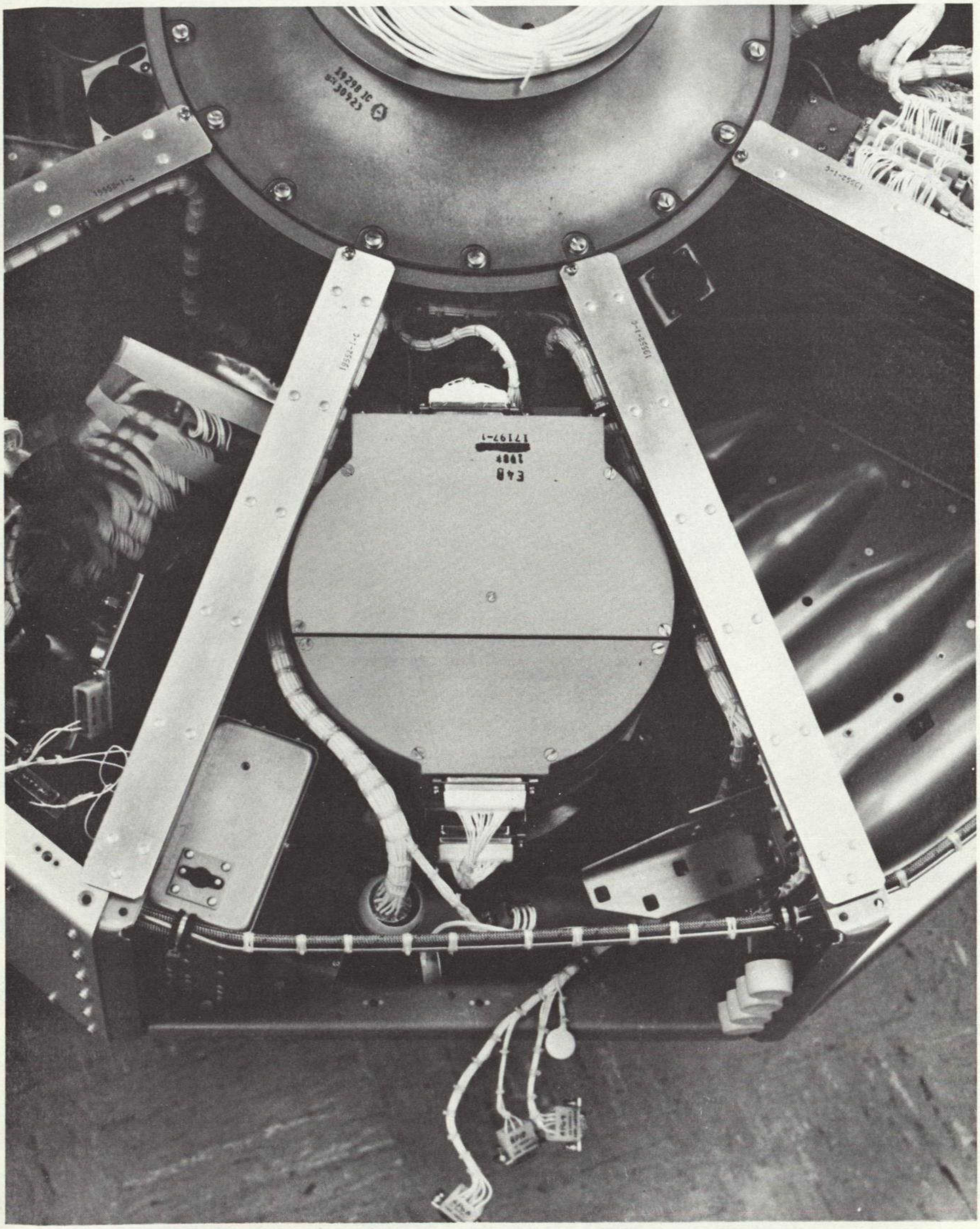


Figure 3-33. Tape Recorders in Compartment No. 8

The following pre-flight preparations would simplify the recorder repair mission

- The new recorder has integral clamps on it for mounting the recorder wheel structure. The compartment cover is integral with the new recorder. The rigid recorder base-plate and cases preclude shimming in order to avoid objectionable mounting strains.
- The new recorder unit has jack cables attached to the 2 Cannon connectors in order to provide adequate cable slack for mating the connectors.
- The new recorder unit has handles on it to facilitate handling.
- Color coding, numbering, stripping, etc. on connectors, units, etc. are used.

Reposition the remote manipulator spacecraft from the OSO adapter ring to a position above the wheel. Grip, as required, the spin gas arms (being careful of the spin-gas pneumatic lines) and the open, cellular structure on the back of the clamped sail. Replace the recorder by performing the following steps:

	TASK	TIME (Minutes)
Step 1.	Reach into the tool bin and mate the special blade-type screwdriver bit to the power tool.	2
Step 2.	Unscrew the 7 slotted head screws securing the compartment cover (Figure 3-33) and store them and their underhead washers on sticky surfaces on the supply bin. (Their self locking nuts are captive on the wheel structure.)	30
Step 3.	Demate and replace the special screwdriver bit and power tool in the tool bin.	1
Step 4.	Slide the compartment cover radially outward from under the two, radial, overhanging, fiberglass rails (Figure 3-34) and store it in the supply bin.	5
Step 5.	Reach into the tool bin and mate the special Allen-wrench to the power tool.	2

TASK		TIME (Minutes)
Step 6.	Unscrew the 6 screws (Figure 3-35 shows 4 of the 6 holes) and store them and their underhead washers on sticky surfaces on the supply bin.	48
Step 7.	Demate and replace the special Allen-wrench and power tool in the tool bin.	1
Step 8.	Unclamp a stored semi-rigid tether and clamp it to the old recorder unit.	3
Step 9.	Extract the old recorder as far as possible from the compartment for good access to the 2 Cannon connectors on it. Plastic cable clamps and lacing have to be removed locally because the connectors are at the narrow end of the wedge-shaped compartment. The tether will keep the old recorder unit in that accessible position.	13
Step 10.	Reach into the tool bin and mate the special blade type screwdriver bit to the power tool.	2
Step 11.	Unscrew the two captive screws on each of the four Cannon connectors.	4
Step 12.	Demate and replace the special screwdriver bit and power tool in the tool bin.	1
Step 13.	Demate the 2 Cannon connectors and store the 2 cables out of the way for the reinstallation phase (tape can be used).	2
Step 14.	Fully extract the old recorder.	4
Step 15.	Unclamp the rigidized tether from the old recorder and store the tether and recorder on the remote manipulator spacecraft which will be deorbited later.	3

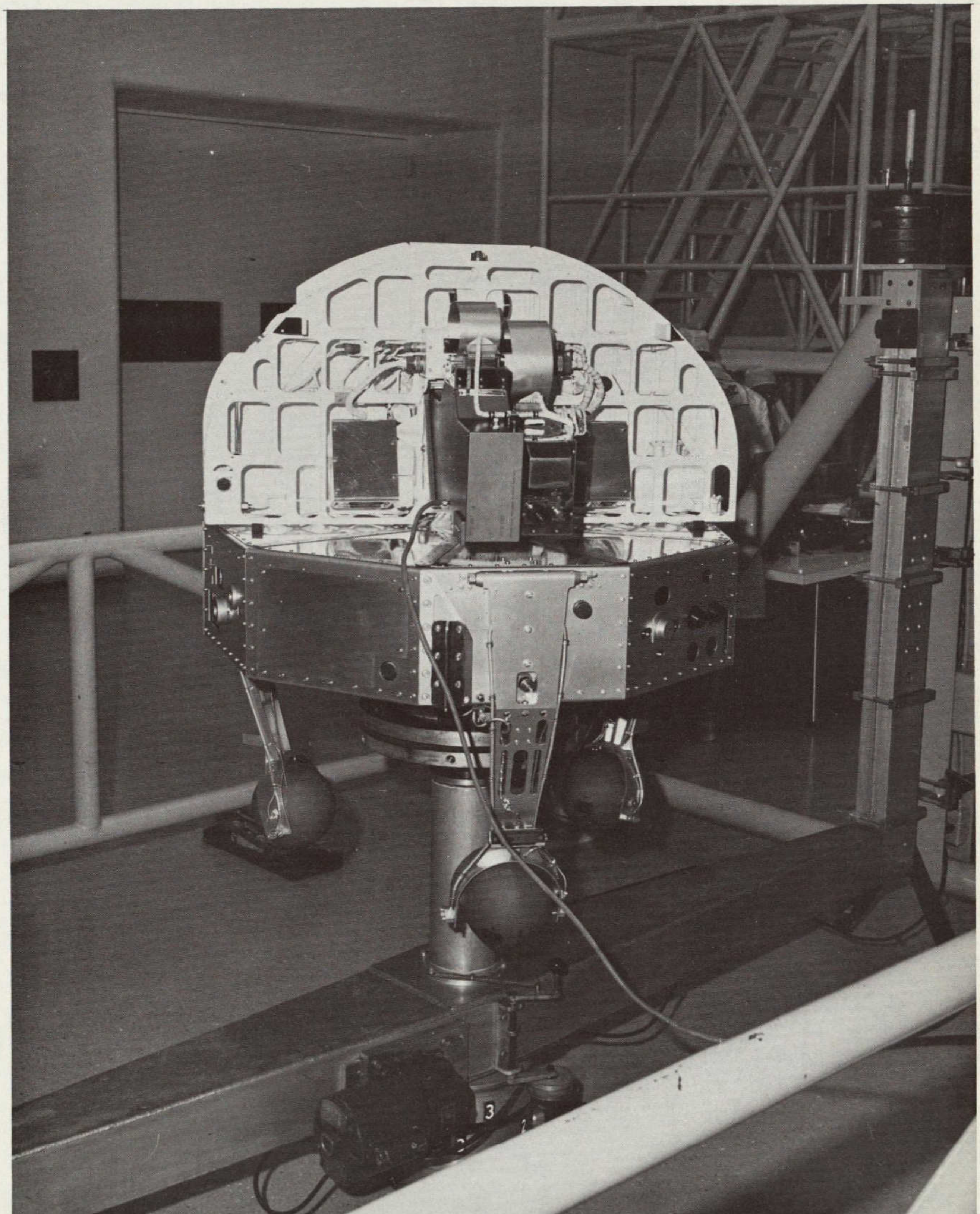


Figure 3-34. OSO-D Showing Compartment Cover Screws and Gas Charge Fitting

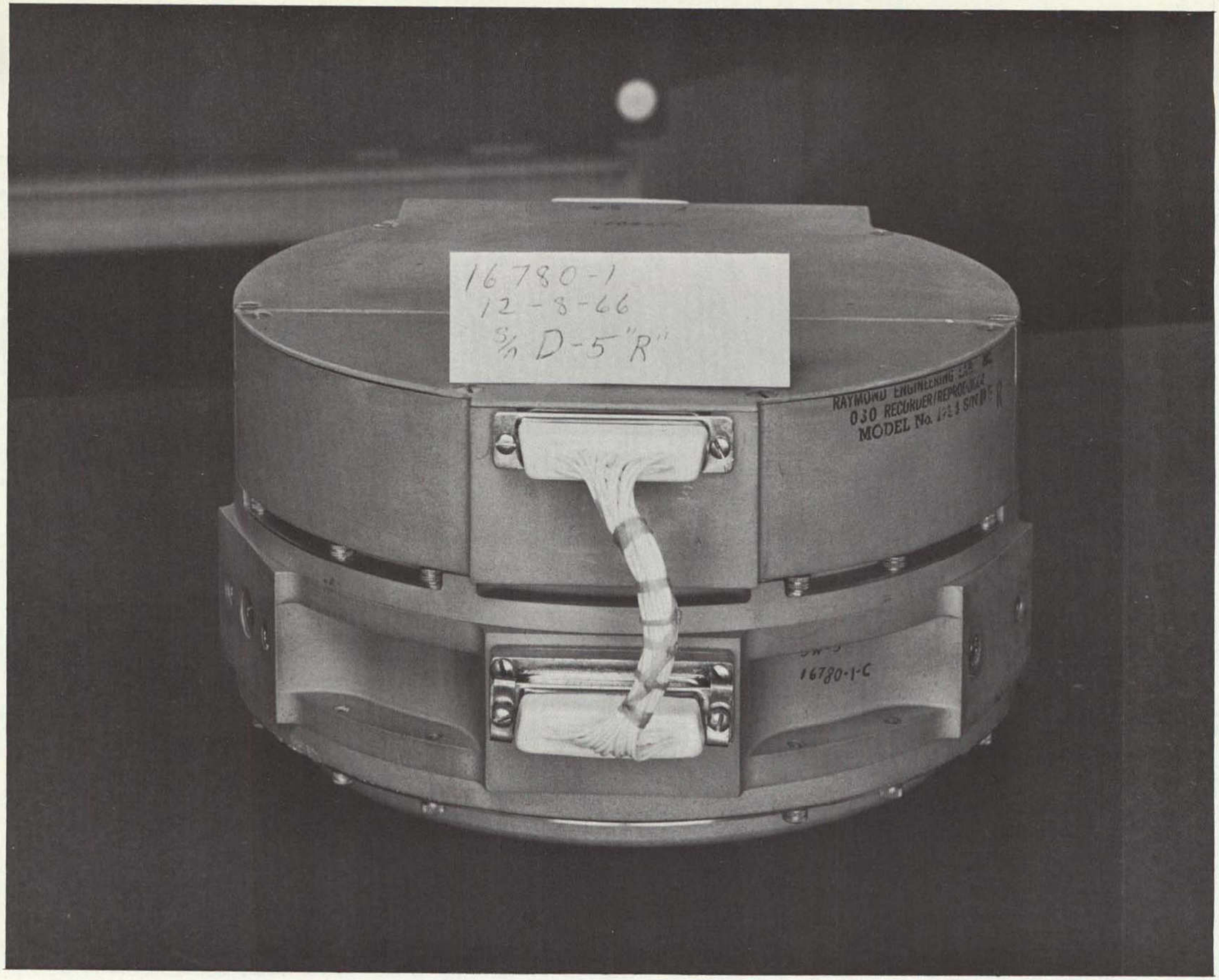


Figure 3-35. OSO-D Recorder

TASK		TIME (Minutes)
Step 16.	Install the new recorder unit according to steps 1 through 15, as required, in reverse fashion. The preflight preparation will facilitate the reinstallation such as integral mounting clamps on the new recorder unit, jack cables, captive cover screws (Allen-head), recorder handles, etc.	114
	Total time to replace the upper recorder.	235

3.2.6.2 Recharge the Wheel Spin-Gas System

The charge fitting for the wheel spin-gas system (3000 psi) is exposed on the side of the wheel (Figure 3-34); thus, no covers or insulation need be removed to gain access to it. The steps to recharge the system are few; they include removal of a threaded cap over the fitting and attachment of the fill nozzle to the charge fitting.

The following pre-launch preparation will simplify the spin-gas recharge task by the RMS:

- The filler nozzle wing-nut has a lead-in to the charge fitting on the OSO. It also orients the nozzle attachment perpendicularly to the charge fitting.

Total time to recharge the wheel spin-gas system	30 minutes
Total time to perform the two OSO-D tasks	265 minutes

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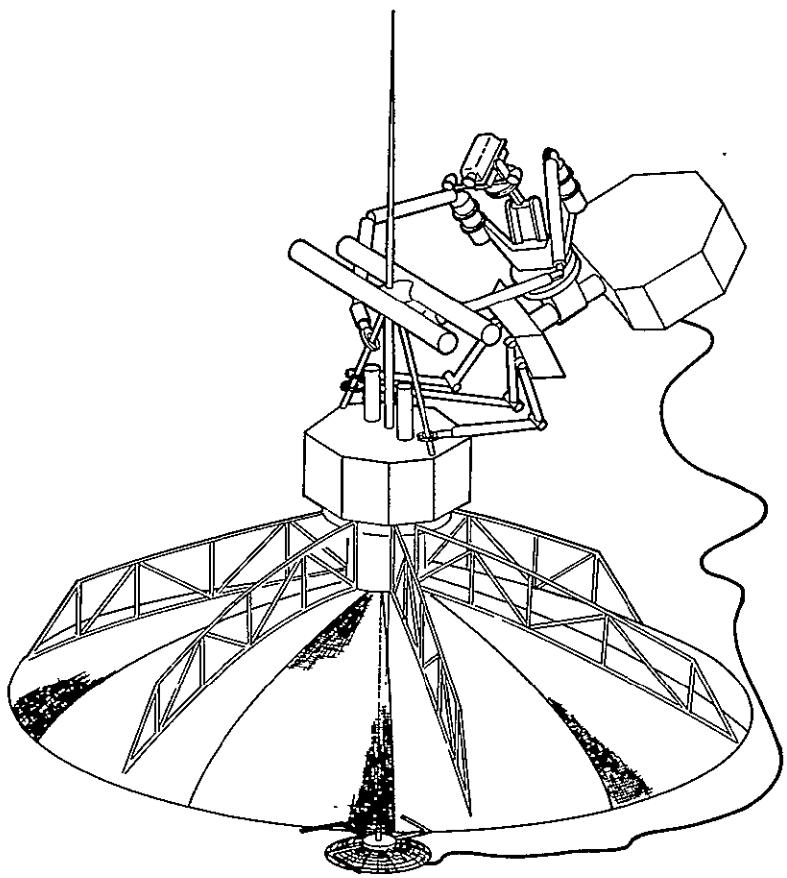
SECTION 4

MISSION ANALYSIS - SATELLITE REFURBISHMENT

On-orbit refurbishment differs from repair in several ways. There is no critical timing factor as exists in repair. Refurbishment missions can be planned and rehearsed well in advance. The nature of the tasks can be defined more precisely in terms of remote manipulator capability. There is a minimum risk that the state of the satellite system will be incorrectly diagnosed prior to the mission. The satellite to be refurbished will usually be cooperative and responsive to ground commands. Nevertheless, there can be peculiarities associated with refurbishment. Thus, two missions are examined; the refurbishment of a direct broadcast satellite into a community television satellite for India and the refurbishment of the Nimbus experiment payloads to later experiment payloads.



DBS Refurbishment

DBS Refurbishment

As in the case of repair, the procedures required to perform the refurbishment are defined, laboratory simulations of key tasks performed, and spare parts, special equipment and tools identified. The mission duration and manipulator requirements also are determined.

4.1 DBS REFURBISHMENT

The refurbishment of the Direct Broadcast Satellite-Voice Broadcast mission (DBS-VBM) was selected because it is an example of a mission that took place at synchronous altitude and involved a satellite which was in the conceptual design stage.

General Electric Space Systems Organization completed a study entitled Voice Broadcast Mission Study (Reference 1). The purpose of the Voice Broadcast Mission Study (VBMS) was to investigate the technological and cost factors associated with satellite designs for direct broadcast of voice programs to home receivers. The investigation included development of satellite conceptual designs for three frequency bands. These were the HF short wave band (15 to 26 MHz), the standard VHF-FM band (88 to 108 MHz), and the UHF-TV band (470 to 890 MHz) for voice-only transmissions to home TV receivers. A prime consideration in the conceptual designs was to minimize the need for modifications or expenditures to the home receiving system.

The study resulted in the definition of four conceptual satellite systems, one for the HF band, two for the VHF band, and one for the UHF band. The refurbishment mission analysis focused on the UHF configuration since it lent itself nicely to a meaningful refurbishment mission; namely, the conversion of the satellite from a voice broadcast satellite to a community TV broadcast satellite for India. This satellite would satisfy the need to supply an instructional, educational, and information dissemination service to that developing nation.

A separate study was performed in-house by the General Electric Space Systems Organization to define the requirements for the community TV broadcast satellite for India and to establish a preliminary satellite configuration. Segments of this study are reported in Reference 2. It was concluded after examination of the two systems that the refurbishment concept was feasible and that the tasks necessary to accomplish the mission should be defined.

4.1.1 SATELLITE CONFIGURATION VBMS-UHF

This satellite will be placed into a synchronous, equatorial orbit at 92°W longitude. It will provide 5 hours per day coverage to each United States time zone. Coverage of one zone at a time permits utilization of a directional high gain antenna. The 92°W longitude station is the midpoint of the central and largest time zone in the United States. Accurate pointing is provided by a satellite interferometer and ground beacon. An artist's pictorial representation and isometric are shown in Figure 4-1. Total orbital weight of the satellite is 2534 pounds. A description of the satellite appears in Appendix B.

4.1.2 SATELLITE CONFIGURATION - COMMUNITY BROADCAST FOR INDIA

This satellite will be placed into a synchronous, equatorial orbit at 77°E longitude. The satellite employs a sun orbit-normal reference system to fully orient the solar array to the sun, and a cooperative ground beacon for pointing the transmitter to the desired location on the earth. This requires a full 360 degrees per day rotation of the solar array assembly with respect to the antenna/body module, accomplished with low voltage DC slip rings. Seasonal inclination of the solar array to track the sun would be accomplished by a ± 23.5 degree motion using a flex harness. The satellite antenna package consists of a deployable 21-foot diameter UHF antenna with a concentric X-band antenna (8.4 GHz)

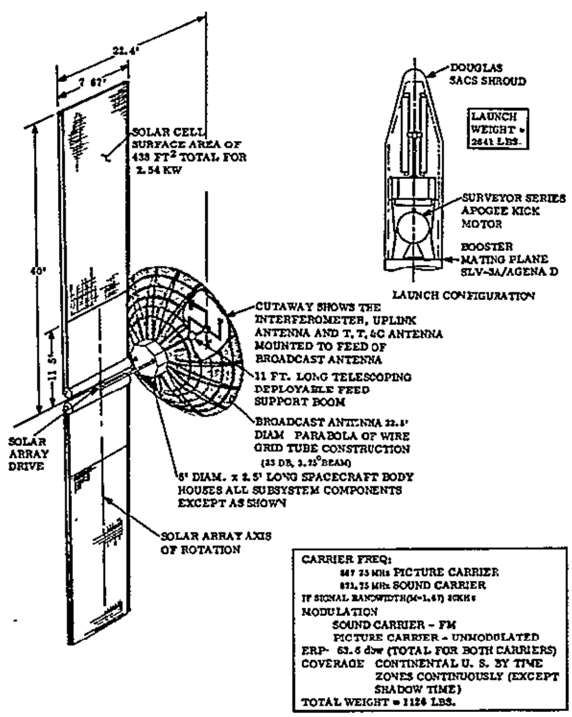


Figure 4-1. Direct Broadcast Satellite

mounted on the back of the prime focus feed. Highly accurate pointing of the feed is accomplished by error signal detection from an RF interferometer system. CFA and TWT power amplifiers are used for the UHF and X-band systems, respectively. The satellite configuration is shown in Figure 4-2. Total orbital weight of the satellite is 752 pounds. The satellite/payload weight summary and design summary appear in Appendix B.

4.1.3 FCC AND CCIR TRANSMISSION STANDARDS

The UHF VBM satellite communications subsystem is designed to transmit according to FCC standards. These standards call for:

- Aural and picture carrier frequency separation: 4.5 MHz
- Frequency deviation: ±25 KHz
- Pre-emphasis time constant: 75 microseconds

These transmissions are usable with receivers having corresponding characteristics and these are located in North America, Japan, and parts of South America. In Western Europe and India, the CCIR standards are used. These standards call for:

- Aural and picture carrier frequency separation: 5.5 MHz
- Frequency deviation: ±50 KHz
- Pre-emphasis time constant: 50 microseconds

4.1.4 REFURBISHMENT MISSION

Sections 4.1.1 through 4.1.3 and Appendix B examined the satellite configurations for the direct broadcast satellite - voice broadcast mission and the community TV service to India satellite mission.

The two configurations are similar in many respects. The conversion of the DBS-VBM into the community TV service mission was analyzed and it was determined that the following steps were necessary.

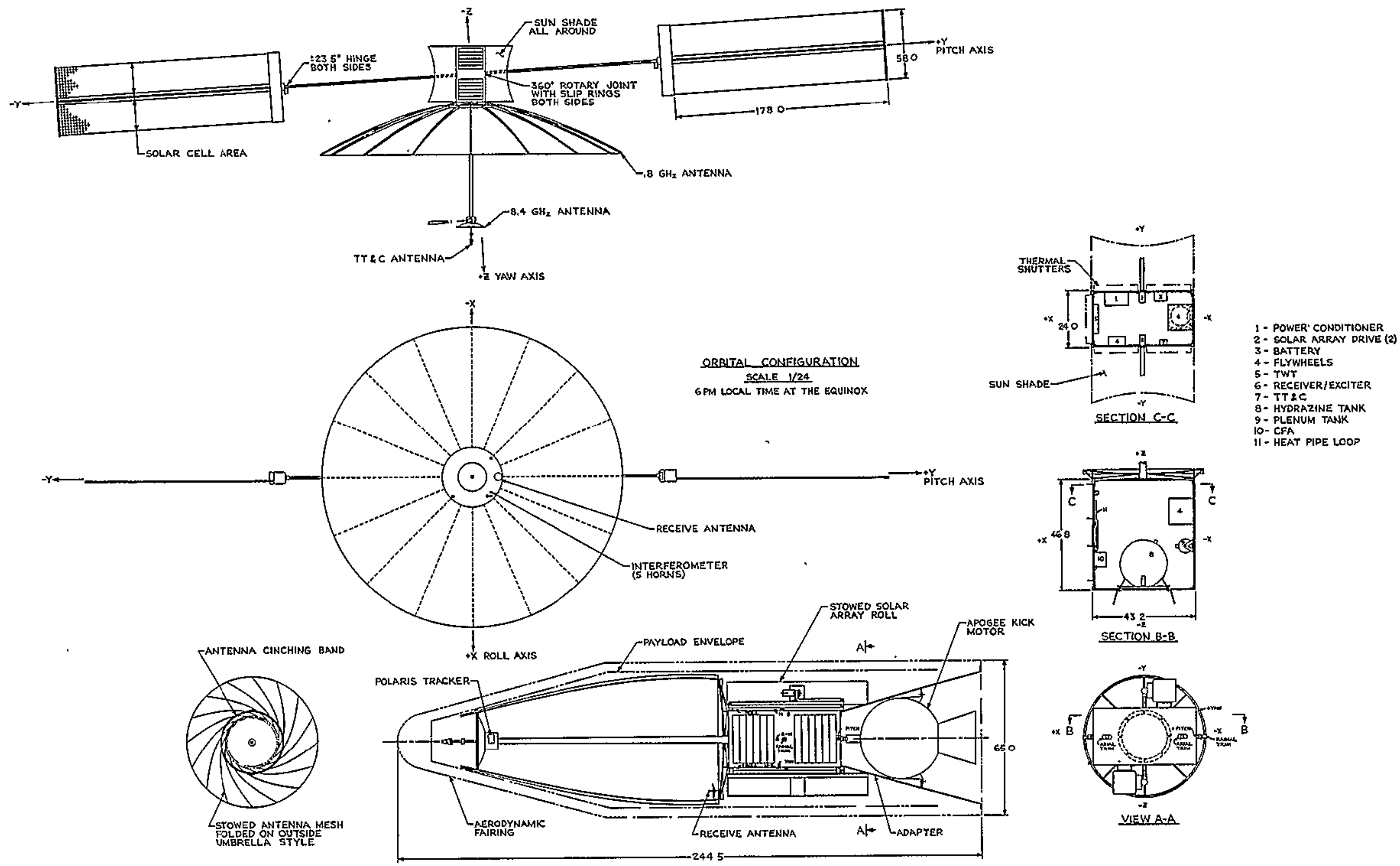


Figure 4-2. Television Broadcast Satellite Community Service to India

a. The TV-aural mode transponder should be replaced. This transponder was designed to extract the carrier signal through one receiver channel and the voice sidebands through a second receiver channel. Since these are narrow band, they would not be able to process modulated video for rebroadcast.

The wideband FM transponder may be designed to process program material conforming to FCC standards (broadcast to USA). There would be a degradation of video program quality if this transponder were used to process material conforming to CCIR standards and a loss of audio material since there is a greater separation between voice and video signals. This is because higher frequency video and voice signals would not be amplified due to amplifier bandwidth limitations.

b. The 22.5 ft parabolic could be used directly. It provides approximately 32 dB gain on-axis at 800 MHz. The HPBW is approximately 4.1 degrees. The on-axis ERP requirement was 58.3 dB which means RF power output is approximately 416 watts. Using a transmitter efficiency of 58 percent, the power into the transmitter is 843 watts. The housekeeping power requirements were 250 watts. The total prime power requirement was 1093 watts. Assuming a 75 percent degradation of the solar array at the end of four years, the required power was 1460 watts.

c. The solar array was rolled up to drop the dc power output to 1.46 kw. This meant reducing the array area to 278 ft^2 or to a length of 18.2 feet.

4.1.5 STEP-BY-STEP DBS REFURBISHMENT MISSION ANALYSIS

	TASK	TIME (Minutes)
Step 1.	Ground command disable solar array drive.	
Step 2.	Ground command inhibit attitude control search mode.	
Step 3.	Ground command shut-off power to transponder.	1
Step 4.	Ground command power off to station-keeping engine.	
Step 5.	Remote Manipulator Spacecraft dock to forward boom of DBS directly behind interferometer.	50
Step 6.	Remote Manipulator Spacecraft reach into supply bin and remove coaxially connected portable deployable antenna.	2

	TASK	TIME (Minutes)
Step 7.	Remote Manipulator Spacecraft deploy antenna by squeezing the restraining latch (similar to self-deploying umbrella).	2
Step 8.	Antenna has mating ring on back to match the interferometer struts. Attach the antenna to the struts by engaging the three spring loaded clamps. Ground command switch Remote Manipulator Spacecraft to newly connected antenna.	5
Step 9.	Remote Manipulator Spacecraft undock and proceed to aft equipment bay.	10
Step 10.	Remote Manipulator Spacecraft dock to solar array support structure.	
Step 11.	Remote Manipulator Spacecraft grasp edge of thermal blanket and release hook and pile securing thermal blanket on upper surface.	1
Step 12.	Roll back insulation and fasten to sides of spacecraft.	8
Step 13.	Remove coaxial connector to antenna.	2
Step 14.	Remove electrical connector from receiver section.	2
Step 15.	Remove electrical connector with telemetry outputs to telemetry subsystem.	2
Step 16.	Remove electrical connector with command inputs from command decoder.	3
Step 17.	Remove electrical connector with power lead.	3

	TASK	TIME (Minutes)
Step 18.	Reach into tool bin and extract power wrench.	1
Step 19.	Insert 1/2 inch socket onto wrench.	1
Step 20.	Remove twenty four 1/2 inch captive screws securing transmitter to mounting bracketry.	20
Step 21.	Replace power wrench in supply bin.	1
Step 22.	Unrigidize a rigidizable tether.	1
Step 23.	Attach tether to transmitter.	2
Step 24.	Lift transmitter with integral heat pipe out of aft equipment bay.	4
Step 25.	Move transmitter behind manipulator spacecraft and rigidize tether.	2
Step 26.	Reach into aft equipment bay and release from receiver module: 1. Coaxial electrical connector from antenna 2. Electrical connector power lead 3. Electrical connector commands leads 4. Electrical connector telemetry leads 5. Electrical connector to transmitter	15
Step 27.	Remote Manipulator Spacecraft reach into supply bin and withdraw a clamp.	8
Step 28.	Clamp a shade in the open position.	
Step 29.	Repeat 27 and 28 four times.	

	TASK	TIME (Minutes)
Step 30.	Remote Manipulator Spacecraft crawl to satellite face containing receiver modules.	4
Step 31.	Unrigidize and grasp rigidizable tether and place next to skin covering bay containing receiver module.	2
Step 32.	Reach into tool bin and extract powered wrench.	1
Step 33.	Insert screwdeiver head into wrench.	2
Step 34.	Unscrew 50 captive screws securing skin to structure.	50
Step 35.	Replace powered wrench in supply bin.	4
Step 36.	Attach rigidizable tether to skin.	2
Step 37.	Move tether back out of way and rigidize.	2
Step 38.	Reach into tool bin and remove special cutting tool.	2
Step 39.	Cut away securing tape and remove any insulation in the way of receiver.	7
Step 40.	Place insulation in supply bin.	3
Step 41.	Place cutting tool into tool bin.	1
Step 42.	Unrigidize and grasp rigidizable tether and attach to reciever module	2
Step 43.	Remove powered wrench from tool bin.	1
Step 44.	Attach 1/4 inch socket to wrench.	1
Step 45.	Unscrew ten captive screws securing receiver module to spacecraft longerons.	8

	TASK	TIME (Minutes)
Step 46.	Replace powered wrench into tool bin.	1
Step 47.	Grasp receiver module and extract from equipment bay	4
Step 48.	Move module behind Remote Manipulator Spacecraft and rigidize tether.	2
Step 49.	Reach into supply bin and remove replacement receiver.	2
Step 50.	Place replacement receiver into equipment bay	5
Step 51.	Reach into tool bin and remove powered wrench.	1
Step 52.	Tighten ten captive screws securing receiver to satellite longerons.	10
Step 53.	Reach into supply bin replace powered wrench and obtain necessary insulation.	2
Step 54.	Install insulation around receiver as necessary.	10
Step 55.	Tape insulation to structure as necessary.	10
Step 56.	Replace tape into supply bin.	1
Step 57.	Grasp and unrigidize tether. Place skin over bay and rigidize tether	3
Step 58.	Reach into supply bin and extract powered screwdriver.	2
Step 59.	Place skin over bay.	1
Step 60.	Screw 50 captive screws securing skin to structure.	50

	TASK	TIME (Minutes)
Step 61.	Unrigidize tether and store	2
Step 62.	Replace powered screwdriver in supply bin.	1
Step 63.	Grasp and unrigidize tether.	1
Step 64.	Release receiver and place into supply bin.	2
Step 65.	Release clamps securing sun shades and place in supply bin.	8
Step 66.	Remote Manipulator Spacecraft move and dock to solar array support structure	4
Step 67.	Install connectors removed in Step 26.	18
Step 68.	Reach into supply bin and extract replacement transmitter.	2

Replacement transmitter has rings for easy gripping by manipulator. Hold transmitter with on manipulator. Grasp powered socket wrench with other.

Step 69.	Insert replacement transmitter into aft equipment bay.	5
Step 70.	Using powered wrench, tighten down twenty-four 1/2 inch captive screws.	24
Step 71.	Replace powered wrench into tool bin	1
Step 72.	Install connectors removed in steps 13 - 17.	18
Step 73.	Unrigidize tether and place old transmitter in supply bin.	2
Step 74.	Release tether and stow.	2

	TASK	TIME (Minutes)
Step 75.	Replace thermal blanket, securing the hook and pile.	4
Step 76.	Refuel Satellite	30
Step 77.	Remote Manipulator Spacecraft undock and proceed to interferometer.	10
Step 78.	Ground command Remote Manipulator Spacecraft control to primary antenna.	1
Step 79.	Release clamps securing secondary antenna to interferometer.	2
Step 80.	Fold antenna as in an umbrella.	3
Step 81.	Stow antenna in supply bin.	2
Step 82.	Remote Manipulator Spacecraft back away from DBS	
Step 83.	Ground command enable solar array drive.	
Step 84.	Ground command enable attitde control search mode. Ground command power on to stationkeeping engine.	10
Step 85.	Ground command power to transponder.	
Step 86.	Commence ground testing of satellite	
Step 87.	At completion of test Remote Manipulator Spacecraft initiate dual burn to transfer to new orbit.	
Total Time to Perform Mission:		494

4.2 NIMBUS SATELLITE REFURBISHMENT

The Nimbus Spacecraft is a large, earth-stabilized spacecraft (see Figure 4-3) launched into a low altitude (500 nm), a near-polar orbit which collects meteorological data over the entire earth on a daily basis. The spacecraft and supporting ground complex are configured to facilitate real-time data collection, processing, and application. A description of the satellite appears in Appendix B. Details of the Nimbus missions were taken from References 3 and 6. Details of the satellite design, which appear in Appendix B, were taken from References 4, 5, and 7. Maintenance photos were taken from Reference 8. Details of the paddle drive hardware were taken from References 9, 10, 11, and 12.

4.2.1 NIMBUS REFURBISHMENT MISSIONS

Two Nimbus refurbishment missions were examined. These were:

1. The on-orbit refurbishment of the Nimus A spacecraft into the Nimbus C version.
2. The on-orbit refurbishment of Nimbus D into Nimbus E.

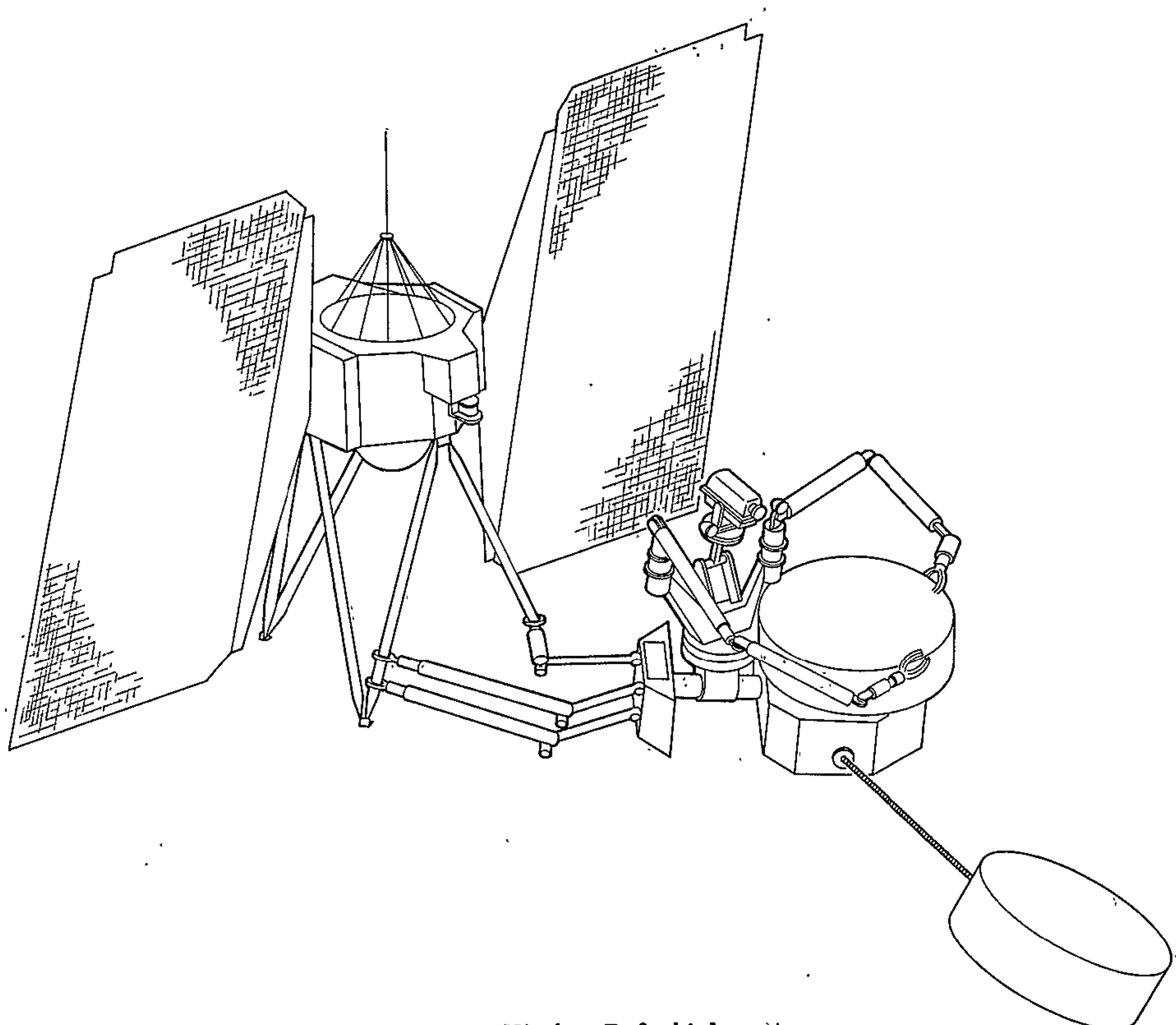
4.2.2 SATELLITE STATE

4.2.2.1 Disabling/Enabling

Before docking, Nimbus was disabled so as not to react to the remote manipulator spacecraft docking actions. The attitude control pneumatics were commanded off. After docking to Nimbus A, power to the spacecraft was disabled by de-mating the four Cannon connectors on the harness that feeds solar array power and battery power into the battery electronics module (Bay 12.1.2) in the Sensory Ring because:

1. The batteries (in the Sensory Ring) cannot be disabled by ground command as on Nimbus B and D.
2. The solar array bus disconnect does not disconnect the batteries.

The thermal blankets over the battery electronics module and the Cannon connectors are removed and replaced in a manner to be described later.



Nimbus Refurbishment

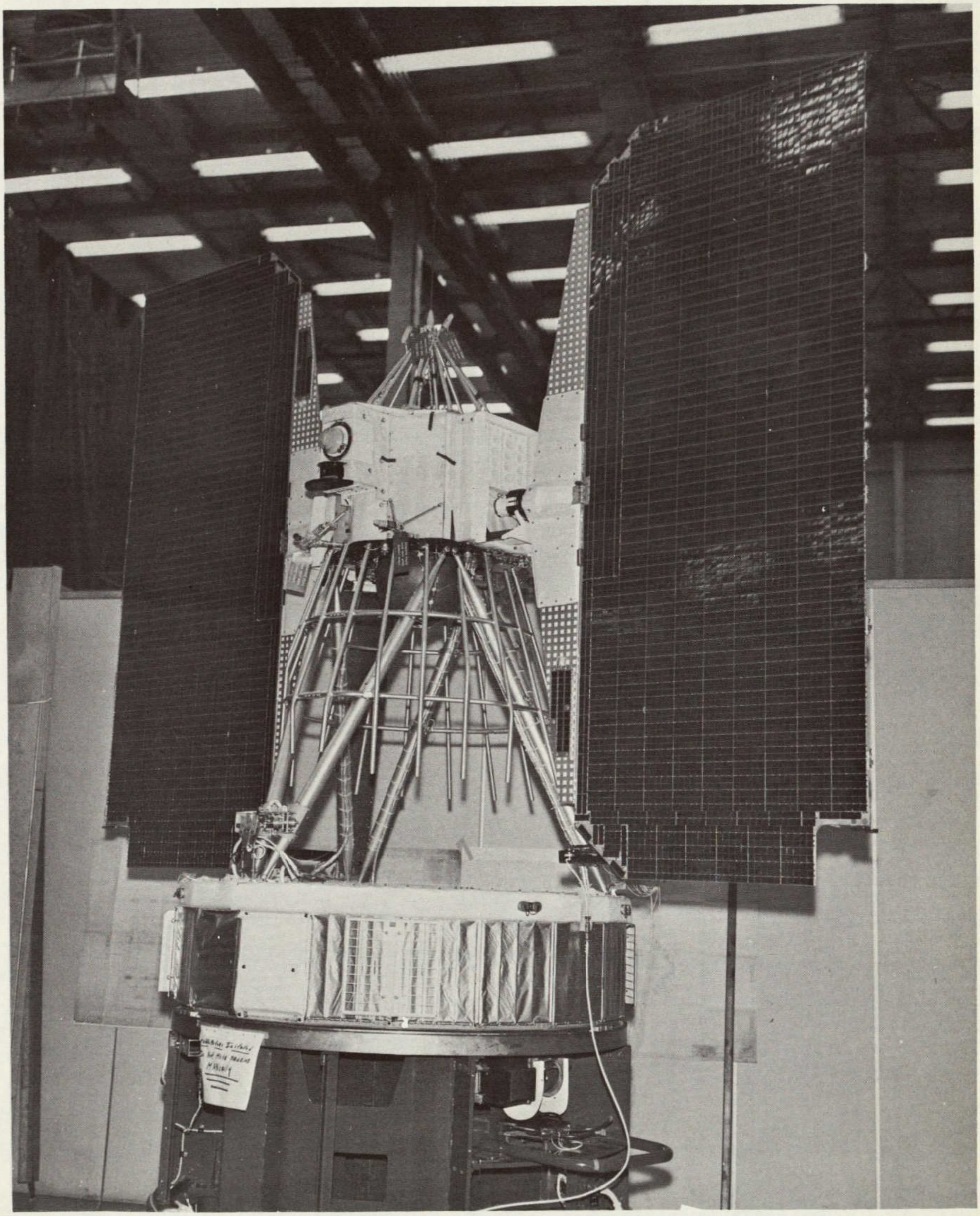


Figure 4-3. Nimbus Spacecraft

Connector caps carried in the remote manipulator spacecraft supply bin are installed to assure no arcing at the battery and array cable connectors.

4.2.2.2 Docking

After the disabling commands, Nimbus A rotates at the maximum of 0.05 degree/sec in pitch and roll and 3 degree/sec in yaw. The pitch and roll rates of 0.05 degree/sec, equivalent to 0.05 in./sec at a 5-ft radius, are practically negligible. Furthermore, a superimposed yaw rate of 3 degree/sec, which is equivalent to 3 in./sec at a 5-ft radius, is sufficiently slow for the remote manipulator spacecraft to reach for and grab any grip-hold as it slowly rotates past. However, the recommended docking technique is to position the remote manipulator spacecraft as near as possible to one of the three junctures of the six truss struts and the sensory ring, and then reach out and grab that juncture.

Although two of the three attitude rates are very slow, the spacecraft attitude can be inverted in inertial space in the interval between disabling and docking. For example, after only one hour at 0.05 degree/sec pitch rate, the Nimbus would be pitched 180 degrees from its operational earth-oriented attitude. Therefore, the remote manipulator spacecraft has only to wait a reasonably short time to approach and grab the Nimbus spacecraft from an angle that has favorable communications and lighting aspects.

After initially docking to the strut juncture, the spacecraft can move to any of many suitable grip-holds such as other struts and the ground-plane between the Nimbus control unit and Sensory Ring, the ground-plane atop the Nimbus control unit, and the solar array paddles. Docking to or in the vicinity of the following should be avoided:

1. The bottom of the Sensory Ring where there are accurately-aligned weather sensors, antennas, and many connectors and harnesses. However, docking to the tapered adapter interface ring on the bottom of the sensory ring is acceptable.

2. The sides of the sensory ring where there are easily damaged thermal shutters and antennas.

3. The Nimbus control unit where there are accurately-aligned attitude sensors and easily damaged sun shades.

4.2.3.2 Prelaunch Preparations

The prelaunch provisions for installing the MRIR Subsystem were as follows:

1. Module cavities at Bays 13 and 14 provided. The four top half-cavities of Bay 13 are empty; the terminal-board support assembly over cavity 13.4a was relocated to any of the other module cavities. The length of the module clamp bar (Figure 4-4) was suitable for holding the MRIR recorder in those cavities. The MRIR recorder size is designated as 4/0 (Figures 4-5 and 4-6 illustrate the module sizes and their designations). The special top baffle over the thermal shutter window is held by captive Allen-head screws that are screwed into the module mounting holes. Normally, the pair of baffles for each bay are removed by unfastening 4 screws at the corners of the baffles with a right-angle powered screwdriver, but their nuts are accessible only through the fragile thermal shutters held open with a foam-rubber wedge.

2. The harness to the MRIR equipment is completely pre-wired; the connectors on it are temporarily attached to harness support posts for the launch environment.

3. The thermal cover and mounting surface on the MRIR Radiometer is designed to facilitate installation of the MRIR Radiometer:

   a. The thermal cover over the vacant mounting area is attached with the C-clips used to attach the thermal blankets on the surfaces of the Sensory Ring because the nylon screws which hold the other thermal covers are too difficult for the remote manipulator spacecraft to remove. A handle is provided on the thermal cover.

   b. A thermal cover is attached to the Radiometer in order to avoid attaching a separate thermal cover in orbit (Figures 4-6 and 4-7). Super-insulation acts as a thermal gasket.

   c. The thermal cover has a guide for the powered screw-driver that torques the radiometer captive mounting screws and washers. The bolt access holes in the thermal cover have spring loaded covers.

   d. The radiometer is mounted on a subplate to contain the captive mounting screws and screws that are screwed into captive nuts on the Sensory Ring. Adequate lead-ins are provided on the fasteners. A rigid base is provided for the Radiometer so that torquing of the mounting bolts will not distort the Radiometer and so that predetermined torques can be repeated in-orbit. Both shimming and yaw alignments are incorporated so that no alignments are required in-orbit. Also provided is an adequate base for handles and mounting hardware and additional heat-sink mass, as well as the required thermal conductivity to the Sensory Ring structure.

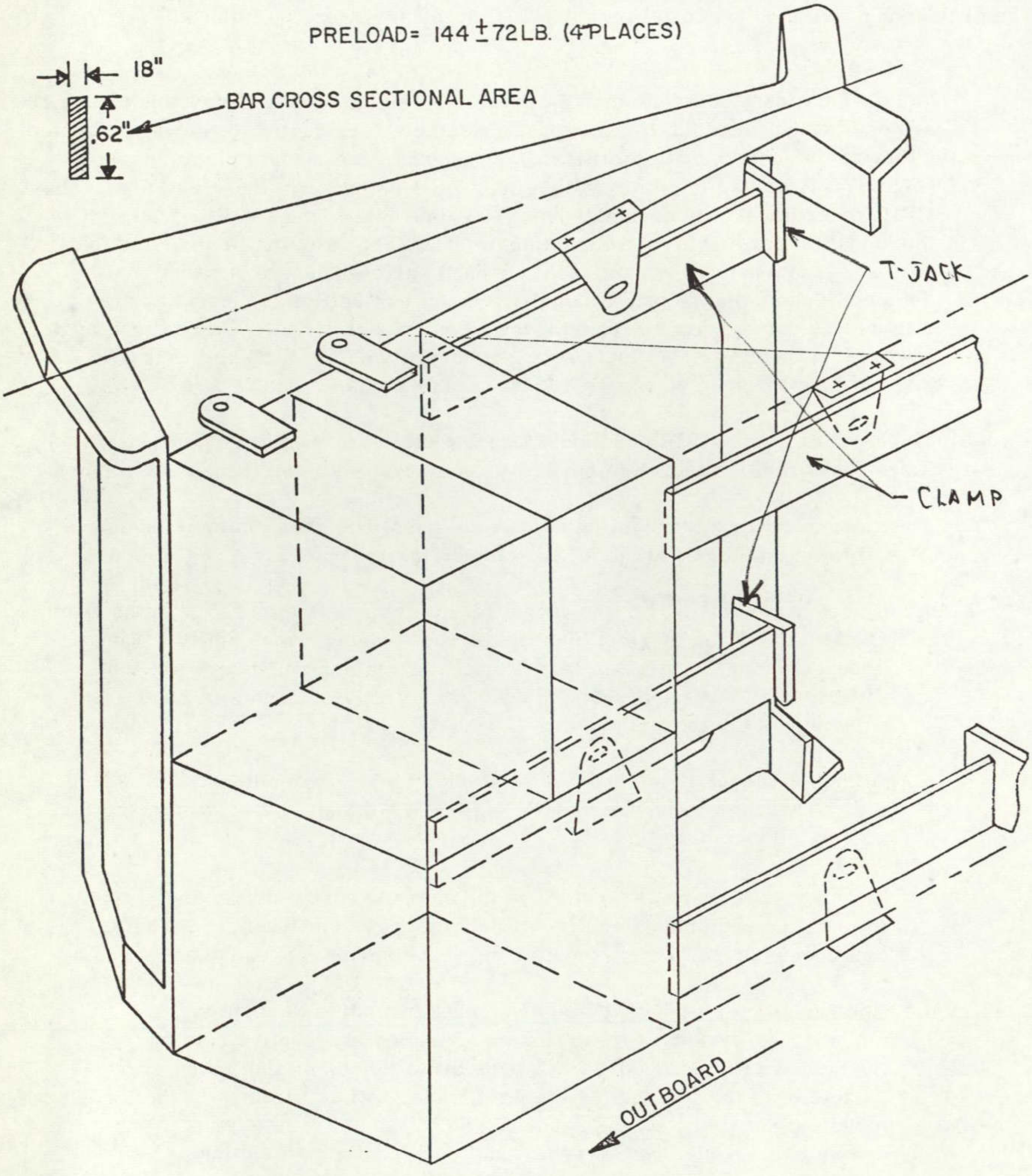


Figure 4-4. Module Preload Arrangement

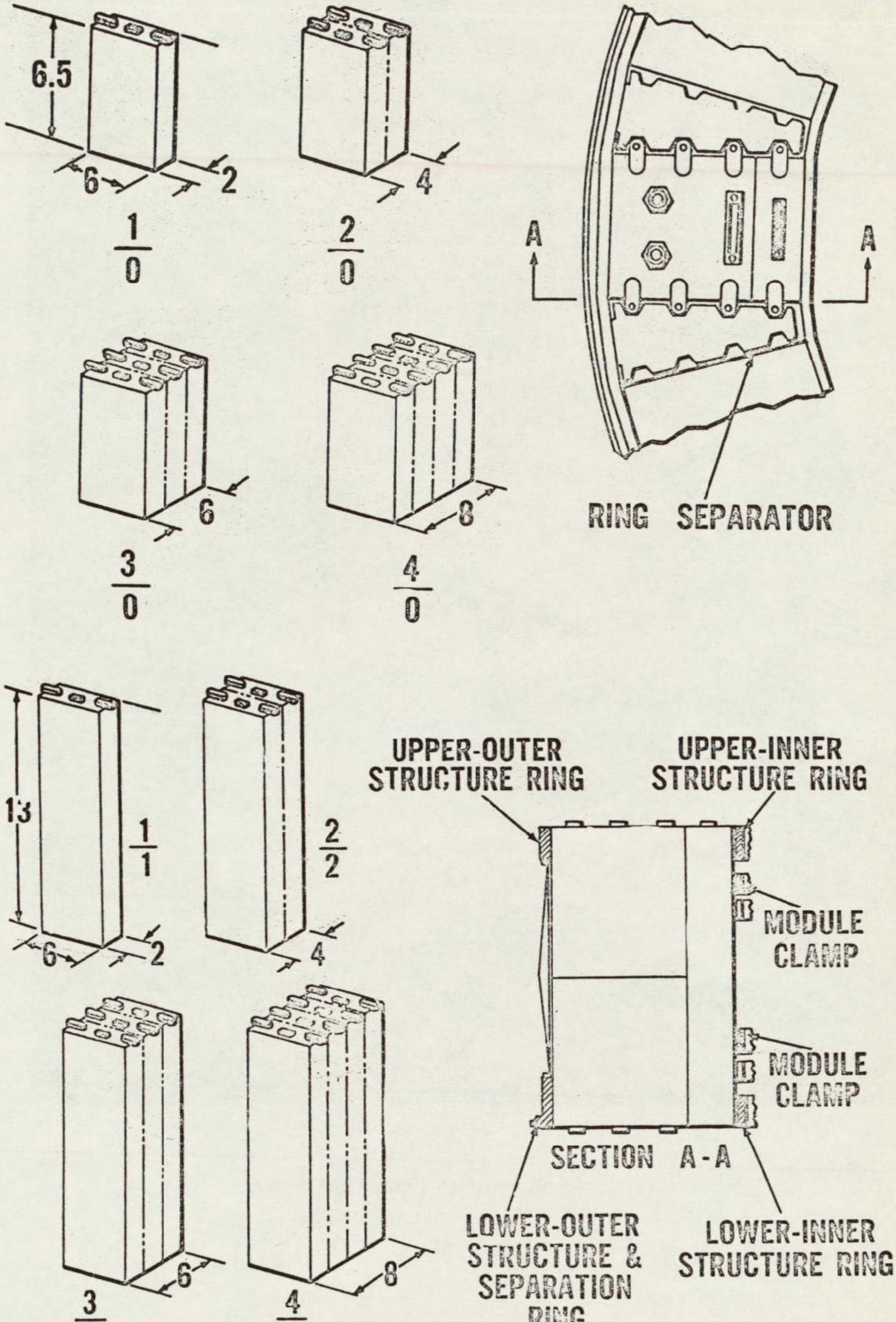


Figure 4-5. Nimbus Module Designation

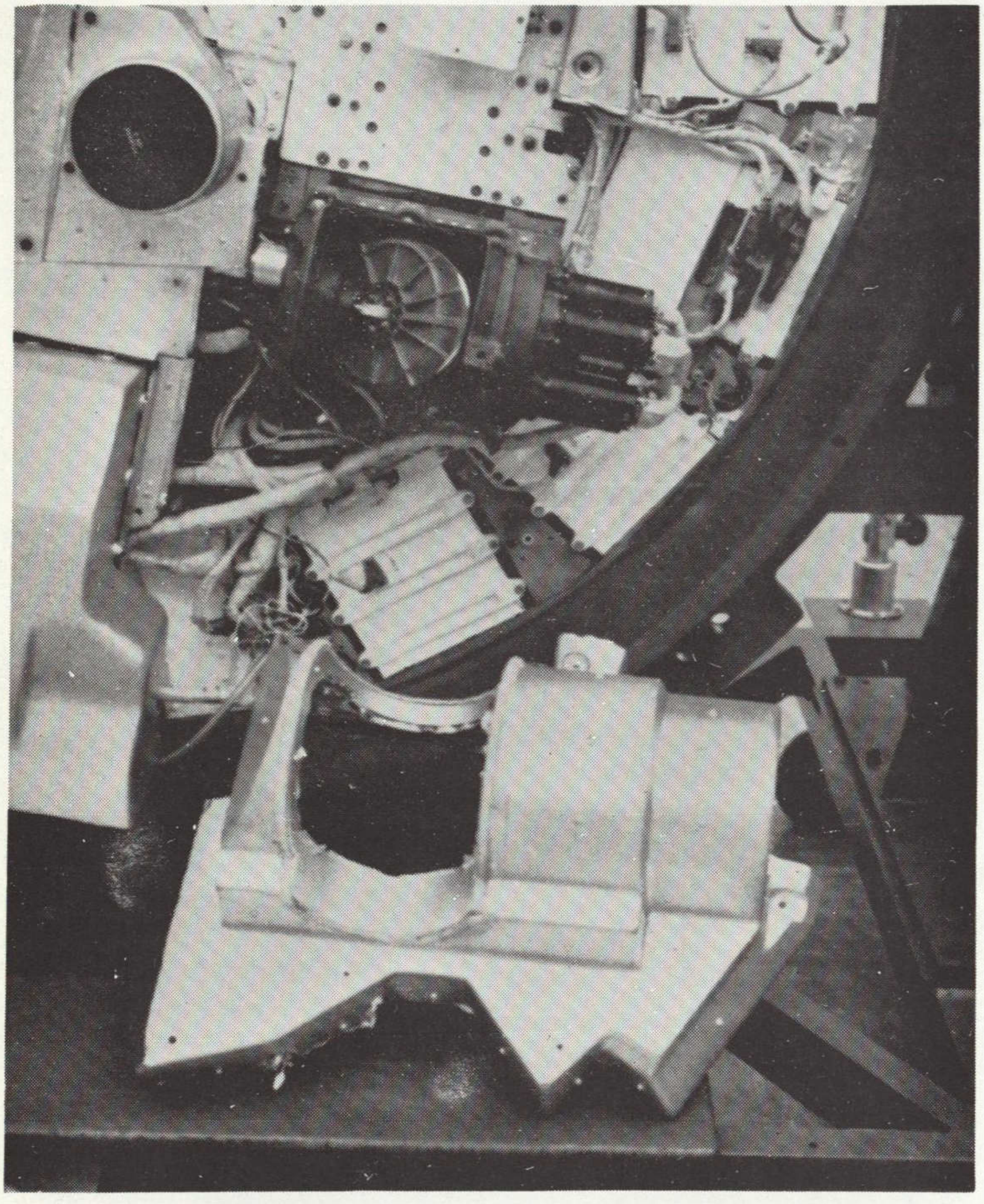


Figure 4-6. MRIR Thermal Cover

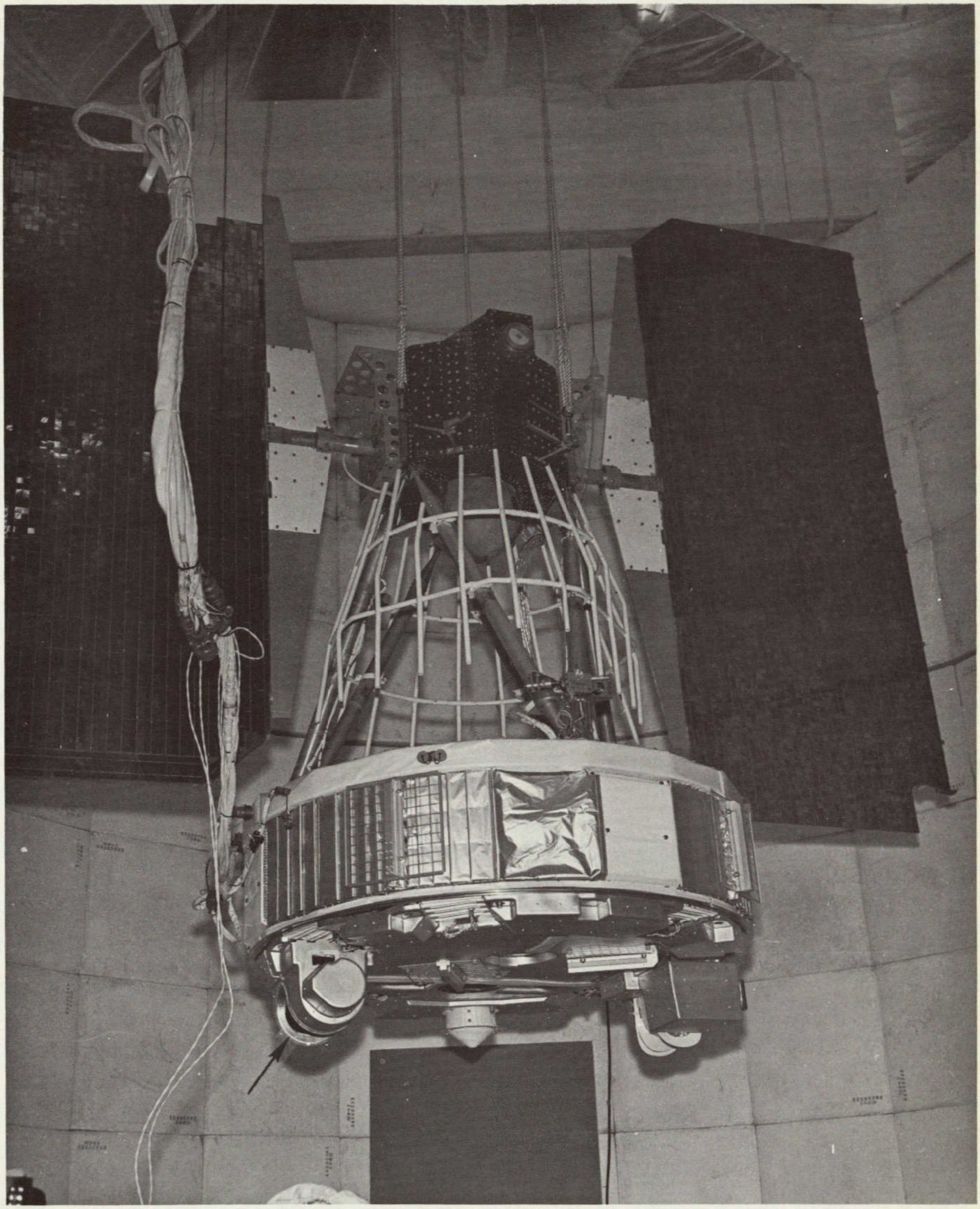


Figure 4-7. MRIR Mounted

The time required to rendezvous and dock with the Nimbus satellite from a distance of 300 ft would be approximately 50 minutes.

4.2.2.3 Nimbus Orientation During Refurbishment

After the Nimbus is disabled, the thorough, passive thermal control techniques (such as the fully automatic and independent shutters on the Sensory Ring and attitude control unit, and the heat-sink function of the massive sensory-ring structure) will control the Nimbus spacecraft temperatures so well that periodic rotation or shades are infrequently, if ever, required. Thus, the attendant problems of blocked communications and incorrect lighting are relieved.

4.2.3 NIMBUS REFURBISHMENT TASKS

4.2.3.1 Refurbishment of Nimbus A to Nimbus C

The refurbishment tasks to convert the Nimbus A spacecraft to Nimbus C are:

1. Replace the solar panels. Radiation degradation of Nimbus solar panels in 6 months is 15 percent, in 12 months is 23 percent.

2. Install the MRIR Subsystem which was not space qualified in time for the Nimbus A launch.

3. Replace the MRIR and AVCS recorders. Nimbus C recorders have exhibited "life" problems.

The overall procedure is to remove the solar paddles, add the MRIR Subsystem, replace the recorders, and install the new solar paddles. Prelaunch provisions are assumed to have been made to Nimbus A prior to launch to facilitate the on-orbit addition of the MRIR subsystem, although minor provisions were assumed for replacing the solar paddles and recorders because they reduce the already long mission duration. Prelaunch provisions are made whenever possible to facilitate the mission.

(7) To provide a structural base for the quick release storage clamp in the supply bin.

4. A bottom edge of each module is chamfered so that the spring-loaded pressure plate is automatically positioned in the radial direction when the modules are inserted in their cavities. These pressure plates hold the modulus while the module lug screws are tightened. The modules are slightly undersize to assure that they will fit into the module cavities. The modules originally had bonded shims which are no longer needed. These shims assured a close fit ($\pm$ 0.005 inch) for the vibration launch environment (Figure 4-8).

5. Some surfaces of the modules are lubricated with a special grease to enhance thermal conduction. These surfaces are the lateral and outboard surfaces and the bottoms of the mounting lugs. The mounting surfaces of the radiometer mounting base are also pregreased.

6. The module-mounting Allen-head screws and washers are captive and the screws have adequate lead-in into the Sensory Ring. All module mounting lugs are made integral with the module as on the MRIR recorder in order to preclude aligning and holding the individual lugs when the lug screws are tightened.

7. The old and new radiometer connectors are relocated, if necessary, by jack-harness fo facilitate mating. Spring-loaded, over-center covers on the Radiometer thermal cover thermally shield the mated connectors.

8. The modules have small handles on their top surface to facilite handling by the remote manipulator spacecraft.

9. The transient suppression filter on the MRIR Recorder module is so attached and supported to the module that it need not be attached to the Sensory Ring in-orbit. The filter connector is a bayonet, rotary type, to facilitate mating in-orbit.

10. Color coding, numbering, stripping, etc., on connectors, modules is used.

11. At bays 13 and 14 the inner and upper structural rings have leaf springs that push the modules radially outwardly at their center toward the thermal shutter assembly. This avoids tightening the T-jacks, which have poor accessibility and lighting, and replacement of the clipped-in-place shutter posts on each side of the thermal shutters is avoided. Otherwise, the leaf springs would have to be clamped onto the inner, top structural ring by the remote manipulator spacecraft. (See T-jacks, Figure 4-4).

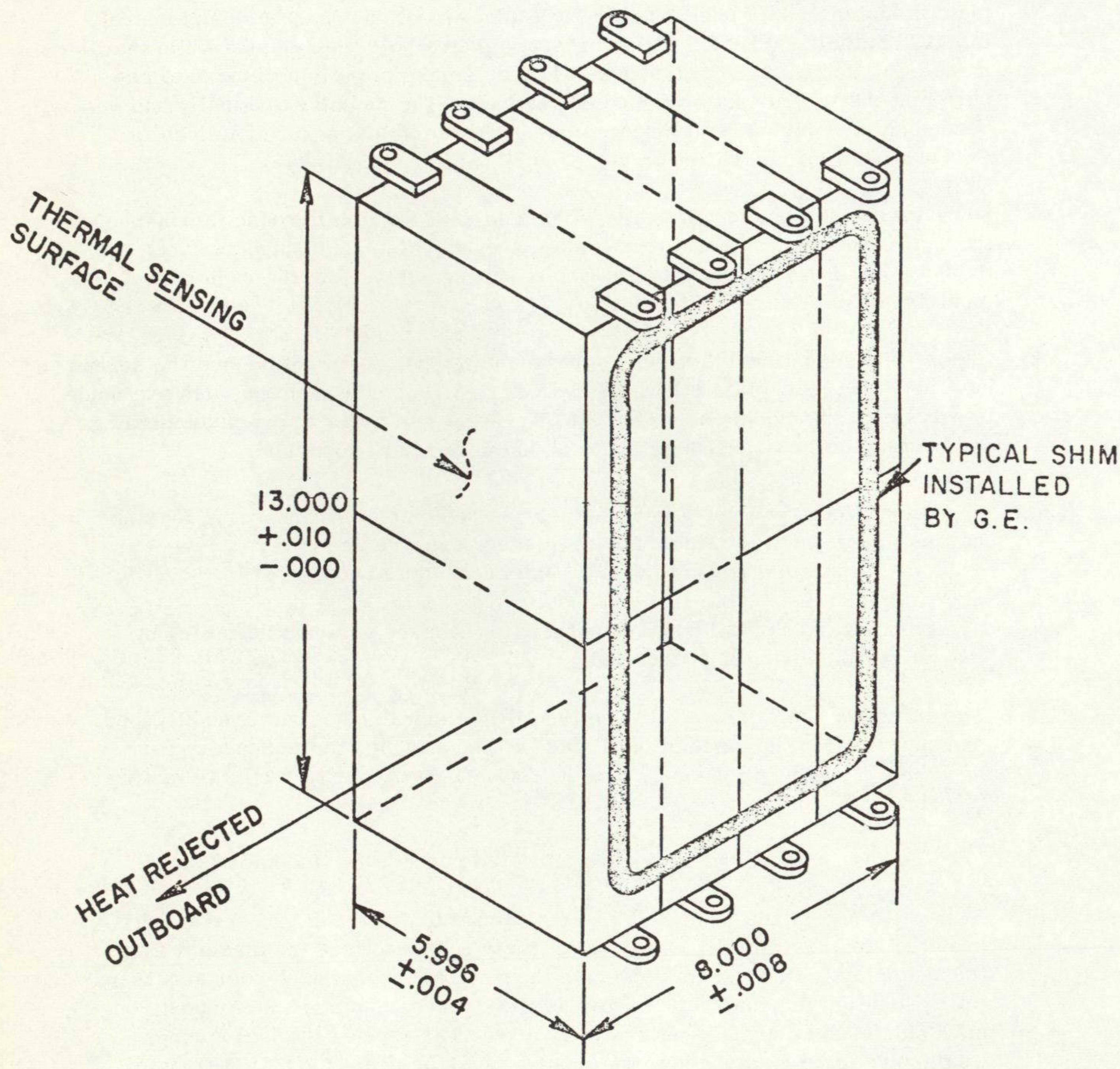


Figure 4-8. Nimbus Module Shimming

12. The MRIR Recorder and Transmitter Electronics (3/0) module and Radiometer Electronics (1/1) module are bonded together with a thermally-conductive bond.

13. Wherever required, hook and pile tape or reflective tape replaces or augments the C-clips used to attach the thermal blankets on the Sensor Ring (one blanket on the lateral surface and seven blankets on the top surface according to Figure 4-9, and solar paddles because reinstallation of the C-clips may be too time consuming even though a special tool with a magazine of C-clips is available. Hook and pile tape also could be used to secure folded-back thermal blankets, and discarded parts (e.g., thermal shutter baffles) in the supply bin.

The prelaunch provisions for replacing the solar paddles are as follows:

1. Wherever required, hook and pile tape or adhesive replaces or augments the C-clips used to attach and hold-down the thermal blankets. Hook and pile tape are used on later Nimbus spacecraft.

2. Add chamfers on the new paddle shaft hub and locking plug holes and assembly hardware.

3. Color coding, numbering, stripping, etc. are used.

4. The new paddle power connector and thermal blankets are temporarily folded-back away from the paddle shaft and held there by hook and pile tape.

5. The new paddle H-clamp has a low modulus elastic material under its four mounting tubs so that the clamp does not have to be held while the paddle is slid onto the driveshaft.

6. The new solar paddle has 10 inches of slack power cable in order to allow more freedom in replacing the solar paddle.

7. The paddles are launched with vertical hinge blocks that prevent the paddles from unfolding when the paddle release mechanisms are activated. During launch, the new solar paddles are mounted on the top of the remote manipulator spacecraft. The paddles are mounted similar to their mounting for launch on the Nimbus satellites:

   a. The vertical paddles are folded for storage (Figure 4-10) within the booster nose shroud (Figure 4-11) along their vertical fold-line and supported at their vertical, free edges, by a stanchion rather than joined at these edges by a tensioned-cable release mechanism. The stanchion allows a rendezvous thruster to exhaust through it. Two tensioned-cable release mechanisms are used which also support each paddle at its low, tip corner.

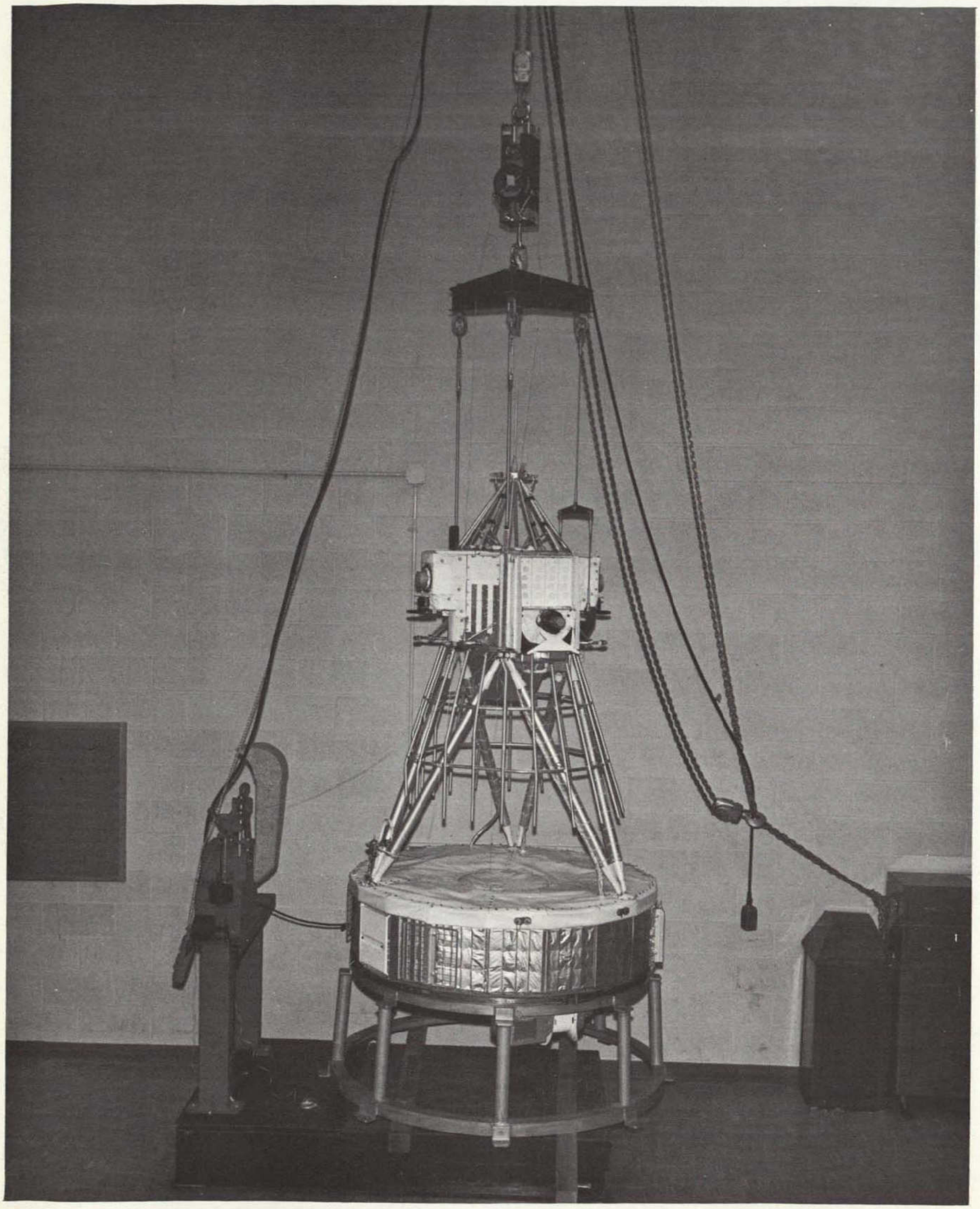


Figure 4-9. Thermal Blanket Attachment

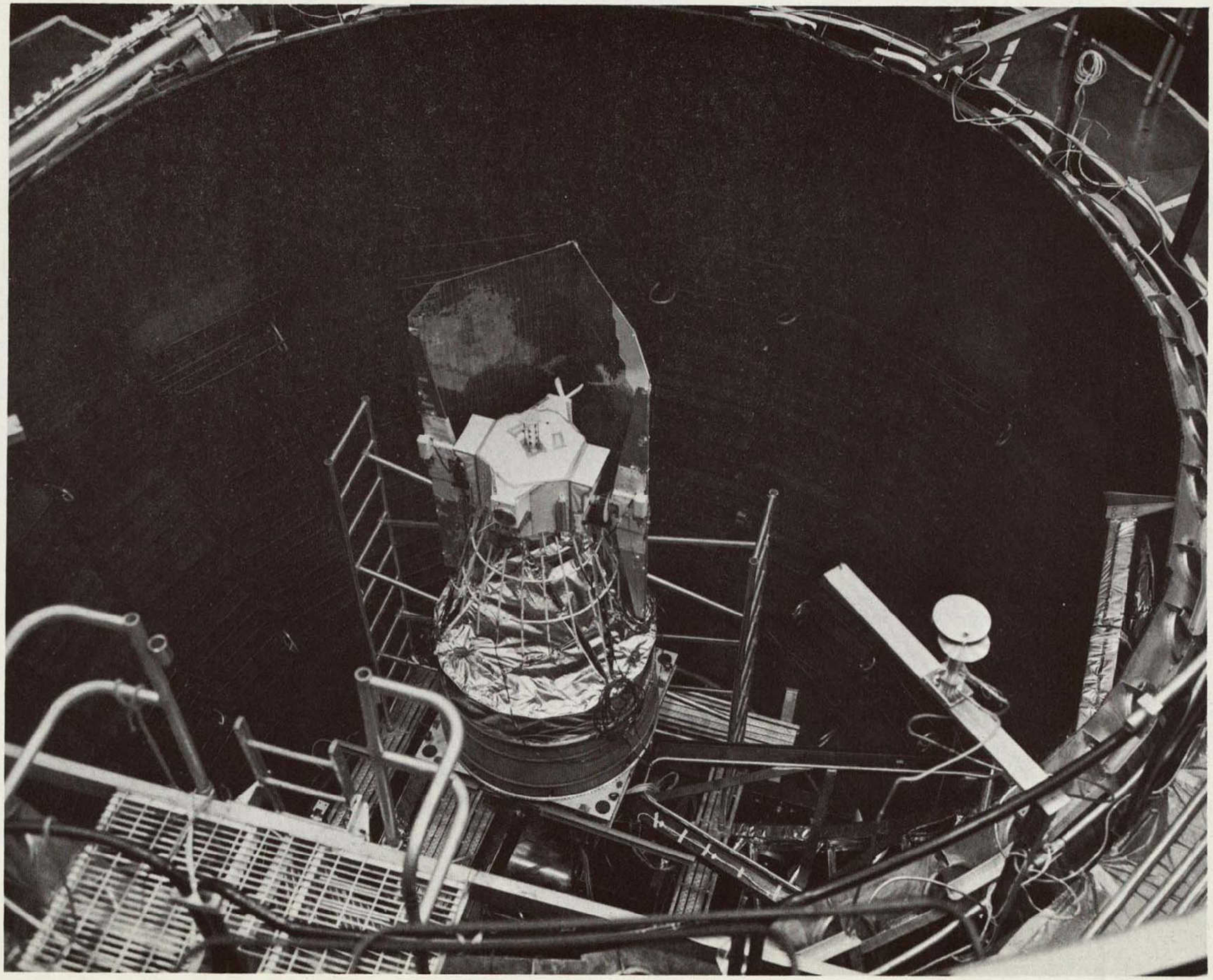


Figure 4-10. NIMBUS Paddles Folded for Launch

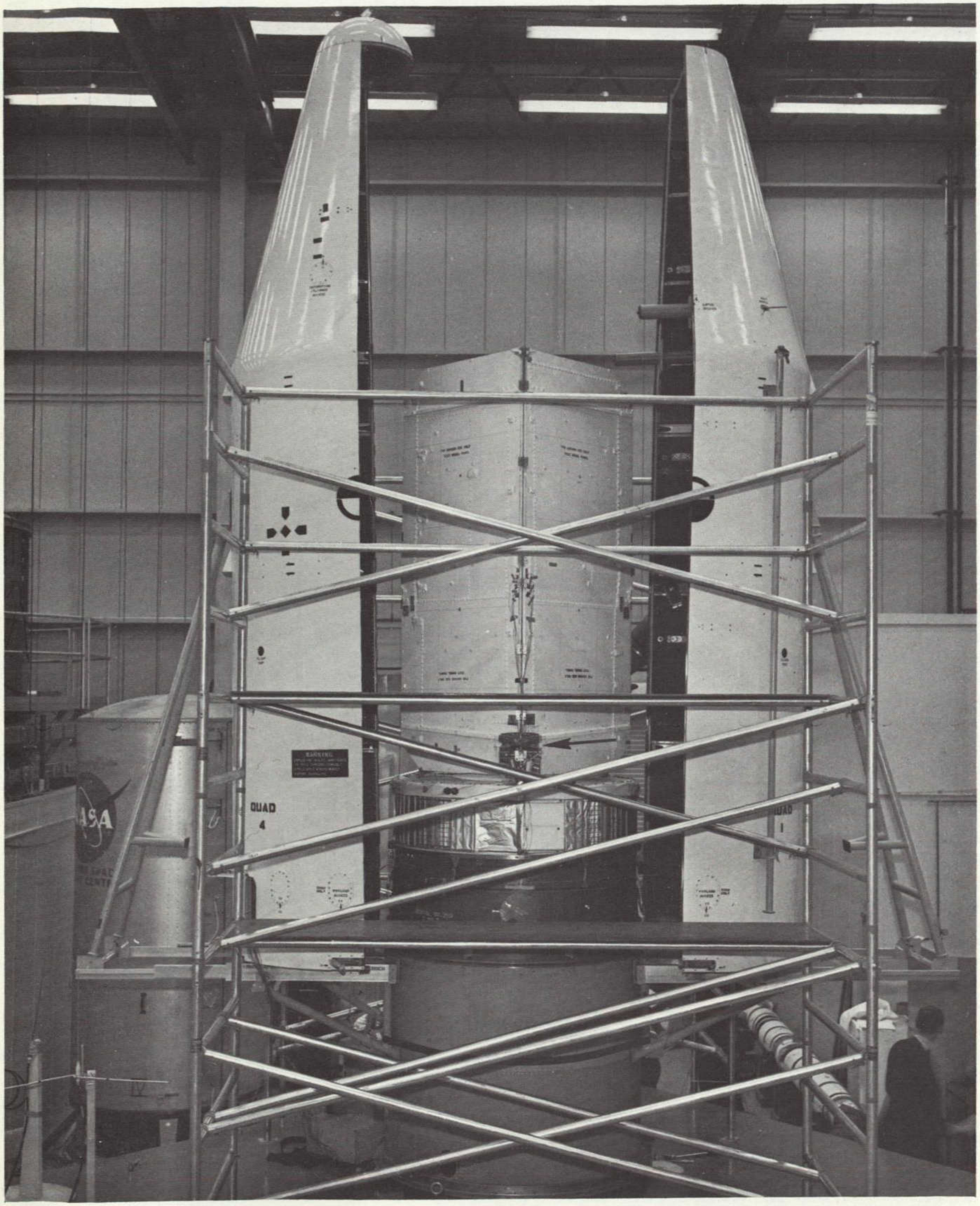


Figure 4-11. Booster Nose Shroud

b. The paddle shafts are clamped to a stanchion on top of the remote manipulator spacecraft.

The end of a rigidizeable tether, partially stiffened for the launch environment by a rigidizing crank accessible to the manipulators, is clamped to each paddle near its shaft; the other end is clamped to the remote manipulator spacecraft. Thus, the paddles can be stored in a trailing position behind the remote manipulator spacecraft by rigidized tethers reeled-out and shaped by the manipulators. The stanchions also are pivoted into a trailing position to allow the remote manipulator spacecraft to operate close to the Nimbus after docking to Nimbus A, complete the following steps on the remote manipulator spacecraft in preparation for installing new paddles:

1. Pull the arming pins at the electrical switch that actuate the explosive cable cutters within the paddle release/mechanisms.

2. Fold back the guard cover over the electrical switch and depress the release switch button.

3. Rigidize the paddle tethers sufficiently using the tether cranks near the top of the remote manipulator spacecraft so the paddles may be slid off the stanchion shafts.

4. Unclamp one paddle by turning the hand-knob on the clamp. The special clamp not only bears on the H-clamp (its four 7/16 inch nuts are loose) but also inserts a pin, shaped like one of the locking plugs, through the paddle hub and the stanchion shaft in order to hold the paddle firmly to the stanchion shaft for the launch environment.

5. Slide the paddle off the stanchion post.

6. Translate the handle in the paddle hinge block so that the paddle erection springs in the paddle hinge may erect the paddle.

7. Store the paddle in a trailing position by shaping the tether with the manipulators as the paddle is placed in its stored position, and fully rigidize the tether.

8. Repeat steps 4, 5 and 6 for the other paddle.

9. Unlatch the latches that lock the three pivotable stanchions.

10. Rotate the stanchions until they automatically latch in the trailing position.

The pre-arranged provisions for replacing the recorders are as follows:

1. Add grip-holds on both the old and new recorders for the tethers to hold and steady the recorders. This provision is mandatory for four reasons: (1) the recorders fit tightly into the Sensory Ring H-structure, (2) the 4 mounting bolts have poor accessibility, (3) the recorder midflanges and connectors are poor as grip-holds, and (4) there are several overlying harnesses.

2. Grease the new recorder mounting pads.

3. Lead-ins are on the 4 Allen-head captive screws (with captive washers) on the new recorder that screw into the 4 captive nuts presently on Nimbus A.

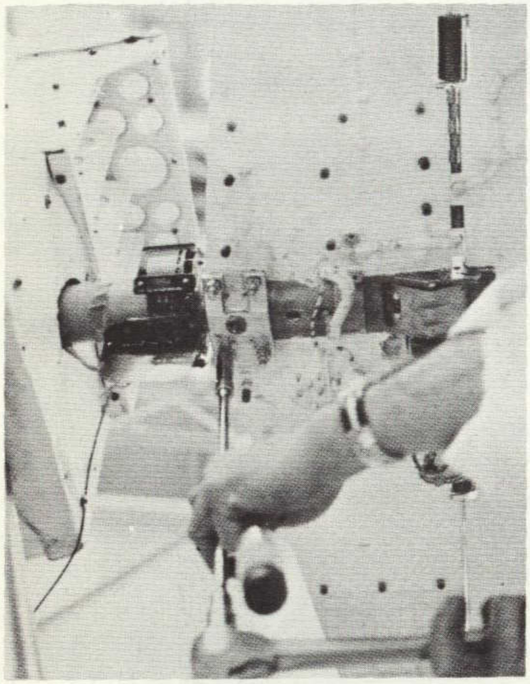
4. The bonded shims, for the old recorder on the H-frame, establish the required mounting plane well enough for the new recorder ($\pm$ 0.0075 inch).

5. Install jack-cables on the old and new recorders.

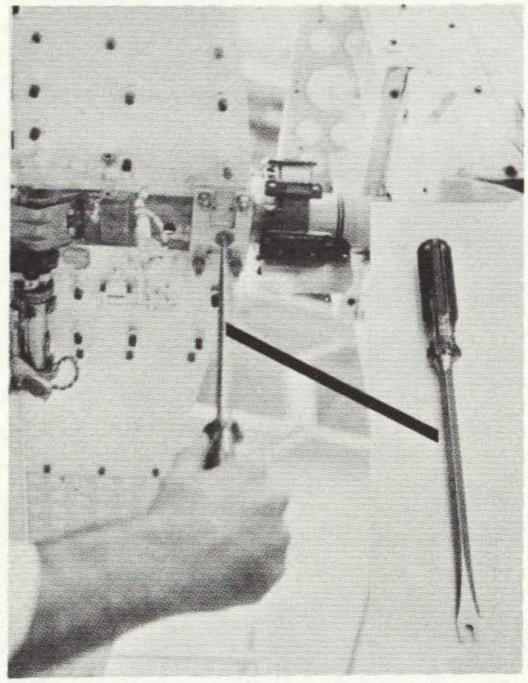
4.2.3.3 Step-by-Step Mission Analysis

	TASK	TIME (Minutes)
Remove Solar Paddles:		
Step 1.	Dock to NIMBUS truss from rendezvous point 300 feet from satellite.	50
Step 2.	Reach into tool bin and attach 7/16 inch socket to power tool.	1
Step 3.	Reach into tool bin for special box wrench to hold the array H-clamp bolt heads.	1
Step 4.	Loosen (but do not take off) four nuts (7/16 inch across flats) on underneath side of array drive shaft. (Figure 4-12).	4

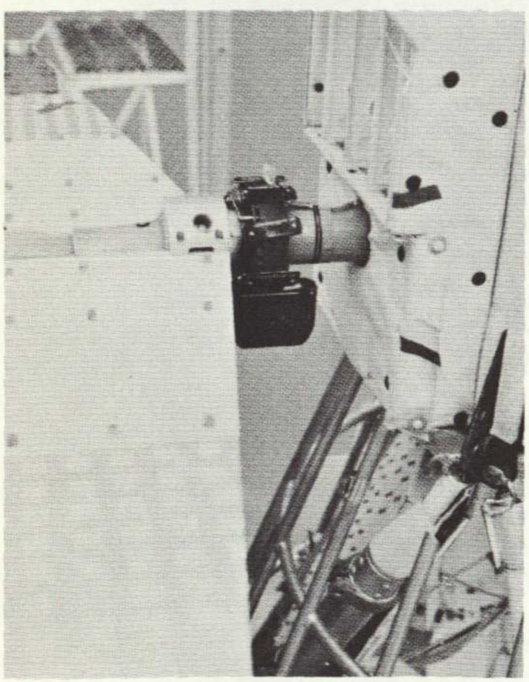
TASK		TIME (Minutes)
Step 5.	Demate and replace 7/16 inch socket, power tool and special box wrench into tool bin.	1
Step 6.	Reach into tool bin for special screw driver bit for unlocking plugs and attach to power tool.	2
Step 7.	Remove three locking plugs (120 degrees apart) using special slotted screwdriver bit (Figures 4-12 & 4-13). To assure safe repositioning of remote manipulator spacecraft to reach the plugs spaced at 120 degrees and to minimize mission time, reposition the paddles by slipping the 165 in-lb clutch. To rotate the solar paddles, grasp the inboard portion of the paddles where there are no solar cells.	6
Step 8.	Place each locking plug into felt-lined slots or sticky surfaces on the supply bin, and replace separated screwdriver bit and power tool into tool bin.	3
Step 9.	Reach into tool bin for electrical cable cutters.	1
Step 10.	Move paddle off shaft just far enough to cut the paddle power cable.	3
Step 11.	Replace cutters in tool bin.	1
Step 12.	Demate power connector (Figure 4-12) and remainder of cable.	2
Step 13.	Attach the old paddle to the remote manipulator spacecraft tether for later deorbiting.	4
Step 14.	Reposition the remote manipulator spacecraft and repeat Steps 1 to 12 in order to remove the second paddle.	27
Total time to remove both solar paddles		104



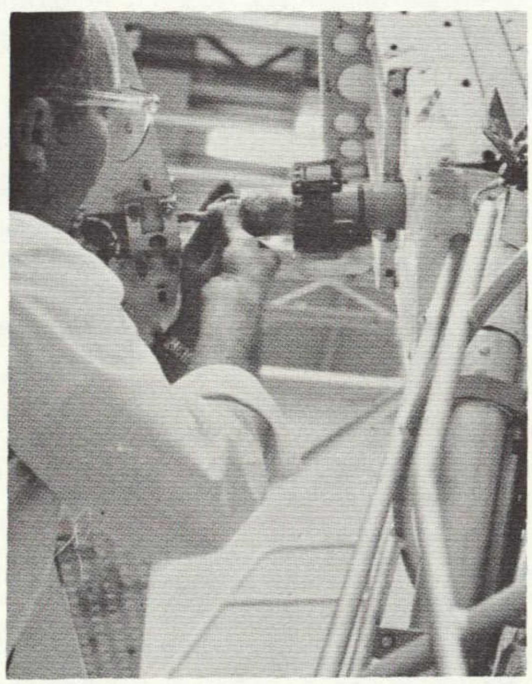
a. Loosening 7/16" Nuts



b. Removing Locking Plug-Bottom



c. Locking Plugs Removed-Top



d. Connector being Demated

Figure 4-12. Removal of Solar Paddles

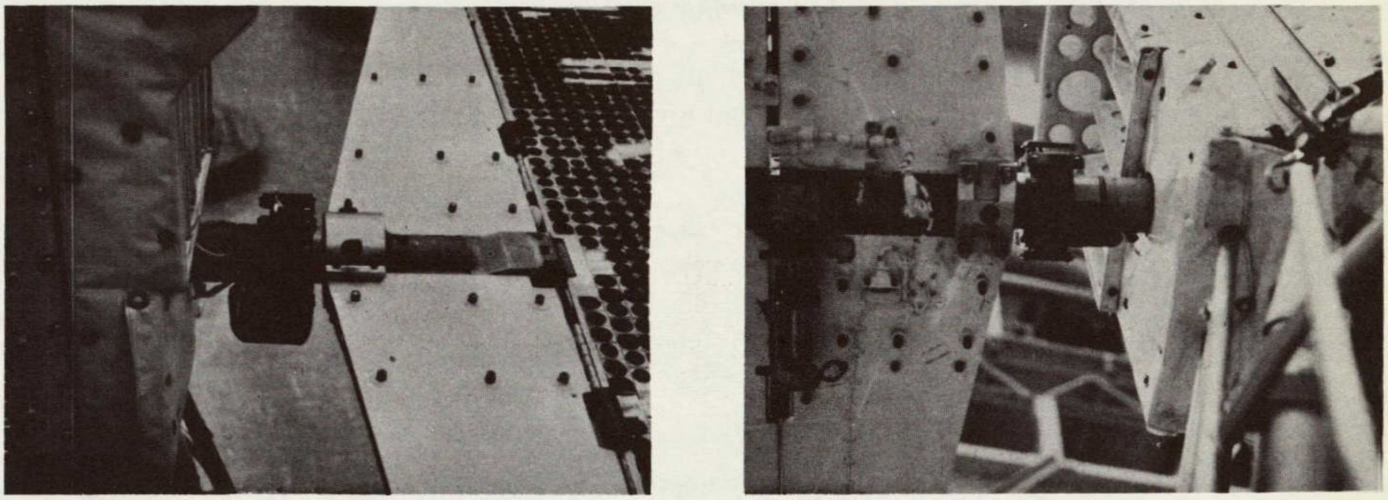


Figure 4-13. Locking Plugs

MRIR Subsystem Installation:

	TASK	TIME (Minutes)
Step 1.	Reposition the remote manipulator spacecraft onto the bottom of the Sensory Ring	4
Step 2.	Reach into supply bin and release clamps retaining an MRIR radiometer (Figure 4-22).	2
Step 3.	Reach into tool bin and mate Allen-wrench bit with power tool.	1
Step 4.	Place MRIR on mounting surface at bottom of sensory ring.	4

TASK		TIME (Minutes)
Step 5.	Open the spring-loaded thermal cover doors over each mounting screw and screw down the three Allen-head captive screws to fasten the MRIR radiometer to the mounting surfaces. The spring-loaded thermal cover doors will automatically close when the Allen-wrench bit is withdrawn.	6
Step 6.	Demate the Allen-head bit and power tool and replace them in the tool bin.	1
Step 7.	Open the over-center doors in the thermal cover over the connectors.	1
Step 8.	Release the 7 Cannon connectors from the retentive hook and pile tape on the Sensory Ring.	2
Step 9.	Mate the 7 Cannon connectors to the MRIR radiometer.	18
Step 10.	Reach into the tool bin and mate the special screwdriver bit with the power tool.	2
Step 11.	Tighten the Cannon connector mating screws.	2
Step 12.	Demate and replace the special screwdriver bit and power tool into the tool bin.	1
Step 13.	Close the over-center doors.	1
Step 14.	Reposition the remote manipulator spacecraft onto the side of the Sensory Ring.	1
Step 15.	Release C-clips on some blankets on sides and top of sensory ring and store slips.	40
Step 16.	Roll back blankets on top of sensory ring.	2
Step 17.	Secure blankets in rolled-back position with tape.	4
Step 18.	Reach into the tool box and mate special Allen-head bit and power tool.	

TASK		TIME (Minutes)
Step 19.	Remove the 2 thermal shutter baffles in all MRIR cavities.	16
Step 20.	Store the baffles in the supply bin for later deorbiting with the remote manipulator spacecraft.	2
Step 21.	Demate the special Allen-head bit and power tool and replace them in the tool bin.	1
Step 22.	Reach into the supply bin and get the integrally bonded MRIR Recorder and Transmitter Electronics module (1/1) and the MRIR T/M Electronics Module (3/0).	2
Step 23.	Place the integrally bonded modules in cavity 14. 1a/b, 2a, 3a, 4a.	3
Step 24.	Reach into the supply bin and get MRIR Recorder module (4/0) after releasing the clamp.	2
Step 25.	Place the MRIR Recorder module in the Sensory Ring Cavity 13. 1a, 2a, 3a, 4a.	3
Step 26.	Reach into tool bin and mate Allen-head bit with power tool.	2
Step 27.	Screw down 16 lugs on the three MRIR modules (two lugs on bottom of the Radiometer Electronics (1/1) module are omitted).	16
Step 28.	Demate the special Allen-head bit and power tool, and replace them in the tool bin.	1
Step 29.	Release the 14 Cannon connectors of the 3 modules from their retention devices.	5
Step 30.	Mate the 14 module connectors, and mate and lock the transient suppression filter Cannon connectors.	30

TASK		TIME (Minutes)
Step 31.	Reach into the tool bin and mate the special screwdriver bit with the power tool.	2
Step 32.	Tighten the Cannon connector mating screws.	28
Step 33.	Demate and replace the special screwdriver bit and power tool into the tool bin.	1
Step 34.	Recover with the thermal blankets and re-attach them with C-clips using a special tool (Figure 7-1) or tape.	19
Total time to install the MRIR subsystem		226

Replace the Recorders (Figures B-14, B-15 and B-17):

It appears easier to separate the truss and the Sensory Ring (held apart with a rigidizeable tether) to replace the recorders than to:

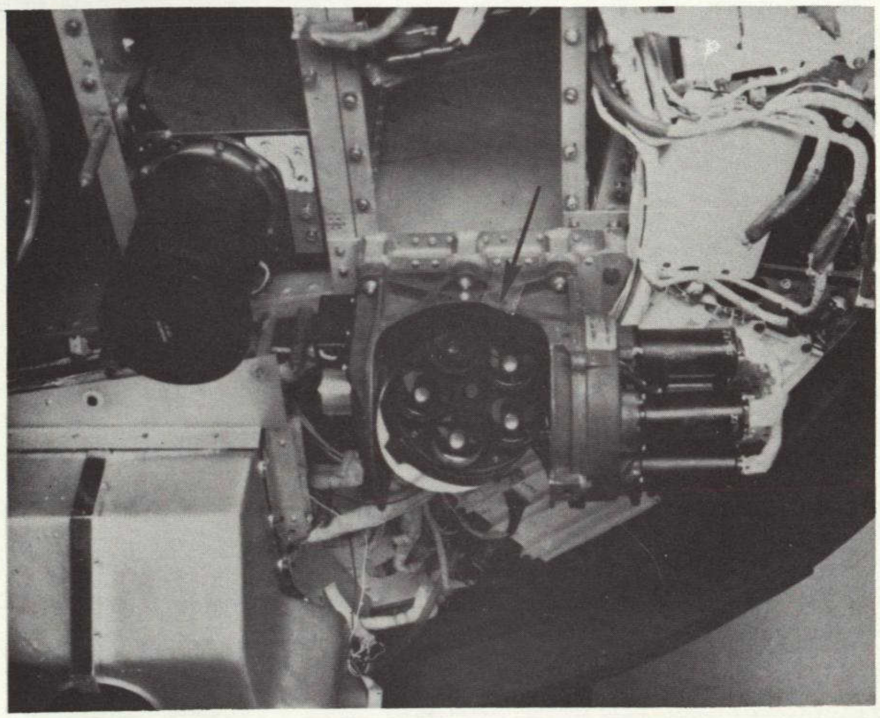


Figure 4-14. MRIR Radiometer

- Reach under and/or through the bottom ground-plane and through the strut truss to remove the recorder Cannon connectors and mounting bolts.

- Reach under and/or through the bottom ground-plane to remove the tightly placed recorders or reposition the remote manipulator spacecraft on the bottom of the Sensory Ring to withdraw the recorders from the bottom of the Sensory Ring after removing the S-Band antennas. Bulky overlying harnesses have to be held aside forcibly (disconnection is very difficult but reclamping the harness is not necessary in-orbit) on either the top or bottom of the Sensory Ring. Furthermore, withdrawing the recorders from the bottom endangers the weather sensors.

Separation and re-attaching the Sensory Ring and the truss requires the same steps as when replacing the Nimbus E Sensory Ring onto the Nimbus D satellite (Section 4.2.3.4) except for a few differences in the number of Cannon connectors (six connectors for the Nimbus A to Nimbus C refurbishment mission and 12 connectors for the Nimbus D to Nimbus E refurbishment). The steps to remove the C-clips attaching the central blanket and two of the six sectored thermal blankets at bays 4, 5, 6, and bays 13, 14, 15 near the recorders are as follows:

Unfasten nylon C-clips and stored them in the supply bin. Fold-back and secure the three pieces of thermal insulation with tape, if required.

The steps to replace the recorders are:

STEP	TASK	TIME (Minutes)
Step 1.	The remote manipulator spacecraft is docked on the top of the Sensory Ring. (Time required to reposition the remote manipulator spacecraft is included in the 100 minutes). The remainder of the Nimbus A (strut truss and attitude control package) is separated and tethered with rigidized tether to the remote manipulator spacecraft (see Section 4.2.4.2).	100

TASK		TIME (Minutes)
Step 2.	Reach into the tool bin for the blade screwdriver bit and power tool, and the right-angle or flexible-drive attachment to be held by other manipulator hand.	2
Step 3.	Mate the bit and attachment to the power tool.	1
Step 4.	Unscrew the 12 screws that hold the six mated Cannon connectors together on the two recorders.	12
Step 5.	Return the demated bit, right-angle or flexible drive attachment, and power tool to the tool bin.	1
Step 6.	Demate the six Cannon connectors and store them out-of-the-way on the Sensory Ring with tape, if required.	6
Step 7.	Reach into the tool bin for special cable cutters and cut the recorder cable lacings in order to get sufficient slack to cut the recorder heater leads (two wires for each of the three heaters on a recorder) where the splices onto the new leads can be most easily accomplished with the splicing tool.	6
Step 8.	Cut the heater leads of the recorders.	2
Step 9.	Replace the special cable cutters in the tool bin.	1
Step 10.	Reach into the tool bin for the special hex-head socket, the power tool, and the right-angle or flexible-drive attachment to be held by other manipulator hand.	2
Step 11.	Unscrew the 4 top bolts and washers (two top bolts for each recorder) and store them in the supply bin on sticky surfaces.	36
Step 12.	Reposition the remote manipulator spacecraft to the bottom of the sensory ring.	1

Step	TASK	TIME (Minutes)
Step 13.	Attach the rigidizable tether sufficiently to hold the recorder in place while the four mounting bolts are removed.	2
Step 14.	Unscrew the two bottom bolts (and washers) and store them in the supply bin on sticky surfaces.	18
Step 15.	Remove the recorder. (Other manipulator hand may have to forcibly hold overlying cables aside).	5
Step 16.	Unclamp the tether and unrigidize it sufficiently to store it on the remote manipulator spacecraft.	2
Step 17.	Reach into the supply bin and attach the thermal shutter baffles to the bottom of the recorder with hook and pile tape.	2
Step 18.	Clamp the old recorder in the supply-bin for deorbiting later with the remote manipulator spacecraft.	1
Step 19.	Reposition the top onto the sensory ring.	1
Step 20.	Repeat Steps 13 through 17 for removal of the second old recorder.	36
Step 21.	Reach into the supply bin for the first new recorder and unclamp it.	2
Step 22.	Install the two new recorders according to Steps 1 through 16 in reverse fashion.	123
Total Time to replace the two recorders.		362

Install New Solar Paddles:

	TASK	TIME (Minutes)
Step 1.	For each paddle, unrigidize the paddle tether sufficiently to retrieve the paddle from its stored trailing position and position it near the paddle driveshaft by reeling in and shaping the tether.	2
Step 2.	Continue to install the paddles according to reverse procedures in the earlier part of this section.	60
Total time to install new solar paddles		62
Total time to refurbish Nimbus A to Nimbus C		754

4.2.3.2 Refurbishment of Nimbus D to Nimbus E

The overall procedure is to remove the Nimbus D Sensory Ring at the strut truss interface and install the Nimbus E Sensory Ring.

4.2.3.5 Prelaunch Preparations

The prelaunch provisions for replacing the Sensory Ring are as follows:

a. The Nimbus E Sensory Ring harness is wired to be compatible with the Nimbus D connectors and wiring.

b. Record of the angular alignments and adjusted lengths of the truss struts allow direct replacement of the Nimbus E Sensory Ring without in-orbit yaw alignment and adjustment of the length of the struts.

c. The lower ground plane is attached to all struts, instead of just four of the six struts, in order to keep the struts in registration for mating to the Sensory Ring fittings. Otherwise, the remote manipulator spacecraft will have to clamp two of the struts to the ground plane in orbit prior to separation of the Nimbus D Sensory Ring, or it will have to hold these two struts, which have rod-ends, in registration during the replacement.

d. Color coding, numbering, stripping, etc. on connectors, modules, etc. are used.

e. Lead-ins on the strut bolts, captive grounding lugs, captive nuts, washers, etc. are used.

f. The Nimbus E Sensory Ring is properly balanced with respect to the remainder of the Nimbus D subsystems.

4.2.3.6 Step-by-Step Mission Analysis

	TASK	TIME (Minutes)
Step 1.	Dock the remote manipulator spacecraft onto the lower ground plane or struts in a position that allows the paddles to rotate.	50
Step 2.	Rotate the paddles to a horizontal position in order that the paddles do not interfere when the Sensory Ring is separated from the strut truss (Figure 4-15)	2
Step 3.	Unfasten hook and pile tape and strip back and secure enough thermal insulation on the top of the Sensory Ring to provide free access to the six connectors on the harness from the attitude control unit and the 10 bolts that attach the strut truss and solar paddle deployment mechanism to the Sensory Ring.	12
Step 4.	Reach into the tool bin and mate the special screwdriver bit to the power tool.	2
Step 5.	Demate the six Cannon connectors on the cables between the attitude control unit and the Sensory Ring. Some of the harness lacing and clamps may have to be cut to free the connectors. Store the cables in an out-of-the-way location with tape, if required.	20
Step 6.	Demate and replace the special screwdriver bit and power tool in the tool bin.	1
Step 7.	Reach into the tool bin for a special pliers and remove the cotter pins in the 8 bolts of the strut truss and the paddle deployment mechanism. (Figures 4-15 and 4-16).	12

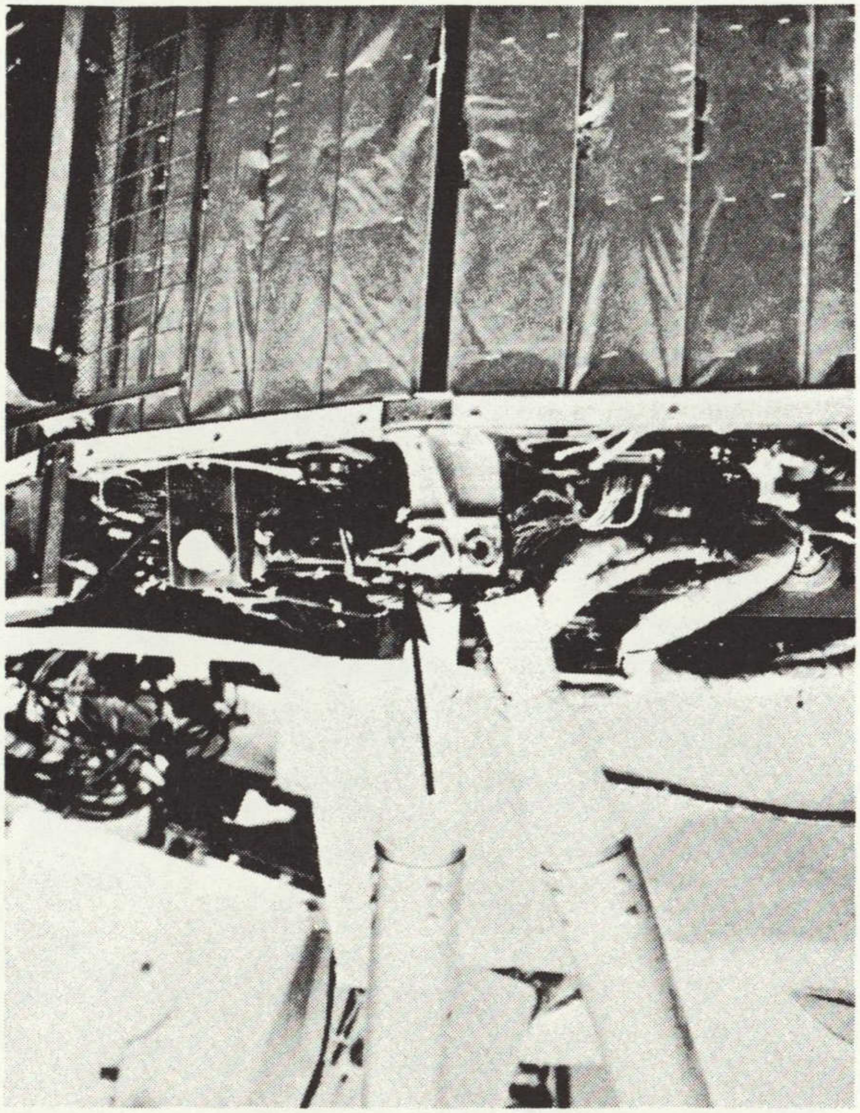


Figure 4-15. Sensory Ring Strut Bolts

TASK		TIME (Minutes)
Step 8.	Return the special pliers and cotter pins to the tool bin. The cotter pins and parts thereof, if cut, are stored on sticky surfaces in the supply bin.	8
Step 9.	Reach into the tool bin for the special hex-head socket and power tool and mate them.	2
Step 10.	Reach into the tool bin for a special box wrench for the 2 cable cutter assembly bolts (Figure 4-16).	1
Step 11.	Loosen and extract the 4 bolts.	12
Step 12.	Store the bolts, washers, and nuts in the supply bin permanently on sticky surfaces in the tool bin, because they have performed then paddle release function	2
Step 13.	Repeat Steps 9 through 12 for the six 3/8 inch strut bolts (Figure 4-15) and deflect plastically the grounding straps away from the bolt assembly area.	20
Step 14.	Clamp rigidizeable tethers to the Nimbus D Sensory Ring and the strut assembly and rigidize the tethers. (The Nimbus E Sensory Ring, launched in a "pan-cake" position on the top of the remote manipulator spacecraft, is rigidly tethered by a rigidizeable tether in a trailing position behind the remote manipulator spacecraft).	3
Step 15.	Reach into the tool bin for the special bolt extractor and clamp it to a strut to push the the last bolt out, if required.	2
Step 16.	Separate the Nimbus D Sensory Ring from the strut assembly.	5
Step 17.	Clamp the Nimbus D Sensory Ring to the remote manipulator spacecraft with a rigidized tether for later deorbiting with the remote manipulator spacecraft.	3

TASK		TIME (Minutes)
Step 18.	Position the Nimbus E Sensory Ring at the strut truss interface holes by partially unrigidizing the tether, reshaping, and adjusting its length.	30
Step 19.	Install the Nimbus E Sensory Ring according to Steps 3 through 16 (but do not replace release mechanism bolts) in reverse fashion except that the 6 castellated nuts and nut washers are replaced with self-locking nuts (having captive washers) and that the 6 new bolts (having captive underhead washers) have lead-ins to align the rod-ends on the truss struts.	100

Total time to refurbish Nimbus D to Nimbus E — 287

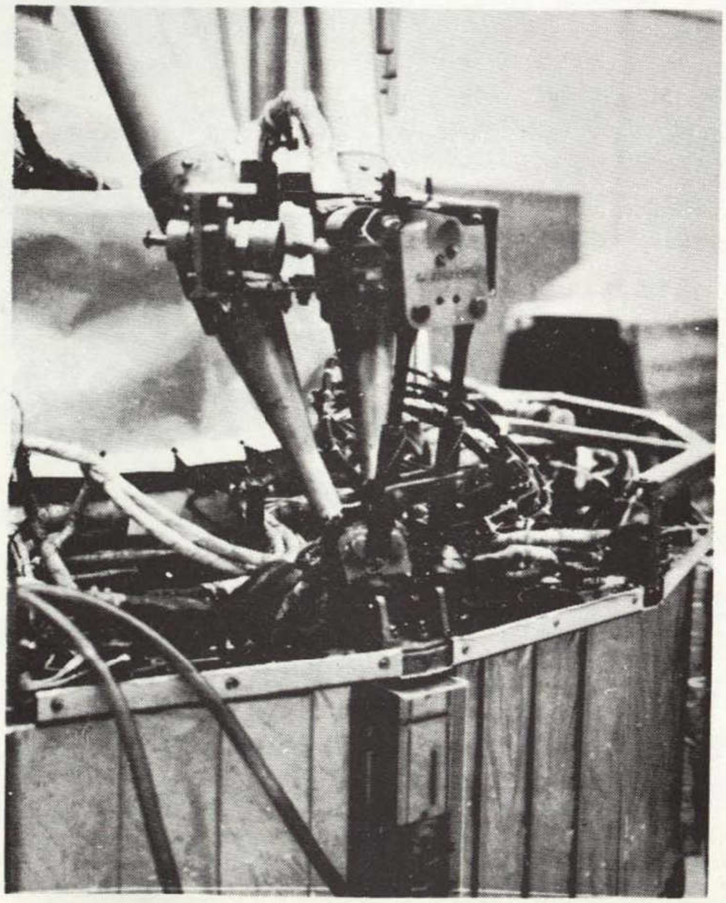


Figure 4-16. Cable Cutter on Paddle Release Mechanism

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SECTION 5

LABORATORY SIMULATIONS

Estimation of manipulator task times is an inexact science. The more empirical data there is, the higher the confidence level of the time estimates. Therefore, many of the critical maintenance tasks were simulated in the laboratory.

5.1 LABORATORY SETUP

The general setup is shown in Figures 5-1 and 5-2. The M-8 mechanical master-slave manipulators were used since their force capabilities most closely match that of the projected space manipulator. Because of the importance of the video link in establishing task times, a TV camera (30 degree field-of-view, 450-line resolution) and 21-inch monitor were used. The experimenter manually located the camera at a distance from the work site that allowed as wide a view as possible while providing the necessary detail. The monitor was situated directly in front of the master control. While performing the tasks, the monitor was used exclusively, except when the angle of view for a specific task was too small to also include all of the tools and storage locations.

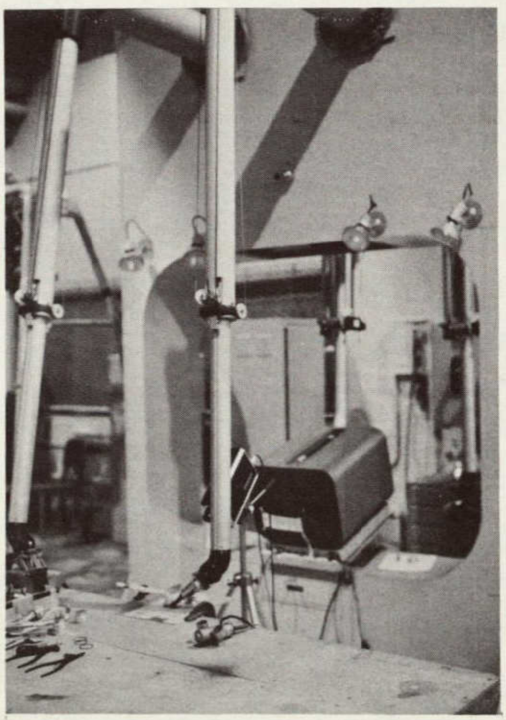

Figure 5-1. Slave Arms of M-8 Manipulator

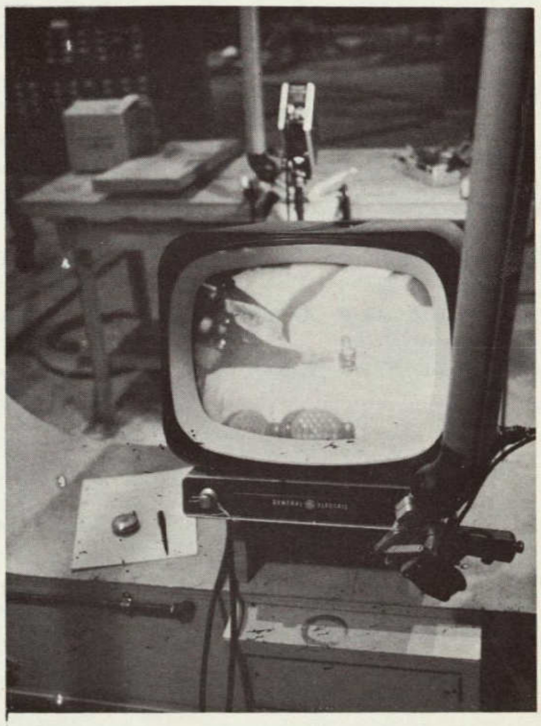

Figure 5-2. Operator TV Monitor

The experimenter practiced until he felt he found the optimum routine. He then timed the tasks, although for lack of time he probably had not reached the flat part of his learning curve for that task. Also, it should be noted that the experimenter was not a manipulator operator by profession.

5.2 TASKS SIMULATED

The simulations described in the following paragraphs, and others performed prior to this study, were used to assist in determining the task times in the four previous missions.

5.2.1 NYLON C-CLIP (NIMBUS)

The early Nimbus used nylon-c-clips to secure the outer thermal blankets. It was found that the standard manipulator jaws could grasp the clip and tear it off (Figure 5-3). The average of 10 trials was 21 seconds. Not included in this average is about one fourth of the clips that sprung out of the jaws as they became free. Adding a sharper, roughened thumb to the jaws would reduce this problem as well as speed up removal time. The TV camera was 1.5 ft from the clip. The clip was at room temperature. If the clip becomes brittle at low temperatures, removal procedures might have to be changed, as debris could be expected from broken clips.

5.2.2 CANNON ELECTRICAL CONNECTOR (NIMBUS)

These connectors are used almost exclusively on Nimbus. Removal requires loosening two captive screws and then pulling the connector loose. (See connectors at left of Figure 5-4.) Removal time was found to depend upon the orientation of the cabling. If the cable did not interfere with the screws, average removal time (5 trials) was 39 sec and replacement time was 37 seconds. However, in about the worst condition, where the cable passed nearly over one of the screws, the removal time was 46 sec and replacement time was 108 seconds. In this situation the cable was not harnessed down. Even longer times are anticipated if the cabling has preset twists and little slack is available. The relatively fast times resulted from use of a power screwdriver with a finder bit to remove and replace the screws.

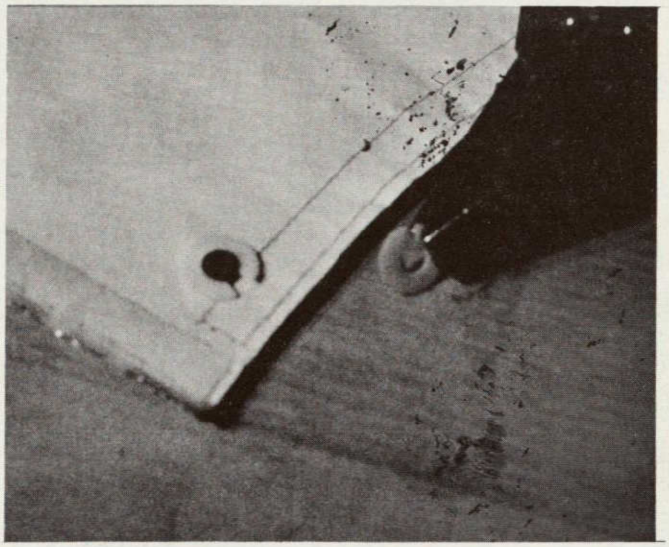

Figure 5-3. Nimbus Nylon C-Clip Removal

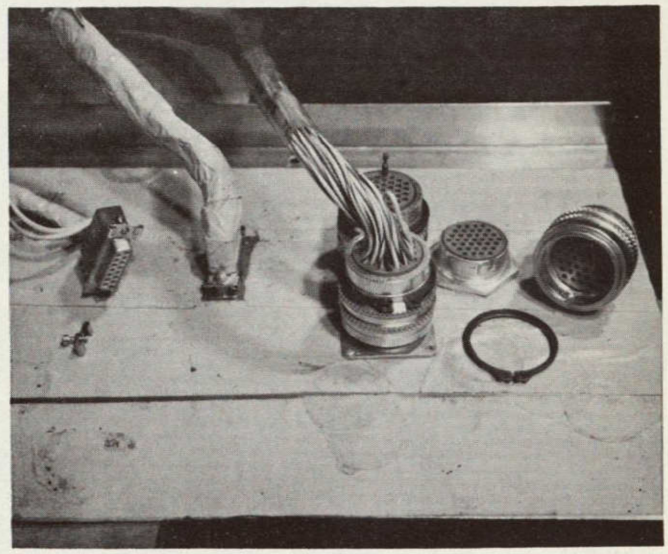

Figure 5-4. Cannon (Left) and Deutsch Electrical Connectors

5.2.3 TAPE RECORDER REMOVAL (NIMBUS)

Figure 5-5 shows the location of the two recorders and the nuts and bolts that have to be removed. A mockup of the recorders was made to: (1) see what tools were needed (2) determine camera locations in this typical module removal task, and (3) approximate task times. It was determined that ordinary box or openend wrenches modified so the manipulator could grasp them and loosen the bolts were adequate. However, one of the wrenches should be replaced by a power-driven wrench to speed up the time of nut removal. This was not done. The TV camera had to zoom in to a closeup to help

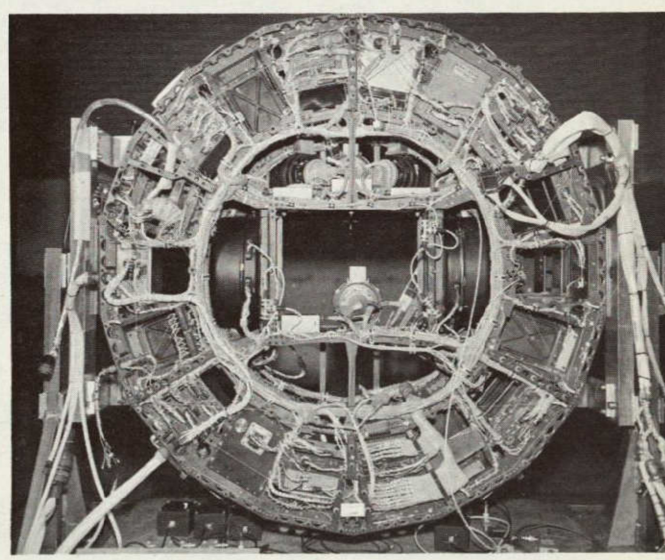

Figure 5-5. Tape Recorder Location

put the wrench over the nut or bolt. Exclusive of the camera relocation time, it took about 9-1/2 minutes to remove a single nut and bolt. However, only two rehearsals were performed and shorter times are expected with more practice.

5.2.4 DEUTSCH ELECTRICAL CONNECTOR (OAO).

The most common connector on the OAO is the quick disconnect type. There are about 22 on the SDHE unit (Figure 5-6). Properly set, it takes from 4 to 6 lb to push the connector on or pull it off. However, the simulation showed that the connector's quick disconnect mechanism could become uncocked if not carefully removed and, if so, it was found difficult to replace. Thus, a special tool was designed to allow resetting the quick disconnect portion.

The connectors are secured from coming loose during launch by snap rings. It was found that an average time of 67 sec for each connector was needed to remove the snap rings for the grouping of three connectors shown in Figure 5-4. Here, the TV camera was in as close as possible (about 1 ft) to obtain the resolution needed to see the small (about 0.07 inch) hole into which the snap ring pliers fit (Figure 5-4).

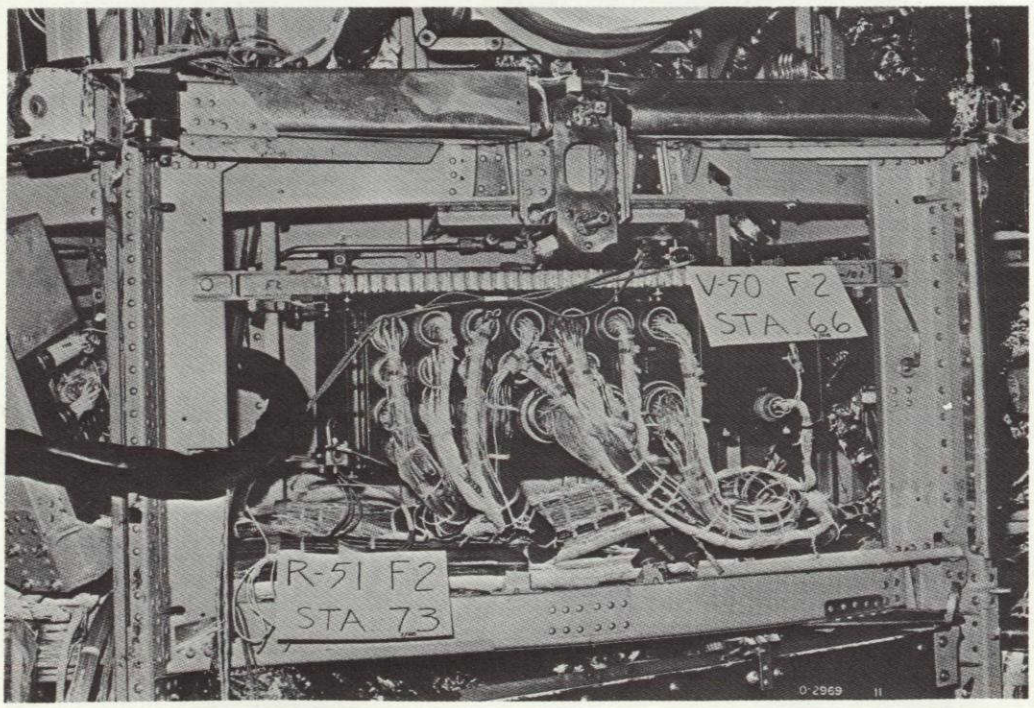

Figure 5-6. SDHE Unit

Subsequent removal of each connector was averaged only 5 seconds. Replacement took much longer since it was necessary to obtain proper alignment of the key and keyway (average 47 sec). Several times were shorter than 30 sec, but not all the connectors had cabling attached which could cause additional troubles if the preset twists were in the wrong direction. The connector's quick disconnect mechanism was reset by hand before starting the test as the desired special tool was not made. Replacement of the snap ring was not thought necessary for satellite orbital operations.

5.2.5 NITROGEN RECHARGE TASK (OAO)

Figures 5-7 and 5-8 show the actual N_2 supply inlet and the laboratory mockup. The simulation was made to determine the time to remove the cap, place a hose fitting and reverse the steps. The times were slightly adjusted to account for: (1) part of the simulated task required direct viewing, (2) the wrench was not modified for manipulator use, and (3) the cap was not tethered. The complete task had an average time of 207 sec, with later trials below 3 minutes.

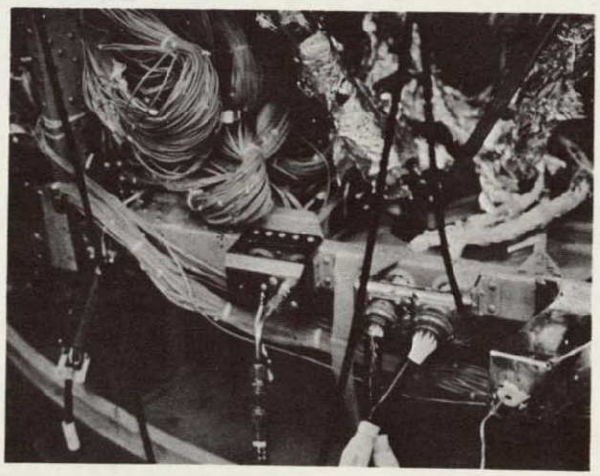

Figure 5-7. Actual Nitrogen Supply Fittings

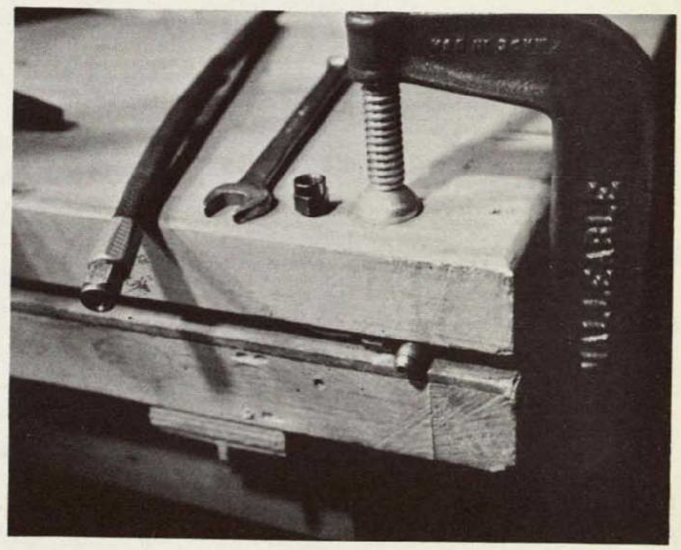

Figure 5-8. Simulated Nitrogen Resupply Task

5.2.6 INSULATION AND TAPE

Most satellities require removal and replacement of superinsulation material or thermal blankets to gain access to components. Manipulator experience was obtained in handling several pieces of material and pressure-sensitive polyester silverized tape in the lab. No task was simulated, but experience was gathered in techniques and problems associated with it.

It was found that the presence of gravity biased the experiment greatly. The tape was difficult to handle and a dispenser is needed. The tape tool should allow easy cutting of the tape without replacement of the hands. Also, a fitted finger is needed on one jaw with a compliant edge to press the tape down. Finally, specular reflections from the silvered insulation caused TV viewing problems.

SECTION 6

MANIPULATOR TOOLS

Optimum selection and design of tools based on trading off task time against system weight is not possible in the short amount of time available in the study. Many hours of creative engineering would have to be put into this portion of the manipulator system design. However, analyses of the Nimbus, OSO, DBS, and OAO maintenance missions give a good indication of typical tooling problems.

The variety and number of tools (as well as replacement fasteners and maintenance ancillaries, such as assembly fixtures) are kept to a minimum to minimize the size and weight of the remote manipulator spacecraft and to enhance the response time of the maintenance mission. (Prelaunch provisions on the satellite also are kept to a minimum for the latter reason. The tools required to maintain the four satellites are described in each maintenance step of Sections 3 and 4. These manipulator tools are described in the following paragraphs.

6.1 GENERAL TOOLS

This category includes those tools similar to conventional manual tools, except for minor or novel modifications which adapt them to manipulator use. A basic question is how these tools will be held by the manipulator; i.e., will the manipulator use the general purpose parallel jaw end effector to grasp the tools or will an easily interchangeable special purpose end effector be used for each tooling task? The latter gives a firmer grasp of the tool than by gripping it with parallel jaws but will take more time to install and remove. The trade-offs between weight, task times, and other factors require further study.

The most essential tool is a general purpose power tool. The tool consists of an electric impact head similar to those previously designed for astronaut EVA. The main purpose of this battery powered tool is to provide the peak torques (e.g., to break screws loose) and to allow rapid removal of nuts, screws, etc. The power tool should be designed to facilitate changing of screwdriver bits, wrench sockets, and other tools it may drive. This could be accomplished by keying the tool to fit into the tool box in the proper orientation when changing bits, sockets, etc. The tool bits and sockets should be designed to retain the fastener and washers and, if possible, to prevent space debris. The tool may be designed to be essentially reactionless but this usually involves more size and weight. The spacecraft tethering system should allow the manipulator to provide some reaction force and torque removal. However, time delay effects could establish that a reactionless tool is essential.

The inventory of general tools for each mission includes only those needed for the mission. Modifications to the standard tools may be done ahead of time and include such things as:

1. Fastener Finders

   a. Ball-ended design on Allen wrenches that preclude perpendicularity of wrenches with the fasteners (Allen-head fasteners common to both present fasteners and each other on the four satellites are used as replacements wherever possible).

   b. Tapered and elongated engagement designs on wrenches.

   c. Tubular finders on blade and Phillips head screwdriver bits.

2. Snap-On/Off Features

3. Manipulator-jaw Finders - On many manipulator-held tools.

4. Some tools have inherent or integral attachment devices for storage in the tool bin such as toggle-action clamps, VELCRO, and wedging surfaces.

5. Some tools have multiple functions such as pliers that can grip and lock (e.g., cotter pins), cut cables (e.g., Nimbus solar paddle power cables) and lacing, cut OAO Al skin panels and crimp the Nimbus recorder heater leads.

6. Distinctive markings and color identification.

7. Fastener retention devices in the tools such as sticky surfaces, spring clips, compliance materials, magnetization, etc.

6.2 SPECIAL TOOLS

For each satellite maintenance mission, it was found necessary to design special tools to expedite the maintenance tasks. Examples from the missions analyzed included:

1. Nimbus C-Clip Gun (Figure 6-1)

   The replacement of the nylon C-Clips that hod the outer thermal blankets in place requires a special tool which has a magazine for several C-clips. The manipulator just presses the gun down over the post to install the C-clip.

2. OAO Al Deutsch Connector Tongs (Figure 6-2)

   The four Deutsch connectors on the two battery packs are in a narrow confined space. A special pair of tongs is used to reach in and grasp the connectors. The tongs incorporate a toglle action to lock onto a connector so that the manipulator need not grasp and pull off the connector simultaneously. The jaws of the tong are designed to grasp the outer shell of the connector when demating and mating the connector.

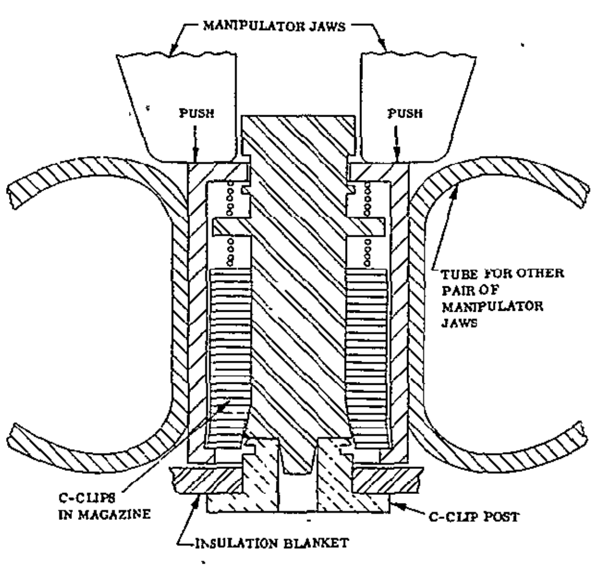

Figure 6-1. Nimbus C-Clip Gun

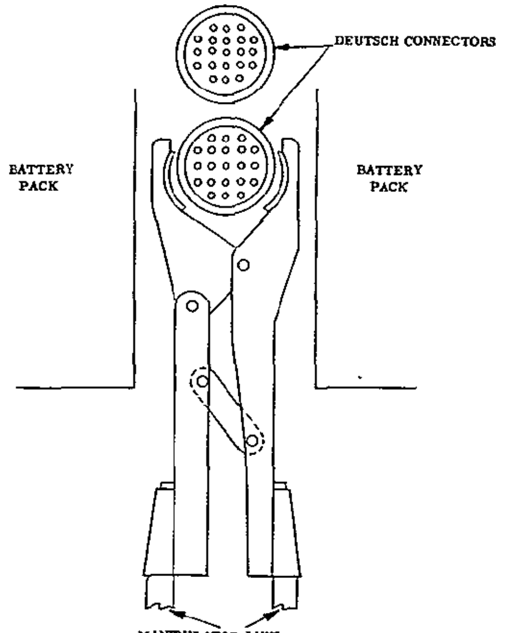

Figure 6-2. OAO Al Deutsch Connector Tongs

3. OAO Al Deutsch Connector Reset Device (Figure 6-3)

During the repair simulations, it was noticed that occasionally the ball retainer in the female connector was not fully reset when the connector was demated. As a result, the balls were not depressed fully by the retainer--the protruding balls prevented mating the connectors. Therefore, a tool was conceptually designed to reset the connector. The manipulator simply pushes each connector down over the reset tool so that upon withdrawal, the retainer is dragged outwardly and depresses the balls.

4. Nimbus Cable Splicer

The heater cables on the Nimbus Tape recorders do not have accessible connectors. The cables are cut and then the heater cables of the new recorder are spliced by a crimping tool in which the cables are inserted and crimped with sharp points to complete the electrical circuit.

5. Nimbus and OAO Super-Insulation Shears

The shears are similar to the electric scissors which children use for cutting paper. The advantage of this type of shears is that the manipulator need only coordinate and direct the movement of the shears relative to the work without

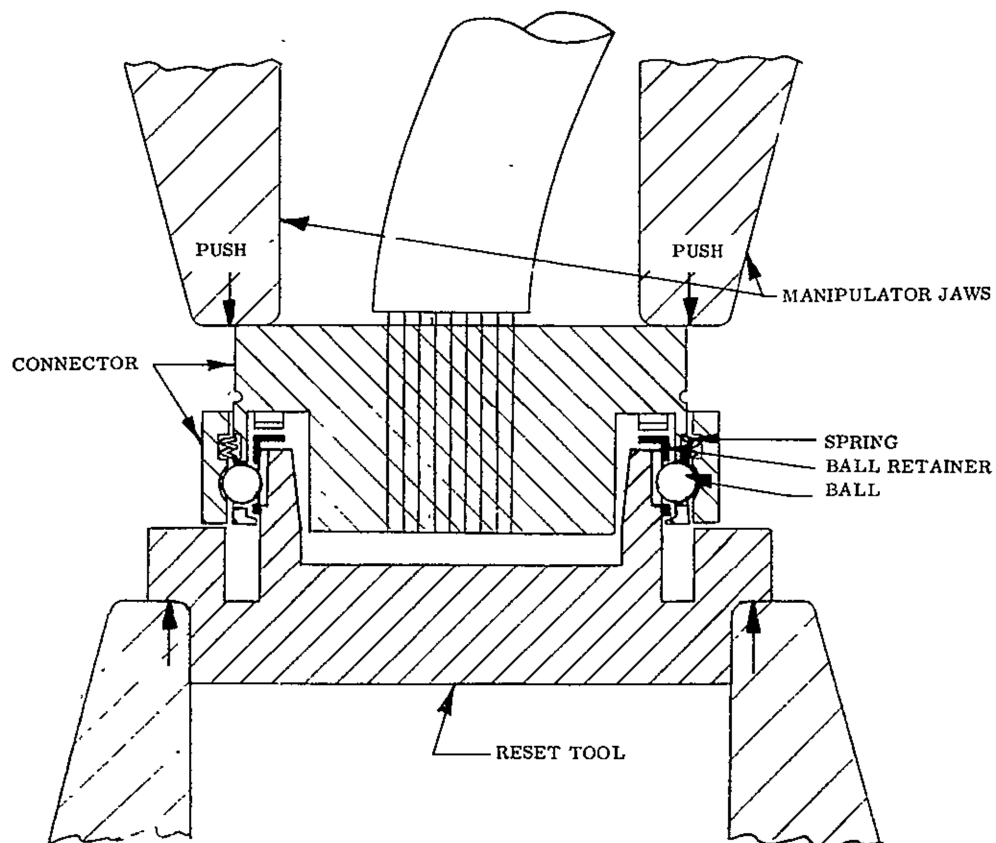

Figure 6-3. OAO Al Deutsch Connector Reset Device

going through cutting motions. A switchblade enclosed in the body of the shears is deployed to start a cut in the multi-layers of tough mylar superinsulation and tape. The length of the switchblade and shear blades and the tips are blunted to avoid damage to underlying parts.

6. OAO and Nimbus Tape Dispenser

It is difficult for the manipulator to tape and hold the insulation in place without a dispenser. The dispenser is held by the jaws of the manipulator. At the first squeeze of the dispenser handle, an inch or two of tape is dispensed to start and pull out the tape. The second squeeze cuts the tape.

7. Nimbus Strut-Truss Bolt Extractor

A bolt extractor similar to a gear-puller is used to extract the bolts (and install them if drift pins are ineffective).

8. OSO Sail Clamp

A special clamp is used to clamp the sail to one of the three spin-gas arms.

SECTION 7

SATELLITE DESIGN RECOMMENDATIONS TO FACILITATE SPACE MAINTENANCE

The following maintenance design criteria and recommendations have been compiled while performing the step-by-step maintenance tasks on each of the four missions. These guides have been supplemented by pertinent criteria and recommendations from other in-house sources and experience. The application of these design criteria and recommendations during the conceptual and development stages of a satellite is an important step in the achievement of a satisfactory maintainability level.

Many of the design recommendations herein are applicable to on-pad maintenance of a satellite. In fact, the on-pad maintenance function while the vehicles are in the armed, radiative or fueled state could be an early application of a remote manipulator system.

The categorization of the maintenance design criteris is chronological by mission phase.

No distinctions have been made whether the design recommendations could be dependent on the level of maintenance effort.

The criteria and recommendations herein are assessed qualitatively regarding their need, their acceptability with regard to reliability, cost, weight, etc., and their impact on present-day design techniques.

A. Pre-Docking Phase

Even if these maintenance design recommendations are not needed for docking to a stationary or slowly turning satellite, some are required for subsequent phases in order to re-stabilize the satellite if the docking and maintenance phases disturb the satellite too much, and to stabilize and service the satellite during the subsequent phases. Many of them are concerned with propulsion subsystems that require command control.

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
1. Additional reserves of attitude control and orbit maintenance propellant to stabilize and passivate satellite prior to docking.	Minor	x		Moderate
2. Additional emergency propulsion systems (probably small, solid rockets) in case above fails	Moderate	x		Moderate
3. Diagnostic thrusts of present propulsion subsystems to test or observe behavior of the satellite	Minor	x		Minor
4. Extra plumbing, valves, etc. to configure appropriate thrust combinations, levels, and pulse rate of a liquid propulsion system for back-up modes of operation.	Minor		x	
5. Termination of malfunctioning propulsion system thrusts, that cannot be commanded off, by tankage blow-out techniques, thrust re-direction, liquid-propellant pump shaft clutches, etc.	Minor	x		Major
6. Additional extendable/retractable inertia booms, YO-YO devices , rods, etc. to alter the moments of inertia of the satellite to despin it.	Moderate	x		Moderate
7. Shift of other satellite masses such as propellants, ballast, or solar arrays to alter the moments of inertia of the satellite to de-spin the satellites.	Moderate		x	

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
8. Locking or snubbing of rotating or translating parts to alter the moments of inertia of the satellite to de-spin the satellites or shorten the docking time.	Minor	x		Minor
9. Jettisoning of satellite masses such as propellant, hazardous equipment, malfunctioning equipment, pre-phase equipment, payloads, R/V, consummables (like food and water), service modules, mission modules, sacrificial equipment (like booms, antennas and solar arrays), etc. in various preparation for docking.	Moderate	x		Moderate
10. Emergency command systems (self-powered) to activate, control or deactivate the parts or entirety of the satellite systems. Back-up manual switches and disconnects on the satellite itself also are recommended.	Major	x		Moderate
11. Strobe markings, flood-lights, identification lights, and read-outs of satellite motion and motion rates.	Major	x		Moderate
12. Provide switching for utilization of satellite sensors such as cameras and radiometers for pre-docking phase inspection.	Minor	x		Minor
13. Safing of explosive devices and other similar stored energy devices.	Moderate	x		Minor
14. Space charge bleeds and other electrostatic bleeds on equipment where the RMS could be shocked or transfer the shock.	Minor	x		Minor
15. Fluid-settling bladders, moveable bulkheads, etc. particularly in large liquid-propellant tanks to minimize inertia transients.	Moderate		x	
16. Utilization of satellite stabilization and orientation systems for pre-docking phase reorientation and inspection.	Moderate	x		Minor

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
B. DOCKING PHASE				
1. Jettisoning of functioning as well as damaged or partially-erected satellite appendages such as antennas and solar arrays, that obstruct or jeopardize docking with the satellite.	Major	x		Moderate
2. Re-positioning of satellite appendages to facilitate docking by enhancing the access to grip-holds (planned and unplanned), and special docking areas such as sacrificial structure and hatches.	Minor	x		Moderate
3. Strengthened and appropriately designed appendages (e.g., inertia booms) for impact docking techniques.	Minor	x		Major
4. Development of docking hardware and areas such as grip-holds, grapple lattices, penetrateable structures, docking cones with energy-absorbing capacity, guide rails and contact switches.	Moderate	x		Moderate
5. On board, extendable tethers, rods, booms, etc., that could reel-in a remote manipulator spacecraft.	Minor	x		Moderate
6. Recesses and barricades for the protection of critically aligned and sensitive components such as attitude control system nozzles and sensors.	Moderate	x		Minor

C. STABILIZATION PHASE

The same satellite design recommendations are required as for the Pre-Docking Phase and Docking Phases.

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
D. REPAIR/REFURBISHMENT PHASE (SAFETY)				
1. In case of restricted access "accident-proof" tunnels and protected runs for	Minor			Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
critical harnesses, hydraulic lines and emergency equipment (e.g., stand-by command systems).				
2. Purging and emptying of connections (propellant, hydraulic, coolant, cycle, bioscience fluids).	Moderate	X		Minor
3. Shielding of high-voltage and radiation equipment and connections.	Minor	X		Minor
4. Access cut-lines and structure for rescue and access to vital equipment.	Minor	X		Moderate
5. Selection of separation systems and other pyrotechnic systems in order to minimize debris and explosive threat to the satellite and the Remote Manipulator Spacecraft.	Minor	X		Moderate
6. Well-shielded power-supplies and automatic or manual capping of power connections.	Minor	X		Minor
7. Interlocks and lock-outs to assure disconnection of electrical energy (e.g., power supplies), mechanical energy (e.g., cocked springs), and chemical energy (e.g., propellants).	Minor	X		Minor
8. Location of satellite equipment so that access to them does not subject the Remote Manipulator Spacecraft to high voltages, excessive temperatures, moving parts, chemical contamination, etc.	Minor	X		Moderate
9. Alerting devices to warn the RMS of impending danger such as radiation.	Minor	X		Moderate
10. Equipment cases on the modules, experiments, and housekeeping subsystem that have two main purposes: o To maximize protection, such as dust proofing, during all phases such as the launch, orbit, and maintenance phases. o To minimize handling during all phases by being modular and integral with the equipment.	Moderate	X		Moderate
11. Safety devices to prevent activation of equipment that could damage the Remote Manipulator Spacecraft such as: automatically deployed solar arrays, antennas, inertia booms, hatches, separation devices, and spin-stabilization devices.	Minor	X		Minor

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
(ACCESS) REPAIR/REFURBISHMENT PHASE				
1. Multiple work platforms and docking hardware such as grip-holds and grip-rails.	Minor	X		Minor
2. Single-motion, quick-disconnect, highly-accessible services connections (propellants, electrical cables), fasteners, and other hardware on hatches, access panels, assembly systems, etc. that react on themselves rather than the Remote Manipulator Spacecraft when activated. Push-pull motion is preferred for the manipulator, even if the manipulator can be programmed or indexed for other motions such as circular or lateral motions. However, torquing is perhaps the primary tool function expected for space maintenance and assembly operations. It is unlikely that assembly and disassembly techniques will be developed which do not rely upon some sort of torque whether light as for knob adjustments or heavy as for bolt break-away torques.	Moderate	X		Moderate
3. Integral disassembly and assembly fixtures such as pivoted racks, equipment drawers, and hinged access panels. Furthermore, large, sensitive, exactly-aligned equipment although weightless in orbit requires assembly fixtures rather than just tethers to accelerate, decelerate, guide, restrain and temporarily hold the equipment.	Minor	X		Moderate
4. Accessible shut-off valves, by-pass valves, and plumbing.	Minor	X		Minor
5. Large and frequent hatches and access doors as well as "through-panel" maintenance techniques which are enhanced by the usually non-monocoque construction of satellites.	Minor	X		Moderate
6. Tapered and rounded access-opening frames, compartment structure and adjacent hardware to prevent hang-ups of manipulators, tools or satellite equipment during maintenance. Closed or foam-filled structural sections would reduce hang-ups further.	Minor	X		Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
7. Integral assembly fixtures, registration/positioning devices, adapters, and temporary thermal protection measures.	Minor	X		Minor
8. Coatings and materials to prevent space-welding.	Minor	X		Minor
9. Extra holes and slots in skin panels and compartment walls for extra cables, fluid lines, and mounting points.	Minor	X		Minor
10. Accessibility must be traded-off only to a slight extent against the module and satellite aspects of size, weight, and complexity. The most frequent obstruction problems can be alleviated as follows: • Avoid overlying cables and harnesses, thermal covers, shutters, and insulation. • Provide ability to swing or reposition solar arrays. • Avoid overhanging connector brackets and support structure. • Hinge protective covers.	Minor	X		Moderate
11. Utilization of tapped-holes, grip-wells and depressed handles if protruding grip-holds adversely increase impact damage on the remote manipulator spacecraft and replacement equipment.	Minor	X		Minor

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
12. Avoidance of satellite parts that can flail, entangle, or foul the satellites or the remote manipulator spacecraft by wrapping, wedging, jamming such as: • Lanyards • Dangling umbilicals • Tethers (particularly flexible) • Safety-wire • Flexible and articulated parts unless rigidizeable remotely or manually. Utilization of rigidizeable or yieldable wrappings or inserts in such flexible components which will limit their motion or make their position predictable for service.	Minor	X		Minor
13. Adequate clearances for tools, connectors, modules, fixtures, sub-assemblies, and tether grips.	Minor	X		Minor
14. Adequate tether pads on the equipment and satellite (for temporary storage).	Minor	X		Minor
15. Provisions for positioning, mounting, and locking a moving component in a position for improved access to that component or other components or for securing that component if malfunctioning. These operations should be remote and free of the moving component itself.	Minor	X		Minor

	NEED	ACCEPTABLE YES	NO	EFFECT ON SATELLITE DESIGN
ASSEMBLY, MODULARIZATION AND LOCATION				
1. Minimum sequential assemblies such as for installing skin panels. Minimum sequential assemblies also for carefully aligned parts. Akin to the minimization of sequential assemblies is the best utilization of logical assembly procedures (e.g., removal of covers first) and temporal assembly procedures (e.g., removal of input devices first) all of which may conflict with one another to a degree in the maintenance process.	Moderate	X		Moderate
2. Debris-proof compartmentalization (e.g., fire-walls or equipment isolation) to prevent contamination in case of depressurization, short circuit, etc. Some controllable connecting pathways are required to allow in-orbit purging by an attached device or when enclosed in a "clean-room."	Moderate	X		Moderate
3. Separate major functional groups not only between the payload and housekeeping categories but also within these categories.	Minor	X		Moderate
4. Unitized, rigid structural systems for systems that require fine alignment such as sensors and nozzles. However, the resulting inter-connecting structure may be massive, complex, and interrupt other	Moderate	X		

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
systems. Since the usual techniques of mechanical alignment are not practical in space, the attainment of fine alignment accuracies are limited in space unless new alignment techniques are developed. One approach is to maintain the alignment interface status by inherent mechanical characteristics such as unitized rigid structural systems or by detailed alignment records of the mounting pads which, unfortunately, could be violated by mechanical or thermal damage. Other alignment recommendations are: • Provide alignment surfaces, pins, indices. • Identify satellite axes • Provide two perpendicular surfaces for three-axis alignment. o Use rigid, accessible and large alignment pads.				
5. Modularized subsystems such as electronics modules for ready internal or external access without removing control parts and without draining liquids and coolants. The more frequent thermal interfaces may reduce the thermal responsiveness of the equipment and the heat soak into the module compartment structure. But both thermally conductive grease and radiation control methods (coatings, superinsulation, and automatic shutters) are effective.	Minor	X		Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
6. Optimized and varied sizes of modules and equipments controlled by:	Minor	X		Moderate

a. The volume, shape, and dimension requirements of the subsystem itself and the possible merger of it with other subsystems that have similar functions, thermal control requirements, reliability/accessibility requirements, and alignment requirements.

b. The thermal control requirement aspects of:

- Size and orientation of satellite radiating surfaces that have coatings, superinsulation, and shutters.
- Size of allowable structural openings and hatches for radiative surfaces and shutters. Structural doors can be resorted to for boost loads; structural loads are low in-orbit.
- Heat-sink response of the supporting structure.

c. The number and "plugability" of the mechanical and electrical connections that also require lead-ins and simple, ganged, non-reactive mating action. Small, and few connectors should be a goal. Guide-rails, assembly fixtures, lead-ins facilitate mating.

d. Module weight is a minor limitation during handling because of zero-g conditions in-orbit. Module size could be limited by booster shroud capacity and access hatch size.

	NEED	ACCEPTABLE YES	NO	EFFECT ON SATELLITE DESIGN
e. The module handling shape is a minor requirement unless the shape distorts the size of the satellite or the booster shroud, because the remote manipulator can grip module handles.				
7. Minimization of the number of independent, non-interlocked degrees of freedom of structures, appendages, and equipment during assembly or erection. This reduces the number of the repair alternatives, but it reduces the number of lock-up or mis-shape possibilities. Randomness must not be allowed. Mechanically interlock as many functions as possible so the remote manipulator spacecraft can actuate them all from a few positions on the satellite.	Minor	X		Moderate
8. Mounting of antennas, solar arrays, pointed experiments, etc. on common rotating platforms where allowed in order to minimize cable loops, slip-rings, rotary joints, bearings, etc.	Minor	X		Moderate
9. Decreased emplacement density of modules and experiments (i.e., increased clearance between modules and experiments) in order to facilitate access to and turning of satellite components.	Minor		X	QW
10. Reduction, if not elimination, of fluid lubricants, cycle fluids (as in advanced power supplies and thermal control equipment), damping fluids, etc. that could spread because of a failure.	Minor	X		Moderate

	NEED	ACCEPTABLE YES	NO	EFFECT ON SATELLITE DESIGN
11. Utilization of non-destructive maintenance and assembly techniques such as those that avoid processes that require chemicals, chipping, heat/pressure, curing, welding, cutting, etc. due to the difficulty of performing these processes in the space environment and of cleaning up the wide-spread space debris.	Minor	X		Moderate

DIAGNOSIS

	NEED	ACCEPTABLE YES	NO	EFFECT ON SATELLITE DESIGN
1. Diagnostic indicators and control panels including instruments, indicators (lights, flags, sensitive coatings) press-to-test lights.	Minor	X		Moderate
2. Adequate inspection windows and removeable covers.	Minor	X		Minor
3. Modules composed of elements which are functionally related. Thus, the fault may be located and isolated more rapidly And after the repair the module may be submitted to a functional check-out.	Minor	X		Moderate
4. Leak detection aids such as dyes, depositions, fluorescence, odor, and radiation.	Minor	X		Minor
5. Thermal paints and crystals on temperature sensitive parts, high temperature parts, and even temperature insensitive parts.	Minor	X		Minor

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
6. Equipment designs that are not selectively mated, matched, or balanced to one another.	Minor	X		Moderate
7. Equipment designs that can be checked, operated, and adjusted separately particularly components that terminate the mission from a performance or maintenance aspect such as attitude control systems.	Minor	X		Minor
8. Easily identified and probed test points of increased diagnostic capability.	Minor	X		Minor
9. Conveniently located and logically arranged terminal boards (including flow path diagrams), ., control panels, distribution buses and centers.	Minor	X		Moderate
10. Sufficiently stable adjustment settings to permit a complete scan of test scheduling without adjustment.	Minor	X		Moderate
11. Telemetry system capability to provide fault isolation data down to the module level, at least, to the diagnosticians and then the spacecraft operators.	Moderate	X		Moderate
12. Electrical and mechanical compensation, tuning, and adjustments, if not precluded by initial satellite design, attained by remote controls and monitoring devices.	Moderate	X		Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
13. Extra capacity for acquiring diagnostic data and for modifying hardware such as extra harness wires, connector pins, slip-rings, and mounting hardware.	Minor	X		Minor
14. Compilation of extensive maintenance data banks in order to enhance in-orbit maintenance.	Moderate	X		Moderate
15. Utilization of the latest techniques of maintenance plans, programs, design (such as the isolation of functions for the analysis of failure symptoms), prediction, training, and application.	Minor	X		Moderate
16. Evaluation of the design for mantainability with regard to ease of manipulation and interpretation, validity, reliability, standardization, comprehensiveness, efficiency, discrimination, and diagnosis.	Major	X		Moderate
17. Utilization of the results of trade-off studies between manual and automatic maintenance methods and their applications regarding cost, flexibility, response time, available maintenance action, and type of maintenance function. For example, automation verifies well, trouble-shoots fairly well, but inspects poorly. Automation has limited capability for adjusting, servicing, replacing, and repairing. Furthermore over-emphasis on automation of the maintenance system may actually create more problems than it solves.	Minor	X		Major

	NEED	ACCEPTABLE YES	NO	EFFECTIVE OF SATELLITE DESIGN
18. Appropriateness of the functional level of the equipment with the level of the maintenance analyses and action. For example, a small electronic module probably should be maintained at the GO-NO-GO level.	Minor	X		Moderate
SERVICE AIDS				
1. Additional umbilicals, connections, and capacity for providing electrical power, propellants, reference data, communications data to the satellite from the maintenance vehicle.	Moderate	X		Moderate
2. Identification and assembly aids such as color codes, markings, large numbers, and irreversible assembly keys and shapes on modules, handling pads, access doors, adjacent structure, assembly points, electrical cables, propellant lines, etc.	Minor	X		Minor
3. Peel-off techniques to clean windows and sensors if optical properties are not changed prohibitively.	Minor		X	QR
4. Planned reflectivity, absorptivity, and diffusivity of equipment, components and their background, superinsulation, and coatings. Illumination, sun shades, and thermal shades also must be considered.	Minor	X		Minor
5. Extra mounting pads, racks and services such as electrical power and thermal control for illumination, maintenance, and by-passing of unremoved failed equipment.	Minor	X		Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECTIVE ON SATELLITE DESIGN
6. Attachment and adjustment alternatives such as cleats, tow fittings, turn-buckles, and extra holes in order to straighten, align, and support solar arrays, booms, c.g./mess inertia ballast, piggy-back equipment, and temporary emergency equipment such as power supplies and temperature control devices.	Minor	X		Minor
7. Integral, aligned fittings to temporarily attach stabilization, alignment, rigging aids, and fixtures such as optical sights, collimators, and propulsion packs.	Minor	X		Minor
8. Grip-holds and internal and external hatches for erecting, attaching and purging with a clean room device, or for simply purging the inside and outside contamination away by gas. These are required to purge and isolate against: • Drillings, chip, debris, scrappings from the maintenance actions. • Failure products, char, and debris. • Residual initial debris. • Splashed, leaking, incompletely purged fluids such as propellants, lubricants, damping fluids, cycle fluids, fuel-cell water, and biological waste products. • Mis-diverted exhaust nozzle deposits.	Minor	X		Moderate

	NEED	ACCEPTABLE YES	NO	EFFECTIVE ON SATELLITE DESIGN
• Out-gasses and deposition products.				
• Electrical flash marks.				
• Space dust.				
• Re-entry skip-out char.				
• Joining processes debris.				
The gas fans and gas nozzles should be reacted so as not to accelerate the satellite nor the remote manipulator spacecraft.				
9. Rigid or collapsible, integral, temporary storage containers with controlled environment for use during maintenance.	Minor	X		Moderate
10. Extra clamps and other assembly/disassembly aids (e.g., tape) stored on board for maintenance.	Minor	X		Minor
11. Adhesive areas, velcro, magnets, clips, etc. inside outside the satellite to temporarily retain tools, fasteners, small parts and debris.	Minor	X		Minor
12. Manually and automatically adjusted alignment stops for in-orbit adjustments.	Minor	X		Moderate

		NEED	ACCEPTABLE YES	NO	EFFECT ON SATELLITE DESIGN
13.	Coated or pre-conditioned electrical and mechanical connections to minimize debris generated at taxing surfaces particularly during assembly and disassembly.	Minor	X		Moderate
14.	Instructions, flow-charts, calibration charts, warning placards (e.g., high voltage and extreme temperature warning signs) on the equipment and satellites appropriate to the remote manipulator spacecraft.	Minor	X		Minor
15.	Installation of special protection devices not carried on the remote manipulator spacecraft because of weight limitations, such as thermal, sun/light, and radiation shades.	Moderate	X		Moderate
16.	Adoption of maintenance programs that may require less overall maintenance capability, such as certain programmed maintenance programs, thereby reducing the implementation of maintenance design practices and recommendations that are too heavy, large, costly, etc.	Moderate	X		Moderate
17.	Utilization of design criteria based on the launch and in-orbit maintenance position and environment in the design of satellites and satellite equipment.	Moderate	X		Moderate
18.	Installation of thermal control and lighting systems to facilitate in-orbit maintenance such as thermal control blankets on equipment temporarily stored outside the satellite during a maintenance operation.	Minor	X		Moderate
19.	Provisions for refurbishment equipment such as pre-installed harnesses, empty mounting spaces, pre-aligned mounting pads, c.g./mass moment balancing, and thermal balancing.	Minor	X		Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
20. Utilization of the alignment of equipment mounting pads to register and align refurbishment equipment.	Minor	X		Minor
21. Installation of attachment hardware and pads for de-orbit burn-up or re-entry hardware such as ballast weight, trim surfaces, de-orbit propulsion systems, or re-entry systems.	Minor	X		Minor
REPLACEMENT AIDS				
1. Commonality and standardization of components, connectors, grip-holds, fasteners, tools, etc. on the satellite and even between satellite programs such as solar paddles, antennas, housekeeping subsystems, hatches, and adapters. For example, maintenance of the modular level (75% to 80% of the total maintenance effort) consists of removing/replacing nuts, bolts, and connections.	Moderate	X		Major
2. Selection of materials and coatings that can be processed in-orbit. However, these processes must not create space debris.	Minor	X		Moderate
3. Standardization of module bays and mounting areas.	Moderate	X		Major
4. "Manipulator-engineered" equipment and fasteners: o Shape and size of grip-holds, knobs, tubular tool finders o TV resolution and end-effector on resolution. o Fastener finders and fastener size (e.g., heads)	Moderate	X		Moderate

	NEED	ACCEPTANCE YES	ACCEPTANCE NO	EFFECT ON SATELLITE DESIGN
o Single-motion, minimum reaction, locked-in-place tools.				
o Single-motion operation of tools, fasteners, connectors, latches, etc. because multiple motions are difficult for the remote manipulator spacecraft such as pushing and twisting simultaneously.				
o Testing by manipulator.				
o Resistance to the grip of the end-effector and strength of the manipulator.				
o Resistance to the unprotected edges of the end-effector.				
5. Assembly, fixture, and tool lead-in guides, tapered shafts and hubs, integral drift pins, etc.	Minor	X		Moderate
6. Large radius edges or protected edges particularly at assembly interfaces in order to minimize accidental impact damage.	Minor	X		Minor
7. Fasteners, connections (mechanical and electrical), locking devices, and locomotion hooks that have a minimum of actuation force, actuation torque, push-off, and tool slippage.	Minor	X		Minor
8. One-handed operations required of the remote manipulator spacecraft in as many in-orbit maintenance tasks as possible in order to:	Minor	X		Moderate
o Reduce hatch and access opening size and clearances.				
o Reduce manipulator working volume.				
o Reduce manipulator impact damage and hang-ups.				
o Simplify and shorten the maintenance operations.				
o Allow a one-handed, partially-functioning remote manipulator to still function.				

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
9. Thermal blankets and super-insulation should be hard-backed and reasonably rigid if not integral with the equipment they protect for several reasons:	Minor	X		Moderate

- o To reduce damage to the insulation by edges on the satellite equipment and the remote manipulator spacecraft.
- o To reduce the number of insulation fasteners and attachment tape which frequently require special tools.
- o To reduce the assembly time.
- o To provide a firm insulation for the end-effect or to emplace over the equipment.

spacecraft which does not have a tactile sense.

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
10. A new line of electrical connectors to avoid:	Minor	X		Moderate

- o Rocking motion during mating and de-mating.
- o Push-off forces.
- o Combined mating motions.
- o Lack of suitable grip-holds and storage-holds.
- o Pin damage and shock damage.
- o Lack of easy keying.

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
11. Devices to hold cables out-of-the-way during maintenance operations. Examples are hook and pile tape, wide-mouth spring-actuated clamps, yieldable cable coverings, or internal stiffners, and spring-loaded cable retractors.	Minor	X		Minor
12. Minimization of the number of maintenance task steps: o Pre-installed captive fasteners, gaskets, shims, thermal grease (impregnated pads), clamps, insulation, fixtures, caps, covers, etc. o Utilize lead-ins, chamfers, mating finders, tapers, guides, etc. on fasteners, connectors, tools, fixtures, mechanical parts (hubs, base plates) modules, and other assemblies. o Utilize gauged fasteners, connections, actuators, locks and pins and fewer but stronger such elements. o Minimize in-orbit calibration, alignment, test, and checkout of satellite equipment and maintenance equipment.	Moderate	X		Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
13. Electrical harness recommendations:	Moderate	X		Moderate

- o Locate harnesses where damage to them is remote but are still suitably accessible to a degree for modifications and maintenance.
- o Provide flexible loops and slack of connectors.
- o Avoid large number of wires and stiff wires in order to attain sufficient flexibility, low positioning forecast and torques, and adequate access to the connectors appropriate to the number of connectors required.
- o Lace cables rather than enclose them in tight sleeves in order to enhance cablė flexibility. Protection can be maintained by slight increases in the coverings on individual wires.
- o Code wires and particularly harnesses not only by numbers and colors but also by patterns and other markings.
- o Plan and provide adequate space for the harnesses, connectors and connector tools.
- o Shape the cable putting at the rear of the connectors suitably for the manipulator end-effector.
- o Use connectors to which pin straighteners or jack-harnesses can be applied in case of pin damage.
- o Segment the harnesses with frequent connectors (compatible with reliability) in order to facilitate removal, test and installation.
- o Group the harnesses and cables according to their functional subsystem.

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
o Protect the wires, cables, harnesses from pinching, abrasion, and cutting by maintenance actions such as removing hatches, equipment, or thermal insulation.				
14. Make connectors, modules, and other equipment in proximity to similar equipment dissimilar preferably by keys and other mechanical interference methods.	Minor	X		Minor
15. Utilization of "off-the-shelf" equipment that is either standardized for a variety of space vehicles previously qualified, and flown, or from increased spares supplies.	Minor	X		Moderate
16. Installation of corresponding assemblies and subassemblies of the same moded number interchangeable both dimensionally and functionally.	Minor	X		Minor
17. Adherence to the satellite design criteria regarding the anthropometrical capabilities and dimensions of standardized remote manipulator spacecraft such as the strength, working reach, manipulative axes of rotafion, TV resolution, end-effector resolution, manipulator dimensions, remote manipulator spacecraft dimensions, and supply bin location.	Moderate	X		Moderate
18. Avoidance of pressure fits, tapered locking surfaces, and other assembly practices that require excessive force, impact and torque applications.	Minor	X		Moderate
19. Provisions for manual operation, deployment or erection of satellite equipment normally automatically actuated such as boost shrouds, adapters, hatches, covers, solar arrays, inertia booms, and antennas. Initial orresidual forces, if any, should react on the satellite rather than on the equipment or the remote manipulator spacecraft.	Minor			Moderate

	NEED	ACCEPTABLE YES	ACCEPTABLE NO	EFFECT ON SATELLITE DESIGN
20. Positioning of a single satellite subsystem in several environments and locations on the satellite so that not only the back-up elements but also the re-supplied elements are not subjected to the same environment or condition that may have caused the initial failure.				
21. Utilization of different design criteria of the replacement equipment due to different launch environments (shock, vibration, thermal protection, areo pressures, dust, rain) from the initial mission launch environment.				

SECTION 8

REMOTE MANIPULATOR SPACECRAFT SYSTEM DESIGN

The approach used to design the system is described herein. Design requirements were taken primarily from the results of the mission analyses described in Sections 3 and 4. The requirements call for a single mission, 10-day life spacecraft which would be compatible with the DSV-2L, two-stage Delta launch vehicle with standard shroud (booster selection was not required by NASA). The ground station requirements call for a re-positionable, re-useable system. Communications between ground station and spacecraft are via a data relay satellite.

Payload requirements for each of the five missions examined varied between 253 and 1237 pounds. The basic spacecraft weight is 968.1 pounds. Payload packaging varied between missions in order to minimize CG travel.

The spacecraft reliability analysis resulted in a reliability prediction in excess of 0.90.

8.1 SYSTEM REQUIREMENTS

The mission analyses described in Chapters 3 and 4 provide most of the requirements for the operational remote manipulator spacecraft system. The remaining requirements are derived from standard launch and on-orbit operations. Following is a list of the 29 most significant system requirements:

1. The system will consist of four items; a remote manipulator spacecraft, a ground control station, ground support equipment at the manufacturer's facility and ground support equipment at the launch pad.

2. The system shall be compatible with the DSV-2L, two stage Delta booster with a standard Delta shroud. (Selection of a launch vehicle was not required by NASA. The General Electric Company selected a launch vehicle in order to establish a launch configuration design constraint by way of the booster shroud. The DSV-2L was selected, because it was the least costly booster available which met the required payload capability.)

3. The system shall be capable of surviving the DSV-2L and T III C launch environment.

4. The basic manipulator spacecraft shall be identical for all missions and differences shall exist only in spare parts, tools, test equipment and quantity of expendables peculiar to each mission.

5. The remote manipulator spacecraft shall be designed to have a minimum orbital life of 10 days.

6. The remote manipulator spacecraft shall be capable of performing a rendezvous maneuver to bring it from the booster separation point to the target satellite (worksite) and it shall be capable of maneuvering around the worksite under command from ground control.

7. The remote manipulator spacecraft shall be capable of docking to and stabilizing the worksite even though the worksite is initially spinning or tumbling. Special docking aids may be used.

8. The remote manipulator spacecraft shall be equipped with two bilateral electric manipulators, each capable of 6 degrees of freedom.

9. Docking tethers will be provided to achieve a passive and rigid mechanical coupling between the remote manipulator spacecraft and the worksite.

10. Communications between ground control station and low altitude remote manipulator spacecraft shall be via an assumed operational data relay satellite system.

11. Communications between the ground control station and a remote manipulator spacecraft at synchronous altitude may be direct or via an assumed operational data relay satellite system.

12. Full-time communications between the remote manipulator spacecraft and a ground control station shall be maintained with the exceptions of outages while slewing the spacecraft antenna from relay satellite to relay satellite.

13. The remote manipulator spacecraft communications subsystem shall be capable of transmitting video, force feedback, and housekeeping telemetry to the ground control station and shall be capable of receiving manipulator and camera position plus system commands.

14. The primary communications subsystem antenna will be steerable and directional.

15. A 4π steradian back-up antenna shall be provided to receive commands in the event of primary antenna failure.

16. The remote manipulator spacecraft shall carry a beacon for ground tracking.

17. The remote manipulator spacecraft video subsystem shall be capable of operating in two modes:

    a. Mode 1: Provide stereoptical visual data during the docking phase.

    b. Mode 2: Provide monocular visual data during the working phase. In this mode a third movable camera can be used to provide closeups and a greater variety of viewing angles.

18. To minimize bandwidth requirements, video system resolution no greater than that determined by the mission analysis and minimum frame rate dictated by human factors considerations shall be used.

19. The video sybsystem shall be operable with either artificial or sunlight illumination. The video subsystem shall include the artificial light source necessary to allow the cameras to operate in shadow and it shall also have protection against inadvertent exposure to direct sunlight.

20. The pan and tilt motion of the video cameras shall be ground commandable and use manipulator actuation as a backup in case of failure.

21. Camera focusing, stereoptical converging and focal length control shall be ground commandable and use manipulator actuation as a backup in case of failure.

22. Electrical energy will be provided for all missions by Ag-Zn primary batteries.

23. Thermal control shall be with passive techniques.

24. Attitude control torques for the remote manipulator spacecraft shall be provided by monopropellant thrusters.

25. The attitude control subsystem shall operate in the following modes:

    a. Initial stabilization and acquisition after booster separation.

    b. Spacecraft stabilization and control during the rendezvous, docking, and orbit change phase.

    c. Stabilization and control of the docked remote manipulator spacecraft and worksite during the working phase.

26. The remote manipulator spacecraft system shall store and transport spare parts and tools for each mission. Passive tethers will be provided to stow articles removed from the target spacecraft.

27. The system ground control station shall house the manipulator master station, video monitors and control station, telemetry processing and display station, command station, data processing equipment and test conductor station.

28. The system ground control station will be van-mounted and designed to be tied into NASA tracking and satellite control facilities as well as an assumed operational data relay satellite system.

29. The system ground control station shall receive and process remote manipulator spacecraft video, force feedback, and telemetry information and shall process and transmit manipulator and camera position commands and the commands required for remote manipulator spacecraft operations.

8.2 SYSTEM DESCRIPTION AND OPERATION

The remote manipulator spacecraft system consists of a ground control station, a remote manipulator spacecraft, Aerospace Ground Equipment (AGE), launch support equipment, and a factory ground station.

8.2.1 GROUND CONTROL STATION

The ground control station is mounted in a van to provide mission flexibility. Before each mission, it is repositioned in the vicinity of the target satellite ground station. It has equipment for telemetry processing and display, for remote manipulator spacecraft command and control, for manipulator control and for mission supervisory functions and is operated by a team of six men. The ground station interfaces with the target satellite control station, with satellite tracking facilities, with a communications center and with a target satellite information center.

8.2.2 REMOTE MANIPULATOR SPACECRAFT

The remote manipulator spacecraft is equipped with position correspondence force feedback manipulators, with an operator-aimed video camera system, with a renedezvous propulsion subsystem, and the usual assortment of housekeeping functions such as command, control, telemetry, tracking, power, thermal control and attitude control. The spacecraft has a minimum 10-day design life in orbit.

8.2.3 AGE, LAUNCH SUPPORT EQUIPMENT, AND FACTORY GROUND STATION

The AGE and launch support equipment will have to be designed to provide satellite status monitoring after mating to the booster on the launch pad. The factory ground station will contain most of the equipment in the ground control station and will be designed to provide system test data from spacecraft assembly through spacecraft test. This equipment will be designed concurrently with the spacecraft and it will also serve as a test bed to determine the ground control station parameters. Designing in this fashion allows the ground control station to be a near optimum design in terms of operational and human factor effectiveness with modifications being made only to the factory test equipment.

8.2.4 SYSTEM OPERATION

The remote manipulator spacecraft has two principal control modes. One mode is for computer-originated command, the second mode is for operator originated commands. The computer generates commands such as tracking antenna pointing angles for relay satellite acquisition and spacecraft attitude commands for renedezvous thrusting. These commands are generated in the ground command station by operational software. The second type command originates in the remote manipulator master station. This station will have to be designed so that an operator will be able to control the slave manipulators by moving his own hands and arms to appropriate locations, and position the spacecraft cameras through a head-aimed device. The operator will also have the ability to generate attitude commands to the spacecraft, commands for switching video cameras and additional thruster firing commands.

8.2.4.1 Launch Operation

The spacecraft is delivered to the launch site and mated to the booster. The ground control station is moved to the appropriate center for hook-up to target satellite ground station, the target satellite information center, and tracking and communication facilities. The booster would be launched to nominally intercept the target satellite both in position and time. Because of booster navigation and guidance errors, the spacecraft is injected with position, time and velocity errors.

8.2.4.2 Rendezvous

In order to accomplish renedezvous, thrusts must be applied to remove boost errors and to apply the appropriate ΔV to bring the spacecraft near the target satellite. The thrust levels and durations are generated on the ground based on tracking data provided through the spacecraft tracking transponder and the ground tracking facilities. The computer-generated thrusts bring the spacecraft to a point approximately 1 mile from the target satellite whereupon visual acquisition is accomplished through the operator on the ground.

At booster separation, the rate integrating gyros null the spacecraft body rates. The tracking antenna is commanded into a search mode to acquire and track the relay satellite. The IR sensors are used before rendezvous thrust commands in order to update the gyros and minimize thrust misalignments.

8.2.4.3 Docking and Mission Performance

After visual acquisition spacecraft attitude is ground-commanded and the optics are used to provide range, range rate, bearing and bearing rate data to the operator and ground computer. Final docking thrust commands are computer-generated with the operator having override capability. The operator visually inspects the target satellite to determine rates, docking point, and docking maneuver. The operator then proceeds and attaches the remote manipulator spacecraft to the target satellite. The ground computer uses the visual docking data to define the docking point, generates a new firing sequence and duration for the manipulator spacecraft control thrusters. This modification is based on the new moments of inertia and mass distribution. The operator then proceeds with the maintenance mission. Relay satellite switching occurs when the line of sight to the relay satellite moves to the end of the tracking antenna gimbal travel. The ground computer generates the antenna slew angles for repositioning to acquire the next relay satellite.

8.2.4.4 Orbit Change

At mission completion, the operator will un-dock the manipulator spacecraft and the modified thrusting sequence is removed. The operator finally commands the rendezvous engines to retrothrust and move the manipulator spacecraft into a new orbit.

8.3 SYSTEM WEIGHT SUMMARY

The remote manipulator spacecraft system is designed to execute five selected maintenance missions with maximum spacecraft commonality. All subsystems except propulsion can be designed to satisfy this requirement.

For purposes of this study, the impulse requirements for rendezvous, maneuvering, stabilization and deboosting the manipulator spacecraft are assumed equal for all five missions. Payload weight requirements differ from mission to mission, ranging from 253 pounds for the OSO mission to 1237 pounds for the Nimbus D to E mission. The mission weight summary is shown in Table 8-1. Table 8-1 shows the effect of mission specific requirements on payload, propellant, and propulsion subsystem weights. Propellant weights can be held constant for the Nimbus A to C, DBS and OSO with propellant and pressurant tanks common for each of these three missions. The heavier payloads in the OAO and Nimbus D to E missions require additional propellant and larger tanks. The weight increment for these heavier tanks causes a corresponding increase in dry spacecraft weight.

The spacecraft requires no changes, even in the propulsion subsystem, for any of the first three listed missions, and is therefore termed the "Basic Remote Manipulator Spacecraft." The weight and balance summary for the Basic Spacecraft is shown in Table 8-2. Note that the weight for the fueled spacecraft ready to fly, but without any provisions for payload, is 968.1 pounds. This basic spacecraft weight appears in the five mission specific weight and balance statements.

Table 8-3 (Mission A), Nimbus A to C Refurbishment Mission Weight/Balance, shows one of the "average" mission weight statements. Note that ballast of 47 pounds is incorporated to control cg location.

Table 8-4 lists (Mission B) DBS Refurbishment Mission Weight/Balance. The basic spacecraft sufficies and the ballast requirements of 136 pounds is tailored to control the cg location.

Table 8-5 (Mission C), OSO Repair Mission Weight/Balance includes the basic spacecraft and the heaviest ballast, 151 pounds, of all five missions.

Table 8-6 (Mission D), OAO Repair Missions Weight/Balance, requires an increase in propellant and tankage of 14 pounds in addition to the capacity of the basic spacecraft.

Table 8-7, Nimbus D to E Refurbishment Mission Weight/Balance, shows the effect of the heaviest payload requirements of all five missions. In all mission specific weights and balance statements, mission specific hardware is required to locate, store, yield, and receive parts, supplies, tools, and fixtures from launch to final disposition on completion of the mission. These weights are identified as tool bin and structure, rack and structure, or as adapter truss. These provisions are estimated on the basis of a constant ratio (15 percent) of the weight of the payload being accommodated.

8.4 RELIABILITY & FAILURE MODES AND EFFECTS

The practicality of using a remote manipulator system for space maintenance functions depends on its overall effectiveness when compared with other approaches such as maintenance by man or complete satellite replacement. This effectiveness depends on manipulator spacecraft cost, weight, and reliability. These three factors can be enhanced by design simplicity, and wherever possible this design approach was used for the remote manipulator spacecraft. Furthermore, most of the hardware has been qualified on previous programs. New designs are required primarily for the video and manipulator subsystems. Redundancy has been limited to those items where improved reliability is essential to mission success and functional redundancy has been used wherever practical.

A preliminary reliability prediction for the remote manipulator spacecraft indicates a mission success probability in excess of 90 percent. This is well above reliability estimates for earth orbiting satellites with missions of 6 months or longer and is primarily due to the shorter 10-day mission for the remote manipulator spacecraft, and low duty-cycle for the video and manipulator subsystems.

A preliminary failure mode and effects analysis (FMEA) was performed and is presented in Table 8-7. The FMEA identifies the most significant failure modes and Table 8-8 contains recommendations for those areas with highest probability of failure.

The mission critical items (i.e., those items in which a single failure would definitely result in mission failure) are as follows:

1. Command Receiver
2. 3-Axis Gyro Package
3. Attitude Control Electronics
4. Propellant Tank
5. Pressurant Tank
6. Ordnance Valve
7. Power Control Unit

Three of these items, the Gyro Package, A/C Electronics and the Power Control Unit have not been qualified on previous programs and require particular attention during detailed design. The Power Control Unit, however, is similar to a previously qualified design.

Table 8-1. Mission Weight Summary (lb)

Mission	Payload Weight		Subtotal	Propellant	Basic Spacecraft	Boosted Weight
	Refurbishment or Replacement Items	Tools, Fixtures, Maintenance Materials and Ballast				
a. Nimbus A to C	165.8	126	291.8		968.1	1259.9
b. DBS	110	189	299		968.1	1267.1
c. OSO	31	222	253		968.1	1221.1
d. OAO	405	89	494	14	968.1	1576.1
e. Nimbus D to E	1090	147	1237	70	968.1	2275.1

Table 8-2. Basic Remote Manipulator Spacecraft Weight and Balance Summary

	Weight (LB)		Sta. (IN)	Moment (IN-LB)
Propulsion Systems				
Rendezvous and Attitude Control		123.3	25.0	3085.0
Attitude Control Reference System		40.4	14.0	566.0
Power Supply				
Power Supply	393.0	408.0	10.0	4080.0
Electrical Power Distribution Module & Harness	15.0			
Communications		124.3		
Antenna (Stowed)	50.0		-6.0	-300.0
Six-foot Erectable Dish with Pedestal, Gimbals, Drives, and Electronics				
Electronics Modules and Omni-Antennas	64.3		14.0	900.0
Command Programmer and Sequencer	10.0		14.0	140.0
Manipulators		104.0		
(2) Arms (Stowed in down position)	58.0		23.0	1337.0
(3) Docking Legs (Stowed position)	18.0		3.0	54.0
(14) Amplifiers	28.0		23.0	643.0
Video		44.1	42.0	1860.0
(2) Cameras and Lights	8.4			
Camera, Light, Tether, Cables	6.2			
Gimbals, Drives, Camera Tray, Parallax/Focus Control	6.0			
(3) Control Units	19.5			
Automatic Light Controls	6.0			
Structure		124.0	12.0	1615.0
TOTAL		968.1	C.G. @ 13.0	12590.0

Table 8-3. (MISSION A) Nimbus A to C Refurbishment Mission Weight/Balance and Tool Volumes

	Weight (LB)					Sta. (IN)	Moment (IN-LB)
Booster Capability With SAC/Nimbus Shroud					1440		
Thor Delta 3L ETR, 500 n.m., 45°							
Boosted Weight					1259.9		
Basic Spacecraft in Orbital Configuration				968.1		13.0	12,590
S/C Payload							
Refurbishment Items			165.8	291.8			
MRIR Recorder Module		9.7				46.0	446.
Radiometer Electronics Module		7.5				46.0	345.
MRIR Recorder/Transmitter Elect. Module		8.2				46.0	377.
Radiometer		7.9				46.0	363.
Radiometer Integral Cover		1.1				46.0	51.
AVCS Recorder		17.4				46.0	803.
MRIR Recorder		16.9				46.0	780.
(2) Solar Paddles		75.5				69.0	5210.
Rack and Structure (15% x 144.2)		21.6				38.0	823.
Tool and Fixtures (Orbital Position)			79.0			62.5	4875.
General Tools and Maintenance Materials*		46.0					
Special Tools and Fixtures		23.0					
C-Clip Dispenser	2.0						
Strut-Truss Bolt Extractor	1.0						
Super-Insulation Shears	2.0						
Tethers (3)	18.0						
Tool Bin and Structure		10.0					
Ballast (to balance off-center refurbished items)			47.0			21.0	990.
						C.G. @ 21.9	27,653

*See Table 8.3-4 for details

Table 8-4. (MISSION B) DBS Refurbishment Mission Weight/Balance and Tool Volumes

	Weight (LB)					Sta. (IN)	Moment (IN-LB)
Booster Capability with Shroud					2050.0		
Titan IIIC, ETR, Synchronous Orbit							
Boosted Weight					1267.1		
Basic Spacecraft in Orbital Configuration				968.1		13.0	12,590
S/C Payload				299.0			
Refurbished Items			110.0			43.0	4,730.
ACS Hydrazine Receiver/		35.0					
Receiver/Exciter		28.0					
Transmitter		33.0					
Rack and Structure (15% x 96)		14.0					
Tools & Fixtures (Orbital Position)			53.0			62.5	3,310.
General Tools and Maintenance Materials		46.0					
Battery-Powered Tool	5.0						
Screwdriver Bits, Sockets, Ball-Ended Allen Wrenches	4.0						
Drive Options (Right Angle & Flexible)	2.0						
Extra Docking Legs (2)	12.0						
Extra, Short, Rigidized Tethers (4)	4.0						
Open-End Box Wrenches (4)	1.0						
Cutter/Crimper/Pliers	2.0						
Tape Dispenser & Tape	2.0						
Spring-Tensioned Cable Retractors, Clamps and Rings (5)	1.0						
Mirrors with Integral Posts and Clamps (4)	2.0						
Clamps (4)	2.0						
Insulation, Retention Grease, Misc.	2.0						
Extra Lights with Integral Clamps & Rheostats	7.0						
Tool Bin and Structure		7.0					
Ballast			136.0			55.0	7,480.
						C.G. @ 22.2	28,110.

Table 8-5. (MISSION C) OSO Repair Mission Weight/Balance and Tool Volumes

	Weight (LB)					Sta. (IN)	Moment (IN-LB)
Booster Capability with SAC/Nimbus Shroud							
Thor Delta 3L, ETR 500, n.m., 45°					1440.0		
Boosted Weight					1221.1		
Basic Spacecraft in Orbital Configuration				968.1		13.0	12,590
S/C Payload				253.0			
Repair Items			31.0			52.0	1,610
New Recorder		13.0					
Gas Re-Supply		14.0					
Nitrogen @ 3000 psi	6.0						
Tankage, fittings, and pneumatic lines	8.0						
Rack and Structure (15% x 27)		4.0					
Tools and Fixtures (Orbital Position)			71.0			62.5	4,440
General Tools and Maintenance Materials*		46.0					
Special Tools and Fixtures		16.0					
Despinner (Figure 3-33)	15.0						
Sail Clamp	1.0						
Tool Bin and Structure		9.0					
Ballast			151.0			52.0	7,870
						C.G. @ 21.7	26,510

*See Table 8.3-4 for details

Table 8-6. (MISSION D) OAO Repair Mission Weight/Balance and Tool Volumes

Item	Weight (LB)					Sta. (IN)	Moment (IN-LB)
Booster Capability with SAC/Nimbus Shroud							
Thor Delta 3L, ETR, 435 n.m., 32^{o}					1650.0		
Boosted Weight				968.1	1576.1	13.0	12,590
Basic Spacecraft in Orbital Configuration							
S/C Payload				508.0			
Repair Items			405.0				
New Battery Unit		175.0				47.0	8,250
New SDHE Unit		48.0				47.0	2,255
New BCSC Unit		50.0				47.0	2,350
Primary and Secondary Gas-Re Supply		79.0				47.0	3,710
Nitrogen @ 3500 psi	32.0						
Tankage, Fittings, and Pneumatic Lines	47.0						
Rack and Structure (15% x 352)		53.0				30.0	1,590
Tools and Fixtures (Stowed Position)			89.0			40.0	3,560
General Tools and Maintenance Materials*		46.0					
Special Tools and Fixtures		31.0					
Deutsch Connector Tongs	4.0						
Deutsch Connector Reset Device	1.0						
Deutsch Connector CIR-CLIP Pliers	1.0						
Super-Insulation Shears	2.0						
Star-Tracker Covers (6)	2.0						
Tethers (4)	24.0						
Tool Bin and Structure		12.0					
Increase in Propellant, Tank and Gas			14.0			24.0	336
						C.G. @ 22.0	34,641

*See Table 8.3-4 for details

Table 8-7. (MISSION E) Nimbus D to E Refurbishment Mission Weight/Balance and Tool Volumes

	Weight (LB)					Sta. (IN)	Moment (IN-LB)
Booster Capability with SAC/Nimbus Shroud							
Thor-Agena D, ETR, 500 n.m. 10° (retrograde)					2610.0		
Boosted Weight					2275.1		
Basic Spacecraft in Orbital Configuration				968.1		13.0	12,590
S/C Payload				1307.0			
Refurbished Items			1090.0				
Nimbus E Sensory Ring		948.0				62.0	58,700
Adapter Truss (15% x 948)		142.0				38.0	5,390
Tools and Fixtures (Stowed Position)			72.0			40.0	2,880
General Tools and Maintenance Materials*		46.0					
Special Tools and Fixtures		17.0					
C-Clip Dispenser	2.0						
Strut-Truss Bolt Extractor	1.0						
Super-Insulation Shears	2.0						
Tethers (2)	12.0						
Tool Bin & Structure		9.0					
Ballast			75.0			-28.0	-2,100
Increase In Propellant Tank, and Gas			70.0			8.0	560
						C.G. @ 34.3	78,020

*See Table 8.3-4 for details

Table 8-8. Failure Modes and Effects Analysis Results

PROPULSION AND ATTITUDE CONTROL SUBSYSTEM

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Mission	Possible Compensating Provisions	Comments
Propellant Tank with Bladder 16 1/2" Dia.	Propellant Storage and Pressurization	Tank Rupture	Loss of Propellant and/or Pressurant	Mission Failure		Qualified on Previous Program
		Bladder Rupture	Mixing of Propellant and Pressurant	Mission Failure		
		Tank Leakage	Loss of Propellant and/or Pressurant	Shortened Mission Based on Leak Rate		
Pressurant Tank 9 1/2" Dia.	Pressurant Storage	Tank Rupture	Loss of Pressurant	Mission Failure		Qualified on Previous Program
		Tank Leakage	Loss of Pressurant	Shortened Mission Based on Leak Rate		
Pressurant Fill Valve	Charge Pressurant	External Leakage	Loss of Pressurant	Shortened Mission Based on Leak Rate		Qualified on Previous Program
Propellant Fill Valve	Charge Propellant	External Leakage	Loss of Propellant	Shortened Mission Based on Leak Rate		Qualified on Previous Program
Ordnance Valve (NC)	Release Propellant from Tank	Failure to Open	No Propellant Flow	Mission Failure	Redundant Valves	Qualified on Previous Program
		External Leakage	Loss of Propellant	Shortened Mission Based on Leak Rate		

Table 8-8. Failure Modes and Effects Analysis Results (Cont'd)

PROPULSION AND ATTITUDE CONTROL SUBSYSTEM Cont'd

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Mission	Possible Compensating Provisions	Comments
Pressure Transducer	Monitor N_2 Pressure	No Output	Loss of N_2 Pressure Data	None	Use of Redundant Transducers	
		Incorrect Output	Incorrect Pressure Data	Could Result in Degraded Mission		Qualified on Previous Program
Temperature Transducer	Monitor Tank Temperature	No Output	Loss of N_2 Temperature Data	None	Use of Redundant Transducers	
		Incorrect Output	Incorrect Temperature Data	Could Result in Degraded Mission		Qualified on Previous Program
Lines and Fittings	Interconnections	External Leakage	Loss of Pressurant and/or Propellant	Mission Degraded to Failed, Depending on Leak Rate		Qualified on Previous Program
		Internal Obstruction	Fouled Solenoid Valves	Mission Degraded to Failed, Depending on Extent		Perform end to end Testing of System
Gyro Package (3 Axis)	Provide Inertial Rate Information	No Output or Incorrect Output	Loss of Stabilization About Failed Axis	Mission Failure	Use of Redundant Gyro or Derived Rate from IR or High Gain Antenna	
IR Horizon Scanner	Provide Roll and Pitch Error Signals	No Output or Incorrect Output	Loss of Accurate Local Vertical	Mission Degraded	Use of Redundant Sensors or Gyro Backup	Mission Failure if IR Failure Occurs in Conjunction with Loss of Data Relay Satellite Tracking. Qualified on Previous Program

Table 8-8. Failure Modes and Effects Analysis Results (Cont'd)

PROPULSION AND ATTITUDE CONTROL SUBSYSTEM Cont'd.

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Mission	Possible Compensating Provisions	Comments
Filter	Filter Propellant	External Leakage	Loss of Propellant	Shortened Mission Based on Leak Rate		Qualified on Previous Program
		Obstruction	Reduced Propellant Flow	None-Increase Burn Time	Redundant Filters	
Solenoid Drivers and Valves	Control Propellant Flow to Thrusters	Failure to Open or Intermittent	Incorrect Flow of Propellant to Thruster	Mission Degraded to Failed, Depending on Ability to Compensate with Other Thrusters.		Not Prevalent Fail Mode Qualified on Previous Program.
		Failure to Close	Continuous Propellant Flow to Thruster	Mission Failure		Series Redundancy is used
Thrusters	Provide Thrust for Rendezvous or Control	Low Thrust	Excessive Propellant Consumption	Mission Degraded to Failed, Based on Ability to Compensate		Qualified on Previous Program
Attitude Control Electronics	Maintain Inertial Reference and Provide Autopilot Function for Spacecraft	No Output or Incorrect Output	Loss of Stabilization	Mission Failure	Use of Redundant Electronics	

Table 8-8. Failure Modes and Evvects Analysis Results (Cont'd)

TELEMETRY, TRACKING, AND COMMAND SUBSYSTEM

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Effect of Failure on: Mission	Possible Compensating Provisions	Comments
High Gain Antenna Including Gimbals, Drive & Electronics	Track Data Relay Satellite for Continuous Contact with Ground Station	Open, Shorted, Failure to Erect Acquire, or Track	Loss of Video and Manipulater Force Feedback	Mission Degraded to Failed Depending on Time Required to Complete Mission	OMNI Antenna used as backup for commands	All items except dish have been qualified on Previous Program
OMNI Directional Antennae	Communication with Ground Station Either Direct or via Data Relay Satellite	Open, Shorted, or Fail to Extend	Loss of Range & Range Rate Data	Mission Failure if occurs prior to Visual Sighting of Target S/C or in Conjunction with Loss of Data Relay Satellite Tracking	Redundant Antenna	Qualified on Previous Program
Transponder	Relay Signal to Stadan Facilities for Computing Range and Range Rate	No Output or Incorrect Output	Loss of Range & Range Rate Data	Mission Failure if occurs prior to Visual Sighting of Target S/C or in Conjunction with Loss of Data Relay Satellite Tracking		Qualified on Previous Program
Multiplexers	Provide Isolation Between Transmitters and Receiver for use of Single Antenna	Loss of Isolation	Possible Damage to Command Receiver	Mission Failure if Receiver is Damaged		Qualified on Previous Program
Command Receiver	Receive and Demodulate Commands	No Output or Output Inadequate to Operate Decoders	Loss of Command Function	Mission Failure Cannot Operate Spacecraft	Redundant Receivers	Qualified on Previous Programs

Table 8-8. Failure Modes and Effects Analysis Results (Cont'd)

TELEMETRY, TRACKING, AND COMMAND SUBSYSTEM Cont'd.

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Mission	Possible Compensating Provisions	Comments
Satellite Command Decoder	Decode Commands, Direct Real Time Commands to Subsystems and Others to Command Programmer & Sequencer	No Output or Incorrect Output	Loss of Command Functions	Mission Degraded to Failed Depending on Criticality of Inoperative Commands	Redundant Decoder	Qualified on Previous Program
Command Programmer and Sequencer	Provide Storage, Timing and Sequencing of Non Real Time Satellite Commands	No Output or Incorrect Output	Loss of Command Functions	Mission Degraded to Failed Depending on Criticality of Inoperative Commands	Redundancy	Qualified on Previous Program
Manipulator/TV Camera Control Decoder	Decode Commands for Manipulator/TV Camera Control	No Output or Incorrect Output	Loss of Command Functions	Mission Degraded to Failed Depending on Criticality of Inoperative Commands	Redundancy	Qualified on Previous Program
TWT Power Amplifiers	Amplify RF Signal	No Output or Low Output	Share Remaining Amplifier	Mission Degraded		Qualified on Previous Program
FM Mod	Modulate TWT Amplifiers	No Output	Share Remaining Amplifier	Mission Degraded	If One Modulator Fails, Share Remaining One.	Qualified on Previous Program
Base Band Multiplexer	Multiplex TV and Force Feedback or Telemetry Signals	No Output, Incorrect Output, or Loss of Channels	Share Remaining Multiplexer	Mission Degraded		Qualified on Previous Program
PCM Commutator (Force Feedback)	Multiplex Force Feedback Signals	No Output, Incorrect Output, or Loss of Channels	Loss of Force Feedback Information	Mission Degraded to Failed Depending on Amount of Data Lost		Qualified on Previous Program
PCM Multicoder (Eng. Telemetry)	Multiplex Telemetry Signals	No Output, Incorrect Output, or Loss of Channels	Loss of Telemetry Data	Mission Degraded to Failed Depending on Amount of Data Lost		Qualified on Previous Program

Table 8-8. Failure Modes and Effects Analysis Results (Cont'd)

VISION AND LIGHTING SUBSYSTEM

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Effect of Failure on: Mission	Possible Compensating Provisions	Comments
Vidicon Cameras with Automatic Light Control (Attached)	Collect Video for Transmission to Ground	No Output	Loss of Video if both Cameras Fail. Loss of Stereo if One Camera Fails	Mission Degraded if Failure Occurs after Docking	Use Close-up Camera if Both Fail	
		Reduced Sensitivity or Resolution	Video Degraded from Failed Camera	Mission Degraded	Use Close-up Camera if Both Fail	
Vidicon Camera with Automatic Light Control (Close-up) and Control Unit	Collect Close-up Video for Transmission to Ground	No Output	Loss of Close-up Viewing	Mission Degraded	Mission can Generally be Completed with Other 2 Cameras	
Vidicon Camera Control Unit	Provide Signals for Operation of Cameras and Process Video Output Data for Transmission to Ground	No Ouput	Loss of Video if Both Control Units Fail. Degraded Operation if One Fails	Mission Failure if Both Control Units Fail, Degraded if One Fails	The 2 Control Units are switchable	
Pan and Tilt Drive	Orient Attached Cameras for Vending	No Drive	Unable to Position Attached Cameras	Mission Degraded	Reposition with Manipulator	
Parallax and Focus Control	Focus Video Image and Adjust Parallax	No Drive	Loss of Direct Focus Capability	Mission Degraded	Use Mechanical Backup Drive by Manipulator	
Lamp and Reflector	Illuminate Work Area	No Output	Operate only with sunlight	Mission Degraded	Replace lamps with Manipulator	

Table 8-8. Failure Modes and Effects Analysis Results (Cont'd)

MANIPULATOR SUBSYSTEM

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Effect of Failure on: Mission	Possible Compensating Provisions	Comments
Manipulator Arm	Perform Maintainance Tasks	No Output of Single Positioning Monitor No Output of End Effector	Limited Freedom of Arm Movement Jaw Cannot Grasp	Mission Degraded. Two Manipulator Arms Available	In Most Cases Manipulator Can Be Positioned to Allow Completion of Task. Make end Effector Removeable and Insert Special Tools	
Tethers	Attach Manipulator to Spacecraft	Unable to Set Mechanical Clamp or Frozen Joint	Cannot Clamp Tether to Target Spacecraft	Mission Degraded	Use Remaining 2 Tethers and Manipulator	
Manipulator Servo Amplifiers	Drive Manipulator Servo Motors	No Output of Single Servo Amplifier	Limited Freedom of Arm Movement	Mission Degraded	In most cases Manipulator can be Repositioned to allow Completion of Task	

Table 8-8. Failure Modes and Evvects Analysis Results (Cont'd)

ELECTRICAL POWER SUBSYSTEM

Name	Function	Assumed Failure	Effect of Failure on: Subsystem	Effect of Failure on: Mission	Possible Compensating Provisions	Comments
Batteries (3)	Source of Electrical Power	Open or Shorted Cell(s)	Available Energy Reduced	Mission can Generally be Completed with 2 Of 3 Batteries	Existing Redundancy is Adequate	Batteries have been Qualified on Previous Program
Power Control Unit	Switching, Monitoring & Distribution of Electrical Power	Open Contacts in Ground/Spacecraft Relay Loss of Voltage or current readings	Loss of Electrical Power Loss of Power Consumption Data	Mission Failure Could Result in Degraded Mission	Use of Redundant Relays in Fail Safe Configuration. Estimate Power Consumption	Similar with Previously Qualified

SECTION 9
SUBSYSTEM DESIGN

The previous sections of this report established the manipulator and spacecraft design requirements for fulfilling the various maintenance missions performed upon the target vehicles. The target vehicles represent broad cross-section of spacecraft, orbits and maintenance activities and therefore a large set of requirements have evolved during the study. This section describes the nine spacecraft subsystems resulting from the mission and spacecraft system analysis.

The nine subsystems:

1. Vision and Lighting
2. Manipulators
3. Communications
4. Propulsion and Attitude Control
5. Attitude Reference
6. Electrical Power
7. Spacecraft Structure
8. Thermal Control
9. Ground Control Station.

Each of these subsystems makes as much use as feasible of existing and presently qualified subsystems and components. When design requirements make it impossible or impractical to use presently qualified components, every attempt has been made to use components that are presently undergoing qualification testing or have a high probability of being qualified in the next few years. As a result:

1. All components of the telemetry tracking and command subsystem are, or soon will be, qualified except the folding dish and erection mechanism portion of the high gain antenna.

2. Efforts are presently taking place with folding antenna designs that are many times larger (30 feet compared to 6 feet) and more accurate.

3. All of the rendezvous and attitude control subsystem components are, or soon will be, qualified except for the 3-axis strap-down gyro package and the attitude control electronics package. It was decided to design this system so that it could take advantage of continuing advances in gyros, thus reducing drift rates and power requirements while offering ever-increasing reliability. The attitude control electronics package is one of the few totally new designs but it too will be composed of many flight proved and qualified components.

4. The manipulators, servo amplifiers and docking tethers are not presently qualified for space use and require development and qualification.

5. The spacecraft structure will be a new design but utilizes a simple non-monocoque construction.

6. Video systems have been qualified and flown but none suits the mission requirements. Whether to modify existing systems qualified for aircraft or to design new is a tradeoff not made by this study. The parallax and focus drive along with the pan and till mechanism will be new design also.

7. The electrical power subsystem will be comprised of qualified silver oxide-zinc batteries and the power control unit will be almost identical to a device previously qualified.

The manipulator spacecraft has a weight of 968.1 lb without tools or refurbishment and repair items. Maximum weight for any of the study configurations is 2275.1 lb on the Nimbus D to E mission. To allow launch into low-earth orbit on a long tank (Thor/Delta), the diameter was held to 56 inches and the profile was kept within the SAC/Nimbus Shroud limits. Synchronous altitude missions are assumed to be launched onboard a Titan III-C. The DBS refurbishment mission is shown with a Titan size shroud.

9.1 VEHICLE CONFIGURATION

A major goal of the remote manipulator spacecraft program has been to design a system offering low cost for a single mission. To achieve this, the same design philosophy has been followed as has been suggested for vehicles to be repaired or refurbished on-orbit. In this manner, the remote manipulator spacecraft stands ready to be modified or updated as the mission requires or as technology permits. On-orbit repair of the manipulator spacecraft is not planned except for the limited action of the manipulators themselves. Many of the satellite maintenance design recommendations (Section 7) are applicable because they are good practice in general and facilitate spacecraft development, assembly, inspection, test, and adaptation to the selected maintenance missions. The following paragraphs highlight the design philosophy:

a. Safety - Protective tunnels and runs for critical harnesses, hydraulic lines and mission critical components.

b. Access

1. Single-motion, quick-disconnect, accessible service connections, fasteners, hatches, access panels, tethers and equipment racks that react on themselves rather than on the actuator. Push-pull motions are preferred to circular or lateral motions and single release latches are preferred to multiple.

2. Integral disassembly and assembly fixtures such as pivoted racks, equipment drawers, and hinged access panels.

3. Manipulator accessible shutoff and bypass valves.

4. Tapered and rounded openings in frames, structures and hardware to prevent hangups of manipulators, tools or equipment during maintenance.

5. Coatings and materials to prevent vacuum welding of such equipment as storage clamps.

6. Extra holes and slots in skin panels and compartment walls for additional harness and plumbing runs.

7. Minimization of:

   (a) Overlying cables and harnesses

   (b) Taped-in-place super-insulation

   (c) Antennas and antenna mounting brackets without hardpoints

   (d) Overlying connector brackets, support structure and protective covers.

8. Avoidance of components that can entangle, or foul the target vehicle by wrapping, wedging or jamming such as:

   (a) Lanyards

   (b) Umbilicals

   (c) Unrigidized tethers

9. Positive locating and locking devices for the moving components such as manipulators, cameras and antennas.

10. Minimization of sequential assembly operations for high failure-rate parts and for parts like skin panels.

11. Unitized, rigid structural systems for components that require fine alignment such as attitude control thrusters. Other alignment provisions include:

    (a) Alignment surfaces, pins, indices

    (b) Two perpendicular surfaces for three-axis alignment

    (c) One surface for two-axis alignment.

12. Decrease in packaging density of components to facilitate access during maintenance.

c. Diagnosis

1. Diagnostic indicators on subsystems along with inspection windows and removeable covers.

2. Equipments that are not selectively mated, matched or balanced to one another and that can be checked, operated, and adjusted separately.

3. Readily accessable diagnostic test points.

4. Telemetry for fault isolation down to module level.

d. Service Aids

1. Identification and assembly aids such as color codes, markings, large numbers, and irreversible assembly keys and shapes on modules, handling pads, access doors, adjacent structure, assembly points, electrical cables and propellant lines.

2. Diffuse surfaces to reduce problems of observation with orbital lighting conditions.

3. Adhesive areas, VELCRO, magnets, clips, etc., inside and outside the spacecraft to temporarily retain tools and small parts during maintenance operations.

4. Instruction, flow-charts, warning placards (e.g., high voltage and extreme temperature warning signs) on the spacecraft.

e. Replacement Aids

1. Standardized spacecraft connectors and fasteners to minimize tool variety.

2. Fasteners, connectors (mechanical and electrical), and locking devices that require a minimum of actuation force or actuation torque.

3. Eliminate all assembly operations requiring probing or feeling.

4. Large wire bundles should be avoided to maintain flexibility and access to connectors.

5. Wires and harness coded by numbers, colors, patterns and other markings.

6. Provisions for manual operation, deployment, or erection of equipments designed to be automatically actuated.

9.1.1 CONFIGURATION ANALYSIS

The remote manipulator spacecraft carries a great variety of repair and replacement equipment at launch. Some of the equipment is nearly as large as the manipulator spacecraft itself (e.g., the two 3-by-8 ft Nimbus paddles that must be launched in their regular attitude). Some of the equipment is nearly as heavy as the spacecraft itself; the 948-lb Nimbus E Sensory Ring must be launched in its regular horizontal attitude. On the other hand, the OSO repair items weigh only 31 lb and take up less than 2 cubic feet. It is this variation in size and weight of maintenance components that has offered one of the more significant design challenges. However, the spacecraft has been designed to successfully cope with this problem and within other difficult limitations such as booster and shroud capacities, and the low-cost, single-mission criterion of the study.

The spacecraft configuration would be different, if it was not required to carry the repair and replacement equipment at launch. Separate logistic launches, multiple remote manipulator spacecraft, master manipulators in manned space stations and manipulator tenders are examples of other modes of on-orbit maintenance missions that would allow other spacecraft configurations.

The configuration of the remote manipulator spacecraft with its various payloads for each of the five maintenance missions is established largely by the following criteria and design decisions:

9.1.1.1 Commonality of the Subsystems

The spacecraft structure, tool crib, and housekeeping subsystems (except for the mounting bracket of the rendezvous thrusters) are common for all five configuration to minimize cost, mission conversion errors, and launch delays. Available booster and shroud capacity allows the commonality of the subsystems on the remote manipulator spacecraft. For example, the resupply gas subsystems are the same for the OAO and OSO maintenance missions although the dry weight of the 3000 psi OSO nitrogen resupply system is only 8 lb versus 47 lb for the 3500 psi OAO system.

9.1.1.2 Booster Capacity and Shroud Volume

The booster capacities and diameters and the shroud payload envelopes are major design criteria. Based on these criteria, the Nimbus A to C refurbishment mission (Figure 9.1-1) established the plan-view baseline configuration because:

1. The length and required launch orientation of the Nimbus solar paddles necessitates that both the paddles and manipulators be adjacent to one another atop a "pancake shaped" remote manipulator spacecraft within the Nimbus shroud.

2. The folded configuration of the paddles requires that the manipulators and video camera yoke be off-center toward the front of the remote manipulator spacecraft for best reach and vision.

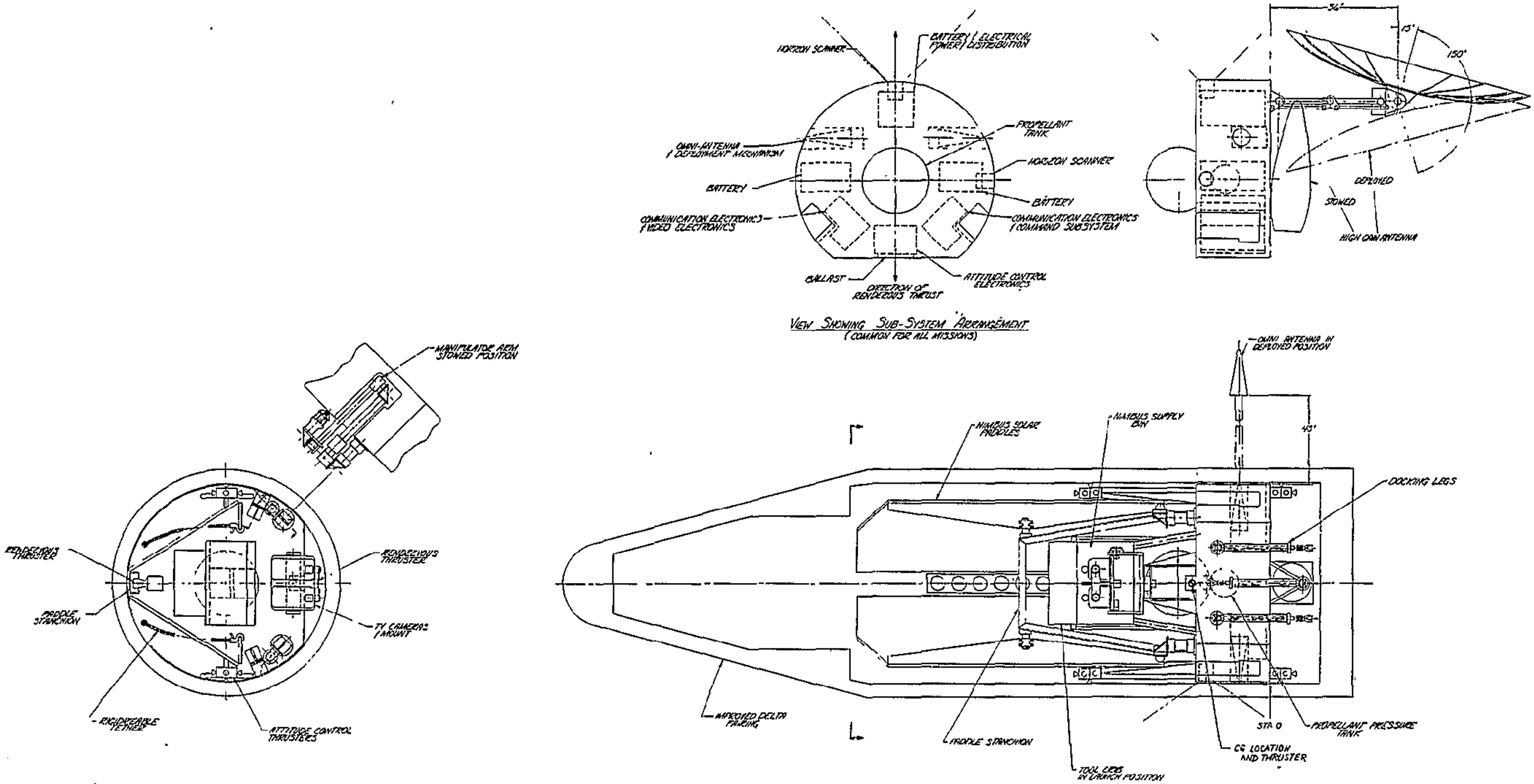

Figure 9.1-1. Nimbus A-C Mission Launch Configuration

The paddles do not support each other on top of the paddle release mechanism as they do on Nimbus launches. Rather, two new paddle release mechanisms and a separate support stanchion are used in order that the rendezvous thrusters can exhaust through an aperture in the stanchion. Paddle hinge blocks prevent the paddles from striking the attitude control thrusters when the paddle release mechanisms are actuated. Deflectors are attached to the hinge-blocks to protect the paddles from two of the attitude control thrusters.

The Nimbus A to C mission configuration is also suitable for the Nimbus D to E refurbishment mission (Figure 9.1-2) even though the Nimbus E Sensory Ring is large, heavy and must be at its regular adapter interface. In fact, the sensory ring is the heaviest replacement equipment and it therefore establishes the elevation baseline because:

1. The travel of the longitudinal CG of the loaded remote manipulator spacecraft must be kept within the bounds of the allowed thrust and mass offsets of the rendezvous thrusters.
2. The nominal location of the longitudinal CG must be at the center of the planar arrangement of the attitude control thrusters when the CG of the sensory ring is located as low as possible.

The baseline vehicle is illustrated in Figure 9.1-3 and 9.1-4. The four clusters of attitude control thrusters (six thrusters in each) can be seen arranged on a structure at the mid-plane of the spacecraft so that their exhausts do not impinge on the satellites being maintained. The CG of the hydrazine tank occupying the central portion of the spacecraft deck must be high enough so that the mass offset of the hydrazine, laterally accelerated at approximately 1/40th G, is less than 5 inches. The video cameras must be located near the middle of the manipulator reach capability, and they in turn must easily reach the tool bin and supply bins.

The baseline configuration is also quite suitable for the OAO, OSO, and DBS missions (Figures 9.1-5, 9.1-6 and 9.1-7) which have smaller and lighter payloads.

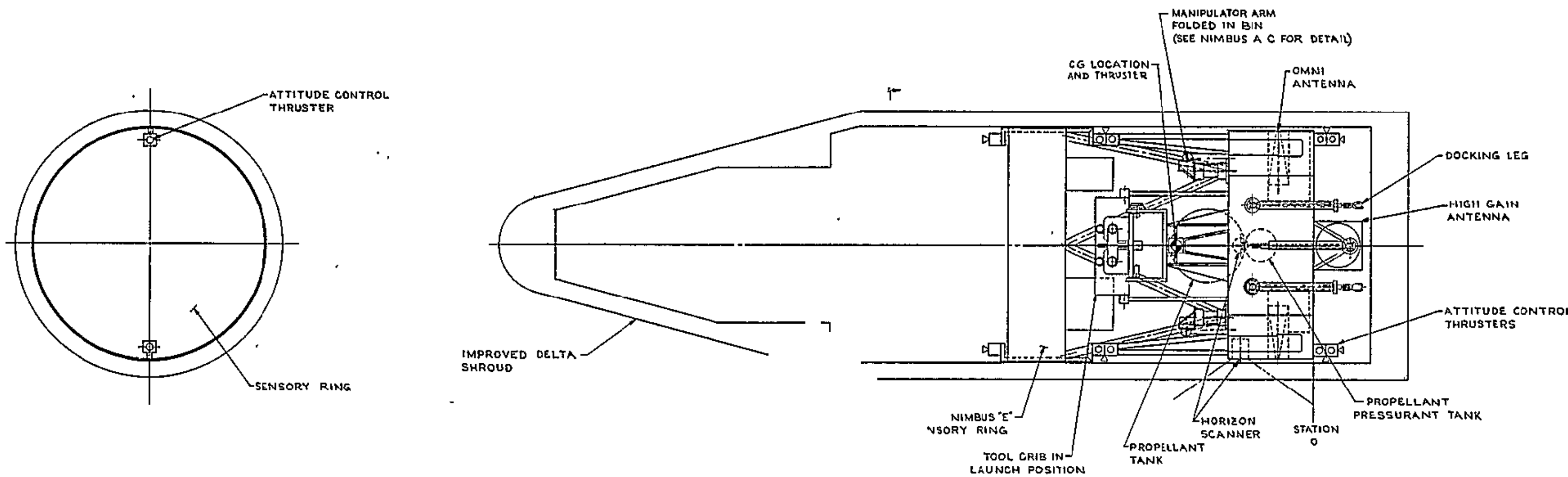

Figure 9.1-2. Launch Configuration-Nimbus D-E Mission

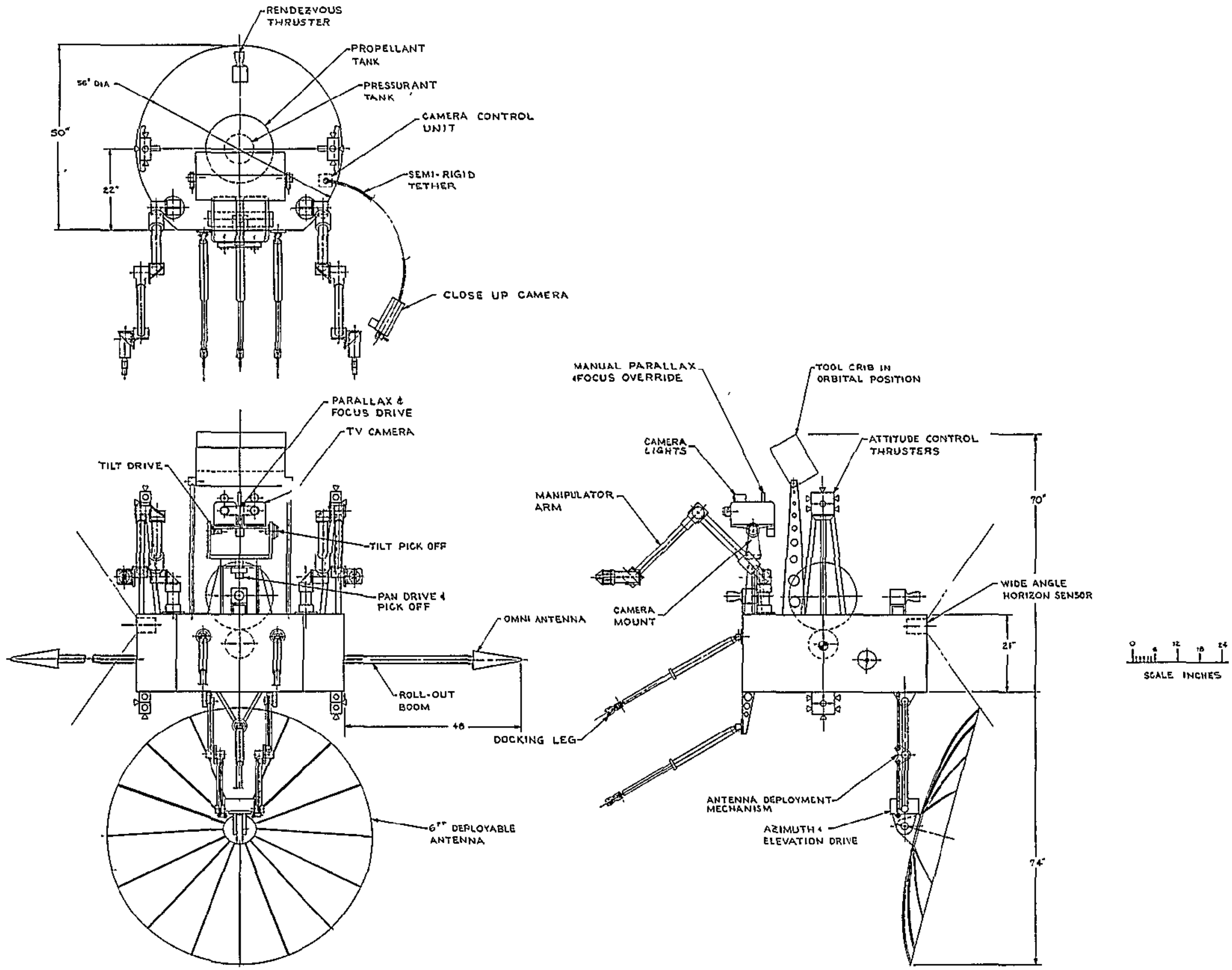

Figure 9.1-3. Basic Manipulator Spacecraft Orbital Configuration

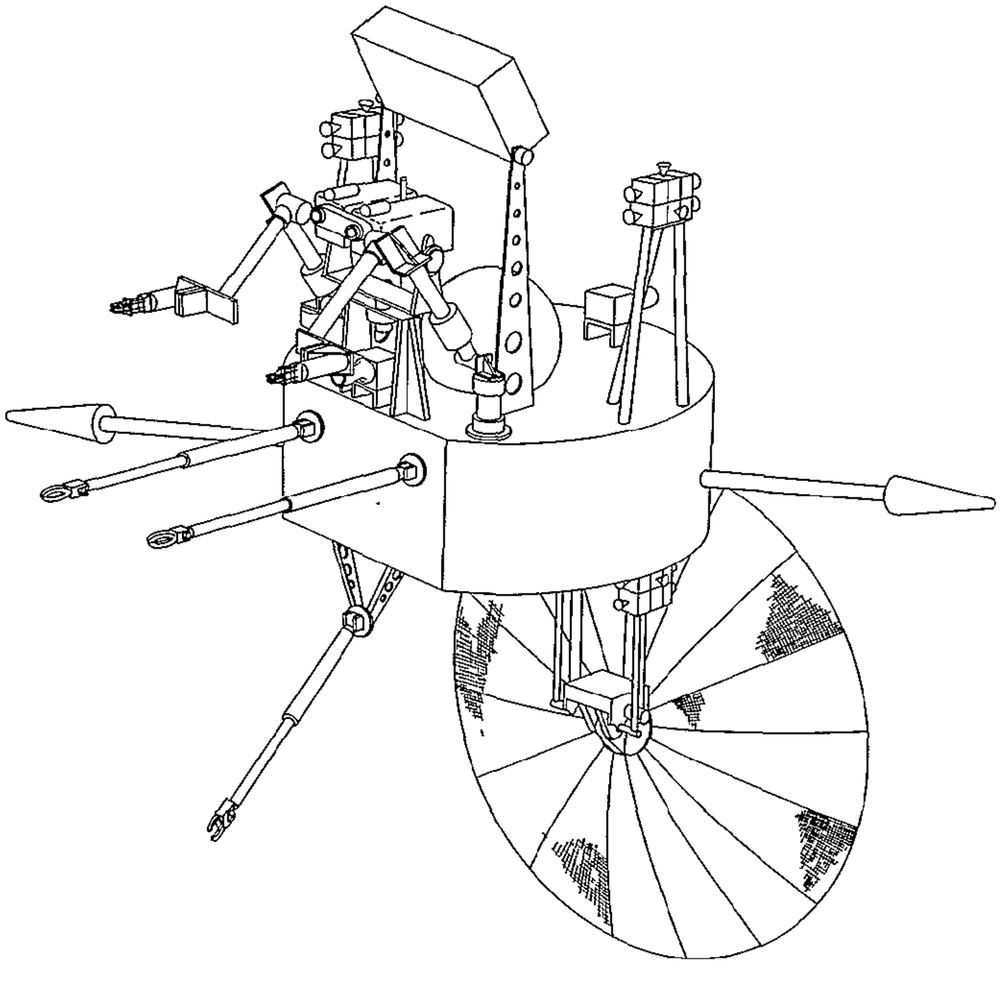

Figure 9.1-4. Pictorial View of Manipulator Spacecraft

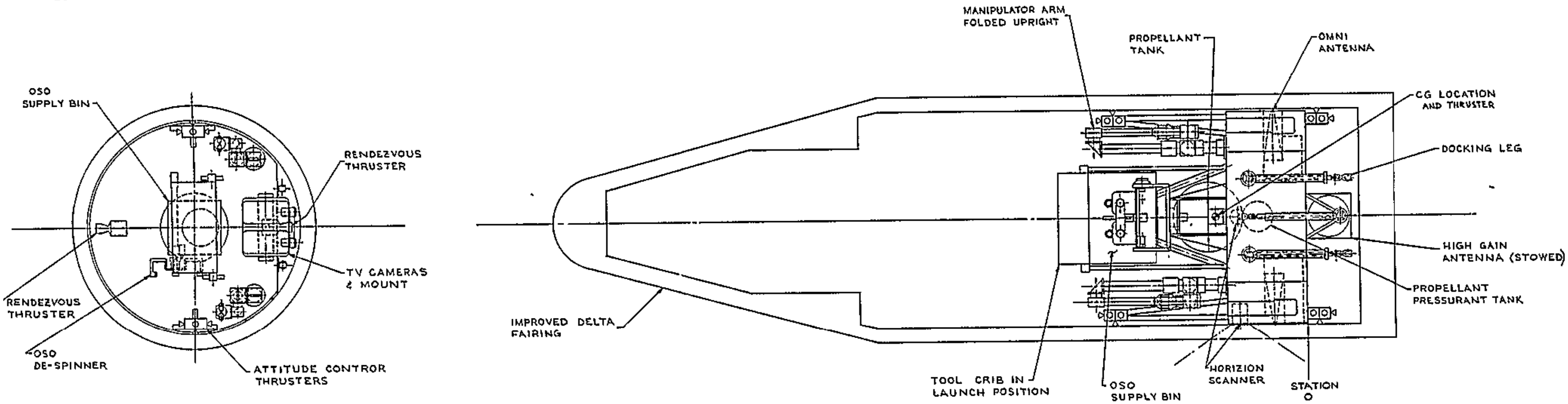

Figure 9.1-5. OSO Repair Configuration

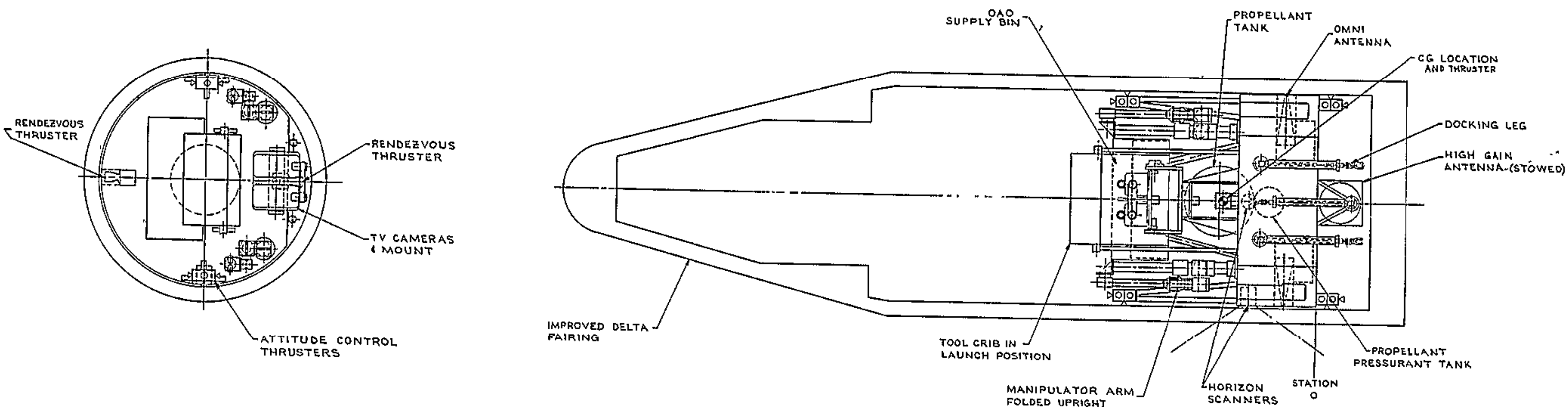

Figure 9.1-6. OAO Repair Configuration

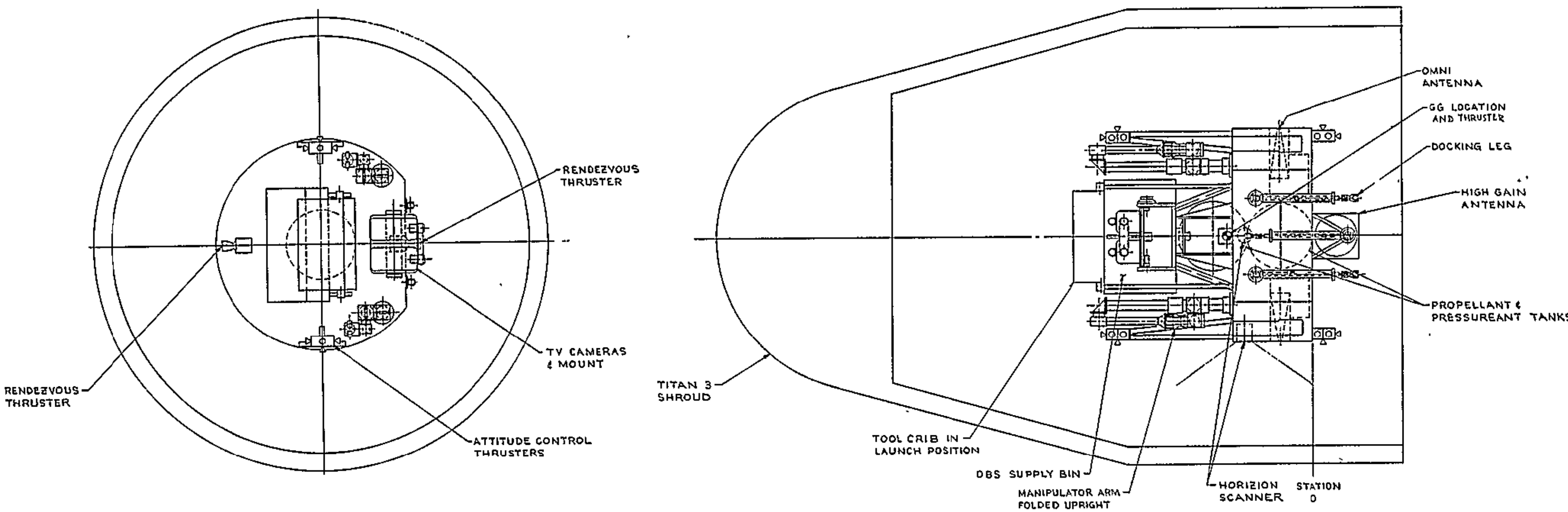

Figure 9.1-7. DBS Repair Configuration

9.1.1.3 Rendezvous and Attitude Control Thruster Locations

To limit the rendezvous thrust offset, the two rendezvous nozzles are repositioned for each mission to pass through the CG of the spacecraft. The CG of the hydrazine propellant tank is located at the middle of CG travel for all missions analyzed. The attitude control thrusters are located so the resultant thrust passes through the center of the hydrazine tank. The resulting thrust CG offset can be overcome by the attitude control thrusters operated in a pulsed mode.

Table 9.1-1 lists an evaluation of locating the opposed rendezvous thrusters either along the longitudinal axis of the remote manipulator spacecraft or perpendicular to that axis. Accordingly, the rendezvous thrusters are positioned at the longitudinal CG of the spacecraft, and the hydrazine tank is mounted near the middle of the CG range.

9.1.1.4 Supply Bin and Tool Crib Locations

The supply bin and tool crib are located near the manipulators and on an elevated deck above the hydrazine tank. Because the weight and volume of the special tools for each of the five missions is a fraction of the weight and volume of the general tools and maintenance materials for all five missions, all are packaged in a tool crib common to the remote manipulator spacecraft for all five maintenance missions. During launch and de-orbit this crib is mounted on top of the supply bin which is changed according to the supply requirements of each of the five maintenance missions. The tool crib can be pivoted so that the operator can easily see and reach the tools. As a result, all of the housekeeping subsystems of the spacecraft such as the attitude control, power supply, communications, and video subsystems are separate from the maintenance equipment and are mounted within the spacecraft where they are well protected thermally and mechanically.

9.1.2 SUBSYSTEM EFFECTS UPON CONFIGURATION

In addition to the weight, volume, CG and location requirements, the following subsystem requirements help establish the overall spacecraft configuration:

Table 9.1-1. Location of Rendezvous Propulsion Tank and Thrusters

Location	*Advantages		*Disadvantages	
Thrusters on Longitudinal Axis	+5	Rendezvous and de-orbit thrust is always thru the rendezvous and attitude control propellant CG, but simultaneous attitude control thrusts are required any way when the high-gain antenna is scanning.	-8	A tractor thruster fouls the remote manipulator spacecraft.
	+1	Thrust is in the same direction as the booster thrust.	-2	Push thruster requires a tall retractable stanchion or a stiffer and stronger attitude control thruster structure.
	+2	Regular Nimbus paddle release/erection mechanism can be used.	-3	The high-gain antenna must be stored considerably off-center.
	+1	Thrust is along customary axis of lowest mass moment of inertia.		
Thrusters on a Lateral Axise	+8	Thrusters are mounted near tanks resulting in better modularization of the subsystem.	-5	The location of the thrusters is not fixed, thereby increasing qualification costs.
	+3	Thrusters are installed in protected locations.		
	+5	The replacement equipment can be stored in more positions.		
	+5	The thrust is perpendicular to and centered on the nominal working position of the manipulators and gimballed video cameras. However, manipulator grasping does not occur during rendezvous.		

*The numbers denote a subjective evaluation of the advantages and disadvantages.

9.1.2.1 Rendezvous Propulsion Subsystem

The thrusters are mounted near the front and rear of the spacecraft so that their plume does not impinge on the spacecraft (particularly the video cameras).

9.1.2.2 Structural Subsystem

The cylindrical base structure of the remote manipulator spacecraft is comprised largely of ribs emanating from an internal, central cylinder and attached to the external cylinder. The external cylinder, along with the bottom skins, thermally and mechanically protect the internal housekeeping equipment. This non-monocoque structure construction has been selected for several reasons:

1. Almost all of the loads imposed by the spacecraft subsystems on the base structure are concentrated loads rather than uniformly distributed loads (e.g., the loads of the manipulator mounting fittings, docking leg mounting fittings, batteries, housekeeping subsystem modules, payload trusses, high-gain antenna, omni-antennas, paddle stanchions, rendezvous thruster brackets, support structure for the Nimbus paddles and sensory ring, and the video camera yoke). The few remaining spacecraft subsystems load the internal central cylinder uniformly, including the hydrzine tank, supply bin support structure, and the tool crib atop the supply bin. The internal cylinder can accommodate the larger pressurant tank required for the Nimbus D to E mission.

2. The ribs provide many, versatile mounting surfaces to control the spacecraft and provide for spacecraft development and growth.

3. The ribs function as heat-sinks and heat distribution aids.

4. The ribbed construction allows the outer surface of the cylindrical base to be scalloped for storage of the manipulators, and designed for thermal shutters and access doors to the housekeeping subsystem.

5. The ribs form compartments that isolate the various housekeeping subsystems against environmental or emergency problems.

6. The ribbed construction transfers the launch loads well between the booster adapter and all of the spacecraft subsystems.

7. The ribbed construction stiffens the spacecraft well both flexurally and torsionally without requiring structural doors or thick skins on the upper and lower surfaces of the cylindrical base.

9.1.2.3 IR Sensors

One of the sensors is mounted high on the aft surface where it can see past the scanning high-gain antenna. The other sensor is mounted 90 degrees away on the side of the spacecraft to look laterally past one of omni-antennas. Electronic blanking may be required.

9.1.2.4 Power Supply

The batteries and other subsystems are mounted in separate groups so that all parts of a subsystem are not subjected to the same spacecraft environment. Furthermore, the number of batteries can be varied according to the duration of on-station maintenance.

9.1.2.5 Communications

The high-gain antenna is mounted at the bottom rear of the spacecraft to achieve a 150 degree included angle of the scanning cone and to offset the weight of the forward manipulators. The antenna is erected by a spring at the middle joint of the erection structure.

9.1.2.6 Manipulator Subsystem

The edges and corners on both the manipulators and the remote manipulator spacecraft are rounded or cogered with resilient material to minimize impact damage to target satellites. Protrusions and gaps that could entrap or jam the manipulators and other equipments are avoided.

9.1.2.7 Tool-Crib Subsystem

The centrally-mounted tool crib that contains both the general purpose tools and materials and the special tools and fixtures for the five missions is mounted above the supply bins.

Specific retention devices for the tools are preferred to a closed-door retention method and some recommendations include:

1. Snap-on/off devices that also attach the tools bits to the power tools
2. Toggle-action clamps
3. Hook and pile
4. Wedging surfaces with compliant liners
5. Sticky surfaces and containers of sticky storage material
6. Spring-clips and spring-straps
7. Magnetization.

For the purpose of commonality all general and special tools are combined into one tool crib and carried on all five missions except the OSO despinner which is carried only on the OSO maintenance mission. Two semi-rigid tethers, stored in the tool bin, are required on each mission to hold a variety of items such as light diffusers, skin panels or solar arrays. Other tether requirements include:

1. The OAO mission requires two additional tethers to hold, at various times, the BCSC unit, two paddles, SDHE unit, new battery unit, and several skin panels.
2. The Nimbus A to C mission requires two additional rigidizeable tethers to hold the paddles during replacement. One tether is required to hold the MRIR unit while it is tightened and then to hold the Nimbus A Sensory Ring when the recorders while it is tightened.
3. The Nimbus D to E mission requires one additional tether to hold the Nimbus D and E Sensory Rings during replacement. The Nimbus E Sensory Ring can be launched with a tether attached.

Two types of tethers are needed. One type is rigidizeable to a fully stiff condition, and the other type is always semi-rigid so it can be positioned by the manipulators.

9.1.2.8 Supply Bin Subsystem

A supply bin for each of the five missions is centrally mounted atop the hydrazine tank. The equipment in both the supply bin and the tool crib is off-center to help center the lateral CG position over the booster thrust axis. Proper launch orientation for each item of replacement equipment is provided. The supply bins are near the manipulators, and some items in them are inclined upward so that the manipulators can easily see and reach the equipment in them. Both the replacement equipment and the old equipment are stored in the supply bin. Therefore, only the old Nimbus A paddles are tethered to the spacecraft for the de-orbit phase. Some features of the supply-bin are:

1. Toggle-action equipment clamps
2. Equipment drawers, racks, guides, and lead-in devices
3. Distinctive markings
4. Doors to thermally and optically protect the equipment.

9.2 VISION AND LIGHTING

To achieve the overall mission objectives, the operator of the manipulator subsystem must be able to observe the area and components that are being repaired or refurbished. Figure 9.2-1 shows the video subsystem block diagram. The selected video components consist of three vidicon cameras; two cameras will be attached to the structure through a yoke arrangement allowing the cameras to pan and tilt. They will be positioned to provide stereo vision to the operator over a range from 1 to 100 feet. The third camera is mounted on a flexible tether and can be positioned for close-up viewing. Each camera will be provided with an illuminating device to provide the light required under shadow conditions. The vidicon cameras will be equipped with an automatic light control mechanism to automatically maintain a constant average illumination on the photocathode of the camera as the scene brightness varies. The Vision and Lighting Subsystem interfaces with both the telemetry downlink and the command uplink. Commands will provide focus and parallax, illumination, power, and pan and tilt control. The telemetry downlink will be used for the video output.

9.2.1 SUBSYSTEM REQUIREMENTS

Subsystem design must satisfy the following requirements:

1. Compatible with orbital altitudes out to synchronous (19,323 nm).
2. Resolution of each camera to be 0.04 inch from an operational distance of about 2 feet.
3. Field-of-view:
   a. Right and left cameras: 10 to 60°
   b. Detachable camera: 25°
4. Illumination to provide acceptable average lighting in a varying light environment.
5. Stereo video from 0 to 400 ft range.
6. Pan 360° and tilt $\pm 75^{\circ}$ from horizontal.

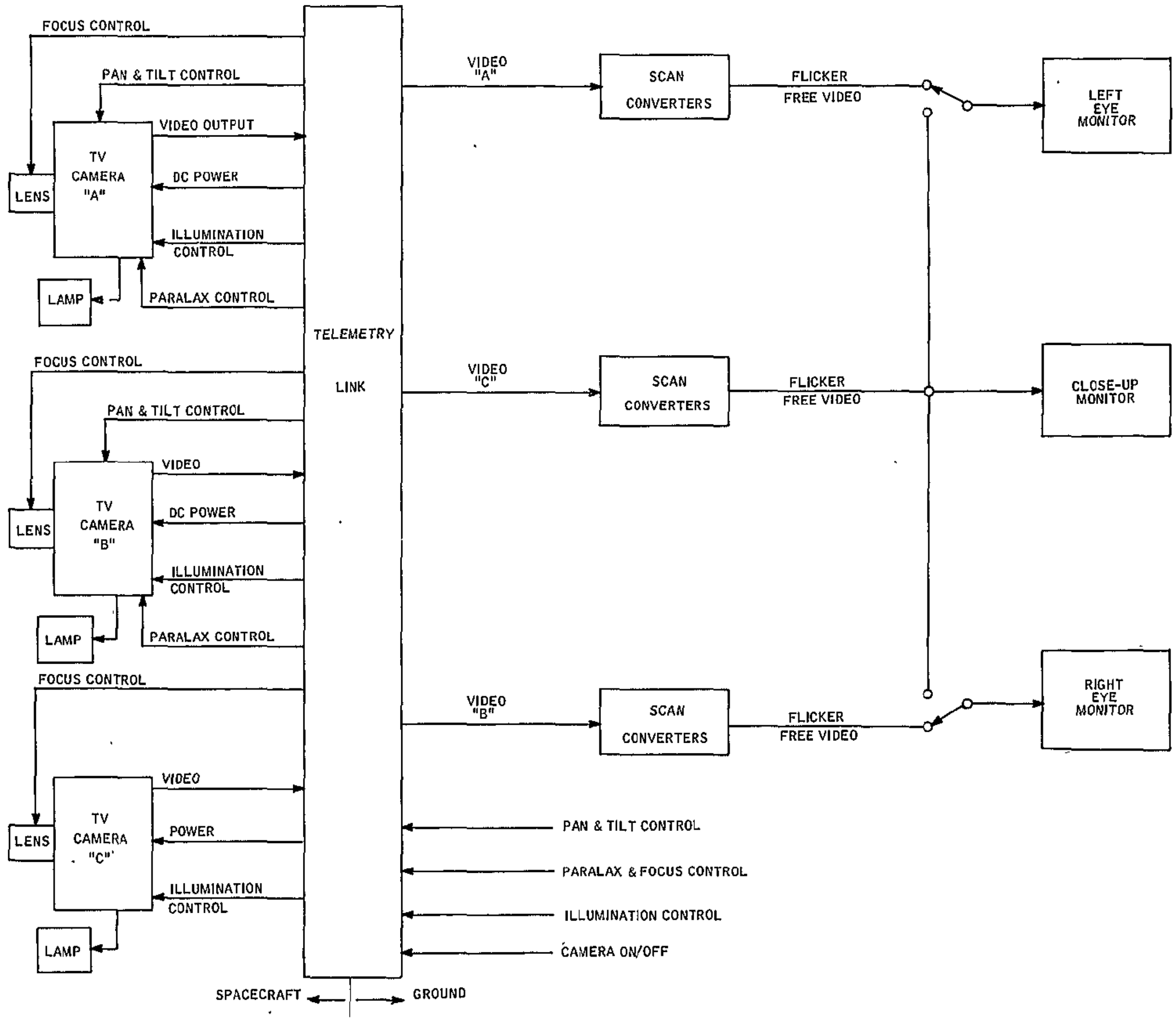

Figure 9.2-1. Video and Illumination Subsystem Schematic

9.2.2 DESIGN APPROACH FOR THE VIDEO PORTION

It has been estimated that the smallest object that must be discerned in the field-of-view will be the small holes (0.040 inches in diameter) in the ends of small snap rings. As an initial estimate, assume that it is necessary to make these observations at a distance of 24 inches. (It is shown in the following discussion that this requirement will be traded off for other system parameters.) Assuming a 25 degree field-of-view, the entire video system can be specified and appropriate tradeoffs can be made in a meaningful manner, keeping in mind that the optical parameters that have to be considered are: (1) minimum object size 0.040 inch, (2) viewing distance 24 inches, and (3) field-of-view 25 degrees. The relationship between subtended angle and distance for objects that are 0.04 inch in diameter is shown graphically in Figure 9.2-2.

The optical resolution that is required for this system can be determined by considering the angle subtended by the object at a specific viewing distance. In this case, the angle 5.7 arc-minutes (arc tan (0.04/24). The optical resolution (R) of the system will be the subtended angle divided by the field-of-view:

$$R = \frac{5.7}{(25)\ (60)} = 0.0038$$

The optical resolution is related in turn to the TV resolution by a factor of two. That is, two TV lines are required to distinguish on optical resolution bit. Consequently, where n is the TV resolution, then $n = \frac{2}{R}$. Let N equal the number of TV lines per frame that are required to produce a TV resolution of n. Then, $N = \frac{n}{(k)\ (k_V)}$ where k is the Kel factor equal to 0.707 and k_V is the vertical duty cycle. From this it is seen that:

$$N = \frac{2}{(k)\ (k_V)\ R}$$

and:

$$N = \frac{2\ (FOV)}{(k)\ (k_V)\ A}$$

where A is the previously computed subtended angle.

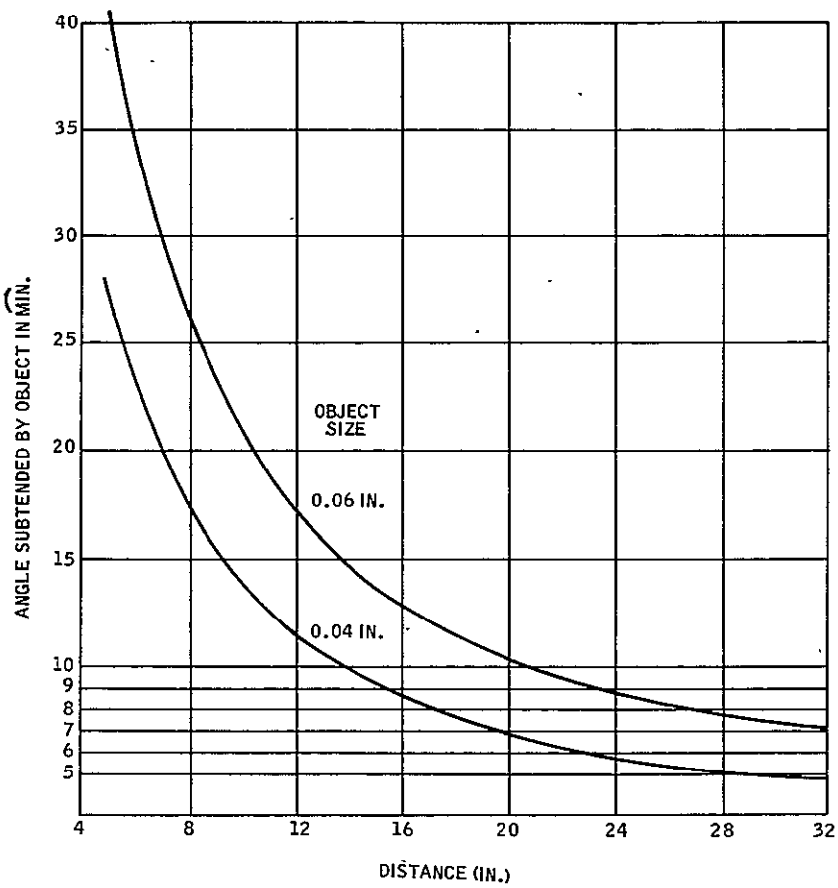

Figure 9.2-2. Subtended Object Angle Versus Distance

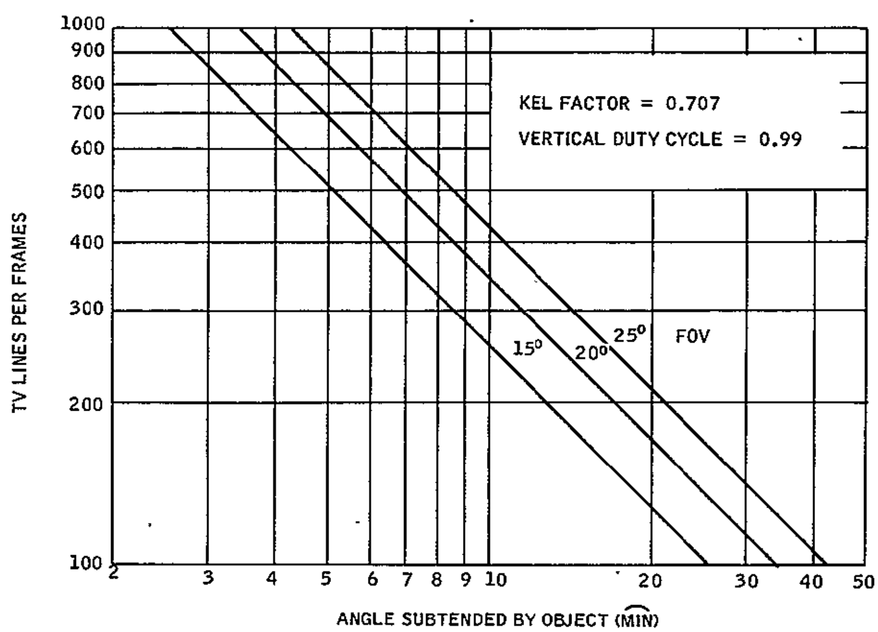

Figure 9.2-3. TVL per Frame versus Angular Resolution

Under normal broadcast TV parameters, the horizontal and vertical duty cycles are specified to be approximately 0.81 and 0.93, respectively. However, since the system under consideration must be a slow scan system to meet the telemetry band width restrictions, it is necessary to redefine these parameters to more reasonable numbers. As a first attempt, let us consider the standard broadcast duty cycles as the parameters. Then:

$$N = \frac{2}{(0.707)\,(0.93)\,(0.0038)} = 803 \text{ TVL/frame}$$

If we assume a slow scan system operating at 10 frames/second, or 1/10 sec/frame, the total time for one horizontal line is:

$$t_H = \frac{(1/10)}{803} = 125 \text{ microsec}$$

From this it is seen that if the horizontal duty cycle were to remain the same as the broadcast value of 0.81, the flyback time would be 10 microsec (this is a reasonable amount of time for the horizontal circuits). However, if the vertical duty cycle were to remain at 0.93, the vertical flyback time would be (0.07) (0.1) = 7000 microseconds. This is an extraordinary amount of time; let us hypothesize a system where the vertical flyback time is chosen to be equal to the time of one horizontal line. The vertical duty cycle would then be equal to $k_v = (1 - \frac{0.000125}{0.1}) = 99.9\%$. By changing the value of k_v from 0.93 to 99.9, the number of TV lines per frame may be re-evaluated to be 745. Although circuitry has been developed that can achieve this value of flyback time for a slow scan TV, it has been found that it is accomplished at the expense of additional circuits and a slight loss of sweep linearity. It is estimated that the optimum compromise for this parameter would be to set the value of k_v at 99 percent; this would have the adverse affect of increasing the total number of horizontal lines to 750 instead of 745.

Since the system considered in this report has a 1:1 interlace, it is understood that the value for k_v will be rounded off to be an integral number of horizontal lines. In this manner, all frames start at the same point in the picture. Figure 9.2-3 shows the relationship between the various parameters that have been discussed.

In order to have the same optical resolution in the vertical and horizontal directions, the video bandwidth must be considered. This relationship can be determined from the fact that if there are n TV lines resolved vertically, there must be (H/V) n TV lines resolved horizontally, where (H/V) is the aspect ratio. The number of cycles/line will have to be 1/2(H/V)n. The video bandwidth (B), is the number of cycles/line divided by the time for one horizontal line. The time per line is equal to $k_h/(N)$ (f), where k_h is the horizontal duty cycle and f is the number of frames per second. Then:

$$B = [1/2(H/V)n]/[k_h/(N)\ (f)] = 1/2\ (H/V)k\ (k_v/k_h)\ N^2\ f$$

Figure 9.2-4 shows the relationship between bandwidth and the number of TV lines/frame. At 10 frames/sec, it is seen that a bandwidth of about 2.1 MHz is required. Tradeoffs between bandwidth requirements and transmitter power indicate that the video bandwidth requirements should be limited to 1.5 MHz to maintain the gain margin in the telemetry link. This restriction necessitates a re-evaluation of the video system parameters. Figures 9.2-2 through 9.2-4 can be used to make the various compromised required to achieve the resolution and bandwidth.

For instance, if a bandwidth limiation of 1.5 MHz is assumed, Figure 9.2-4 indicates that 630 TV lines can be supported at 10 frames/sec and Figure 9.2-3 indicates that this is equivalent to 6.8 minutes of arc within a 25 degree field-of-view and Figure 9.2-2 indicates that this is equivalent to viewing an object diameter of 0.040 inch at a distance of 20.2 inches.

However, a 630 TV line system requires special monitors at the ground station. It must be emphasized that it is more convenient and less expensive to use "standard" equipment for this aspect of the problem than to redesign and modify equipment for a "special situation." With this in mind, Figures 9.2-2 through 9.2-4 are "worked" to achieve a 525 line system. The required bandwidth is 1.05 MHz, the object angle subtended is 8.2 arc-minutes, and the viewing distance is 16.8 inches.

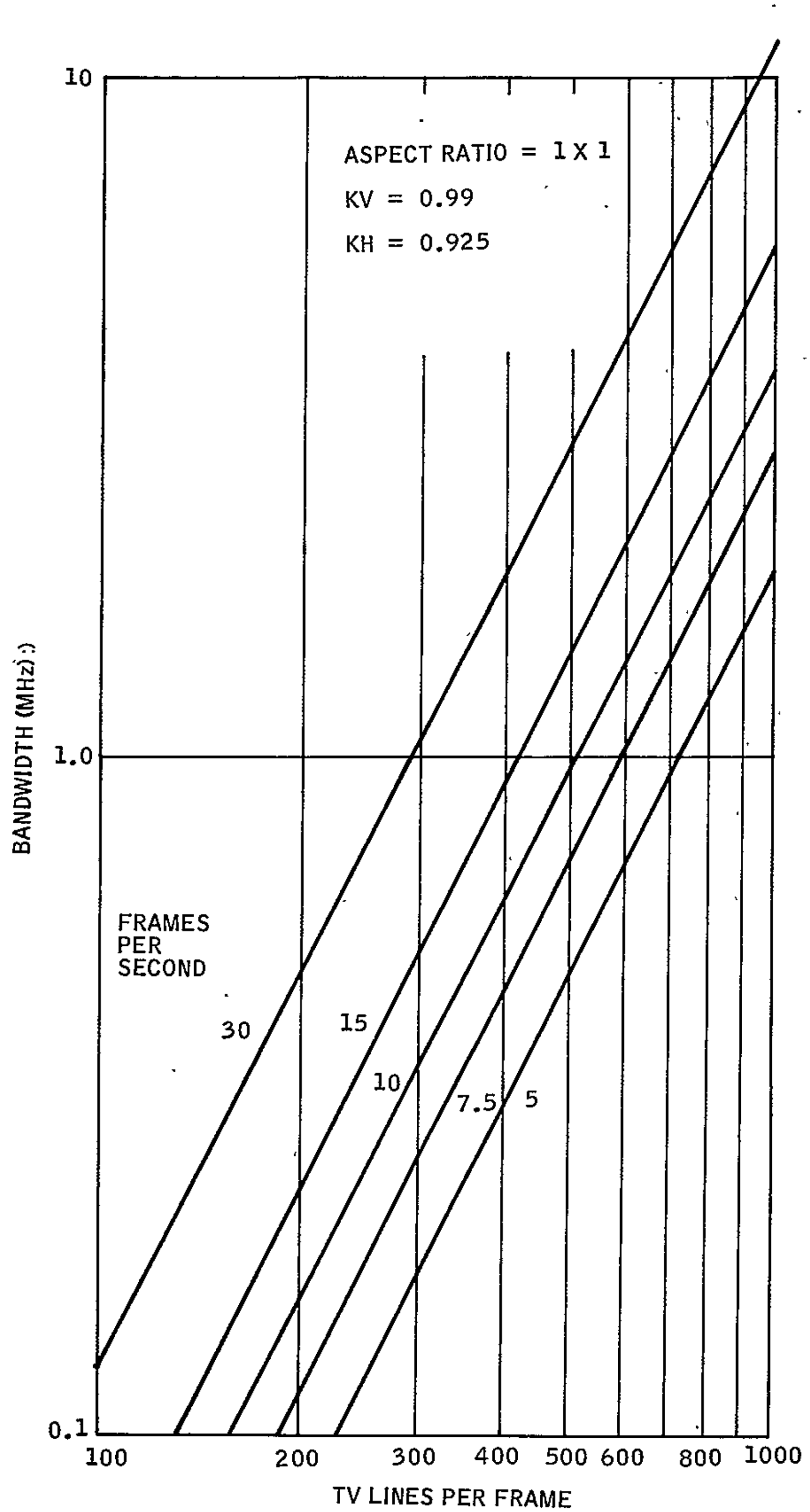

Figure 9.2-4. Bandwidth Versus TV Lines per Frame

It is also interesting to note that the system can be made "more standard" by changing the aspect ratio from 1 x 1 to 4 x 3. This changes the bandwidth from 1.05 MHz to 1.4 MHz which is still within the desired value. A survey of the physical size of the manipulators indicates that there should be no mechanical interference between the TV camera and the manipulators at viewing distance of 16 inches. A 25 degree field-of-view at a distance of 16 inches results in a viewing area diagonal of about 7-1/4 inches; this should be sufficient area for the tasks in mind and the resolution will be commensurate with the requirements.

Another tradeoff that can be made to reduce the required bandwidth is to reduce the frame rate from 10 to 5 frames/second. This compromise puts an undue strain on the charge spreading characteristics of the photo conductor in the TV pickup tube. Although slow scan materials are available for incorporation into pickup tubes, reliability data on these materials is scarce and it does not seem to be expeditious to entertain a long and expensive reliability program in this area when it is not justified.

9 2.3 TELEVISION SUBSYSTEM PARAMETERS

From the preceeding discussion of the video characteristics, the following parameters have been selected as the best compromise for the TV subsystem:

1. Aspect ratio: 4 x 3
2. Frame rate: 10 frames/sec
3. Line rate: 525 TV lines/frame
4. Bandwidth: 1.4 MHz
5. Field-of-view: 25 degrees (nominal)
6. Viewing distance: 16.8 inches (to resolve 0.04 inch diameter)
7. Interlace 1:1
8. Spectral response: S-18
9. Illumination requirements: High light level (approximately 155 ft-candles/ft^2).

9.2.4 DESIGN APPROACH FOR THE ILLUMINATION PORTION

One of the problems that exists in the application of TV to the manipulator spacecraft is that of illuminating the scene to be observed. This problem can be broken into three cases:

Case 1

If the scene under consideration is facing the sun, it will be very bright and may damage the TV pickup tube photocathode. This situation is corrected by using an automatic light control mechanism. The ALC mechanism replaces the need for remote control of the iris, neutral density filters, shutters, and remote lens capping.

The primary purpose of an ALC system is to automatically maintain a constant average illumination of the photocathode of the pickup tube as scene brightness varies over a day-to-night range of conditions. This is accomplished by a closed-loop servo system which positions a continuously variable neutral-density filter near the image plane of the camera lens system, such that the average output video is held to a desired constant value.

Figure 9.2-5 shows an automatic light control mechanism that has been used in low light level TV cameras. This device is capable of controlling photocathode illumination over a 10^7 range.

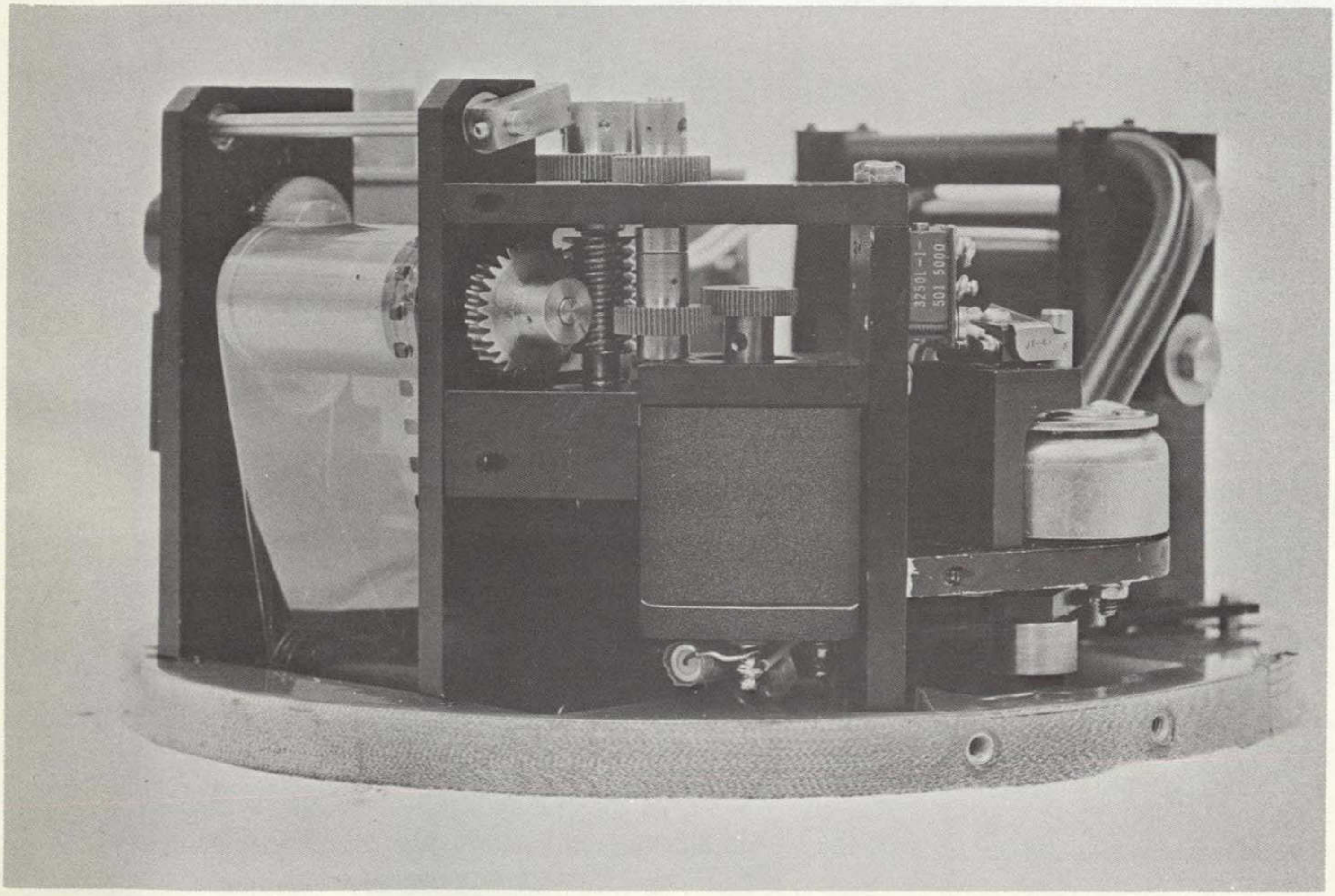

Figure 9.2-5. Automatic Light Control Shutter Assembly

Case 2

If the scene to be observed is on the shadow side of the target, additional light is required to illuminate the scene. Several schemes have been considered to accomplish this effect including:

1. Diffuse collectors feeding light to fiber optics bundles that can be directed by the manipulator arms

2. Reflectors and diffusers that can be positioned by the manipulators

3. Artificial light sources.

Since the orientation of the scene with respect to the sun will not be fixed, the use of collectors, diffusers and reflectors requires continuous reorientation of either the device or the spacecraft and the devices can cause mechanical interference with the telemetry antenna or conversely, the antennas may shadow the device. On this basis, it was concluded to provide artificial illumination and a small incandescent light and reflector will be mounted to the TV cameras to provide the required amount of light for proper operation of the TV pickup tube. Data from lamp catalogues indicates that at 6 watt incandescent bulb produces 41 lumens. If all of this light were focused on a 7-1/4 x 7-1/4 inch area, the illumination is equivalent to 105 lumens per square foot.

Assuming an f/2 lens with 75 percent transmittance, there will be 1.55 foot-candles available for the pickup tube if the surface has a reflectance of only 10 percent. This is quite enough light for most applications.

Case 3

In the event that the average scene is brightly illuminated, but the particular area of interest is in a shadow, the TV cameras will not be able to "see" in the shadow. It is expected that the relative intensities of light will be great and the artificial illumination will not suffice. Under these circumstances, the use of a reflector/diffuser surface positioned by the manipulators appears appropriate. This reflector can be small, about one foot square, and it will not cause any mechanical interference with the antenna since it can be positioned between the body of the spacecraft and the antenna. This case is much different from the situation that exists in Case 2, where a reflector may have to be positioned behind or over the antenna.

9.2.5 VIDEO DESCRIPTION

The video portion of the subsystem is a high resolution stereo TV system that mounts two camera assemblies on a moveable platform to provide pan, tilt, paralax control and focusing capability that is controlled by the manipulator operator. A mechanical backup is achieved by allowing the manipulators to physically position the pan-tilt assembly and adjust the focus and paralax control in case any drive system fails. The paralax and focus mechanism is illustrated in Figure 9.2-6. A third camera head assembly, located within reach of the manipulator is attached to a semi-rigid tether, so that is may be placed in any position for close viewing of the work area. The general camera configuration can be seen best in Figures 9.1-3 and 9.1-4. These are the figures illustrating the basic spacecraft.

The camera assemblies providing stereo vision each consist of a camera head assembly containing vidicon image tube, focus coil and pre-amplifier and also a camera control unit containing the sync and sweep generator, video processor and power supply. A synchronous DC/DC converter provides all necessary internal voltage.

The third camera intended for closeup viewing will have the camera head assembly and camera control unit separated by 4 to 5 feet of flexible tether. This will allow the manipulator to place the camera in close proximity to the work area.

The camera lens have not been chosen but some of the parameters are known. The paired cameras should be able to focus from 10 inches to infinity and have an adjustable focal length to allow the field-of-view to vary from the narrow angle required for rendezvous to the wide angle required for inspection. Field-of-view variation from 10 to 60 degrees is desirable. The closeup camera can use a fixed focal length lens with a 25 to 30 degree field-of-view and the ability to focus down to 12 inches.

The illumination portion of the subystem consits of three 5-watt incandescent lamps with reflectors plus spares, three automatic light control mechanisms, and two reflector/diffusers. One lamp and one ALC is mounted to each of the two main cameras, one lamp and one ALC is mounted to the moveable camera. Reflector/diffusers will be stored within reach of the manipulators for use when required.

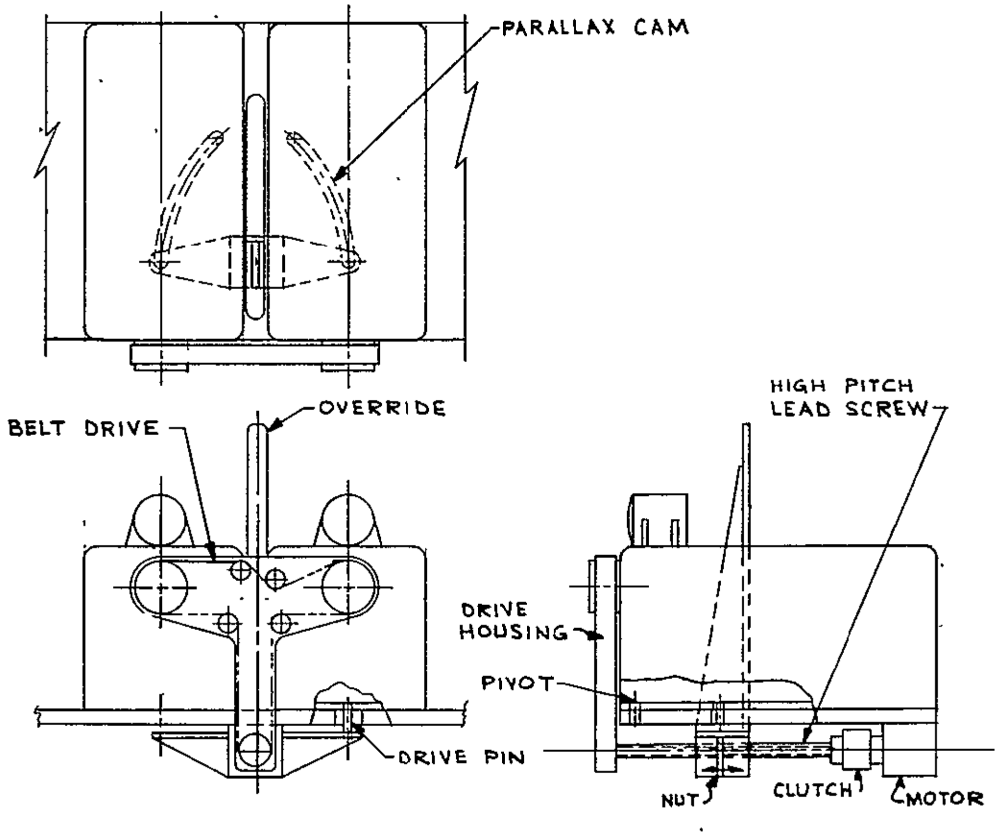

Figure 9. 2-6. Stereo Monitor Presentation

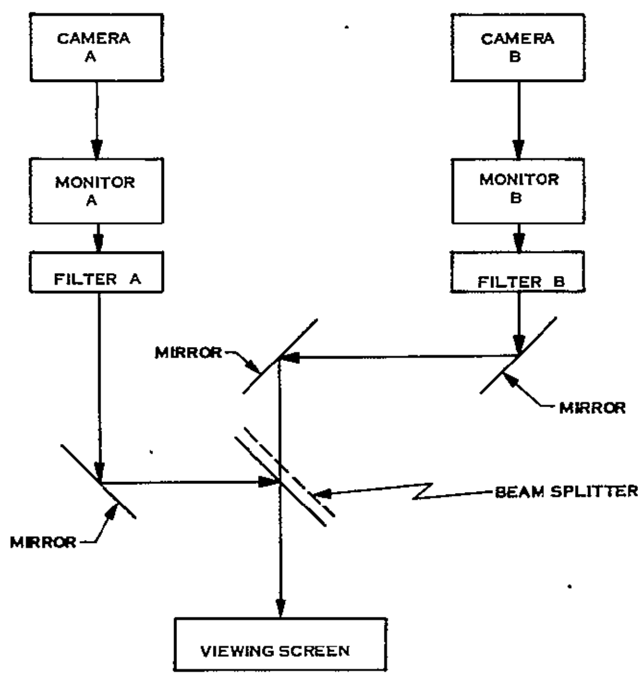

Figure 9. 2-7. Stereo Monitor Presentation

9.2.6 VIDEO MONITORING AT THE CONTROL STATION

Two of the cameras onboard the spacecraft are positioned together so that they can be used to provide the operator with a stereo-video presentation. The intent is to make use of the manipulator and spacecraft operator depth perception and aid them in the docking maneuvers.

.There are various ways to produce the desired stereo presentation. One way is to present the scene from each camera on a separate monitor and then equip the operator with an optical system that presents each eye with a scene from a different monitor. This can be achieved with a binocular-like system that observes a pair of large monitoring screens or it can be packaged into a helmet/head-set device. The helmet can also be part of a servo loop positioning the TV camera. This is advantageous as it points the camera where the operator wants to look and it leaves his hands free to control the manipulators or spacecraft.

Requiring the operator to view a video presentation through a binocular-like device may be fatiguing. Another way to present stereo pictures to an operator is to place a color filter in front of each monitor, optically combine the two images on one screen and present this combined image to an operator wearing colored eye-glasses. Each eye piece would have the same color filter as the corresponding monitor. This can be done with filters of red and blue or red and green. Each eye would then see only the picture from one monitor, the other being filtered out and the stereo effect is preserved. This latter scheme (Figure 9.2-7) is expecially desirable if a large number of observers are involved. The camera control can be driven from a mechnism following the motion of the operator's eyes and a light-valve projection systemfulfills the large size and combining requirements.

Another problem that will exist in the video presentation results from the flicker of a slow scan system. The flicker from a 10 frame/sec system will be both annoying and fatiguing to the operators. The fatigue problem can be relieved by inserting a scan convertor in the video channel to change the frame rate from 10 to 30 frames/second. This results in one picture being present on the monitor for 1/10 of a second but it will be a flicker free presentation. Motions may still appear to be jumpy but the jump size will depend on the velocity of the motion. "Jumping" is inherent at 10 frames/sec but the manipulator operator will not be making "fast" motions, and the need for higher frame rates and "real time" viewing is not necessary.

9.2.7 VIDEO SUBSYSTEM CHARACTERISTICS (WEIGHT AND VOLUME)

Subsystem	No. Required	Envelope (in.)	Wt (lb)
Camera Head Assemblies	3	2 (dia.) x 6.5	1 lb max. (each)
Lens	2 1	3 (dia.) x 4 2 (dia.) x 3	2 lb max. (each) 1 lb max.
Control Unit	3	6 x 6 x 6	5.5 lb (each)
Camera Mount	1	24 x 19 x 4	6 lb
Automatic Light Control	3	7 (dia.) x 3.5	2 lb (each)
Lamps and Reflectors	3	2 (dia.) x 2	0.2 lb (each)
Tether and Cabling	1	0.75 (dia.) x 72	5 lb
Reflector/Diffuser	2	0.13 x 12 x 12	1 lb

9.3 MANIPULATOR SUBSYSTEM

The requirements which the manipulator system must meet depend primarily on the tasks it must do. The baseline manipulator for this study is designed to accomplish the missions analyzed previously in this report.

9.3.1 MANIPULATOR REQUIREMENTS

Table 9.3-1 summarizes the missions and some of the resulting requirements. In general, it can be said that if the manipulator has human strength, reach and dexterity, the system will be adequate. This is the conclusion GE has previously reported (Reference 1). Much of the manipulator design work in the AFAPL study is applicable to the present study.

The methodology used to define the requirements includes:.

1. Inspection of actual spacecraft and mockups. During the inspection of the Nimbus and OAO spacecraft, an articulated mockup of the proposed slave manipulator arm was used to physically check out reach requirements.
2. Checking spacecraft drawings and photos.
3. Use of previous personal experiences. This includes GE industrial manipulator experience and previous space manipulator studies, as well as the informative inputs from the spacecraft designers and technicians.
4. Actual laboratory simulations of tasks defined by the mission. This work is discussed in detail in Section 5.
5. Study of past pertinent manipulator studies and experimental work. As an example, of prime importance to this study is the work on time delays effects previously done at MIT. This is summarized in Appendix F.

The methodology described above led to a set of requirements which the manipulator subsytem should meet. These are summarized in Table 9.3-2.

9.3.2 MANIPULATOR SUBSYSTEM DESCRIPTION

The manipulator subsystem design follows the requirements listed above. It follows the design philosophy of the manipulator design GE produced during the AFAPL Study

Table 9.3-1. Mission Analysis Summary

Mission	Duration (minutes)	Docking Constraints	Maintenance Package Weight (lb)	Maximum Manipulator			Laboratory Task Simulations
				Reach (in.)	Force (lb)	Torque (in.-lb)	
OAO-A1 Repair	986	In uncontrolled tumbling state, docking would not be attempted above 1.5 rpm	405	40	20	40	Yes
OSO-D Repair	265	Special despinning device needed	31	40	15	40	Yes
DBS-VBM/UHF Refurbishment	494	None. Satellite cooperative and stable	110	40	15	40	Yes
Nimbus A-C Refurbishment	754	None. Satellite cooperative and stable	166	40	15	40	Yes
Nimbus D-E Refurbishment	287	None. Satellite cooperative and stable	1090	40	15	40	Yes

Table 9.3-2. Manipulator Requirements

Specification	Suggested Value
Configuration	Two 6-degree-of-freedom arms
Type	Bilateral (i.e., closed loop position control with force feedback)
Reach	40 inch reach, spherical envelope
Response	Slightly less than man's response (about 4 cps BW)
Resolution	0.04 inch
Force	About 15 lb per arm minimum
End Effector	Parallel tong jaws
Video	Monocular (2 cameras) 1 fixed with pan and tilt 1 positionable by manipulator
Special Tools	Several defined
Life	Approximately 10 days in orbit
Tethering	Should allow easy repositioning of manipulator spacecraft
Indexing	Two shoulder joints

(Reference 1). A drawing of the right arm of the slave unit is shown in Figure 9.3-1. In some respects, the design varies from past hot lab electrical manipulators:

1. Because of 0-g environment, no counterbalancing is required and this reduces weight and allows the manipulator greater freedom.

2. The servo package is mounted at each joint and no cable or tape drives are used. This also reduces weight and complexity, gives a stiffer system which helps stability, and enables the arms themselves to be used as heat sinks.

3. The joints are offset to allow compact folding during launch as shown in Figures 9.1-1 and 9.3-1. The motors used are dc torque motors which have not been used in a complete manipulator system as yet, although the AEC's Brookhaven Lab has an arm in construction using this type motor.

9.3.2.1 Typical Joint Design

Figure 9.3-2 shows a detailed view of the shoulder joints. Each of the joints will be essentially identical in design. The goal was to use standard hardware items and to meet the 10-day life requirement. Some of the features of this design and comparisons with the AFAPL design include:

1. A standard harmonic drive unit is proposed in place of a conventional gear train because the AFAPL study has shown that a large weight saving is possible. However, the design shown if Figure 9.3-2 differs from the AFAPL design in that the harmonic drive is not hermetically sealed. The United Shoe Machinery Company has reviewed this concept and said that a silicone grease would work better than a dry film lubricant and that the duty cycle and life requirements seem within the present state-of-the-art.

2. The dc torque motor chosen is a shelf item that would give no problem for the proposed life (with proper treatments like Ball Bros. "Vacu Kote"). Use of the standard harmonic drive allows use of a larger diameter motor and, therefore, in the transport joints, one 140-watt motor is used instead of two 98-watt motors as used in the AFAPL design.

3. A standard film type potentiometer is mounted within the joint. The AFAPL design required a specially configured pot integrated into the joint design.

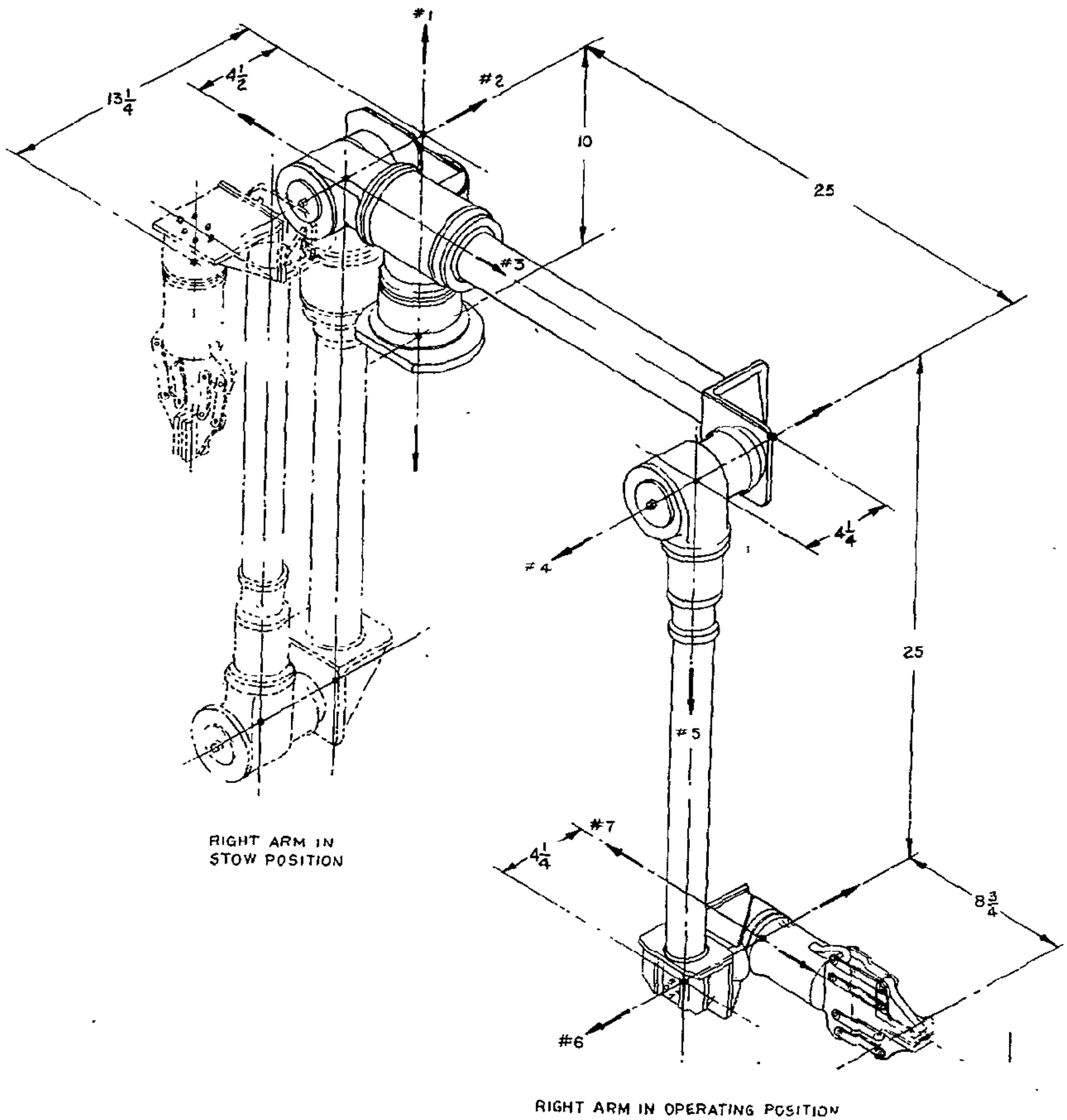

Figure 9.3-1. Isometric of Slave Manipulator

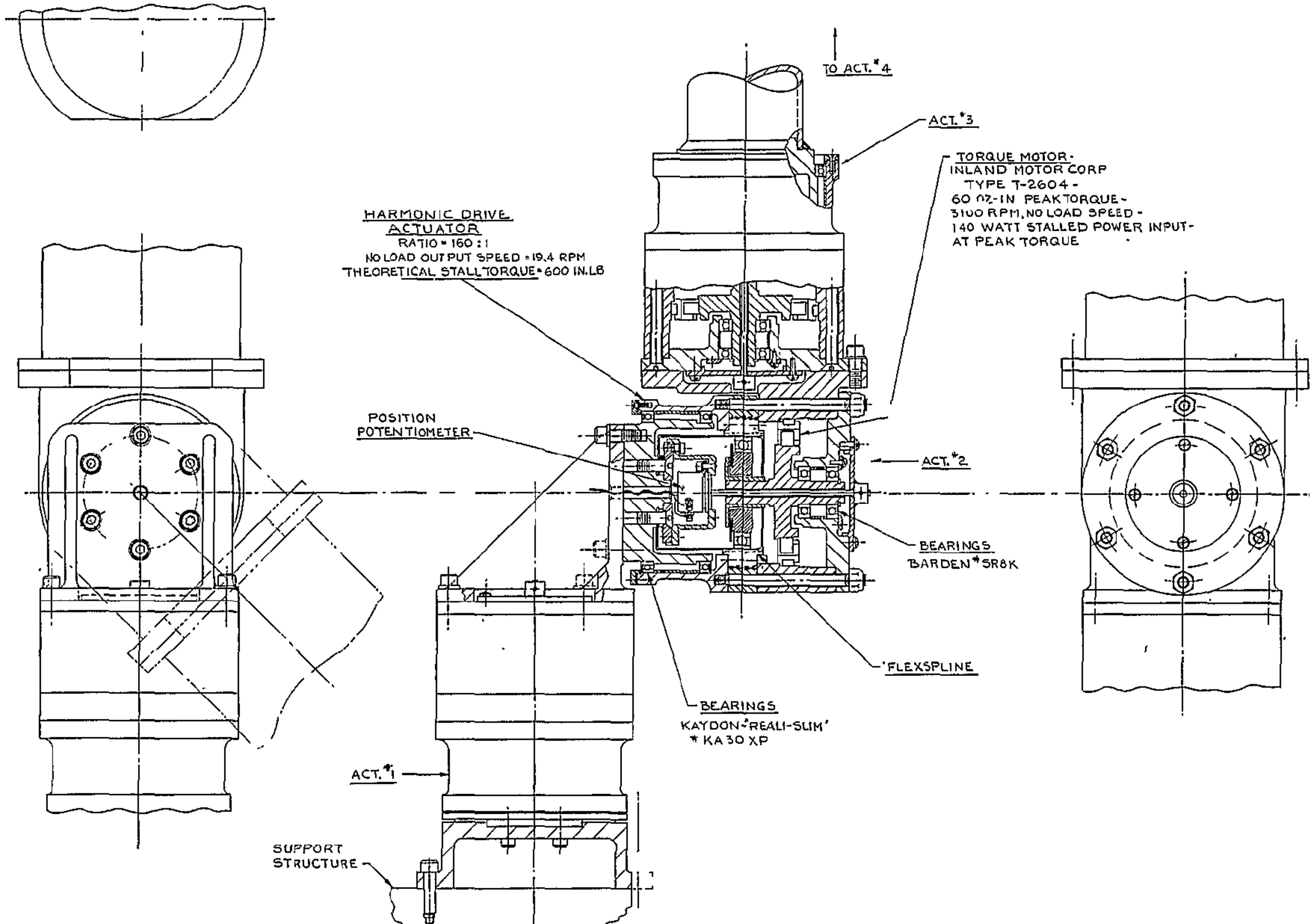

Figure 9.3-2. Shoulder Joint

9.3.2.2 Working Volume and Indexing

Normally, the spacecraft will be tethered so that the shoulder joint is a nominal 35 inches from the work site. The end effector is extended as much as 35 inches above and below the shoulder pivot in the sagittal plane. At an angle, of course, the reach is reduced. To keep the working volume within the spherical envelope of maximum reach, the working volume is as defined in Figure 9.3-3. Ideally, the manipulator will be tethered so that the center of the worksite is below the shoulder joint, giving a better master arm position. To reach all the working volume and also the stowage areas behind the arms, an indexing mode is provided in two of the shoulder joints. The master station can be designed to allow full swing of the master arms so that when reaching for material in storage behind the slave, the right arm does not become the left, etc., if both arms are indexed 180 degrees. When just one arm is indexed, this possible confusion of right for left does not occur, and so both indexing and full master azimuth swing can be provided for.

9.3.2.3 Velocity and Force Capabilities

The baseline manipulator design for the study is less powerful than the AFAPL design. However, it develops comparable torques by reducing the speed about 30 percent. Its speed and force capability as a function of reach is shown in Figures 9.3-4 and 9.3-5. The velocity exceeds the desired 30 in./sec* for reach beyond 25 inches and drops to a minimum of 15 in./sec at 10 inches.

This reduction for close-in work appears reasonable from a human factors standpoint. The effect of transmission time delay may well dictate restricting velocities to below 30 in./sec, especially in the bilateral control mode. The only known experimental work in this area has been with the MIT unilateral manipulators which have a maximum velocity of about 4 in./second.

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* As determined in the development of electric master slave manipulators for "hot lab" applications.

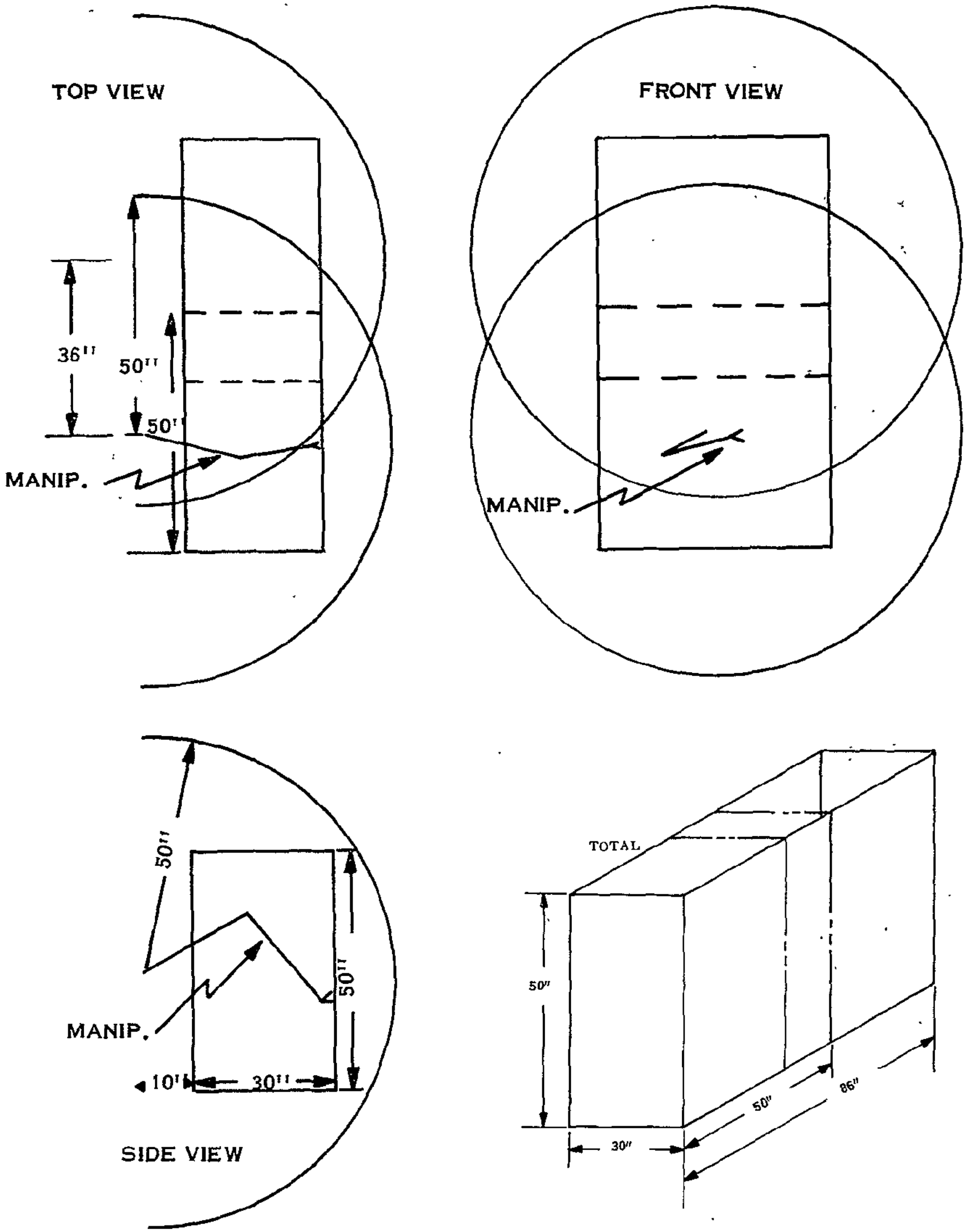


Figure 9.3-3. Total Working Volume for Two Manipulators

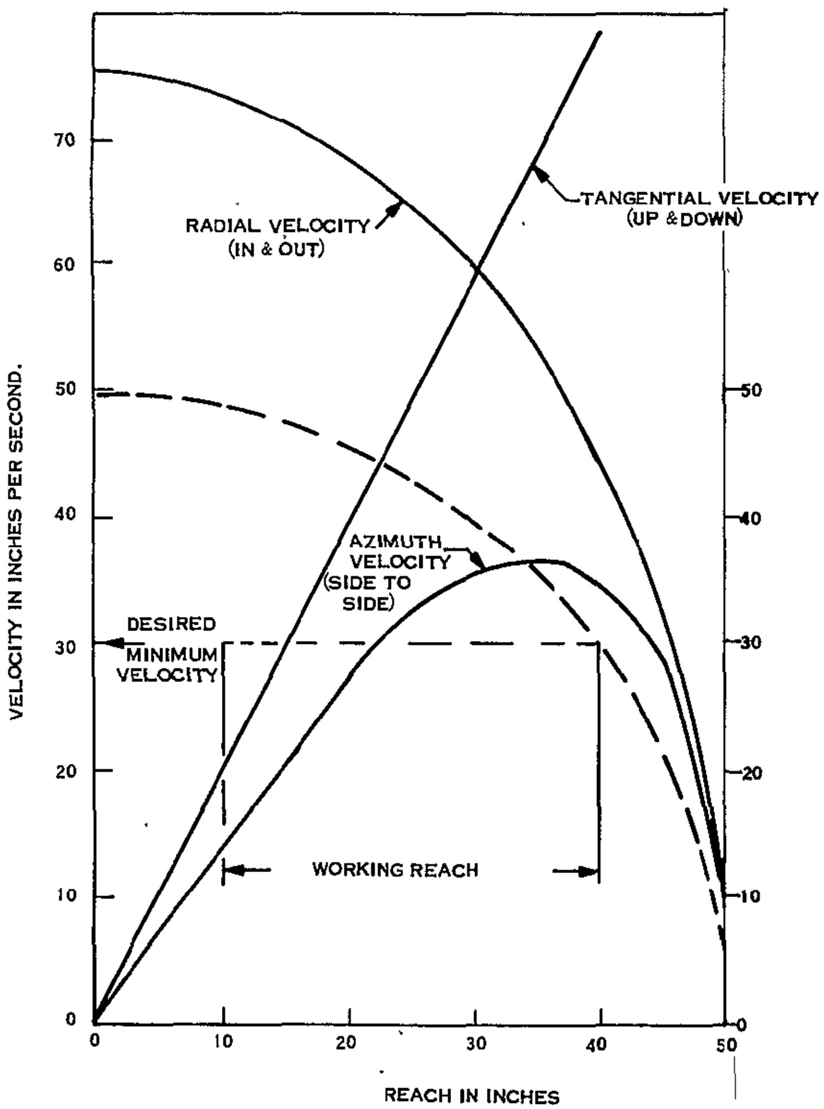


Figure 9.3-4. Velocity Capability versus Reach

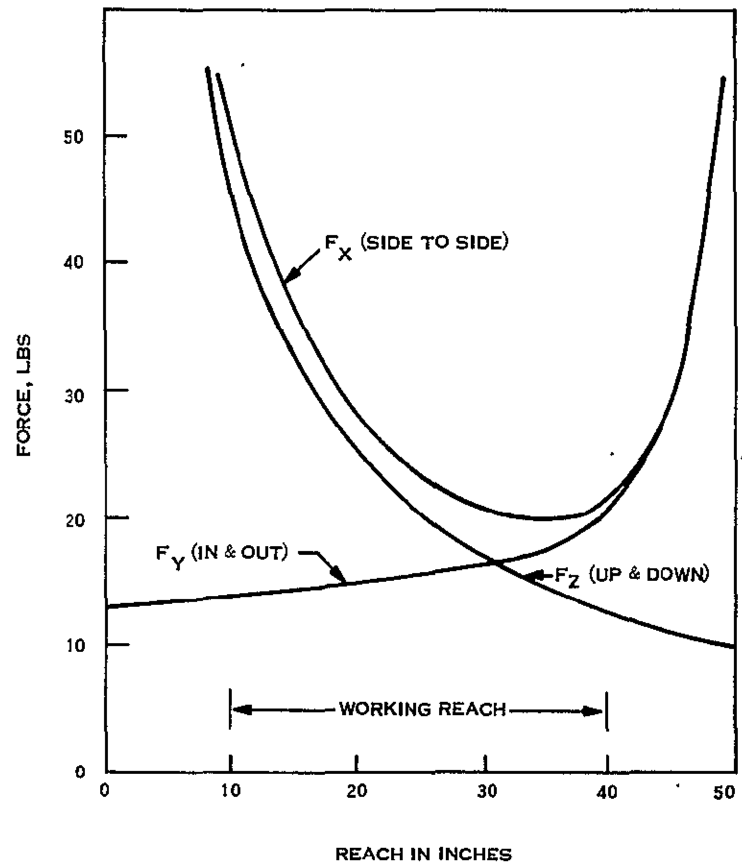


Figure 9.3-5. Force Capability versus Reach

The force capability shown in Figure 9.3-5 exceeds 12 lbs in all three directions and 15 lbs in two directions throughout the working volume. These curves assume a conservative 84 percent transmission efficiency.

9.3.2.4 Heat Transfer

A detailed heat transfer analysis was made on the AFAPL manipulator areas to check that the arms could dissipate the power of the joint motors. Since the maximum power level was reduced by 30 percent, it is assumed that the configuration of Figure 9.3-1 will be able to handle the thermal input. The original analysis was made for a synchronous orbit and for five different conditions. It showed some problem areas such as the need for sunlight protection when working on a large reflecting body.

9.3.2.5 End Effector

It is proposed to use the parallel jaw end-effector as the general purpose hand. This end-effector has almost universal acceptance in active manipulator ground applications. The only known articulated design is that of the Handyman, GE's design of a heavy duty hydraulic bilateral manipulator and it is seen to be heavy for space at this time. In addition, it requires additional servo loops. For space applications, the end effector design should incorporate these features:

1. Easily replaceable jaws that will allow using jaw surfaces that are hard, serrated, or pliable according to task requirements at hand. The shape of the jaws may also be changed.

2. A 3-inch opening of the jaws is suitable for all the grip hold and hardware thicknesses encountered in the maintenance mission studied (i.e., Nimbus strut truss, Nimbus paddles, OAO-A1 paddles, paddle devices and latches, and OSO spin gas arms.)

3. To allow gripping without any power drain, the end-effector should have a self-locking gear train.

9.3.2.6 Tethering Design

The design of the tethers is not to be overlooked since they should provide a solid attachment yet allow fairly easy repositioning of the spacecraft relative to the satellite. Study of the kinematics of the tethering structure led to the design using three tethers. The tethers can telescope to twice their stored length. At the satellite attachment end, ball and socket joints are provided so that alignment of the tether is not critical and, at the manipulator spacecraft end, the configuration has a rigidizable joint to allow the tether to be positioned in azimuth and elevation on command. This repositioning can be achieved by relaxing the tether joints and moving the manipulator spacecraft with the manipulators without disengaging the tethers.

9.3.2.7 Servo Amplifiers

The block diagram for the bilateral servo system is shown in Figure 9.3-6. The manipulator operator positions the master input while viewing a video picture which is delayed by a transmission time delay. The manipulator positioner's signal is transmitted to the slave through the same time delay. The slave loop is a typical dc servo position loop; feedback to the operator comes back in two ways, by a video camera observing slave motion and a signal proportional to motor current (shown in Figure 9.3-6 as alternate B-B'). This signal is proportional to motor torque plus friction effects of the gearbox. Use of an alternative scheme (shown as C-C'), would overcome the friction effect problems by deriving the signal from torque transducers on the slave arm. The C-C' alternate would give the operator the best "feel". The force feedback signal is transmitted to the master station, again through a time delay. The master reproduces the slave force signal and depending upon how well or poorly the operator reacts, the system might go unstable. If the oscillation is at a low enough frequency, adaptive controls can sense the oscillation and reduce master gain. The simplest scheme is for the operator to shut off the master amplifier when he senses an oscillation buildup. The amplifiers will be typical solid-state power bridge devices, running as Class B amplifiers. The spacecraft manipulator servo-amps will be designed for low quiescent power.

9.3.3 WEIGHT, AND POWER

The weight of one arm of the slave manipulator as given in Table 9.3-3 is 29.0 pounds. Fifty-eight percent of the weight is in the housing, tubing, and brackets in which the motor, harmonic drive and other hardware are mounted (i.e., the actual structural components).

Table 9.3-3. Weight Summary for 1 Manipulator

Item	No. Users	Location	Total Weight (lb)
T-2604 Inland DC Motor	4	Index and 3 shoulder joints	1.4
T-1218 Inland DC Motor	4	Tongs and 3 wrist joints	0.6
HDUC25 Harmonic Drive	4	Index and 3 shoulder joints	3.6
HDUC14 Harmonic Drive	3	3 wrist joints	1.2
Parallel Jam Tongs	1	End-effector	0.9
Non Reversing Cearbox	1	End-effector	1.0
Kaydon Reli-Slim Bearings	8	Index and 3 shoulder joints	1.2
Kaydon Reli-Slim Bearing	6	3 wrist joints	0.6
Barden SR8K Bearings	14	All joints	0.7
Markine Film Potentiometers	8	All joints	0.4
Cabling and Connectors			0.5
2-1/2 inch Tubing	1	Upper arm	0.5
2 inch Tubing	1	Lower arm	1.0
Housings (including Nuts and Bolts)	3	3 shoulder joints	7.5
Housings (including Nuts and Bolts)	3	3 elbow joints	3.6
Housings (including Nuts and Bolts)	3	3 wrist joints	3.3
			29.0
Amplifiers	7	Spacecraft	14.0

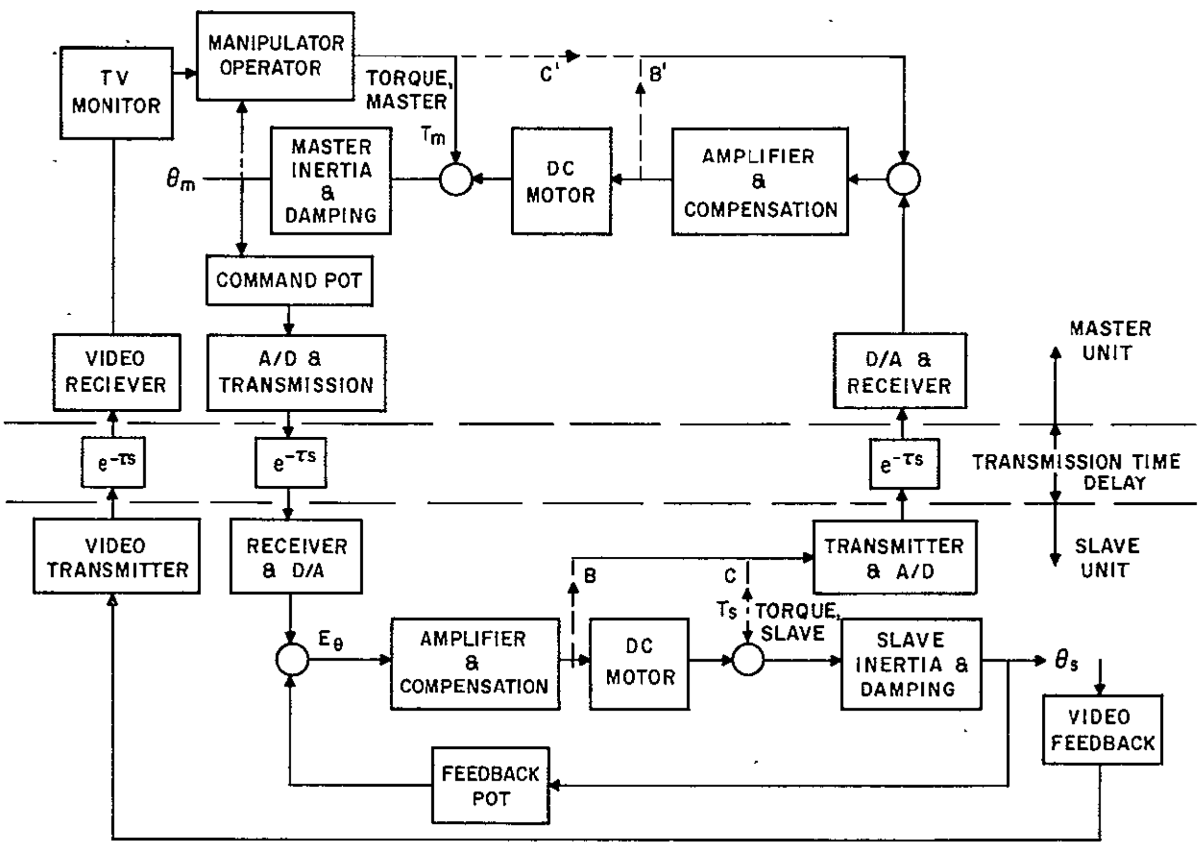


Figure 9.3-6. Servo Block Diagram

More weight could be saved by lightening procedures after a structural stress analysis. Aluminum was assumed as the structural material to keep the cost down from more exotic materials such as beryllium. The weight of the servo amplifiers is based on present commercial items which could do the job. The weight does not include heat sinks which are assumed to be part of the manipulator spacecraft structure.

The estimate of power consumption is based on two approaches: (1) the estimate made in the AFAPL design and (2) actual power measurements made in the GE laboratory. In the Air Force study an assumed task was analyzed and force levels estimated as a function of time. From this, the expected peak load and average working load power estimates were made. The peak estimate was 400 watts for less than 0.1 percent of the task time, while the average was 20 watts.

To check this estimate, an experiment was devised using the E-2 electrical master-slave manipulator. A representative task was performed by the manipulator, and the total power consumed was measured. Figure 9.3-7 shows the task setup -- a solenoid valve to be assembled and disassembled. The task was accomplished using just one arm to simplify the recording and set up time requirements. Figure 9.3-8 is a plot of power consumed versus time, showing subtask headings.

To reduce this to a power profile, several factors need to be considered. First, the measured power is that of the E-2 master plus slave. Quiescent power for both was accounted for by measuring from the motionless arm level (pause level). Since master and slave motors are identical and wired in series, it is assumed one-half the measured power is actual slave power consumed. Second, the task was done in a gravity field. Third, the grip was not self-locking but required power drain to maintain a grip. Fourth, the task was done with direct viewing instead of through a video link. Fifth, the force level of the E-2 arm is only half that of the proposed space design. Sixth, the task was well known and rehearsed and thus, required no inspection time.

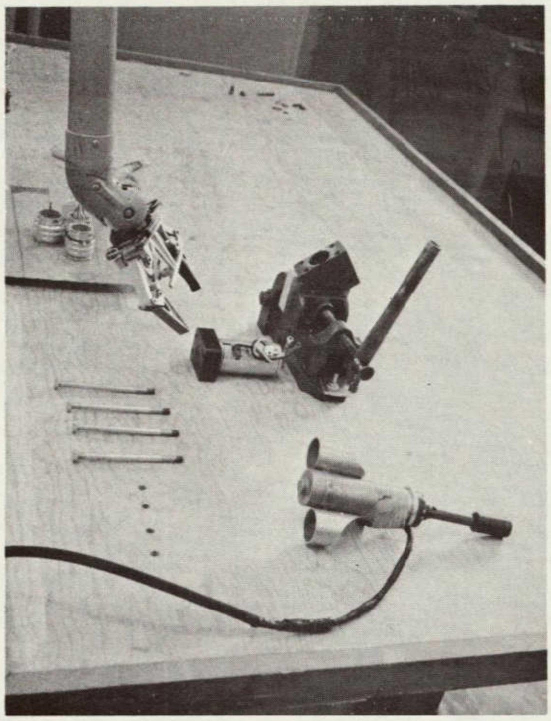


Figure 9.3-7. Task Setup

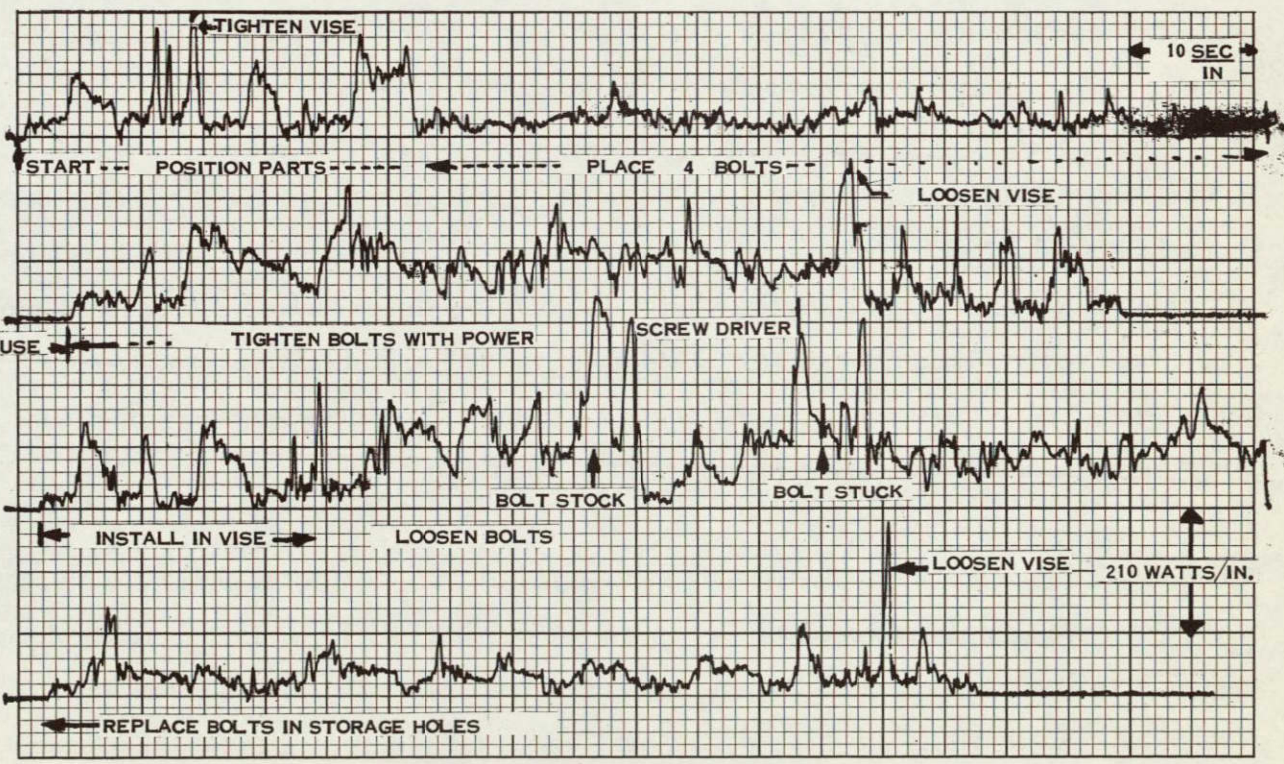


Figure 9.3-8. Power Profile, Assembly and Breakdown of Solenoid Valve

It is assumed that the second and third point will balance out the fourth. Considering the fifth point, note that with the lab simulation, the peak slave power level was 180 watts. The E-2 uses 2-phase ac servo motors of 32.5 watt stall power, whereas the space design uses 140 and 63 watt stall power dc motors. However, by assuming a linear torque-speed curve and ignoring inductive power losses, in both cases it can be shown that the instantaneous power from the amplifier is given by:

$$P = \frac{Tw_o}{8.85} \quad \frac{\text{sec}}{\text{in./lb}} \quad \text{watts}$$

where T is the instantaneous torque and w_o is the no-load speed of the motor. Thus, in extrapolating the power drain to the space design, consideration must be given to comparing w_o and T. The comparison shows the space manipulator drawing similar amounts of power to do the laboratory task. However, the space arm does have higher force capabilities and thus can dissipate more power. For this reason, the peak power level is raised to 300 watts per arm. This level is equivalent to 90 percent of capacity of a shoulder, elbow, and wrist joint. This produces a maximum possible of 670 watts when all the motors on one are stalled. This condition seems to be improbable.

To come up with an average power level, it is possible to average the power profile curve shown in Figure 9.3-9. An estimated amount of inspection time (20 percent of task time) was added to the profile to account for the sixth point mentioned above. Using this factor, the average power becomes 43 watts. This is for the one arm; however, it is assumed that this is a reasonable total for two arms doing the same task as the second arm would be doing minor tasks with low force levels, such as positioning.

Added to the above power levels should be an estimate of the additional power level required by the dc servo amplifiers which are less than 100 percent efficient. Design rules of thumb suggest a 10 percent addition to the average power level is a nominal value and that a 5 percent addition is within the state-of-the-art. Using the 5 percent figure, the average power drain will then total about 23 watts per arm. It is felt that the laboratory measurements are applicable and the results of the AFAPL study confirm this.

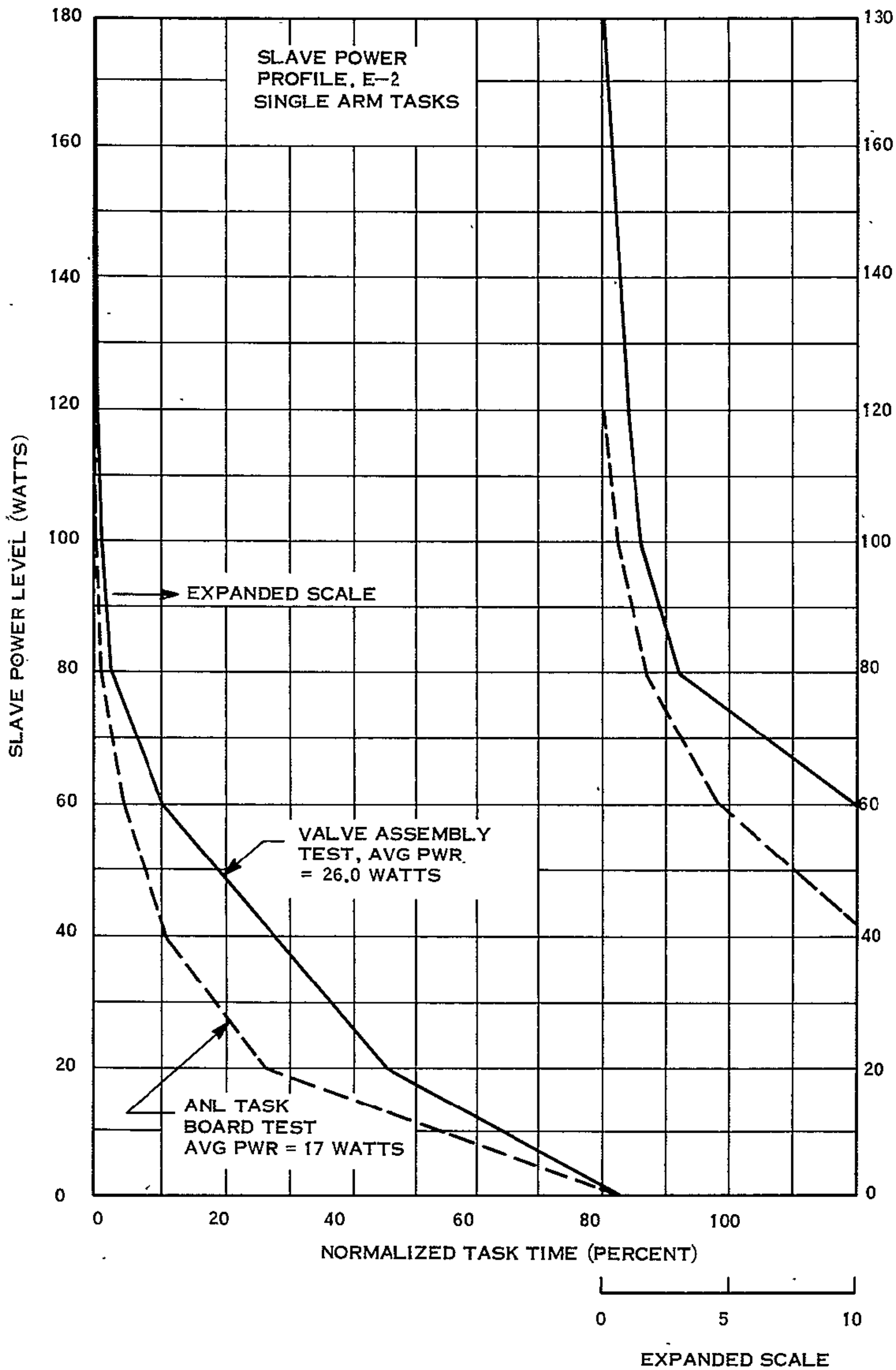


Figure 9.3-9. Average Power Curve

Total power drain (energy) required will be the product of the estimated task times without time delay and the average power level. Power requirements are summarized in Table 9.3-4.

Table 9.3-4. Manipulator Power Requirements Summary (2 Manipulators)

Expected peak load, less than 0.1% of task time	600 watts
Average power	43 watts
Expected amplifier quiescent power	2 watts
Index joint average power	1 watt

9.4 COMMUNICATIONS SUBSYSTEM

The Remote Manipulator Spacecraft is unmanned and to accomplish its mission, a means must be provided for communicating with and receiving data from it. This section presents the communications requirements, describes the functional design of the satellite subsystem, calculates predicted performance, gives estimates of physical characteristics, and describes further tradeoffs and problem areas.

9.4.1 GENERAL REQUIREMENTS

The design of the communications subsystem must satisfy the following requirements.

1. Provide continuous two-way communication between the Remote Manipulator Spacecraft and a ground station.
2. The design must be compatible with missions having low (500 nm) to synchronous (19,323 nm) altitude orbits.
3. The spacecraft must be able to transmit the following signals to the ground station:

Channel	Type of Signal
2	Analog TV (1.4 MHz baseband bandwidth)
14	Force feedback (7-bit word, 10 words/sec)
50	Analog housekeeping data (1 sample/sec)
160	Analog housekeeping data (1 sample/16 sec)
100	Digital ON/OFF housekeeping data (1 sample/16 sec)

4. The ground station must be able to transmit the following signals to the spacecraft:

Channel	Type of Signal
14	Manipulator control (12-bit word, 200 words/ sec)
2	TV camera control (15-bit word, 65 words/ sec)
1	Spacecraft operational commands

5. A means of tracking the Remote Manipulator Spacecraft to an error volume of 100 mile radius must be provided.

6. The design must have the minimum cost compatible with satisfying the performance requirements.

Continuous communications between spacecraft and the ground station are assumed to be provided by using a synchronous-altitude relay satellite* to relay signals between the spacecraft and a ground station. The characteristics of an appropriate relay satellite are listed in Table 9.4.1. (References 2 and 3).

9.4.2 COMMUNICATION SUBSYSTEM REQUIREMENTS

The communication subsystems on-board the spacecraft must satisfy the following functional requirements:

1. Range and range-rate signals from as many as three STADAN facilities must be received, coherently translated in frequency by a fixed ratio, and retransmitted to the STADAN facilities.

2. Manipulator and TV camera control commands from the ground station must be continuously received, decoded, and distributed.

3. Spacecraft operation commands from the ground station must be received, decoded, and distributed or stored until execution.

4. TV signals, force feedback signals, and engineering telemetry data must be continuously transmitted to the ground station.

5. Total peak power consumption shall be less than 200 watts and total average power consumption less than 100 watts.

6. Carrier frequencies, RF bandwidths, and signal power levels must be compatible with the characteristics (Table 9.4-1).

9.4.3 COMMUNICATION SUBSYSTEMS DESCRIPTION

Based on the general communication requirements and spacecraft communications subsystem requirements, a functional design has been performed (Figures 9.4-1 and 9.4-2). The radio subsystem

*Any synchronous altitude satellite used to relay communications between the manipulator spacecraft and ground station will be referred to as a Data Relay Satellite (DRS).

Table 9.4-1. Relay Satellite Characteristics

No. Relay Channels		
2 downlink (spacecraft-DRS-ground station)		
2 uplink (ground station-DRS-spacecraft)		
Channel Bandwidth		
10 MHz each downlink		
1 MHz each uplink		
Transmitter Power		
20 watts to ground station		
10 watts to spacecraft		
Antenna Gain (db)		
34.2 (Transmit to spacecraft)		
44.0 (Receive from spacecraft)		
28.0 (Transmit to ground station)		
28.0 (Receive from ground station)		
Frequency Assignment	Channel 1	Channel 2
DRS to Manipulator System	1819.8 MHz	1831.8 MHz
Manipulator System to DRS	2272.5 MHz	2287.5 MHz
DRS to Ground Station	7250 to 7750 MHz Band	
Ground Station to DRS	7900 to 8400 MHz Band	

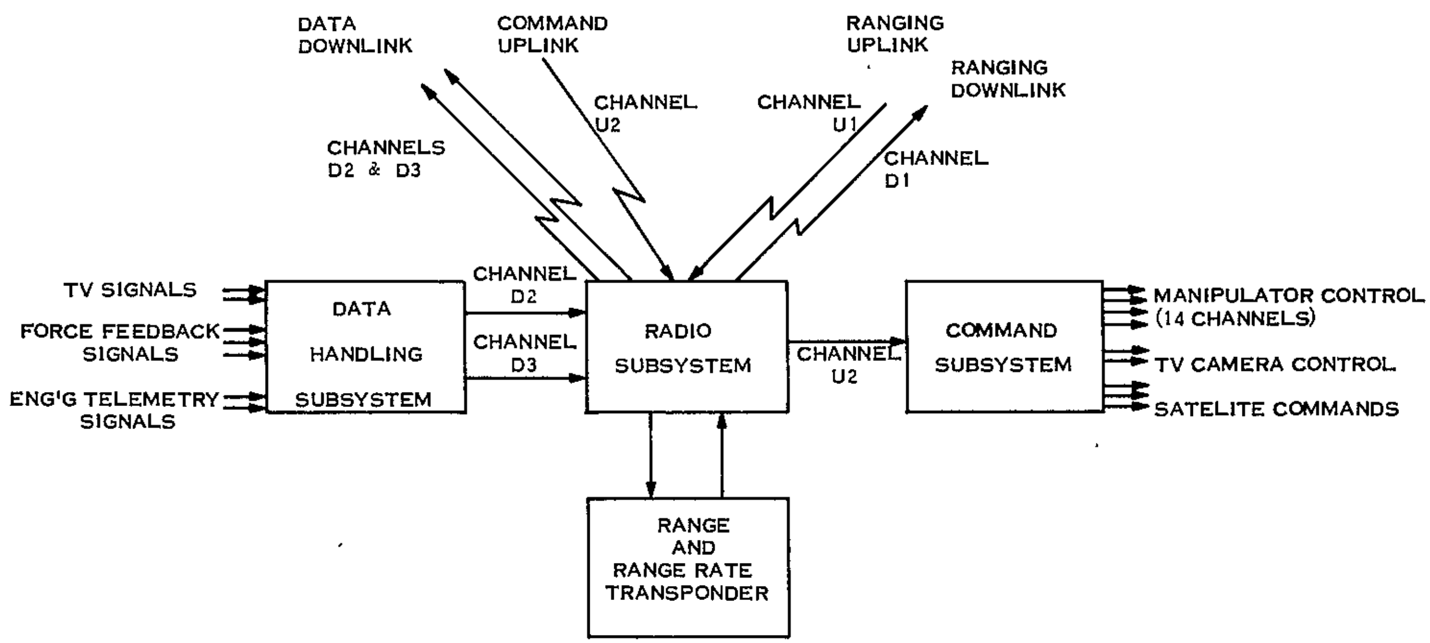


Figure 9.4-1. Communication Subsystem Functional Flow

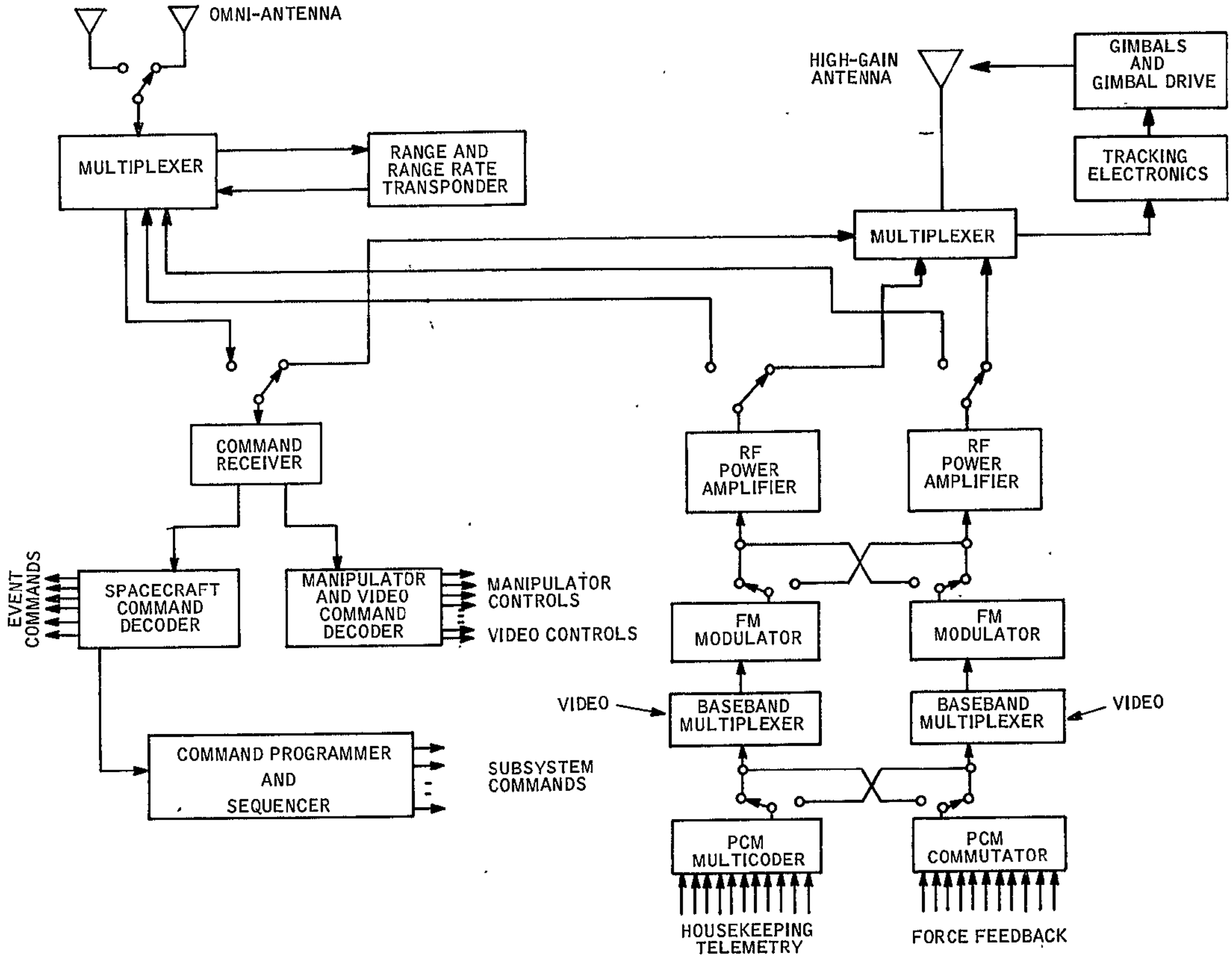


Figure 9.4-2. Radio Subsystem Block Diagram

subsystem receives RF signals on two frequency channels and transmits RF signals on three frequency channels via two omni-directional antennas and one high-gain tracking antenna. The command subsystem processes the received signal from the radio subsystem to recover and distribute the manipulator control, TV camera control, and satellite command data. The data handling subsystem processes the two TV signals, 14 force feedback PCM signals, and the engineering telemetry to provide two composite baseband signals to the radio subsystem for transmission to the ground station. The range and range-rate transponder receives up to three STADAN ranging signals in the same frequency channel from the radio subsystem, coherently translates the frequency of each to the proper downlink frequency, and sends the composite ranging signal to the radio subsystem for transmission to the STADAN facilities.

9.4.4 RADIO SUBSYSTEM

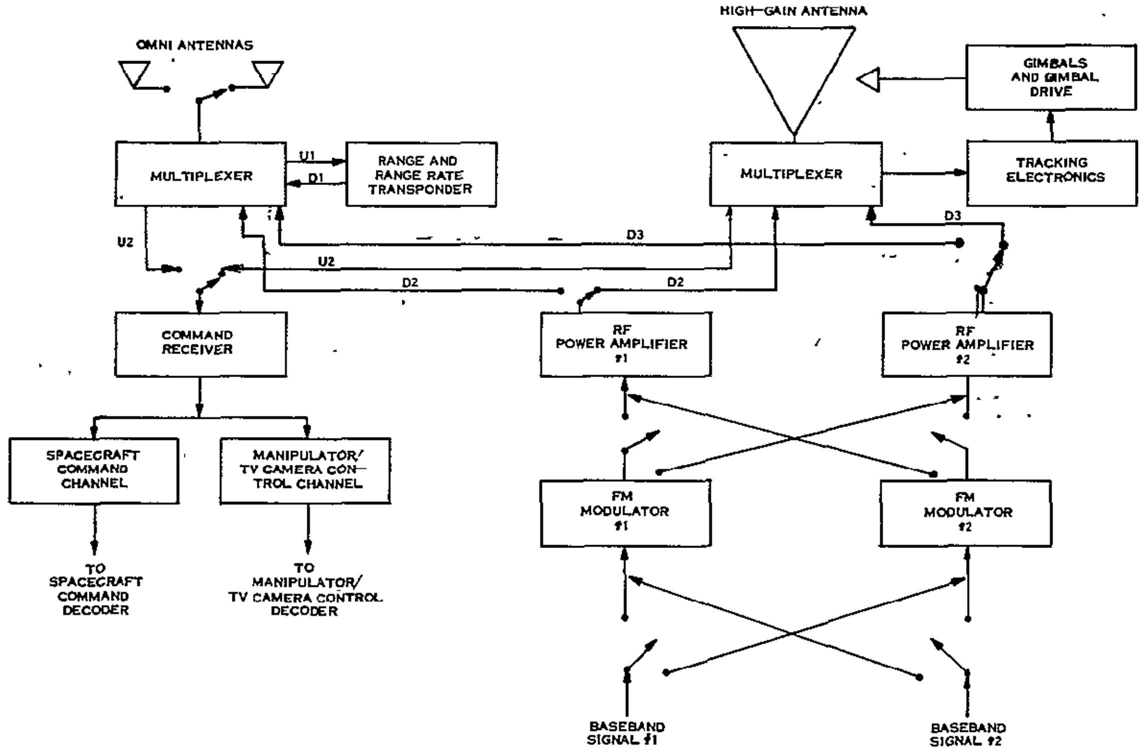
The radio subsystem consists of a high-gain tracking antenna (Figure 9.4-3), two omnidirectional antennas, two multiplexers, two RF power amplifiers, two FM modulators, one AM command receiver, and various RF switches, cables and connectors. The radio subsystem is designed to simultaneously receive uplink signals of two different frequencies and transmit downlink signals of three different frequencies. The frequency assignment is:

1. Uplink

Frequency	Designation	Use
2253 MHz	U1	Range and Range Rate
1831.8 MHz	U2	Manipulator Control, TV Camera Control, Satellite Commands

2. Downlink

Frequency	Designation	Use
1700	D1	Range and Range Rate
2272.5 MHz	D2	TV Signal, Force Feedback
2285.5 MHz	D3	TV Signal, Engineering Telemetry



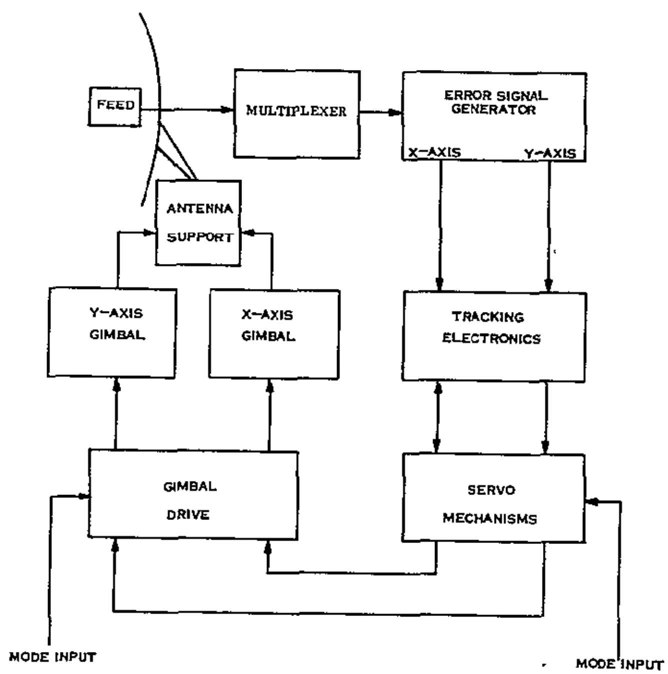


Figure 9.4-3. High-gain Antenna Tracking System

The high-gain tracking antenna provides the capability of transmitting TV signals to the ground station and receiving high bit rate commands to control the manipulators and TV cameras. It consists of a 6-ft erectable parabolic dish, RF feed assembly, gimbal and gimbal drive, servo electronics, and tracking electronics. The tracking electronics operate on the uplink signal to keep the high-gain antenna boresight pointing to the uplink signal source and the downlink signal destination, since both signals normally traverse the same path in opposite directions.

There are three modes of operation-acquisition, etc. In the acquistion mode, the antenna is dithered in a pre-set pattern until the signal from the relay satellite is acquired. The auto-track mode keeps the antenna boresight aligned with the DRS line-of-sight (LOS). The slew mode is used just before the spacecraft-DRS LOS is blocked by the earth to rapidly change the pointing direction of the antenna until a new DRS is acquired. Normally, following the slew mode, the auto-track mode can begin again. However, the acquisition mode will commence if a new DRS signal has not been acquired within 10 seconds after slew. Typical characteristics for the antenna are as follows:

1. Peak gain:

   28 dB at 1700 and 1831.8 MHz
   30 dB at 2253, 2272.5 and 2287.5 MHz

2. 3 dB beamwidth: 5^{o} total

3. Maximum allowable initial pointing error:

   5^{o} Auto-track mode
   20^{o} Acquisition mode

4. Average pointing error: 0.5^{o}

5. Polarization: Circular

6. VSWR: 1.5

7. Maximum slew rate: 5^{o}/second

8. Maximum angular excursions:

   150^{o} X-Axis
   360^{o} Y-Axis

9.4.4.1 Omni-directional Antennas

Two hemispherical-coverage antennas, each located on opposite sides of the satellite, provide essentially uniform unity-gain reception and transmission by the satellite. Mutual interference between each antenna precludes operating them simultaneously; therefore, switching between them occurs initially at one-minute intervals until the ranging transponder or the command receiver indicates the presence of a received signal. Loss of received signal recommences the switching mode. Switchover can also be effected via ground command. Each antenna consists of a concial spiral mounted on a 4-ft long boom to minimize interference caused by the spacecraft.

Typical electrical characteristics for an S-band conical spiral are:

1. Gain: 0 dB
2. 3 dB beamwidth: Hemispherical
3. Polarization: Circular
4. VSWR: 1.75
5. Axial ratio: 6.0 dB (loss 0.4 dB)

9.4.4.2 RF Power Amplifiers

The RF power amplifiers amplify inputs from the FM modulators to a power level suitable for transmission to the ground station. Each amplifier consists of a traveling-wave tube amplifier (TWTA) plus self-contained power conditioner (28 vdc input). Following are the required characteristics of the TWTA:

1. Output Power: 5 watts (saturated)
2. Input Power: 50 milliwatts
3. Efficiency: 25% overall
4. Center Frequencies:

   2272.5 MHz (No. 1)
   2287.5 MHz (No. 2)

9.4.4.3 Modulators

The input to each modulator is a composite baseband signal. The modulator consists of a wideband, linear frequency modulator and preamplifier. Following are the required characteristics of each modulator:

1. Output power: 50 milliwatts
2. Modulation: True FM
3. Center frequencies:

   2272.5 MHz (No. 1)
   2287.5 MHz (No. 2)

4. Input voltage level: 5 volts (peak-to-peak)
5. Frequency deviation: 2.0 MHz/VRMS
6. Input bandwidth: 1.5 MHz

9.4.4.4 AM Command Receiver

The uplink command signal is received and stripped from the RF carrier by the command receiver. The receiver is an AM double super-heterodyne type at 1831.8 MHz. The required characteristics of the receiver are:

1. IF bandwidth: 400 kHz
2. Sensitivity: -135 dBW
3. Audio bandwidth: 200 kHz

9.4.4.5 Radio Subsystem Interface Definitions

1. Inputs

   a. One composite data signal from the data handling subsystem containing TV signal and force feedback PCM data

b. One composite data signal from the data handling subsystem containing TV signal and engineering telemetry PCM data

c. Range and range rate signal from range and range rate transponder

d. Commands from command subsystem.

2. Outputs

a. Command signal to command subsystem

b. Range and range rate signal to the range and range rate transponder

c. Engineering telemetry monitor signals.

9.4.5 COMMAND SUBSYSTEM

Figure 9.4-4 is a block diagram of the command subsystem and Figure 9.4-5 is a block diagram of the manipulator TV/camera control decoder. The input signal is a PCM/FSK-AM type (Reference 4) with a subcarrier frequency of 100 kHz. The bit rate is 40,000 bps, which includes a 10 percent allowance for addressing and synchronization.

Figure 9.4-6 is a block diagram of the satellite command decoder. The input signal is a PCM/FSK-AM-type (Reference 4). The subcarrier frequency is 10 kHz. The bit rate is 128 bps. The decoder logic operates on the received command words to decide what command has been sent. Real time commands are transferred directly to the subsystems. Non-real time commands are directed to the command programmer and sequencer, where they are stored until execution.

Figure 9.4-7 is a block diagram of the command programmer and sequencer. Stored commands and time tags are loaded into the CP&S from the ground. When the clock and time tag agree, the commands are executed. Inputs from on-board sensors will cause initiation of automatic operation modes if a failure is indicated.

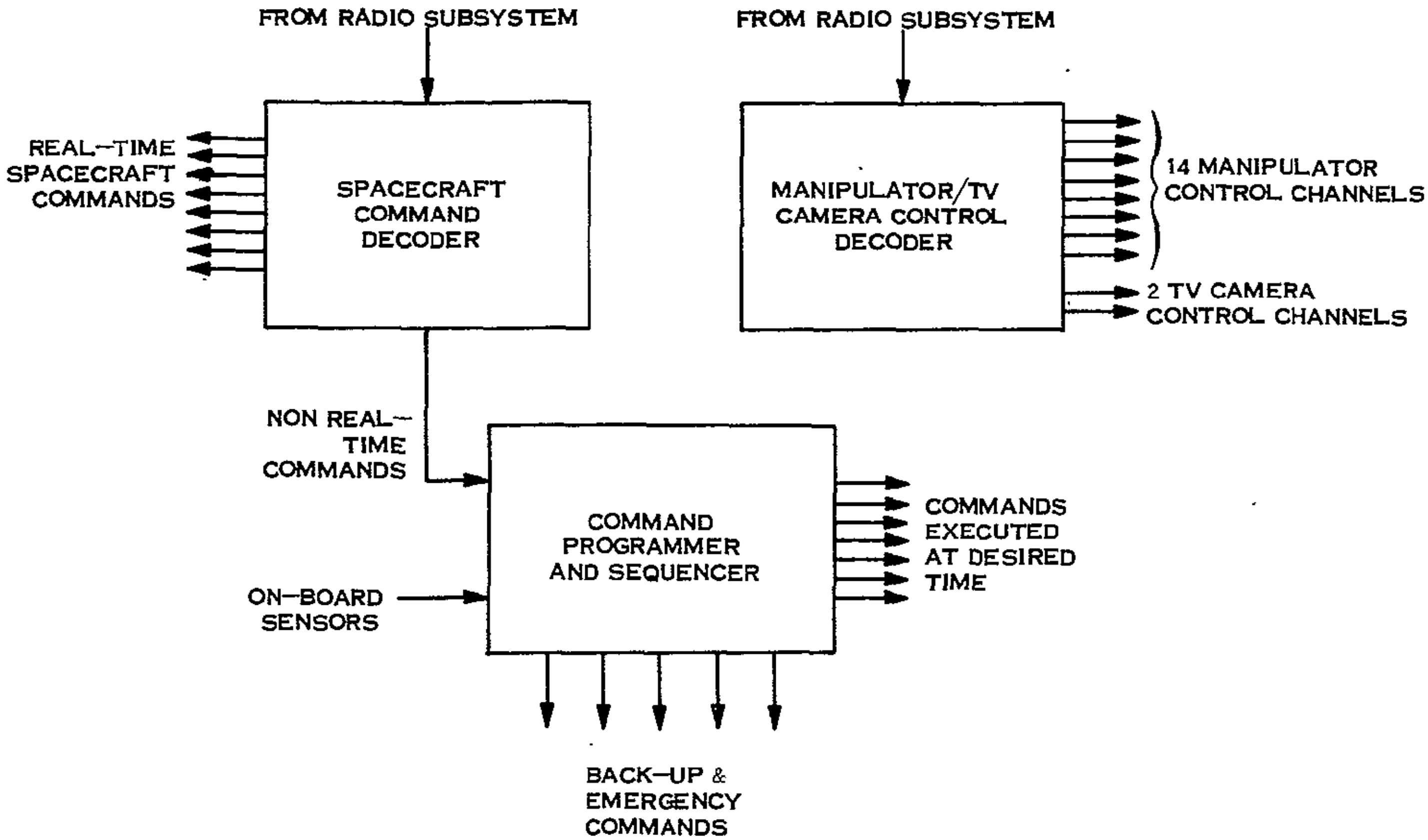


Figure 9. 4-4. Command Subsystem Block Diagram

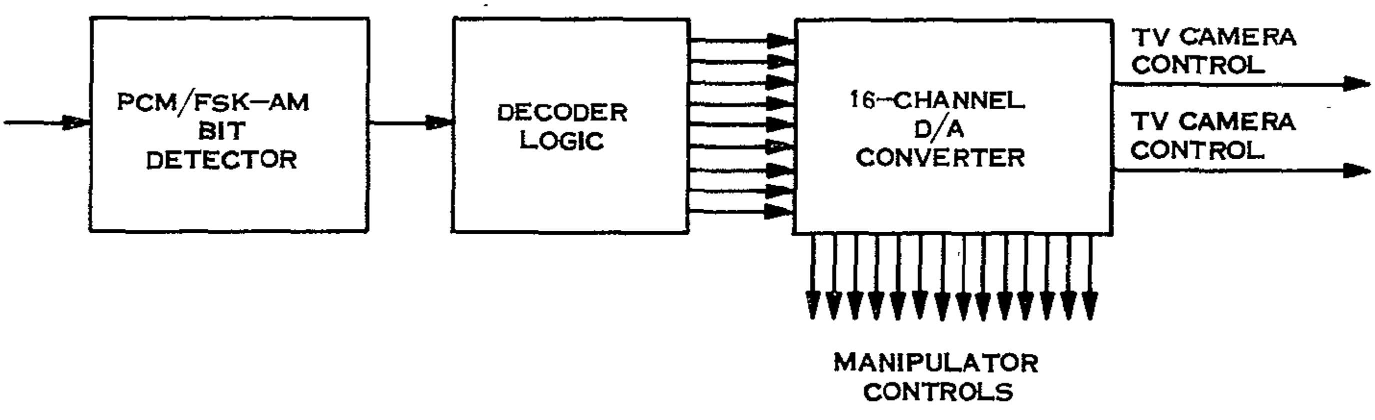


Figure 9. 4-5. Manipulator/TV Camera Control Decoder Block Diagram

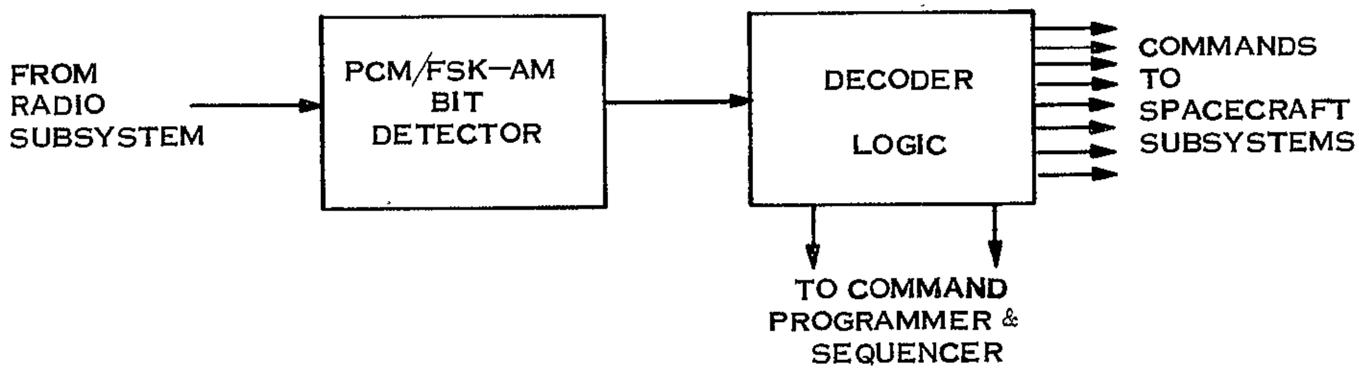


Figure 9. 4-6. Satellite Command Decoder Block Diagram

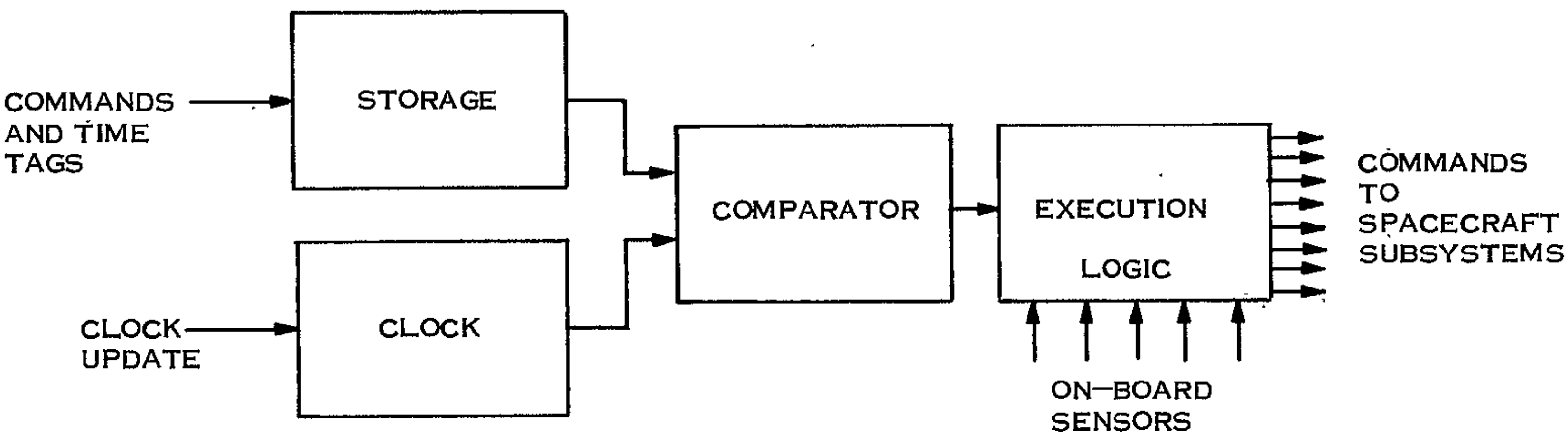


Figure 9. 4-7. Command Programmer and Sequencer Block Diagram

9.4.5.1 Command Subsystem Interface Definition

1. Inputs

   a. Receive command signal from radio subsystem

   b. On-board sensor readings.

2. Outputs

   a. 14 manipulator control signals

   b. 2 video control signals

   c. Satellite operational commands

   d. Telemetry monitor signals.

9.4.6 DATA HANDLING SUBSYSTEM

Figure 9.4-8 is a block diagram of the data handling subsystem and Figure 9.4-9 is the block diagram of the baseband multiplexers (the amplifiers provide gain such that the TV signal has a 4.86 volt peak-to-peak level and the PCM/PSK signal a 0.14 volt peak-to-peak level in the linear summer). Figure 9.4-10 is a block diagram of the PCM multicoder for the engineering telemetry data.

For the force feedback data, the A/D converter and PCM subcommutator are not required. Each PCM multicoder converts the multichannel input into a serial PCM data stream. This data stream phase-shift keys a sinusoidal oscillator of frequency 1.45 MHz. The data bit rate out of the force feedback multicoder is 1000 bps. The data bit rate out of the engineering telemetry multicoder is 500 bps. Each phase-shift keyed subcarrier is multiplexed with a baseband TV signal for transmission. Figure 9.4-11 is a plot of a typical power spectrum of the composite baseband signal.

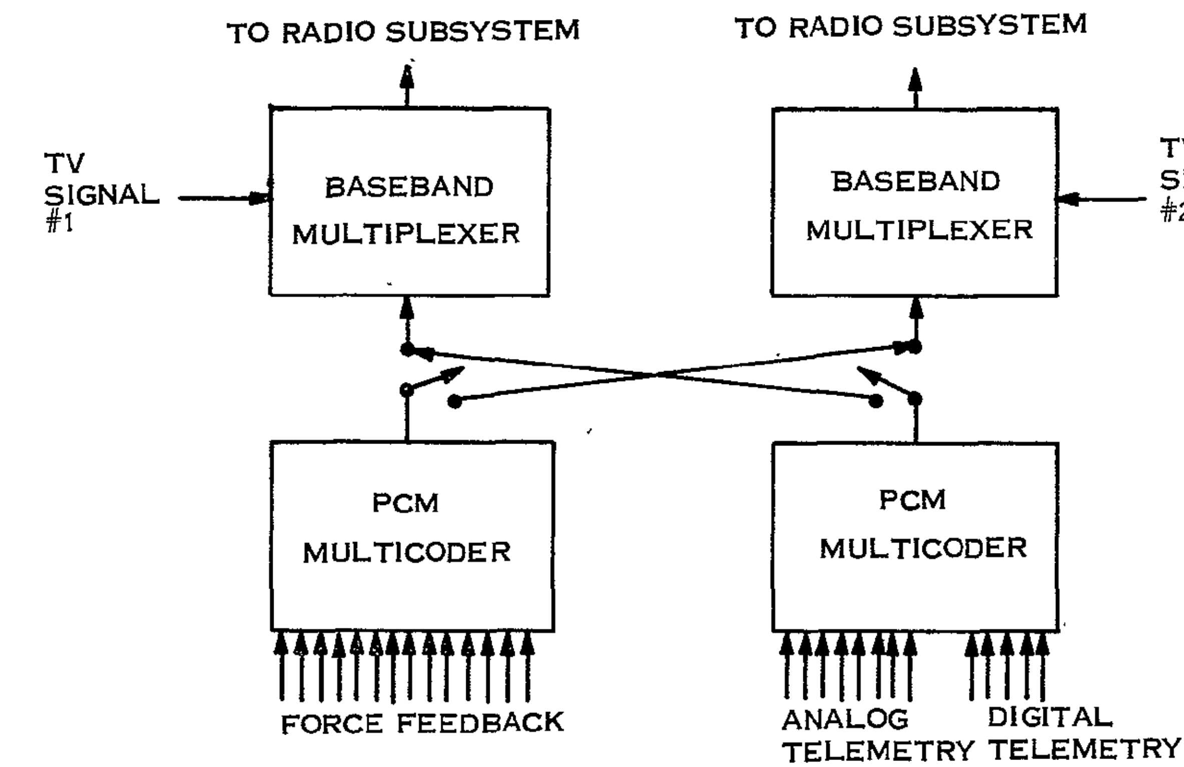


Figure 9. 4-8. Data Handling Subsystem Block Diagram

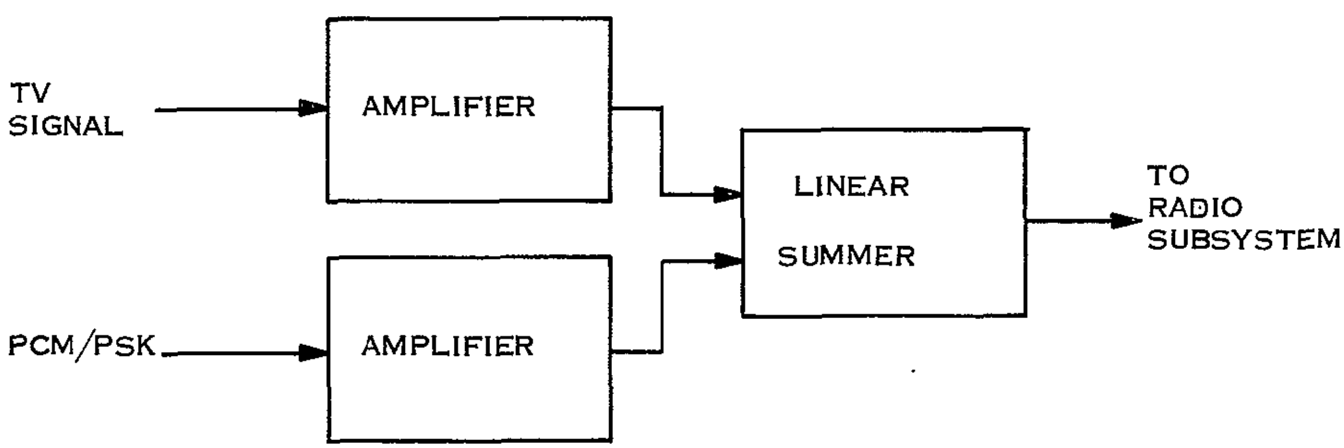


Figure 9. 4-9. Baseband Multiplexer Block Diagram

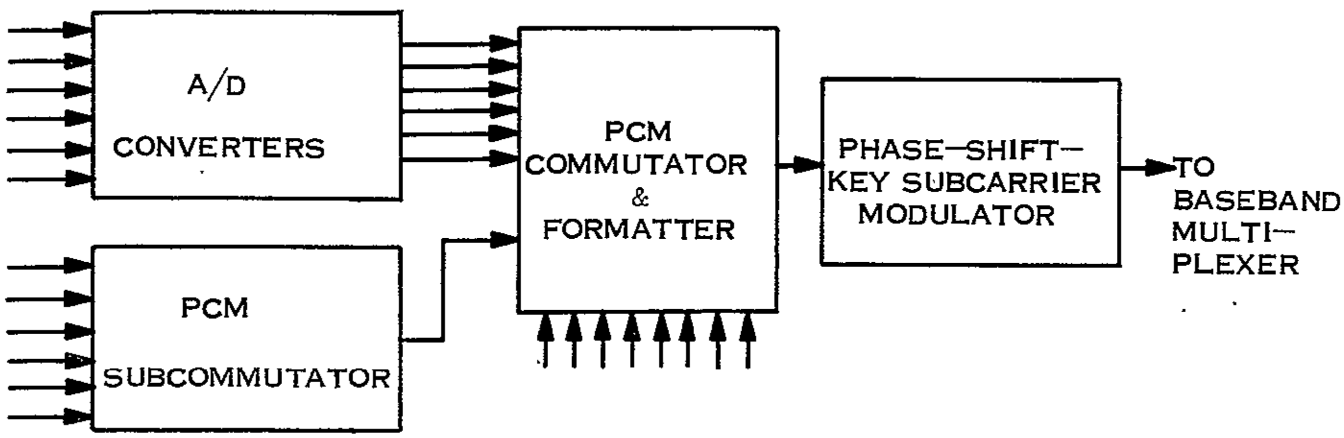


Figure 9. 4–10. PCM Multicoder Block Diagram

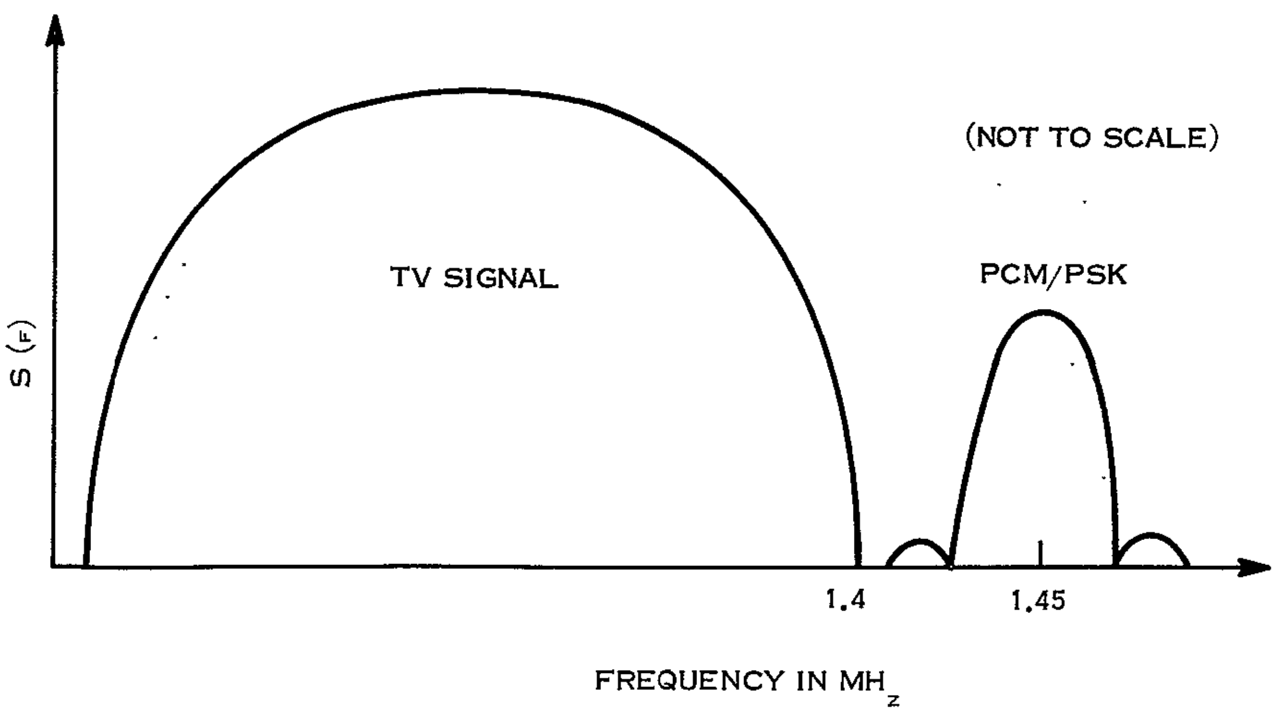


Figure 9. 4–11. Composite Baseband Power Spectrum Plot

9.4.6.1 Data Handling Subsystem Interface Definition

1. Inputs
    a. 14 force feedback signal channels
    b. 2 TV signals
    c. Engineering telemetry signals
    d. Operational commands.
2. Outputs
    a. 2 composite baseband signals
    b. Telemetry monitor signals.

9.4.7 RANGE AND RANGE RATE TRANSPONDER

A simplified block diagram of the S-Band transponder is shown in Figure 9.4-12. The transponder has a reference oscillator frequency (f_o) of 28.4167 mc which is used in a double conversion to generate an I-F signal of frequency ($80\,f_o - f_T$) where f_T is the ground transmitter frequency (in a zero Doppler case). This I-F signal is then separated into three frequency bands, centered at 1.4 mc, 2.4 mc, and 3.2 mc. Thus three different ground transmitter frequencies may be used, each one producing its own unique I-F frequency, allowing the transponder to be used simultaneously by one, two, or three ground stations. Each of the three channel filters consist of bandpass amplifiers, limiters, and squelch gates. The output of the channels in use are summed and used as subcarriers to phase modulate the transponder transmitter. The squelch gates keep unused channels from modulating the transponder transmitter with noise. It should be noted that the transponder transmitter frequency is derived from the reference oscillator (f_o) which was used to generate the subcarriers. This technique permits the ground stations to extract coherent 2-way doppler.

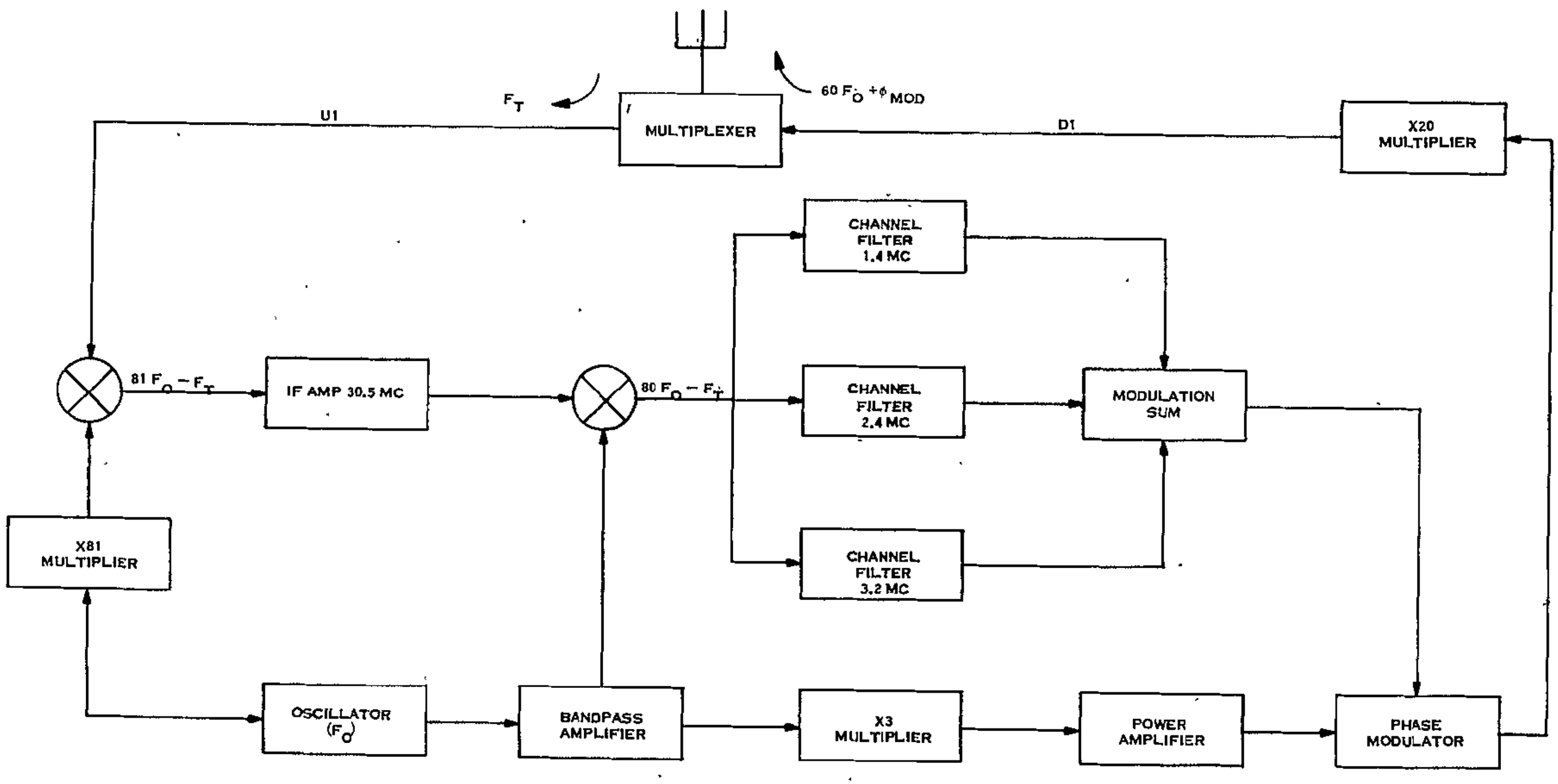


Figure 9.4-12. S-Band Transponder, Simplified Block Diagram

9.4.8 COMMUNICATION SUBSYSTEMS OPERATING MODES

Table 9.4-2 lists the various operating modes throughout the mission of the manipulator spacecraft. This applies for missions of all altitudes, assuming that even at synchronous altitude, the spacecraft will use a relay satellite.

9.4.9 COMMUNICATIONS SUBSYSTEM PERFORMANCE

This section lists the performance parameters, the required values for the various communication links, and the performance margins on each link. The performance requirements for each communication link and the link parameters will dictate the amount of effective radiated power (ERP) required from the Remote Manipulator Spacecraft and from the ground station. This will also depend on whether or not the communications signals are being relayed by a relay satellite. This section calculates the required C/N_o, received carrier power-to-noise spectral density ratio, at the receiver input for each signal.

9.4.9.1 Tracking

The tracking performance is a function of the carrier power-to-noise power density ratio at the STADAN facility tracking receiver input. Since the noise accumulated in the uplink signal is ngligible in comparison with downlink noise, the signal-to-noise density ratio for the overall tracking link is given by:

$$C/N_o = \frac{P_T G_T G_R L_m L_{fs} L_p}{KT_{eq}}$$

where:

P_T = transmitter power in the satellite

G_T = satellite antenna gain

G_R = STADAN antenna gain

L_m = circuit losses

L_{fs} = free space loss

Table 9.4-2. Manipulator Spacecraft Communication Subsystems Operating Modes

Time of Occurrence	Signal	Channel	Antenna	Destination	Relay or Direct
Following Liftoff through 1st Correction Burn	R R S/C TM	U1 D1 U2 D2 or D3	O O O O	A ST A ST	Direct Direct Direct Direct
Following Acquisition of DRS by Manipulator S/C High-Gain Antenna	R R TV1 + TM S/C	U1 D1 D2 U2	O O H H	A ST G A	Direct Direct Relay Relay
Following Visual Sighting of Target Satellite through End-of-Mission	TV1 + TM TV2 + FF S/C + M/T	D2 D3 U2	H H H	G G A	Relay Relay Relay
Loss of DRS by High-Gain Antenna	R R S/C TM	U1 D1 U2 D2 or D3	O O O O	A ST A ST	Direct Direct Relay or Direct Direct

Key to Symbols

Signals
- R - Range and Range Rate
- S/C - Manipulator Spacecraft Operational
- M/T - Manipulator/TV Camera Control Commands
- TV1 - TV Signal 1
- TV2 - TV Signal 2
- FF - Force Feedback
- TM - Engineering Telemetry

Channels
- U1 - Uplink: 2253 MHz
- U2 - Uplink: 1831.8 MHz
- D1 - Downlink: 1700 MHz
- D2 - Downlink: 2272.5 MHz
- D3 - Downlink: 2287.5 MHz

Antennas
- H - High-Gain Tracking
- O - Omini-Directional

Destination
- A - Manipulator Spacecraft
- ST - STADAN Facilities
- G - Ground Station

L_p = propagation loss

K = Boltzman's constant = 1.37×10^{-23} joules/^oK

T_{eq} = equivalent temperature of STADAN receiver

For C/N_o = 50 dB, the following errors occur (Reference 7):

Maximum Range Error = ± 8 meters

Maximum Range Rate Error = ± 0.01 meters/sec

The uplink signal from the ground station contains the spacecraft operational commands and the manipulator/TV camera control commands. Two subcarriers, one at 10 kHz for spacecraft commands and one at 100 kHz for the manipulator/TV camera commands, are frequency-shift-keyed by the PCM data streams, amplitude modulated by sinusoids at the respective bit rates, and linearly summed. The sum signal amplitude modulates a carrier. At the spacecraft, the carrier is removed by the AM command receiver and the PCM/FSK-AM signals are detected in the respective command decoders. The usual measure of performance is the probability of bit detection error in the decoders. Figure 9.4-13 shows P_e, the probability of bit error, versus E/N_o, the equivalent signal-to-noise ratio in a bandwidth equal to the data rate. Typically we require $P_e \leqq 10^{-5}$, so that $E/N_o \geqq 13.5$ dB. To obtain C/N_o from E/N_o, we use:

$$E = CT_b = \frac{C}{R_D}$$

where

T_b = bit duration (100% duty cycle)

R_D = bit rate

thus

$$C/N_o = R_D\,(E/N_o)$$

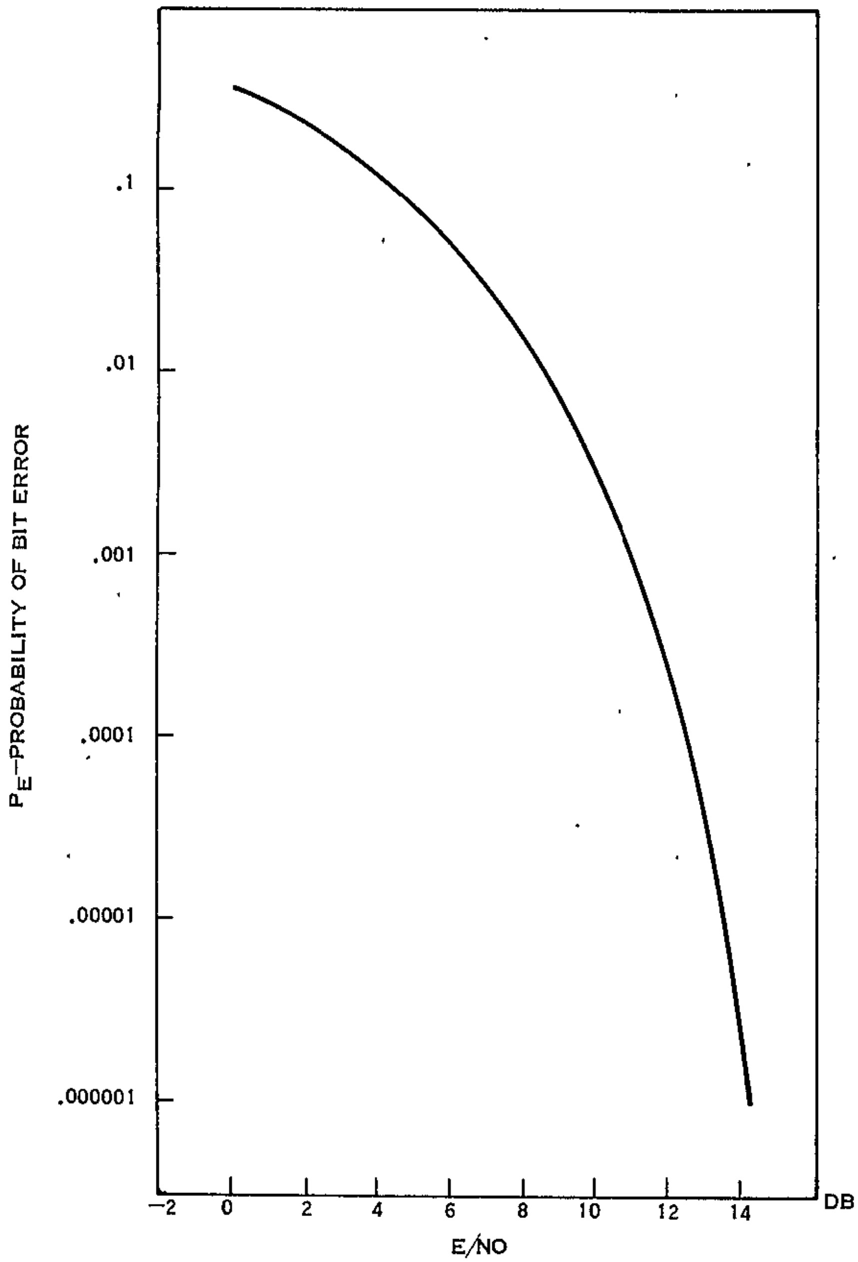


Figure 9.4-13. P_e versus E/N_o

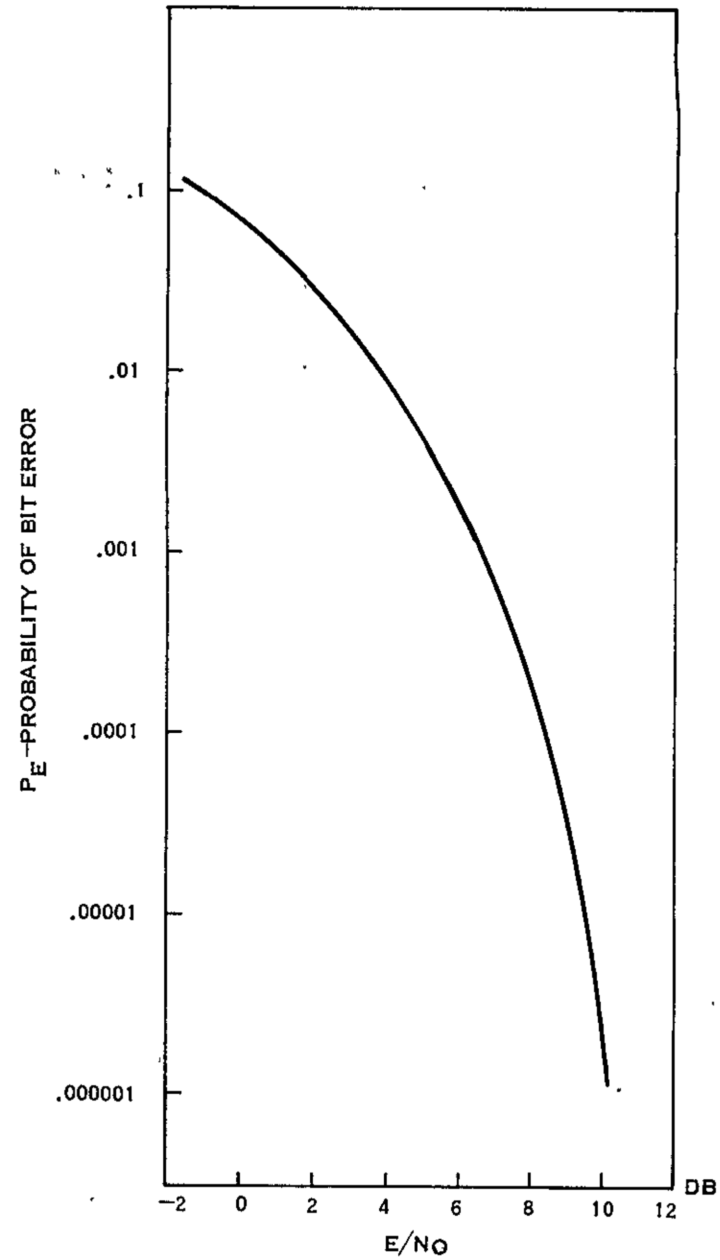


Figure 9.4-14. P_e versus E/N_o for PCM/PSK

Clearly, the required C/N_o for the high bit rate channel will be the more stringent requirement. Moreover, if we assume modulation indices such that the total power is split evenly between the high-bit rate and low-bit rate signals, we have:

$$(C/N_o) \text{ required} = (13.5 + 46 + 3) \text{ dB} = 62.5 \text{ dB-Hz}$$

Note that if this value of C/N_o is provided, the high bit rate decoder is at threshold but the low bit rate decoder is almost 36 dB above threshold.

9.4.9.3 Downlink

The downlink signals to the ground station contain the analog TV signals, PCM force feedback signals, and PCM engineering telemetry signals. Each TV signal is multiplexed with a subcarrier which has been phase-shift-keyed by one of the PCM data streams. Each composite signal FM-modulates a carrier frequency for transmission. At the ground station, the carrier is removed in the receiver, the video signal is recovered with a lowpass filter, and the PCM/PSK signal is detected in a telemetry decoder. The TV signal received quality is expressed in terms of a test-tone-power-to-noise-power ratio into the video display by:

$$S/N = \frac{3}{2} \left(\frac{\Delta f_v}{f_v} \right)^2 \frac{I(W)}{f_v} (C/N_o)$$

where

S/N = test tone/noise ratio

Δf_v = peak carrier frequency deviation due to TV signal

f_v = bandwidth of TV signal

$I(W)$ = noise weighting improvement

The following values apply:

S/N	$= 30$ dB	(Requirement)
Δf_v	$= 3.4$ MHz	(Limited by RF bandwidth)
f_v	$= 1.4$ MHz	(525 lines, 10 frames/sec)
$I(W)$	$= 2.7$ dB	(for $f_v = 1.1$ MHz)

thus

$$(S/N_o) = (30 - 2.7 - 9.5 + 61.4 \text{ dB}) = 79.2 \text{ dB}$$

The PCM data quality is expressed in terms of P_e, the probability of bit error. Figure 9.4-14 shows P_e versus E/N_o for PCM/PSK. However, our link is really PCM/PSK/FM with a performance relationship given by:

$$C/N_o \cong (E/N_o)_{PCM/PSK} R_D \left(\frac{f_s}{\Delta f_s}\right)^2$$

where

$(E/N_o)_{PCM/PSK}$	$= E/N_o$ required for a given P_e in a PCM/PSK link
R_D	= bit rate
f_s	= subcarrier frequency
Δf_s	= peak carrier frequency deviation caused by PCM/PSK subcarrier

The following values are applicable:

$(E/N_o)_{PCM/PSK}$	$= 9.2$ dB (for $P_e = 10^{-5}$)
R_D	= 1000 bps

$f_s \qquad = 1.45$ MHz

$\Delta f_s \qquad = 100$ kHz

thus

$$C/N_o \quad = (9.2 + 30 + 21.9) \text{ dB-Hz} = 61.1 \text{ dB}$$

One additional requirement exists for the downlink signal; the carrier receiver must operate about its threshold value which is given by

$$(C/N_o) \text{ threshold} = 10 \text{ dB} + 3 \text{ dB Margin} + 10 \log (B_{if})$$

for a discriminator, approximately 3 to 7 dB less for a FMFB receiver, and up to 10 dB less for a phase-lock receiver (Reference 6). B_{if} is the IF bandwidth of the signal. For $B_{if} = 10$ MHz

$$(C/N_o) \text{ threshold} = 83 \text{ dB (maximum)} = 73 \text{ dB (minimum)}$$

depending on the type of receiver used. Thus, the required downlink C/N_o per channel is determined by the receiver threshold value unless a phase-lock receiver is used at the ground station.

9.4.9.4 Relay Link Requirements

When a relay satellite is required to relay communication signals betwee the manipulator spacecraft and the ground station, the performance on the intermediate links must be higher than that for the overall link to allow for the additional accumulation of noise at the data relay satellite. For a relay link (Reference 5),

$$C/N_o = \frac{1}{1/(C/N_o)_1 + \left[1 + 1/(C/N_o)_1\right] / (C/N_o)_2}$$

where

C/N_o = overall C/N_o

$(C/N_o)_1$ = C/N_o on first leg alone

$(C/N_o)_2$ = C/N_o on second leg alone

Good engineering practice is to require

$$(C/N_o)_w = C/N_o + 1 \text{ dB}$$

$$(C/N_o)_s = C/N_o + 7 \text{ dB}$$

where

$(C/N_o)_w$ = C/N_o on weaker link

$(C/N_o)_s$ = C/N_o on stronger link

Then the desired overall C/N_o is obtained. The weaker link is that between manipulator spacecraft and the data relay satellite.

9.4.9.5 Link Calculations

The link margins were computed for each of the communications links:

Link	Max. Range (nm)	Margin dB)*
Spacecraft to Grd. Sta.	22,600	4.0
Grd. Sta. to Relay	22,600	25.1
Relay to Spacecraft	23,300	11.0
Grd. Sta. to Spacecraft	22,600	13.2
Spacecraft to Relay	23,300	3.5 to 7.3
Relay to Grd. Sta.	22,600	3.1 to 10.0

*Variations in margin result from different types of FM demodulators.

9.5 PROPULSION AND ATTITUDE CONTROL SUBSYSTEM

The remote manipulator spacecraft has as its specific mission the repair and refurbishment of other satellites while in earth orbit. To accomplish this a propulsion system is required that will provide the impulse necessary for rendezvous, maneuvering, docking, and stabilization. The system selected to perform all of these functions uses a hydrazine monopropellant operating in a blowdown mode. The thrusters used include two 26-lb thrust rendezvous engines (Figure 9.5-1), eight 2-lb thrusters (Figure 9.5-2), and sixteen 1/2-lb thrusters (Figure 9.5-3). The location of the rendezvous engines is varied from mission to mission so that the thrust is through the CG. The smaller thrusters used for attitude control and maneuvering have fixed locations. They are operated in a pulse width modulated mode and are operated in coupled parts providing pure torque to overcome the effects of any offset in the CG location. Their location was chosen to minimize plume impingement on the target vehicle.

9.5.1 SYSTEM REQUIREMENTS

The remote manipulator space vehicle requires propulsion for rendezvous, for attitude control and for docking, undocking and maneuvering around the target satellite. The 3 sigma ΔV requirements for rendezvous as a function of altitude are:

Altitude (nm)	ΔV for Rendezvous (ft/sec)
100	184
200	182
430	194
1000	200

These ΔV's compensate for the DSV-2L launch vehicle errors in apogee (altitude and velocity), inclination and orbit period for the given altitudes. A tracking error of 18.5 ft/sec must be added to these ΔV's for total rendezvous velocity requirements.

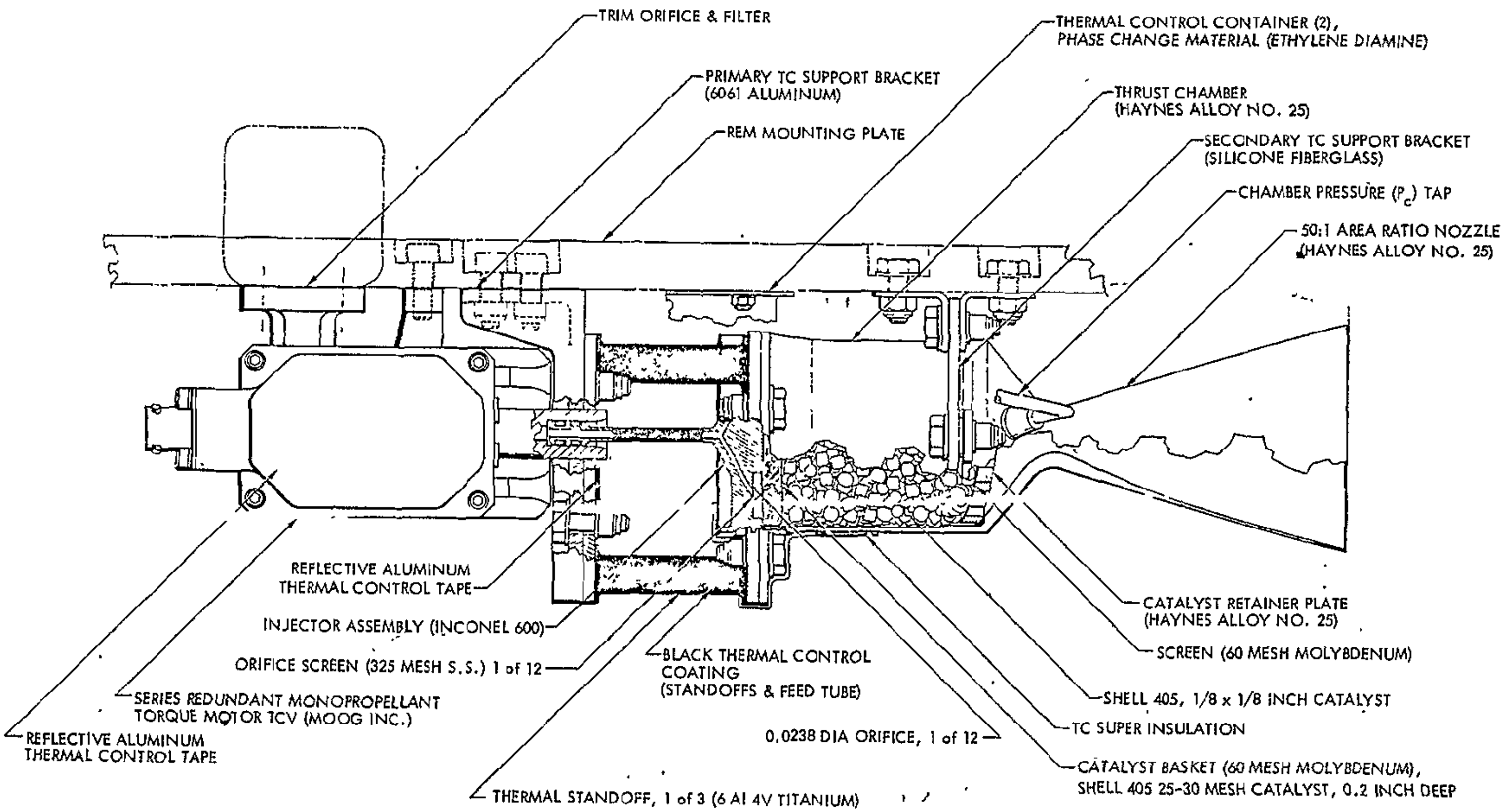


Figure 9.5-1. 26-lb Monopropellant Hydrazine Rocket Engine Assembly for Rendezvous

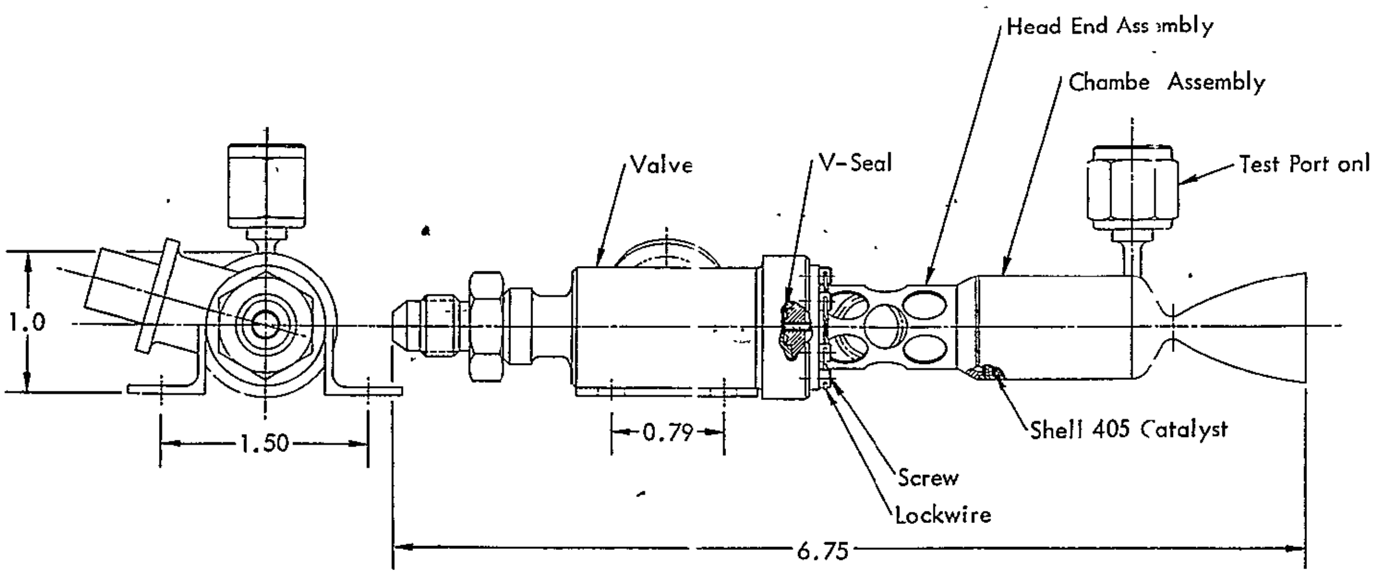


Figure 9.5-2. 2-lb Monopropellant Hydrazine Rocket Engine Assembly for Attitude Control and Maneuvering

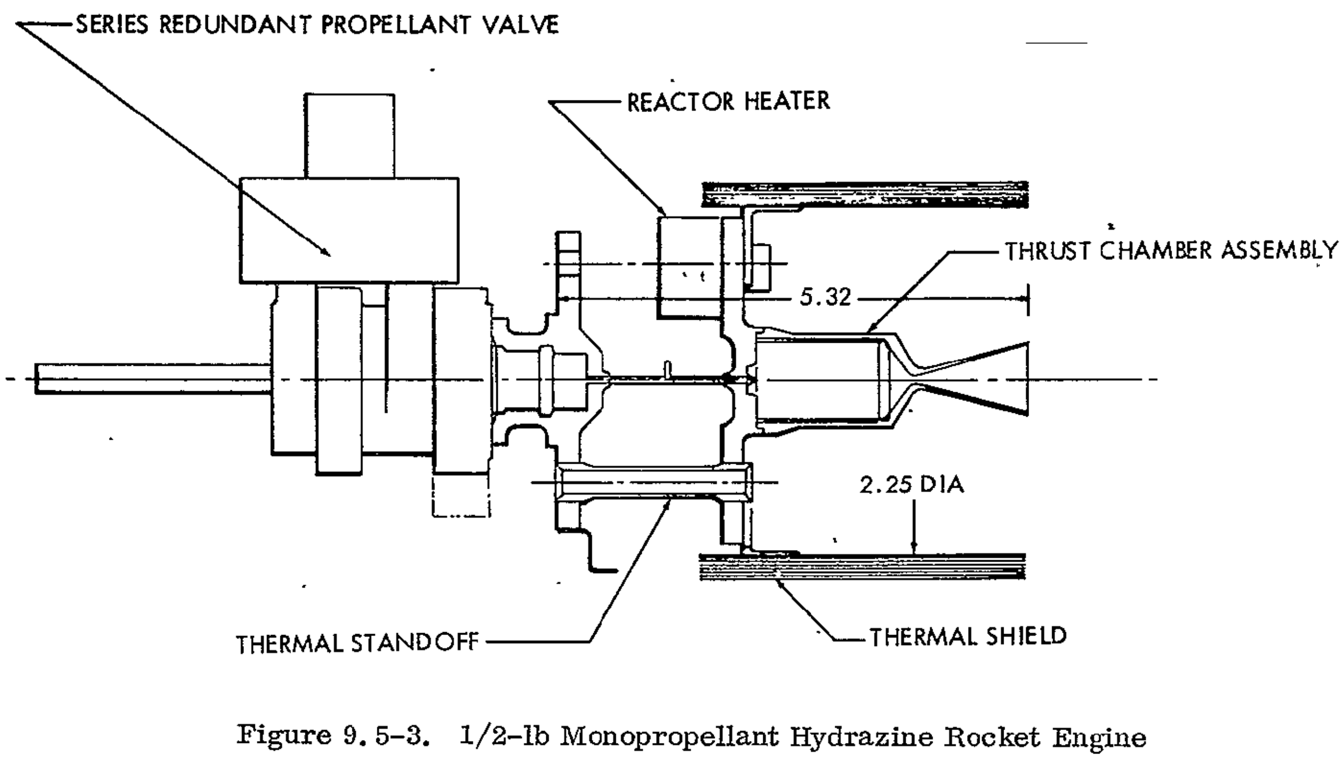


Figure 9.5-3. 1/2-lb Monopropellant Hydrazine Rocket Engine Assembly for Attitude Control and Maneuvering

9.5.1.1 Propulsion Requirements

The propulsion requirements for other spacecraft functions include:

1. Docking: 1 ft/sec approach and stop (10 times for a total of 20 ft/sec)

2. Undocking: 5 ft/sec (once)

3. Separation Tipoff Rates: 1°/sec (0.0174 rad/sec) in all axes

4. Rendezvous Engine Burn: Thrust vector misalignment of 1/2 inch radius about the CG

5. Attitude Control

   Limit cycle excursion ($\pm 3^{\circ}$ nominal)

   $\pm$0.05 rad roll, pitch

   $\pm$0.1 rad yaw

   Thruster on-time minimum: 50 millisec

   I_{roll}, I_{pitch}

   2250 slug ft^2 (docked)

   75 slug ft^2 (undocked)

   I_{yaw}

   560 slug ft^2 (docked)

   75 slug ft^2 (undocked)

   Thruster moment arm: 4.5' roll, pitch, yaw

   Thruster force (total for translation): 4 lb
   Thruster force (total for rotation)

   2 lb roll
   1 lb pitch, yaw

   Mission duration: 10 days

The propellant required for each of the 7 identified functions:

1. Rendezvous Impulse

   Vehicle basic weight: 968.1 lb

   ΔV: 200 ft/sec

   Tracking error: 20 ft/sec

   $$\text{Impulse} = \Sigma\ \Delta V \times \frac{W}{g} = 220 \times \frac{968.1}{32.1} = 6600 \text{ lb/sec}$$

2. Docking

   $$\text{Impulse} = 20 \times \frac{968.1}{32.2} = 600 \text{ lb/sec}$$

3. Maneuvering about Target Vehicle

   Maximum circular velocity (radian/sec.)

   $$F = \frac{W}{g}\ \frac{V^2}{r} \qquad \therefore V = \sqrt{\frac{(F)\ (g)\ (r)}{W}}$$

   Assume 10 ft/radius circle and a maximum thrust of 4 lb

   $$V^2 = \frac{(4)\ (32.2)\ (10)}{968.1 \text{ lb}} = 1.33\ \frac{ft^2}{sec^2}$$

   $$V = 1.152 \text{ ft/sec}$$

   $$\text{Time for 1 rev} = \frac{20}{1.152} = 54.5 \text{ sec}$$

   Impulse required for 5 revs = 5 x 4 x 54.5 = 1090 lb/sec

   Total Docking Impulse = 1090 + 600 = 1690 lb/sec

4. Undocking (Separating from target vehicle at 5 ft/sec)

   $$\text{Impulse} = \frac{5 \times 968.1}{32.2} = 150 \text{ lb/sec}$$

5. Initial Stabilization (Tipoff Rate Removal) and Alignment for Rendezvous Thrusting

a. Tipoff rate removal (angular acceleration $= \alpha = \frac{FR}{I}$

$$\alpha_R \text{ (roll)} = \frac{(2)(4.5)}{75} = 0.12 \text{ rad/sec}^2$$

$$\alpha_P = \alpha_Y \text{ (pitch and yaw)} = \frac{(1)(4.5)}{75} = 0.06 \text{ rad/sec}^2$$

$$\text{Engine on time} = t_{EO} = \frac{\Delta\dot{\theta}}{\alpha}$$

$$t_{EOR} = \frac{0.0174}{0.12} = 0.145 \text{ sec}$$

$$t_{EOP} = t_{EOY} = \frac{0.0174}{0.06} = 0.290 \text{ sec}$$

$$\text{Total Impulse} = \Sigma F \cdot t_{EO}$$

$$= 2(0.145) + 2(1)(0.290) = 0.290 + 0.580 = 0.87 \text{ lb-sec}$$

b. Alignment for Rendezvous Thrusting

$$\text{Rotation Rate } (\dot{\theta}) \geq \alpha \, t_{EO_{min}}$$

$$\dot{\theta}_R \geq 0.12 (0.05) = 0.006 \text{ rad/sec} = 0.344^{\circ}\text{/sec}$$

$$\dot{\theta}_P = \dot{\theta}_Y \geq 0.06 (0.05) = 0.003 \text{ rad/sec} = 0.172^{\circ}\text{/sec}$$

Assume $\dot{\theta} = 1^{\circ}$/sec $= 0.0174$ rad/sec $=$ Tipoff rate

Then, Total Impulse = 2 Total Impulse for Tipoff rate removal

c. Summation

Total Impulse = 3 total impulse for Tipoff rate removal

$$= 3(0.87) = 2.61 \text{ lb-sec}$$

6. Stabilization During Rendezvous Engine Firing

Rendezvous engine firing time

$$\frac{I_T}{F} = \frac{6600 \text{ lb/sec}}{23 \text{ lb (avg. thrust)}} = 287 \text{ sec}$$

Assume the 1/2-inch engine misalignment from the CG is located at 45° to the pitch and yaw axes. The disturbance (L_D) will then be:

$$L_{DP} = L_{DY} = F_D \cdot R_D = 23 \text{ lb} \times \frac{1}{24} \text{ ft} \times 0.707 = 0.679 \text{ ft-lb}$$

$$L_{DR} \cong 0$$

The control torque (L_C) in pitch and yaw is:

$$L_{CP} = L_{CY} = F_C \cdot R_C = 1 \text{ lb} \times 4.5 \text{ ft} = 4.5 \text{ ft-lb}$$

The vehicle will be operating in a limit cycle mode prior to rendezvous engine firing. The behavior of the vehicle will depend upon the conditions existing at the initiation of firing (θ_O, $\dot{\theta}_O$), the switching logic parameters and the deadband limits. After a short period of time (less than 30 sec) the vehicle will settle into a new limit cycle in pitch and yaw near one of the deadband limits with the duration of the control nozzle operation equal to the ratio of the disturbance to the control torque:

$$\frac{L_D}{L_C} = \frac{0.679}{4.5} = 0.151$$

The impulse requirements during the transient will be relatively small, on the order of 10% of the steady-state operation.

$$\text{Total Impulse} = 1.1 \times F \cdot \text{Number of axes} \cdot \frac{L_D}{L_C} \cdot t_{EO}$$

$$= 1.1 \times 1 \times 2 \times 0.151 \times 287 = 95 \text{ lb-sec}$$

7. Limit - Cycle Stabilization

$$\text{Total Impulse} = \frac{F \cdot T_{TOT} \cdot \dot{\theta}^2_{LC}}{\frac{\Delta\theta_{DB}}{2} \alpha}$$

$$= \frac{F \cdot t_{EO} \cdot \epsilon_{LC} \, T_{TOT}}{\Delta\theta_{DB}}$$

a. During Rendezvous Coast

Assume 2 days maximum = 2 x 3600 x 24 = 178, 800 sec

$$\dot{\theta}_{RLC} = \alpha_R \frac{t_{EO}}{2} = 0.12 \times \frac{0.05}{2} = 0.003 \text{ rad/sec} = 0.172^\circ/\text{sec}$$

$$\dot{\theta}_{PLC} = \dot{\theta}_{YLC} = 0.06 \times \frac{0.05}{2} = 0.0015 \text{ rad/sec} = 0.086^\circ/\text{sec}$$

$$I_{TR} = \frac{2 \text{ lb} \times 0.05 \text{ sec} \times 3 \times 10^{-3} \text{ rad/sec} \times 178,000 \text{ sec}}{0.1 \text{ rad}}$$

$$= 534 \text{ lb/sec}$$

$$I_{TP} = \frac{1 \text{ lb } (.05 \text{ sec}) (1.5 \times 10^{-3} \text{ rad/sec}) \times 178,000 \text{ sec}}{.1 \text{ rad}}$$

$$= 66.7 \text{ lb/sec}$$

Total Impulse Coast = 794.2 lb/sec

b. During Docking

Assuming axes of docked assembly are parallel to vehicle control axes, 8 days maximum (686, 000 sec) and degraded thrust on control nozzles:

$$\dot{\theta}_{RLC} = \alpha \frac{t_{EO}}{2} = \frac{F.R.}{I} \cdot \frac{t_{EO}}{2} = \frac{1.6\ (4.5)\ (0.05)}{2250 \times 2}$$

$$= 8 \times 10^{-5} \text{ rad/sec} = 4.59 \times 10^{-3} \text{ deg/sec}$$

$$\dot{\theta}_{PLC} = \frac{0.8\ (4.5)\ (0.05)}{2250 \times 2} = 4 \times 10^{-5} \text{ rad/sec} = 2.3 \times 10^{-3} \text{ deg/sec}$$

$$\dot{\theta}_{YLC} = \frac{0.8\ (4.5)\ (0.05)}{560 \times 2} = 1.61 \times 10^{-4} \text{ rad/sec} = 9.22 \times 10^{-3} \text{ deg/sec}$$

$$I_{TR} = \frac{1.6 \times 0.05 \times 8 \times 10^{-5} \times 686{,}000}{0.1} = 44 \text{ lb/sec}$$

$$I_{TP} = \frac{0.8 \times 0.05 \times 4 \times 10^{-5} \times 686{,}000}{0.1} = 11 \text{ lb/sec}$$

$$I_{TY} = \frac{0.8 \times 0.05 \times 1.61 \times 10^{-4} \times 686{,}000}{0.2} = 22.2 \text{ lb/sec}$$

Total Impulse = 77 lb/sec

For the case where the docked assembly principal axes are not parallel to the vehicle control axes, there will be an increased fuel requirement for two reasons: the controls about the assembled axes will be reduced in effectiveness and cross-coupling between axes will increase the control requirements about the other axes. The increase will be a function of the ratios of inertias and the misalignment angles. Assume a 45 degree misalignment in all three axes causes a 50 percent reduction in effectiveness and a 50 percent cross-coupling. This will double the impulse requirements.

∴ Total Impulse = 154 lb/sec

c. <u>Summation</u>

$$\text{Total Impulse} = I_{T\ \text{undocked}} + I_{T\ \text{docked}}$$

$$= 794.2 + 154 = 948.2 \text{ lb/sec}$$

9.5.1.2 Total Propellant Requirements

The propellant requirements (W_P) for each function will depend upon both the total impulse (I_T) and the specific impulse (I_{SP}) of the engine used in that mode. The following calculations are based upon expected values of both.

$$W_P = \Sigma \frac{I_T}{I_{SP}}$$

1. Rendezvous: $\frac{6600}{225}$ lb-sec = 29.4 lb
2. Docking, Undocking, Initial Stabilization and Attitude Control during Rendezvous Engine Firing: $\frac{1690 + 150 + 2.6\ 95}{200} = \frac{1937.6}{200} = 9.7$ lb
3. Limit Cycle: $\frac{948.2}{100} = 9.5$ lb

The total propellant requirement is:

$$29.4 + 9.7 + 9.5 = 48.6 \text{ lb}$$

There are other causes for increased fuel consumption such as thrust mismatch between nozzle pairs, nozzle misalignment, attitude control fuel used during docking, undocking and manuevering, allowance for leakage and unusable fuel trapped in the tank. In addition, external and internal disturbance torques will be acting on the vehicle and the inertias will change as the arms are moved. Assume a safety factor of 30 percent to allow for all these sources.

$$\therefore\ W_{PT} = 1.3\ W_P = 1.3\ (48.6) = 63 \text{ lb}$$

9.5.1.3 Propellant Tank Requirement

Since the system functions in a blowdown mode, the propellant and part of the pressurant gas is contained in one tank separated by a bladder. The rest of the pressurant gas is contained in a separate tank. The system blowdown pressure ratio requires that the total volume be approximately 50 percent propellant.

Volume required to contain the 63 lb of fuel is approximately 0.91 cubic ft. Pressurant gas will occupy another 0.85 cubic ft. Total volume required in the tank system is 1.76 cubic ft. Two off-the-shelf tanks were chosen to provide the required volume, one 16.5 I.D. tank manufactured by P.S.I. and one 11.0 diameter tank by the same company. For the heavier payload missions, larger tanks (up to 18.0 in.) are used.

9.5.1.4 Power Requirements

1.	Pressure Transducer (1)	0.15 watt
2.	Temperature Transducer (1)	0.15 watt
3.	Squib Valve (1)	35 watts for 10 millisec
4.	Control Valves (Rendezvous) (2)	35 watts (max.) at any one time
5.	Control Valves (Rendezvous) (2)	14 watts

Additional power will be required, above that given for the component operation, to supply strip heaters for those engines and lines that are outside the thermal controlled environment of the spacecraft. This additional heat will require 25 additional watts.

9.5.2 DESIGN APPROACH

The selected rendezvous engine is a 26-lb thrust hydrazine unit developed by Rocket Research Corp (RRC) for use in another program (Figure 9.5-1). This engine was selected because of its development status and because it can provide small impulse bits to achieve the accurate velocity correction required for rendezvous. This thrust level will provide approximately 1 ft/sec acceleration to the vehicle ΔV of 0.05 ft/second.

Engines with thrust levels of 2 lb and 1 lb were selected for attitude control and maneuvering. These thrust levels were selected to provide the translational and rotational acceleration necessary to achieve the desired command authority for docking in the thruster configuration chosen for the manipulator spacecraft. Tests have shown that good docking capability requires the vehicle to possess translation accelerations of 0.1 to 0.3 ft/sec and rotational acceleration of 1 to 3 deg/second.

The 2-lb thrust engines are manufactured by TRW (Figure 9.5-2) and qualified on the Intelsat III programs. Because there is no currently qualified engines in the 1-lb thrust range, the 1-lb thrust is achieved by pairing 1/2-lb thrust engines. The 1/2-lb thrust engines are manufactured by Rocket Research Corporation (Figure 9.5-3) and qualified on a classified program. The thrust levels of 2 and 1/2 lb are maximums in this application and the thrust will depreciate to approximately 1.2 and 0.27 lb at the end of the blowdown cycle. Figure 9.5-4 shows location and thrust levels of engine of the proposed propulsion system.

9.5.3 DESCRIPTION OF SUBSYSTEM

A schematic of the subsystem is shown in Figure 9.5-5. It includes the following components:

1. Two 26-lb thrust engines
2. Eight 2-lb thrust engines
3. Sixteen 1/2-lb thrust engines
4. One gas fill valve (JPL)
5. One propellant fill valve (JPL)
6. One normally closed pyro valve (Pyronetics)
7. One filter (Vacco)
8. One propellant tank (16.6 in. dia.) (Pressure System Ind., S/N 80081-1)
9. One pressure transducer (Bourris, S/N 541)
10. One temperature transducer (Transonics, S/N 4086A)
11. Required plumbing.

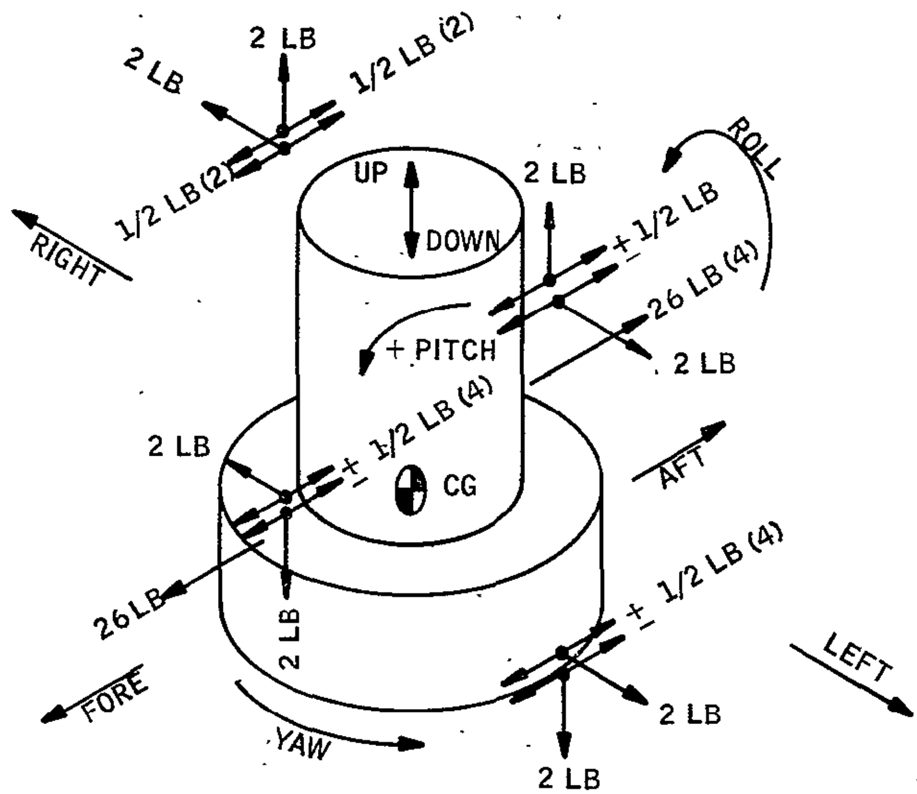


Figure 9.5-4. Rendezvous and Attitude Control Thruster Location

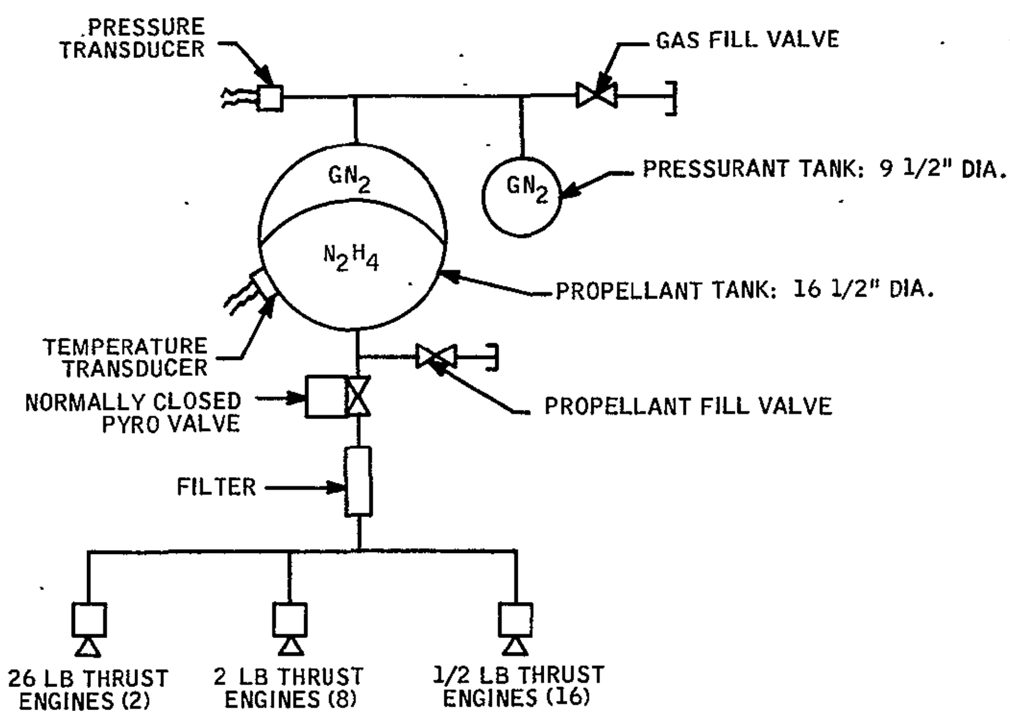


Figure 9.5-5. Propulsion Subsystem Schematic

9.5.4 WEIGHT SUMMARY

Item	Wt/Unit (lb)	Units	Total Weight (lb) Mission A, B, C	Mission D	Mission E
Tankage	16.8/29.2	1	16.8	16.8	29.2
Ordnance Valve	0.4	1	0.4	0.4	0.4
Fill and Drain	0.3	2	0.6	0.6	0.6
Filter	0.2	1	0.2	0.2	0.2
Thruster (26-lb)	3.8	2	7.6	7.6	7.6
Thruster (1/2 to 2-lb)	0.9	24	21.6	21.6	21.6
Pressure Transducer	0.4	1	0.4	0.4	0.4
Temperature Transducer	0.4	1	0.4	0.4	0.4
	Hardware Component Weight =		48.0	48.0	60.4
	Lines and Brackets = 15%	=	7.2	7.2	9.1
	Dry Hardware Weight		55.2	55.2	69.5
	Propellant Weight		63.0	77.6	123.0
	Gas Weight		1.2	1.3	2.0
	Total Subsystem Weight		119.4	134.1	194.5

The subsystem weight varies because of variations in manipulator payload weights. The variations in total spacecraft weight for each mission are listed below.

Mission	All-Up Remote Manipulator Spacecraft Weight (lb)
A (Nimbus A to C)	1259.9
B (DBS)	1267.1
C (OSO)	1221.1
D (OAO)	1576.1
E (Nimbus D and E)	2275.1

9.5.5 PROBLEM AREAS

9.5.5.1 Engine Exhaust Plume Impingement

To minimize plume impingement heating and reaction on the target vehicle the thrusters were located a minimum distance equal to the radius of the manipulator spacecraft from the target vehicle. This radius is approximately 3 feet. At this distance, a 1/2-lb thrust engine plume has the following characteristics:

1. Density: 10^{-7} slug ft^3
2. Pressure: 1.4×10^{-4} psi
3. Temperature: $26^{o}K$
4. Velocity: 7200 ft/sec
5. Composition: NH_3, H, N_2

The effects of ammonia on the target vehicle were not examined; however, spacecraft have been flown utilizing hydrazine engines and have not experienced and difficulty because of ammonia in and around the spacecraft. Therefore, at this time, there is no reason to believe that the ammonia will present any problems.

9.6 ATTITUDE CONTROL REFERENCE SUBSYSTEM

The Attitude Control Reference Subsystem (ACRS) has as its principal components an inertial reference unit consisting of three body-bound rate integrating gyros and an electronics package designed to provide all necessary computation, amplification, integration, addition, logic and switching functions.

The remote manipulator spacecraft attitude relative to the target spacecraft is displayed to the spacecraft operator by way of the stereoptical video presentation that is part of the manipulator master station and originates at the paired video cameras mounted on the manipulator spacecraft. Deviations in the inertial rates of the spacecrafts are sensed by the gyros and the resulting error signals are transformed by the electronics package into firing commands to the appropriate thrusters. Stable attitude rates (nominally zero with respect to inertial space) are automatically maintained without ground operator attention. When changes in attitude are required, the operator must maneuver a control stick generating rate commands proportional to the stick travel. The flight commands are transmitted to the manipulator spacecraft and cause the inertial reference null to rotate. The electronics package generates the required thruster commands to rotate the spacecraft and follow the changing null position. When the operator requires changes in spacecraft velocity, he must again maneuver a control stick to generate thruster commands.

The thrusters operate in a pulse-width-modulated mode and the thruster pulse width and therefore vehicle acceleration will be proportional to the control stick travel. The body-bound gyros will sense any rotations that occur due to translation-thrust/CG-offset. The error signals from the gyros will be added to the translation thrust commands and the thruster pulse-width modulation will be differentially varied to decouple rotation from translation. The operator control stick is spring loaded and detented at the zero command position so that upon release of the control stick, the transmitted commands will return the spacecraft to an attitude hold operating mode and maintain the attitude acquired by the spacecraft at the moment the rate command returned to zero.

Additional sensors that are part of or are used by the attitude control reference system are illustrated in Figure 9.6-1. These sensors include a pair of IR horizon scanners, tracking antenna gimbal pickoffs and the video system gimbal pickoffs. The IR scanners provide local vertical information and their output is used directly by the ACRS as one of the inputs for the initial acquisition mode. The tracking antenna gimbal pickoff information supplies spacecraft yaw information but requires processing in the ground station. It is also used after docking to provide attitude error information to the ACRS. The video system gimbal pickoffs are used by the ACRS in a backup mode to determine the local vertical and ground track by observing relative motion of earth features and position of the earth's horizon.

9.6.1 ACRS MODES

The ACRS is in use during the entire mission and it must provide a variety of functions. At injection, it must measure and null the tipoff rates. It must then hold the spacecraft in place while the tracking antenna searches for and acquires the relay satellite. When the orbits of the manipulator and target spacecraft are determined, the commanded rendezvous attitudes must be held while the rendezvous propulsion system burns take place. This may require two to four burns with pointing held to within one degree. Docking has probably the most active and challenging requirements. The ACRS must respond to the operator's translation and rotation commands while decoupling rotation from translation. The ACRS response and the thrust levels of the attitude control thrusters must be relatively high to provide a 1-to -3 degree/sec command authority. As the two vehicles dock, the center of gravity and principal axes are suddenly changed. The change in inertia along with a small but finite docking impulse can cause the high gain tracking antenna to lose lock. With the loss of antenna lock, the video, telemetry and command system is also lost so an automatic target satellite despin and reacquisition mode is required. After the relay satellite is reacquired, the maintenance phase of the mission can begin and the ACRS must switch the thruster logic to provide for attitude control with a widely offset CG (Figure 9.6-2).

In this mode the 2-lb thrusters, heretofore used for translation, provide attitude control torques for roll, ptich and yaw. The attitude of the spacecraft pair is mostly a function of relay satellite and sun position during the maintenance phase so the tracking antenna gimbal

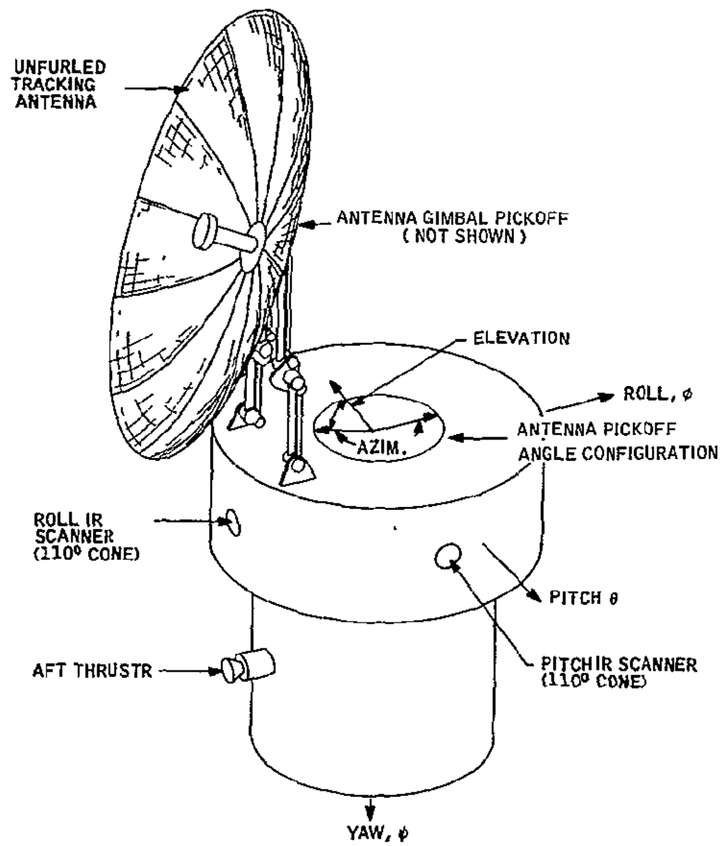


Figure 9.6-1. Location of IR Scanners and Tracking Antenna

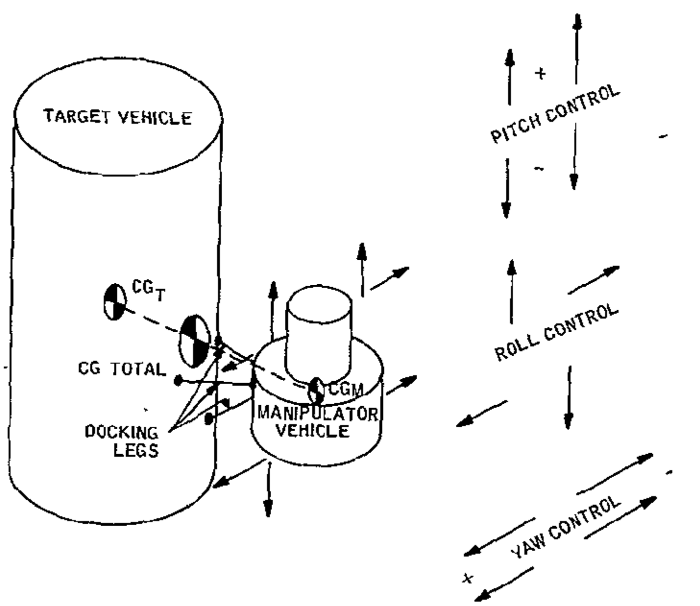


Figure 9.6-2. Thruster Operation for Attitude Control With Offset CG

pickoffs can be used as attitude error sensors during this phase. The attitude control system switching lines can be computed on the ground to take sun location into consideration. Each of the relay satellites will come into and disappear from view and the tracking antenna must be driven from one to the other. This will require driving the antenna in both azimuth and elevation while the ACRS provides the antenna with a stable platform. A listing of the various ACRS modes appears in Table 9.6-1.

9.6.2 ACRS DESIGN APPROACH

One ACRS design requirement called for continuous operation for the entire 10-day mission, if need be. The 10-day continuous opeation coupled with a primary battery for a power source caused the ACRS power consumption to be a significant consideration to keep the total weight down. Weight and volume of the ACRS was another consideration and led to the use of a body-bound subsystem over a gimballed subsystem. The desire to minimize total system weight and cost led to an analysis of the need for a rendezvous radar system. The analysis appearing in Appendix E revealed that the target and manipulator spacecraft ephemerides and the manipulator spacecraft attitude would be known to sufficient accuracy to achieve rendezvous without a rendezvous radar. The analysis indicated that with either a three or four impulse rendezvous scheme the manipulator spacecraft will be within a 0.1 nm (3σ) error volume. Rendezvous using the video system as a primary instrument from 0.1 nm is assumed possible based upon Gemini experience.

The need for accurate pointing to direct the rendezvous thrusters and also to direct the tracking antenna during relay satellite transfer caused the IR horizon scanners and the antenna gimbal pickoffs to be included in the ACRS. The horizon scanners will enable the spacecraft attitude to be set in pitch and roll to the local vertical. Knowing the relative position of the manipulator and data relay satellite establishes a tracking antenna search cone angle. Once the tracking antenna is locked on to the data relay satellite the azimuth pickoff can be used to establish the manipulator spacecraft's yaw attitude with respect to orbit plane, the earth or inertial space. The azimuth angle accuracy is reduced and becomes ambiguous at high elevation angles but operational constraints can insure that yaw attitude measurements are taken only at low elevation angles.

TABLE 9.6-1. Outline of Operational Phases and Attitude Control Modes

Operational Phase	Attitude Control Mode	Sensors Used	Comments
Initial Stabilization	Inertial Reference	Gyros (Position & Rate)	Antenna Search
Inertial Reference Update	Fine Sensing	Pitch & Roll IR Scanners Antenna Azimuth Angle Gyros for Rate Info.	Bias, if required, when switching to inertial reference
Orbit Correction Thrusting	Inertial Reference	Gyros (Position & Rate)	Bias as required
Docking	Rotational Rate Translational Accel.	Gyros & Rate Bias each loop returns to angular position mode at zero control angle. Proportional control for linear acceleration.	Visual observation via TV for attitude information.
Target Satellite Despin	Inertial Reference	Gyro (Position & Rate)	Inertial Ref. removes rates primarily. Reacquisition required.
Reacquisition	Fine Sensing With	IR Scanners and gyros Antenna Azimuth Angle	Required after spin. Available in case of inadvertent loss of relay satellite.
Maintenance	Coarse Antenna Pointing	Azimuth and Elevation Angles	Resolve angles to A/C Ref. frame on ground, and transmit firing commands.
Relay Satellite Antenna Transfer	Fine Sensing with Acquisition Logic	IR Scanners & Gyros Antenna Azimuth Angle	Antenna slews to calculated position of new satellite and searches.

The manipulator spacecraft is required to carry the repair and refurbishment loads for each of the missions and the total system weight and its CG also vary as a consequence. While every attempt has been made to keep the CG variation small, it still varies over a large distance. The rendezvous thruster mounts are varied to follow the CG shift and thereby reducing the total gas consumption that would be required to overcome any thruster/CG offset. The attitude control thrusters; however, are fixed in place and there will be thrust/CG offsets whenever a translation command is issued. To overcome this problem, the attitude control thrusters are operated in a pulse-width modulated mode.

The thruster pulse-width magnitudes will be set as an inverse function of the thruster CG offset when translation maneuvers are commanded. The different thrust levels will allow the vehicle to translate while maintaining small rotational torques. The rate gyros will sense any residual rotation and their signals will be added to differentially modify the pulse-width and null out the spacecraft rotation.

9.6.3 ACRS DESCRIPTION

9.6.3.1 Inertial Reference Mode

Figure 9.6-3 is a block diagram of the ACRS operating in the inertial reference mode. This is the mode used during initial stabilization, rendezvous thrusting, tracking antenna transfer and docking maneuvers. The output of the three rate gyros is integrated and compared with a position signal resulting from integrating the rate commands received from the ground. The attitude and attitude rate signals are then summed and threshold detected before being sent to the thruster firing logic circuits.

The particular configuration shown is based upon spring restrained rate gyros to take advantage of their cost and low power requirements and small size. If attitude accuracy becomes more important or if a torque rebalance gyro becomes available that is competitive from a cost and power consumption standpoint, the system would be changed slightly. The rate commands would then be used to torque the gyros directly.

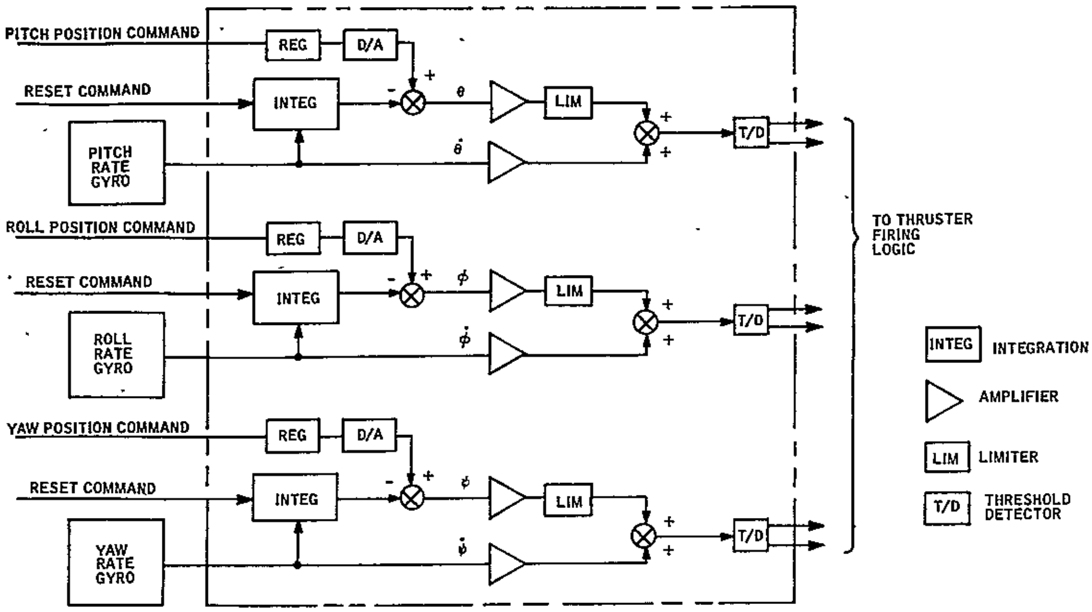


Figure 9.6-3. Attitude Control Reference Subsystem - Inertial Mode

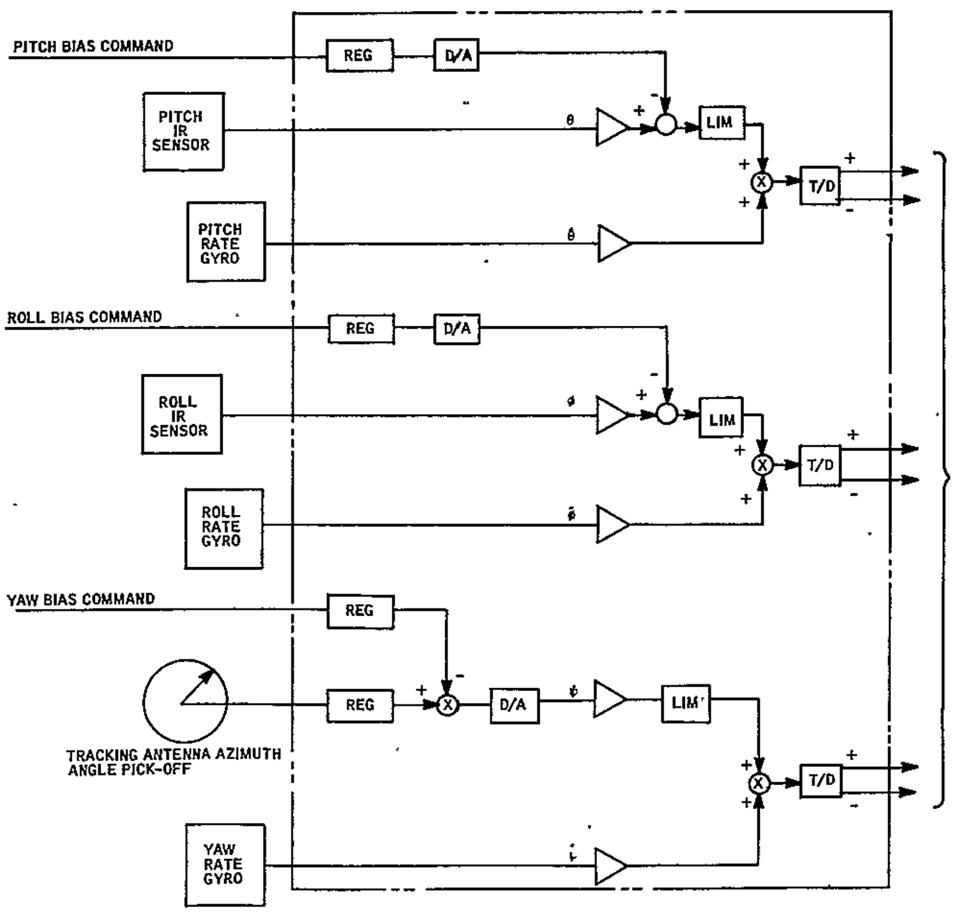


Figure 9.6-4. Attitude Control Reference Subsystem - Fine Sensing Mode

9.6.3.2 Fine Sensing Mode

Figure 9. 6-4 is a block diagram of the ACRS in the fine sensing mode. In this mode, the IR horizon scanners provide pitch and roll error which is added to the pitch and roll rate gyro signals to produce an error signal. The tracking antennas azimuth pick-off is summed with a yaw bias command generated by the ground station, and this signal is summed with the output of the yaw gyro to produce an error signal.

This mode without the antenna input in azimuth is used for both initial acquisition at injection, and it will be commanded automatically by the command programmer if the tracking antenna should lose lock at any time. The reacquisition logic will use earth-sky presence from the horizon scanners to determine if a useable earth signal is present. Without an earth presence signal in the proper quadrants of both IR scanners, a roll search bias will be added to cause the vehicle to roll until the earth presence requirements are fulfilled.

When the IR scanners are locked on and the errors are nulled, the antenna azimuth search can be commanded by the ACRS electronics package. A pre-inserted antenna elevation angle will be required. The elevation angle can be updated from ground stations via the ommi-antenna command link until lock-on occurs. At lock-on, the azimuth angle command loop can be closed causing the entire manipulator spacecraft to slew around to the proper yaw altitude while the antenna tracks the data relay satellite.

9.6.3.3 Coarse Attitude Control Mode

The purpose of the coarse attitude control mode (Figure 9.6-5) is to allow the spacecraft attitude error deadband to be limited only by the tracking antenna mechanical limits of $\pm$ 75 degrees in elevation. This mode is designed and will be used principally to reduce propellant consumption, thereby reducing contamination potential and also saving fuel.

One of the unique aspect of this design is that resolution of the antenna azimuth and elevation angles into roll and pitch error signals to drive the attitude control thrusters is done on the ground. This is feasible because of the $\pm$ 75 degree excursion of the elevation gimbal

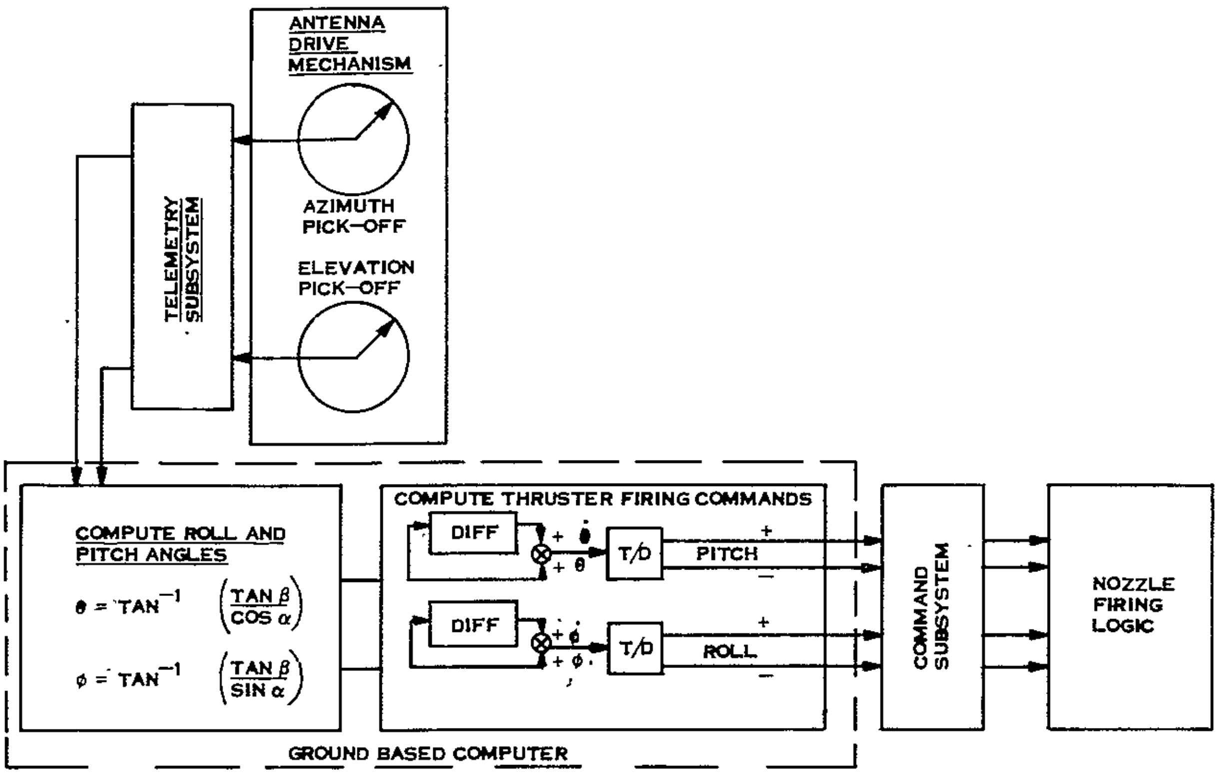


Figure 9.6-5. Attitude Control Subsystem in Coarse Antenna Pointing Mode

and the continuous communication by way of a data relay satellite. The block diagram indicates that the switching lines are set at $\pm$ 50 degree in elevation but both the elevation and azimuth switching points can be varied as necessary. One reason for changing the switching points will be to achieve the best orientation of the vehicles relative to the sun.

9.6.3.4 Thruster Firing Logic

The control torque vector requirements are the most critical portion of the attitude control problem and a unique design concept is required to solve it. During the portion of the mission when the manipulator spacecraft is not attached to the target satellite, the attitude control sensor axis set is aligned, nominally, to the vehicle principal axes of inertia. This is the case with most satellites to minimize coupling effects between axes.

After docking, however, the manipulator spacecraft may be attached to vehicles which are many times its mass and inertia. The principal axes of inertia of the combined vehicles may be severely displaced from the control axes. Further, the center of gravity of the combined vehicles will in most cases lie outside of the manipulator spacecraft envelope. If a simple CG offset were to exist (e.g., along the roll axis), the thruster firing logic readjustment can be relatively simple. One technique is to disable one-half of the pitch and yaw couples that can contaminate the target vehicle. However, it is necessary to permit almost random orientation of the manipulator spacecraft with respect to the target vehicle to facilitate the maintenance tasks.

As long as the inertial cross-coupling ratio is less than one (i.e., response to a sensed error is greatest about the axis with which it is sensed), the control system will eventually settle to a null but settling time and gas consumption will be reduced as the cross-coupling is reduced. Therefore, it is necessary to develop a means of providing variable adjustment of the direction and magnitude of the attitude control torques in response to sensed errors. This variable ability can also be applied to the undocked spacecraft during translational maneuvers when thrust/CG offset conditions exist.

A portion of the thruster firing logic is shown in Figure 9.6-6. The computation of the required thrust magnitudes and thruster combinations is done by the ground based computer and transmitted to the manipulator spacecraft. There are two memory units for each sensory axis which can retain the combination of thrusters and the thrust magnitude for each direction of pure rotation about each of the control axes. This is accomplished by sending digital address and magnitude commands to each memory unit. The address command consists of three binary sequences. The first two are inputs to the address matrix. The address matrix transmits an enable signal to the appropriate "AND" gate shown in Figure 9.6-6. The third sequence of the address command is the magnitude to be stored in the magnitude register. The last bit of each address command is used as a reset signal to enable the address matrix to receive the next address command. The sequence is repeated until all of the required thruster logic and magnitude for one axis and sense have been satisfied. A similar process is repeated for all address matrix and magnitude registers until firing commands have been stored for all six axes and sense combinations.

Figure 9.6-6 shows how the stored signals are used when attitude control error signals in excess of threshold limits are presented. A magnitude signal is only transmitted when an error and an address enabling signal are present to drive the threshold switch. The output of the threshold switch is transmitted to a D to A converter and then to a summing junction.

The summed analog magnitude signal coming from the summing junction is applied as the input to the analog-to-pulse width generation (APWG). The APWG consists of a fixed frequency multivibrator whose pulse width is varied as a function of the magnitude of the analog input. In order to avoid time lag problems, the frequency is selected to be 10 Hz. Consideration, of the minimum response time of the solenoid valves and transport lags in the nozzle feed lines results in an estimated minimum pulse width of 10 millisec with a 10:1 variation ratio of time-average thrusting available.

9.6.4 ACRS WEIGHT AND POWER SUMMARY

The following listing of weights, sizes and power requirements is based upon presently available hardware and experience gained in using these or similar components

Item	Quantity	Size (in.) (each)	Total Weight (lb)	Power (watts) Average	Power (watts) Peak
Electronics Package	1	18" x 12" x 8"	20	15	15
3-Axis Gyro Package	1	6" x 7" x 4"	10	15	30
IR Sensor	2	5" x 4" dia.	8	10	32
Solenoid Drives	26		7.8	0.3	9
		Totals	45.8	40.3	86

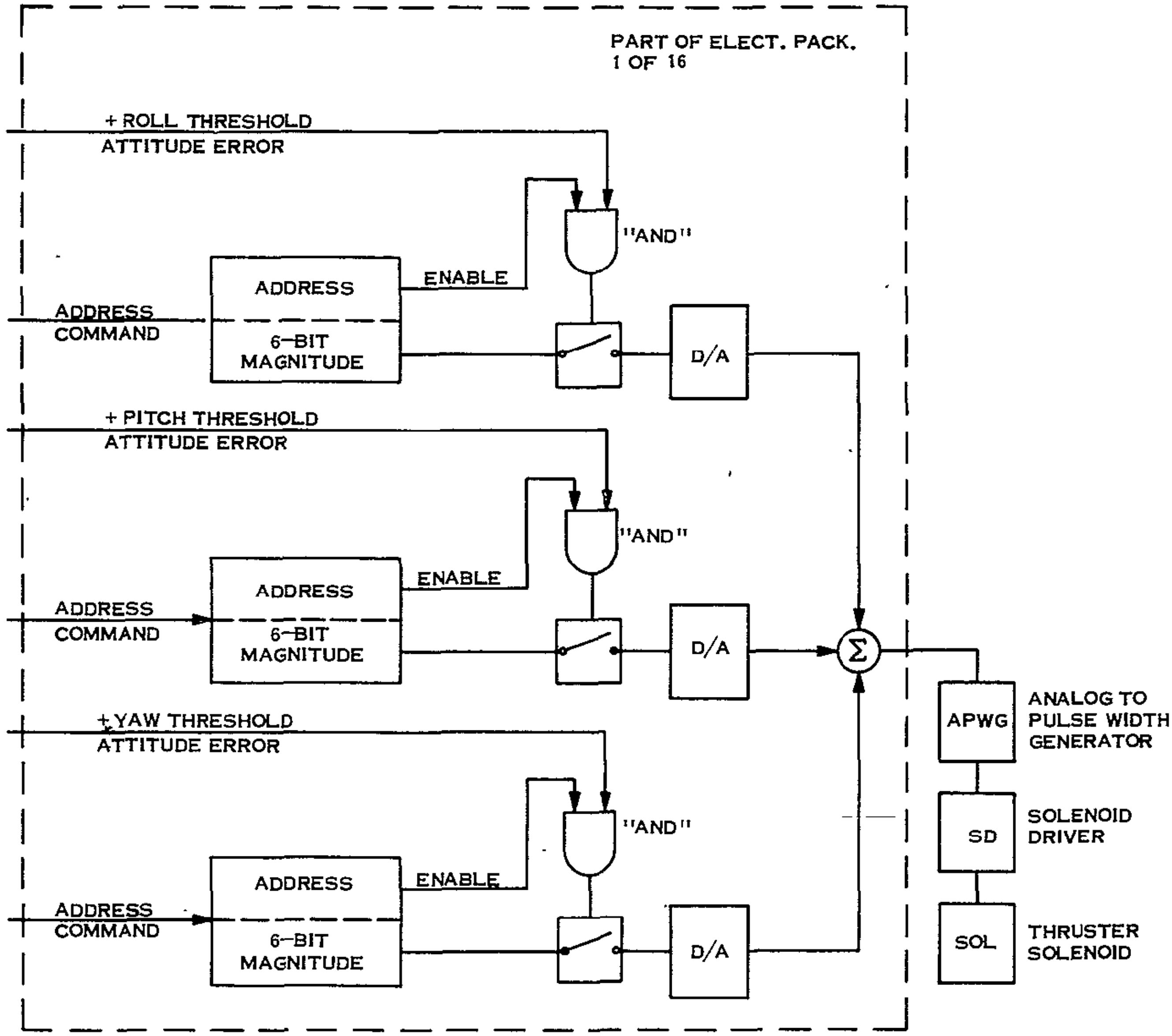


Figure 9.6-6. Thruster Firing Logic for One Thruster Solenoid

9.7 THERMAL CONTROL SUBSYSTEM

The spacecraft thermal control subsystem is largely passive for several reasons:

1. The thermal control requirements are modest. For example, there are no severe thermal deflection requirements as on OAO. In addition, the short 10-day mission-life requirement allows higher equipment operating temperatures.

2. The peak power consumption is 750 watts and the average power consumption is 100 watts for the 10-day OAO repair mission. If detailed thermal control analysis shows that the thermal output of an equipment bay is too concentrated (such as the batteries), for the heat-sink capacity of the spacecraft structure, automatic thermal shutters similar to those on Nimbus can be installed on the peripheral faces of the sectored structural bays.

3. In case of unexpected thermal control problems, the spacecraft and the attached satellite can be oriented by the ground operator because of the versatile view angles of the IR earth sensors and high-gain antenna. Furthermore, the spacecraft can erect and position thermal and light shades, diffusers, and reflectors to thermally control the manipulator and target spacecraft and the replacement equipment during the brief transportation period. It is felt only the OSO satellite may require re-orientation and shielding; Nimbus, OAO, and DBS satellites have adequate passive thermal control measures such as automatic shutters, thermal radiation blankets, reflective finishes, and other thermal coatings.

4. Almost all of the replacement equipment is thermally protected because it is inside the supply bin. The exposed replacement equipment can withstand the thermal environment as well as it does during the regular missions of the satellites:

   a. The Nimbus C solar paddles obviously can withstand the thermal environment of the repair mission.

   b. The Nimbus E sensory ring can withstand the thermal environment due to insulation blankets on its top, thermal covers on its bottom and automatic thermal shutters on the sides. The bottom could be protected thermally by a radiation blanket.

The spacecraft thermal control subsystems consists of the following:

1. Coatings of appropriate absorptivity/emissivity coefficients on the super-insulation radiation blankets, as on the Nimbus sensory ring blankets, and other critical surfaces such as the attitude control thruster structure. This open structure is exposed on nearly all sides; thus, thermal deflection is reduced. Highly-reflective finishes are avoided because they may cause the video cameras to bloom.

2. Thermal isolation devices such as plastic insulations are used on all subsystem mounts including replacement equipment mounts.

3. The manipulator joints that encase the servo motors may have to be fabricated from special materials if coatings cannot adequately control their temperature. Beryllium is a possibility because of its exceptionally high thermal diffusivity wherein the product of the specific heat and thermal conductivity is quite large per unit of mass. Furthermore, the exceptionally high elastic modules (40×10^6 psi) causes low load deflection.

The only active thermal control measure contemplated is the one-watt heater for each of the two earth IR sensors and the 25 watts that may be necessary for the thrusters. The 3-axis rate gyro package will have heaters for warmup but these will not be needed unless the gyros are turned off for short periods.

9.8 ELECTRICAL POWER SUBSYSTEM

The EPS supplies electrical energy from the main power source for the manipulator spacecraft subsystem loads. The EPS (Figure 9.8-1) is configured to provide the most effective design for meeting vehicle electrical requirements with reasonable weight and high reliability. The recommended design makes use of a considerable amount of experience gained with existing hardware and procedures.

The electrical energy source is a set of 3 silver oxide-zinc batteries, providing approximately 445 ampere-hours of energy per battery. The interface between ground power and internal power is provided by the Power Control Unit (PCU). This unit also provides battery isolation for failure protection and telemetry for energy management and monitoring of critical subsystems.

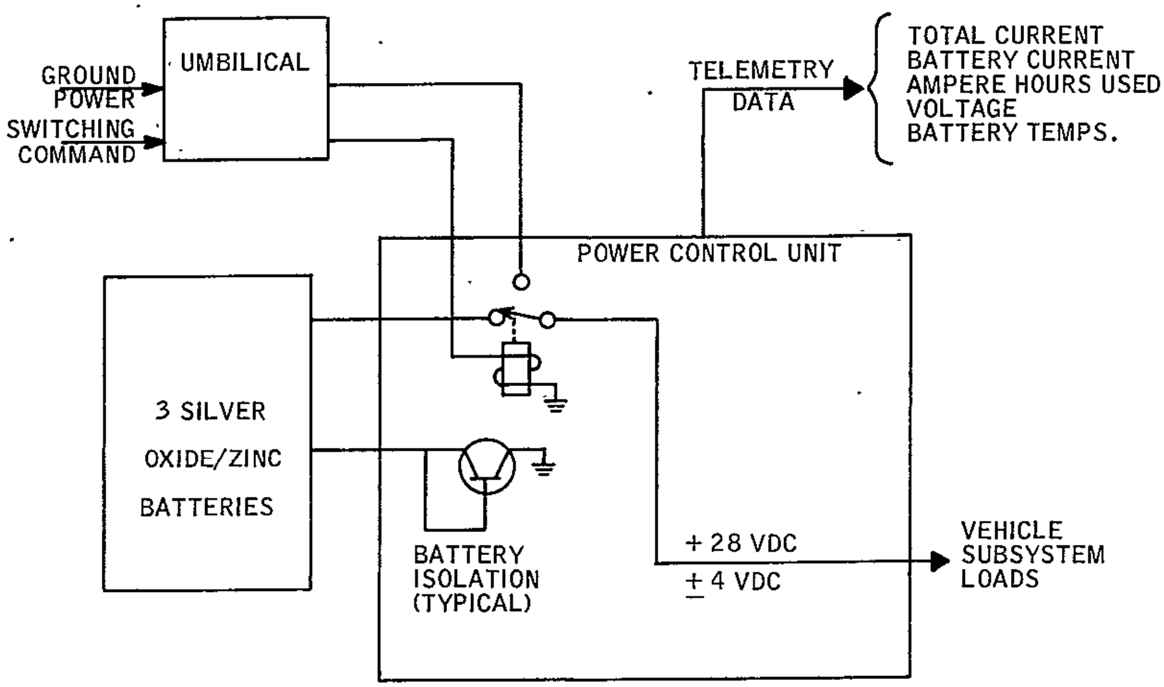


Figure 9.8-1. Electric Power Subsystem Block Diagram

9.8.1 EPS REQUIREMENTS

The EPS must be capable of supplying all vehicle power requirements over the duration of the mission as well as energy for initial ground tests and prelaunch operations conducted on internal power.

The mission requiring the most manipulator power is the OAO repair mission which is 16 hours long. It will be assumed that the manipulators will be used two hours each day for eight days. The peak power required may briefly equal 750 watts and the maximum energy required by the manipulators is 736 watt-hours.

The OAO repair mission is used for determining the power capability of the EPS, since it requires the longest duration of effort and therefore offers the most conservative power subsystem design. The power requirements for the various subsystems are discussed in the following paragraphs.

9.8.1.1 Communication Subsystem

During the 10-day duration of the mission, the video system will be on 2 hours/day for a total of 20 hours. The video system will be on during manipulator operating periods and the telemetry system must transmit two video channels plus manipulator force feedback signals as well as the housekeeping telemetry that will be transmitted continuously. For the 2 hour/day manipulator operating period, two telemetry transmitters will operate and the prime power required will be 76.5 watts (average). During the other 22 hours/day portions of the TT&C system, one of the two transmitters will be shut down and the power requirement reduced to 53.5 watts.

The high gain tracking antenna drive will operate at a 5-watt average level but produce peak requirements of 60 watts three times a revolution as the antenna is switched from one data relay satellite to another. At its maximum slew rate of 5 degrees/sec, it will take approximately one-half-minute to complete the transition from one data relay satellite to the next. The tracking beacon required 25 watts but it will only be energized an average of 8 times/day for about 10 minutes/station pass.

9.8.1.2 Video Subsystem

The video subsystem is assumed to be on for each of the eight 2-hour work periods plus two hours on the first and last days for initial and final inspection. The prime power requirements are 15 watts/camera, 5 watts for automatic light control, and 10 watts for illumination on the dark side of the orbit. At altitudes of interest, the orbital period is approximately 100 minutes long and 40 minutes of this is spent on the night side. To simplify the video power requirement, it will be added as 44 watts average for a 2-hour period each and every day.

9.8.1.3 Attitude Control Subsystem

The attitude control subsystem includes a 3-axis rate gyro package, a pair of IR horizon scanners, an electronics package, solenoid valves and fuel supply. The total prime power requirement while operating is 40 watts average. Peak power levels during initializing (firing of squib valves) may reach 150 watts for periods of 10 to 15 millisec and power to energize the thruster solenoids may reach 35 watts; but the two will not occur simultaneously, and the total energy required is low. The power requirements of the squib valves of the thruster solenoids will be included in the power profile as an average of 0.5 watts.

9.8.2 SYSTEM POWER PROFILE

Figure 9.8-2 shows the manipulator spacecraft prime power profile for a 24-hour period. This profile is derived from the requirements of Section 9.8.1. This power requirement is assumed to repeat each day for the 10 days of the orbital mission. Actually, the total power requirements of the first and last days may be somewhat less because manipulator activity on these days will be minimal. The first day will be devoted to rendezvous and docking and the last day would see the manipulator leaving the target vehicle, observing the orbital operation of the target satellite and then de-orbiting itself. Each period is arbitrarily started with the manipulator activity and this continues for two hours.

The total electrical energy required for the 10-day OAO mission is calculated to be less than 26,000 watt-hours and peak power requirements occurring during manipulator operation may be as high as 750 watts.

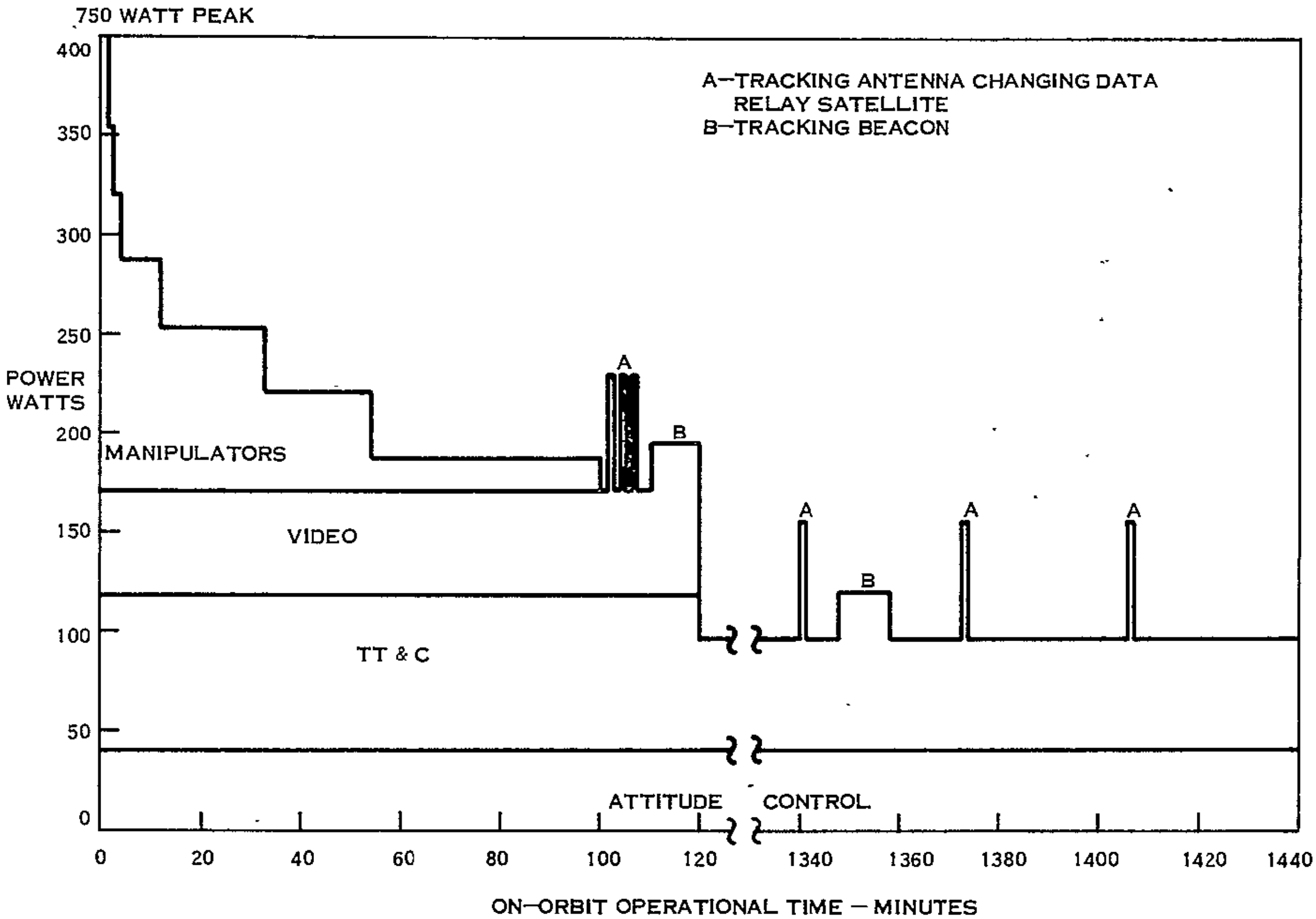


Figure 9.8-2. Manipulator Spacecraft Power Profile

Table 9.8-1. Power Source Selection

Faction	Fuel Cell System	Silver Oxide-Zinc Batteries
Availability	Development problems	Off-the-shelf
Integration	Structural design required	Battery wells
	Radiator/coolant loop	
	Tankage vibration	
	Fuel levels and flow rates	
Power Capability	Requires an additional topping-battery for peak load energy	Adequate
Reliability	Depends on current development	Flight-proven

9.8.3 EPS DESIGN APPROACH

The recommended manipulator spacecraft power source was selected after considering the tradeoff between implementing a fuel cell system and primary silver oxide-zinc batteries. Photovoltaic systems were not considered due to the relatively short length of the mission, the power levels required, the unfavorable interfaces required with the vehicle and the low earth orbit mission profile. Table 9.8-1 shows the perinent factors in the tradeoff between the fuel cell system and the battery system. These factors indicate that the battery system must be selected, because of development uncertainties and integration difficulties associated with implementing the fuel cell system.

9.8.4 DESCRIPTION OF EPS

The EPS is capable of supplying an average of 37,500 watt-hours within a voltage range of 24 to 32 volts. Initially the voltage will be 32 volts under a nominal load due to the "peroxide peak" characteristic of silver oxide-zinc batteries. As energy is withdrawn from the batteries, the voltage will decay to approximately 28 volts. This point will be reached after approximately 10 to 20 percent of the available energy is removed. For the remainder of the mission the voltage will be nearly constant at 28 volts except for periods of short duration high peak loads when the voltage may dip to approximately 26 volts.

Only nominal environmental control is required since the batteries will operate satisfactorily over a temperature range of 50°F to 90°F. Maximum performance will be obtained at a temperature of approximately 80°F. Thermal dissipation is low, an average of approximately 10 watts for each battery at nominal loads.

9.8.4.1 Power Source

The battery power source is an 18-cell configuration, employing the Eagle-Picher Co. type MAP-2598-7 cell. This cell has been manufactured for several years and has extensive flight history. Sixteen-cell batteries have been flown on a number of programs, such as the electrical power source in the Agena. The Eagle-Picher MAP 4138-3 battery which employs this cell was used in the Gemini capsule. This cell was also used in a similar battery in the Agena docking vehicle. The 18-cell battery reliable activated life is 35 days at a stand and discharge temperature of 80°F; greater life is achievable by storing and discharging at reduced temperatures. Battery capability is a nominal average of 44 ampere-hours (37,500 watt-hours). However, depending on the activated stand time and temperature during stand and operation this may be degraded to a minimum average of 425 ampere-hours (36,000 watt-hours).

9.8.4.2 Power Control Unit

The PCU is the central power distribution point and provides telemetry data, including total vehicle current, total ampere-hours consumed, main bus voltage, and individual battery current. These data are used to provide the capability of monitoring power source performance and projecting capability throughout the mission. This unit is similar to one developed and flight qualified for another program.

9.8.5 EPS CHARACTERISTICS

1. Weight and Volume

Battery Envelope:	18.5 x 11.4 x 8.2 in.	131 lb (max) (363 total)
Power Control Unit Envelope:	6.2 x 5 x 8.25 in.	10 lb

2. Telemetry - The following telemetry points will be required:

   a. Individual battery current

   b. Main bus current

   c. Ampere hours expended

   d. Bus voltage

   e. Battery temperature

3. Command - The following command is required:

   a. Relay closure for each battery when switching from ground power to internal power.

9.9 GROUND STATION

The approaches presented in this subsection, along with the equipment types listed, should be regarded as part of a feasibility study and subject to refinements and improvements. It must be recognized that the ground operations required for the remote manipulator mission are very different from almost all earlier ground stations; an optimum degree of flexibility is required. In most ground stations, missions are preplanned and practiced extensively; the most likely emergencies are anticipated and practiced. With manipulator spacecraft, however, one of the most important functional aspects is "flexibility", the ability to change plans of attack with all ground station personnel performing as a team to quickly take up a different plan for maintenance of a spacecraft. The ground station layouts and requirements presented were created with these factors in mind.

The ultimate objective is to obtain an operational trailer that is equipped with the optimum electronic gear organized in the most efficient manner; the trailer equipment will be influenced the most by experience gained during factory checkout phases.

Experience gained in the factory ground station will aid in developing the best possible trailer design. It should be noted that a man-machine system consisting of a man, his controls, and an object being controlled is being developed. A sketch of a feasible trailer approach is presented in Figure 9.9-1.

9.9.1 FACTORY GROUND STATION

In testing a spacecraft of any type during system integration, one of the best ways to connect to the spacecraft is through the RF links it carries. A ground station with receivers, transmitters, decoders, and processors thus is required and this equipment is often identical to the operational ground equipments. For data analysis, the factory ground station will probably be equipped with line printers to provide historical records of all test data.

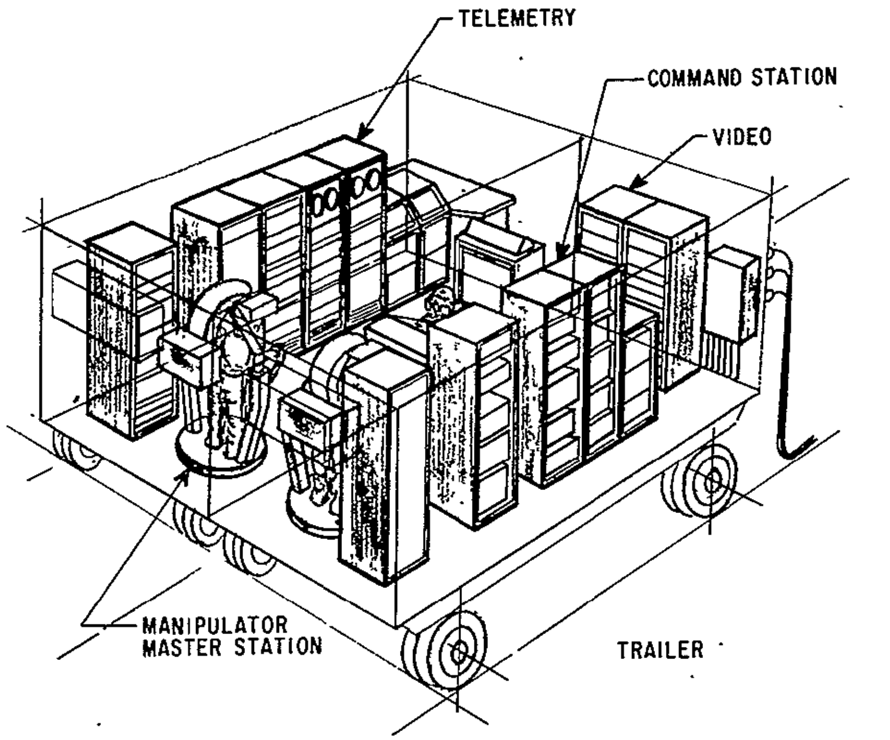


Figure 9.9-1. Ground Station Trailer Design

9.9.1.1 Possible Equipments for the Factory Ground Station

The following equipment will be required:

1. Two manipulator stations with one manipulator shared for training purposes, but both available for system tests. The data conversion equipment necessary to digitize manipulator controller motions and provide interface conversions will be located in associated racks.

2. A telemetry receiver rack consisting of receiver, bit sync, decommutator, digital-to-analog converters, display devices and perhaps a small processor for organizing telemetry into useable parameters, with limit checking.

3. A command station for sending the discrete commands to the spacecraft. A separate part of the command station will be for communications with "outside" areas, and will be included as part of the trailer development.

4. A test controllers station containing a few critical functions necessary to control ground test personnel during system tests.

5. A video monitor station containing the video receiver equipment.

9.9.1.2 Flexibility

The factory ground station equipment will be designed for rearrangement so that several arrangements can be tried and evaluated to develop the best layout.

9.9.1.3 Closed Circuit TV

A standard closed circuit TV should prove valuable for video transmission between the operational trailer and a target vehicle center where drawings, photos, models, parts, etc., will be stored. The manipulator operator can look at part drawings, photos, etc., to assist him in his work on the target spacecraft. If such a video system is included, it should be developed in conjunction with the factory ground station.

Closed circuit TV to the target vehicle center is desirable. In addition to supplying intelligence to the trailer, it provides an additional means for bringing the target vehicle project people close to the maintenance mission. In some circumstances target vehicle managers will have to decide on the spot what can and cannot be tounched on the target vehicle.

9.9.1.4 Delay

An electronic delay mechanism to simulate long distance transmission will be required.

9.9.1.5 Launch Support

The factory ground station will be temporarily located in a trailer for use in launch support.

9.9.2 TRAILER GROUND STATION

The design start will be after experience has been gained with the factory ground station (perhaps while testing an electrical systems model). Some of the reasons for configuring the ground station in a trailer are:

1. This equipment usage is of an "as required" nature and long periods may occur between usages.

2. No setup or installation problems will arise later.

3. A trailer allows rapid transportation to any operation site.

4. Prior to remote manipulator launches, the trailer can be returned to the factory for final compatibillty verification

9.9.2.1 General Communications

Figure 9.9-2 illustrates the ground station interfaces. A tie-in to a target satellite data bank provides immediate access to design details of the target satellite. A tie-in to the tracking facilities provides real time ephemeris data of both remote manipulator spacecraft and target satellite. A tie-in to the target satellite control station provides immediate accessibility to target satellite status. Finally, communications services are provided through a tie-in to synchronous data relay satellite facility.

9.9.2.2 Power

A power distribution panel on the trailer will interface with the local electrical distribution system.

9.9.2.3 Personnel

Trailer personnel requirements appear to be fulfilled by:

1. Two manipulator operators

2. One test conductor with overall responsibility and authority for the mission performance

3. One housekeeping telemetry monitor

4. One communications system operator

5. One target vehicle monitor

6. Other personnel will be located at remote sites.

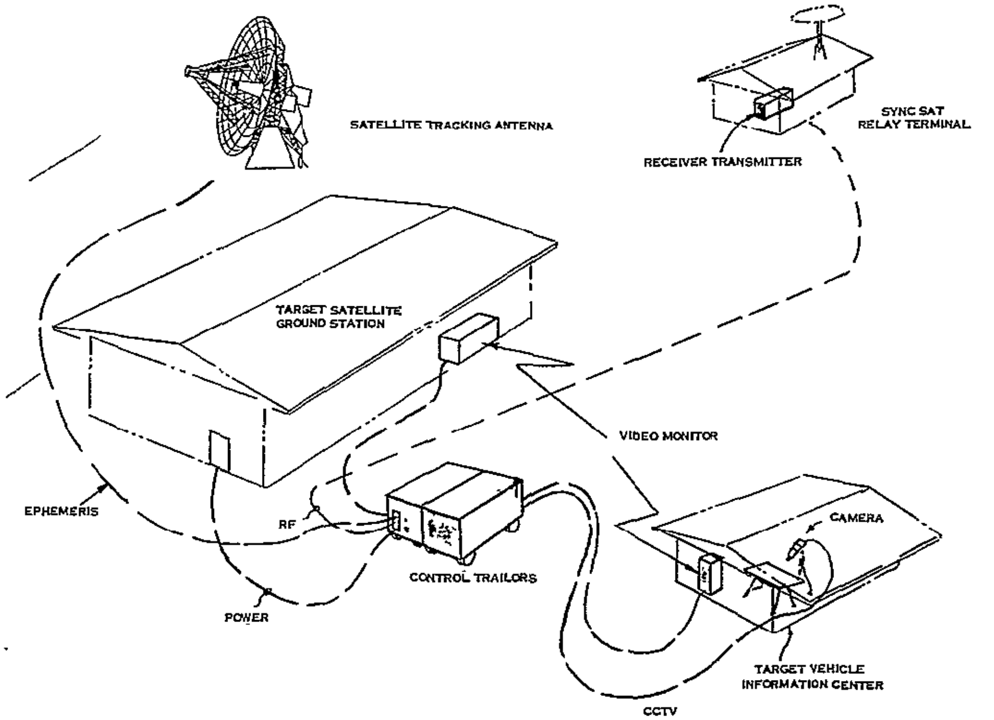


Figure 9.9-2. Ground Station Interfaces

9.9.2.4 Trailer Equipment and Layout

A plan view of the trailer layout is shown in Figure 9.9-3. The equipment is described below:

1. Telemetry

   Two racks were deemed suitable to house telemetry equipment. Manipulator requirements of 50-1 per second analog, 100 every 16 seconds digital and 160 every 16 seconds analog channels were assumed. Some telemetry panels include:

   a. Bit synchronizer

   b. Serial to parallel converter

   c. Digital to analog converter

   d. Interface converters (with processor)

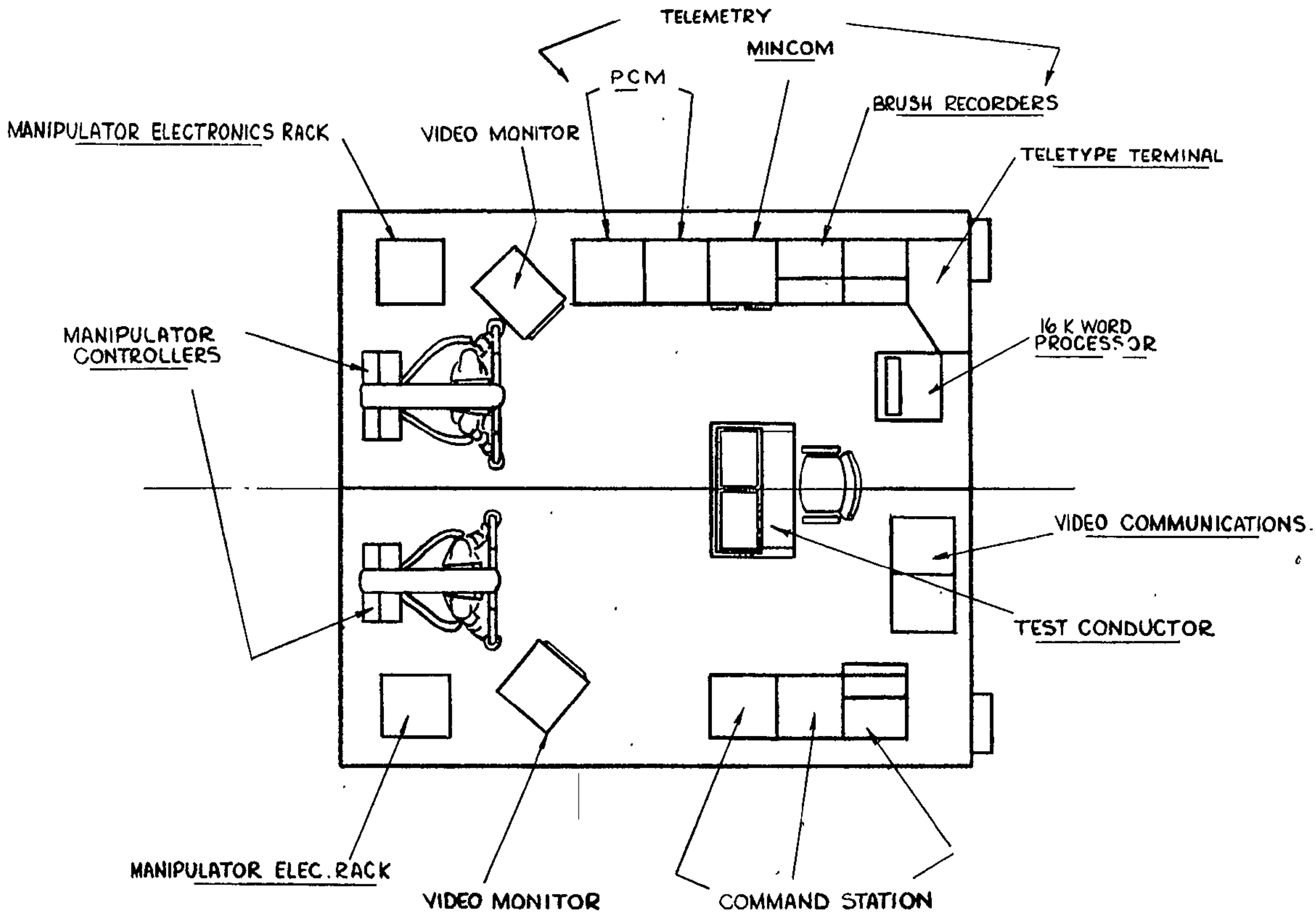


Figure 9.9-3. Ground Station Layout Plan View

e. Decommutators

f. Power supplies

g. Control panel

h. Signal simulator panel

i. MINCOM (or equivalent) recorder (a third rack)

j. 2 Brush recorders

2. Communications Equipment

a. Transmitters and receivers

b. Filters

c. Signal processing

d. PSK detector

e. Subcarrier modulator

f. Multiplexer

g. Video/manipulator encoding

h. S/C command encoder

3. Test Conductor Console - This console will contain mission critical displays and controls for the test conductor. Displays and controls can include:

a. Range and rate for docking

b. Counters showing computer time-to-fire for thrusters

c. Critical temperatures and voltages

d. Attitude control pressures, temperatures, thruster firing rates, etc.

e. Ground communications control

4. Processor Requirements - A preliminary look at processor requirements points to a machine in the DDP 516 class, with a total of about 16,000 words of memory and a paper tape reader input.

5. Video Monitors - At least two manipulator spacecraft and CCTV video monitors are recommended for general trailer use. The manipulator operators will each have their own video presentation as a part of the master manipulator station.

6. Command Station - Two consoles and a desk are estimated as the equipment needed.

9.9.3 AGE

A partial list of other ground equipment for use with the remote manipulator spacecraft includes:

1. Systems Test Console
   a. Power supplies, monitoring and recording, and power control.
   b. Monitoring and recording of critical temperatures (such as batteries)
   c. Monitoring of other critical functions.
   d. Provide special controls such as gyro preheat, etc.
2. RF Test Console
   a. VSWR measurements
   b. Antenna tuning
   c. RF power measurements
   d. RF sensitivity measurements
3. Test Targets (for Video System)
4. Special Thermal-Vacuum Equipment
   a. Heater arrays (with power supplies)
   b. Thermocouples, special cabling, etc.

5. IR Scanner Stimulators - To provide a known stimulus for the controls scanners.

6. AGE and Spacecraft Simulators - For launch support. These equipments are sent to the launch pad prior to launch to check out pad wiring.

7. Blockhouse Console (for Launch Support)

REFERENCES (Section 9)

1. "Remote Manipulators and Mass Transfer Study," General Electric Company for the Air Force Aero Propulsion Laboratory, AFAPL-TR-68-75, 1968.

2. "Orbiting Data Relay Network," Final Report, Astroelectronics Division, RCA, Contract No. NASW-1447.

3. NASA RFP No. 525-71787-335 plus Amendment No. 1.

4. Aerospace Data Systems Standards, PCM Instruction Command System Standard, Goddard Space Flight Center, NASA.

5. R. P. Pahmeier, "Effects of Non-Optimum Repeater IF Bandwidths on Link Performance," General Electric Company, Space Systems Organization PIR 1J41-ATS-489.

6. Schwartz, Bennett, and Stein, <u>Communication Systems and Techniques,</u> pp. 154-163.

7. Goddard Range and Range Rate Design Evaluation Report, Vol. 1, MCD, GE Company, Contract No. NAS 5-9731.

SECTION 10

MANIPULATOR SUBSYSTEM COST AND SCHEDULE

Estimates of the time schedule, manpower requirements and hardware costs needed to produce the master and slave manipulator arms described in Section 9 are presented in this section. The estimate for the arms includes all components in the servo loop; i.e., amplifier, motor, gear reduction, and structure. It does not include all ground or space station support items (e.g., D/A converters), the video subsystem hardware development, or the passive tethering devices.

10.1 DEVELOPMENT SCHEDULE

In Table 10-1 the development of the manipulator is broken down into four phases; (1) preparatory studies, (2) hardware design, (3) space qualification, and (4) production. The time estimate to finish the qualification program and start delivery of production slave arms is less than four years. Included in this time is the training of the manipulator operators so that at delivery of the arms, the system would be ready to fly and perform the intended function. The phases are described in the following paragraphs.

10.1.1 PREPARATORY STUDIES

Before the hardware can be developed, various concepts will have to be investigated, such as alternative kinematic configurations, various servo components and system studies. Also, the actual task definition will need further investigation (i.e., what future missions and type of functions will be expected of the manipulators). These two sub tasks will lead to setting down of the manipulator specifications, similar to the suggestions in Section 9. Also, requirements as to needed advanced development studies will be outlined and needed long-term hardware ordered. Portions of this phase are currently underway at GE.

The advanced development studies will show by laboratory simulation the problem areas and capabilities of the manipulator system. The basic tool will be an off-the-shelf electric master-slave manipulator, such as an E-4A, modified for time delay studies. Initial work can be done by mounting the arms on a gimbal riding on an air-bearing floor. A video link

Table 10-1. Development Time Schedule

		Calendar Years After Go Ahead						Number of Arms Required	
		0	1	2	3	4	5	Master	Slave
PREPARATORY STUDIES	Conceptual Studies								
	Task Definition								
	Manipulator Requirements								
	Advanced Studies Requirements							2	2
	Task & Time Delay Simulation								
	Docking & Tethering Simulation								
	Video Requirement & Evaluation								
	Ground Station Prototype								
HARDWARE DESIGN	Preliminary Analysis								
	Servo Analysis								
	Servo Component Design								
	Prototype Slave Joint Layout								
	Prototype Joint Testing								
	Prototype Arm Layout								
	Master Arm & Station Layout, Manufacturing and Assembly							4	
	Prototype Fabrication and Assembly								2
	Prototype Test & Evaluation								
	Slave Final Design								
	Fabricate Engineering Model								2
	Engineering Model Preliminary Test								
SPACE QUALIFICATION	Fabricate Quality Test Arms								4
	Monitor Quality Test Program								
PRODUCTION	Begin Delivery of Flight Arms								

will also be used. Actual space hardware can then be worked on providing for real life effects of time delay on stability, base line time and motion factors, tooling requirements, development of techniques (operator training), etc. The manipulator simulator can also be mountedon a 6-degree-of-freedom lab simulator which will allow docking and tethering experience. During both of these experiments, various video systems (camera placement, stereo, color, etc.) can be tried out for operator preference and effectiveness. Also, human factor studies on design of the master ground station can be run.

10.1.2 HARDWARE DESIGN

A brief preliminary analysis stage precedes the actual design. This analysis should bring the design engineers up to date as to the most preferred components, materials, concepts, etc., applicable to the manipulator design. The servo analysis and design of the amplifiers will parallel the actual first layout of the arms. The first mechanical layout will be of a typical joint and probably two slave joint concepts will be carried through the first breadboard test. During the breadboard test, layout of the whole arm will be started and completed when the breadboard test indicates the best design features. Only one layout, and no preliminary testing is deemed necessary on the master arms. This part of the design can be done at a leisurely pace, waiting for all imputs from the preparatory study phase. The first prototype slave arm should complete its testing at about the 30-month point on the schedule. It is anticipated that bugs will show up and a redesign to the flight model will be necessary. Delivery of the engineering model is scheduled at about the 40-month point.

10.1.3 SPACE QUALIFICATION AND PRODUCTION

After the redesign of the slave arm is complete, an engineering model (for further operator training, etc.) and two pair of slave arms (for spacecraft qualification testing) will be manufactured. The spacecraft tests will be monitored by the manipulator engineers and only slight modifications to the manipulator design are presumed to be necessary.

10.2 MANIPULATOR COSTS

In Table 10-2 the cost of each phase listed in Table 10-1 is estimated. Manpower is broken down into engineering (including scientific, consulting and managerial functions), technical (including drafting and technician) and shop (including machine, electrical and assembly work). Material costs include computer usage and rental of such things as space simulators besides the actual material purchased. Labor rates used are those projected for 1973.

Table 10-2. Manipulator Cost Summary

Phase	Manpower (Man-Years)			Cost ($K)		
	Engineer	Technician	Shop	Manpower	Materials	Total
Preparatory Studies	7.0	4.0	0.6	127.7	500	627.7
Hardware Design	15.8	17.2	18.8	794.0	302	1096.0
Space Qualification	0.6	0.6	10.2	120.1	174	294.1
Production	0.2	0.2	5.0	55.3	87	142.3

Master station cost, for two sets of arms, is based on extrapolation of related technology for which GE has past cost experience. Slave arm cost is derived from a shop estimate to produce a single shoulder joint. The estimate was more than double to consider added control and rigid inspection techniques. Extrapolation to eight joints, plus end effector and amplifier, gave the present estimate. This is checked against the known cost to build the Brookhaven arm, which is similar in concept to the projected space arms though designed for earth environment.

The last line of Table 10-2 presents the recurring cost to manufacture one set of slave arms. A slight increase in efficiency is assumed from the prototype manufacture. This price would not drop drastically for additional pairs of slave arms. The reason for this is that the manipulator consists of essentially two different types of joints (shoulder and wrist) with eight joints in each category. Thus, the initial order will be for a relatively large number of units and the initial hardware costs will not be too much higher than a large volume order. Also, because of the similarities in joints, machine shop efficiency should be good before the pair of arms is delivered. The estimates for varying numbers of flight arms are summarized in Table 10-3.

The costs of Tables 10-2 and 10-3 are strictly labor and materials with no adders such as overhead. Section 11 gives overall costs.

Table 10-3. Recurring Cost vs. Number Built

Quantity	Labor (Man-Yr)	Material ($K)	Labor ($K)	Total ($K)
1st Pair Flight Arms	5.4	87	55.3	142.3
5 Additional Pair of Flight Arms	5.0	81	51.5	132.5
20 Additional Pair of Flight Arms	4.5	76	46.7	122.7

SECTION 11

SYSTEM COSTS AND SCHEDULE

An estimate was made of the cost of the system described in Sections 8 and 9. These included development, recurring and sustaining costs and were based on the development plan of Figure 11-1.

11.1 DEVELOPMENT PLAN

A total of four years is necessary to develop the system. The first 20 months are used to the finalize the design and begin assembly of mockups and of the engineering prototype spacecraft. Fabrication of the qualification vehicle begins at the end of the second year and spacecraft qualification is completed by the middle of the fourth year. The first flight unit fabrication begins at the start of the third year and flight acceptance testing is completed at the end of the fourth year.

The four year estimate considers the time necessary for component, subsystem, and technology development as well as the complete system testing phase. Expenditure of additional funds would be necessary to significantly shorten the schedule.

11.2 COSTS

The estimated system costs were arrived at from a "bottom up" estimate of the designs described in Sections 8 and 9. The development costs cover the program costs through the testing of the qualification spacecraft. This includes the costs of the ground control station, AGE and launch support equipment, factory ground station, simulation and training, engineering prototype spacecraft and qualification spacecraft. The development costs are detailed in Table 11-1. The recurring costs include the cost of fabricating and testing the first flight spacecraft and the necessary launch support. These costs are detailed in Table 11-2.

The sustaining costs cover the cost of relay satellite rental and ground control station setup and operation. The rental costs to the public for using the COMSAT relay satellites are established by FCC tariffs. The costs of uplink transmission depend on the type of signal transmitted and the origin and destination of the signal. The public pays the charges not to Comsat Corporation, but to a common carrier, such as RCA Communication, or Western Union International. These sustaining costs are listed below:

Item	Cost Per Flight ($K)
Relay Satellite Services	218.0
Ground Control Station Operation	43.2
Total	261.2

An estimate for the entire program was made and is detailed in Table 11-3. The labor rates are based on projected 1973 labor rates. All contract adders are listed. Also listed are multiple unit costs for the next 10 spacecraft.

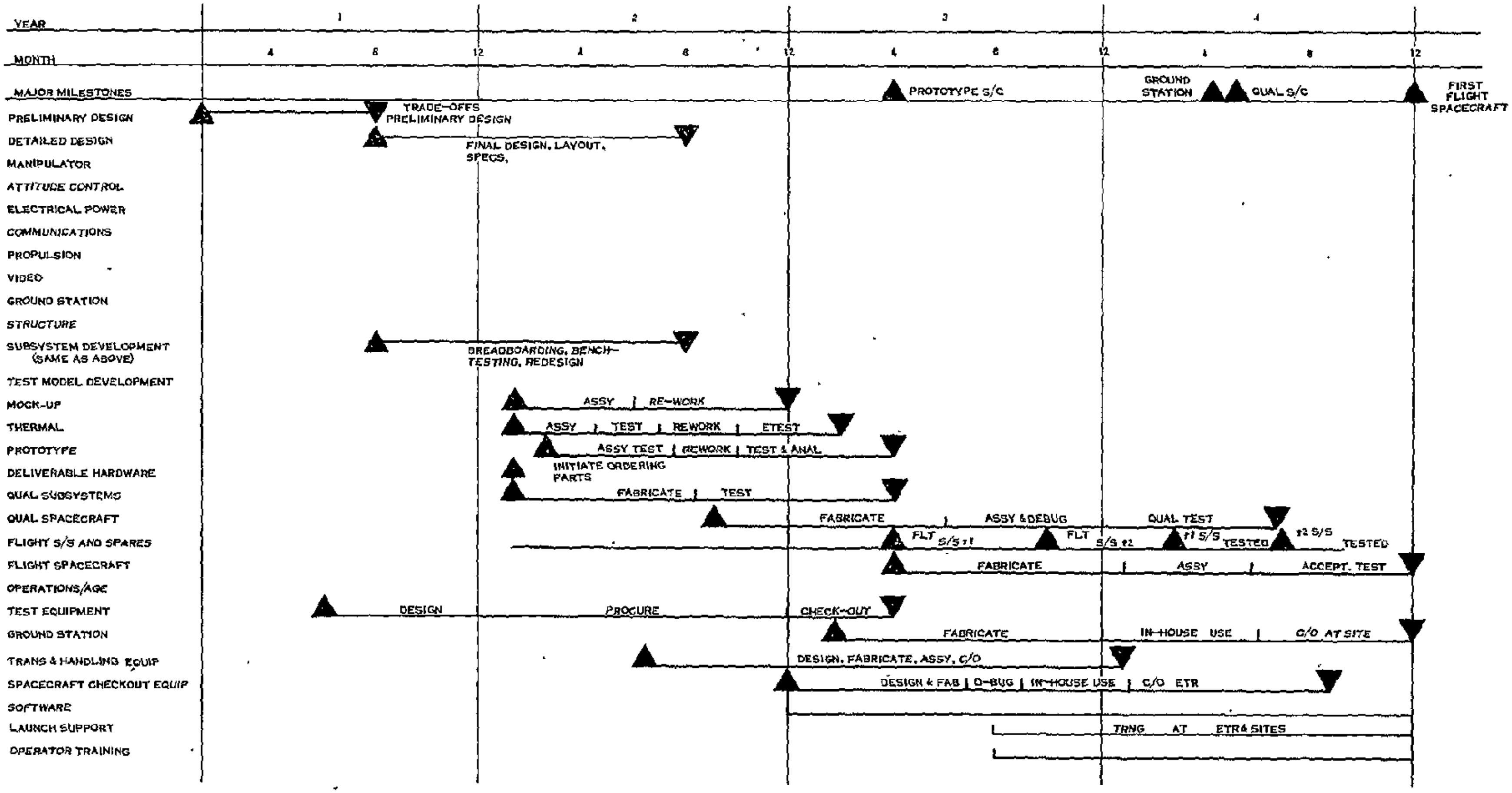


Figure 11-1. System Development Plan

Table 11-1. Manpower Estimate - System Development

Section	Manpower (Hours)			
	Engineer	Draftsman	Hourly	Technician
Research and Engineering	99480	59488	-----	54080
Manufacturing	9360	-----	63232	53664
Product Assurance and Testing	58408	-----	18304	66144
Program Management	35500	-----	-----	17280
TOTAL	203180	59488	81536	191168

Table 11-2. Manpower Estimate - Recurring (First Flight Unit)

Section	Manpower (Hours)			
	Engineer	Draftsman	Hourly	Technician
Research and Engineering	12688	2288	-----	4576
Manufacturing	6864	----	22880	16016
Product Assurance and Testing	20124	----	5616	25168
Program Management	12500	----	-----	5624
TOTAL	52176	2288	28496	51380

Table 11-3. Remote Manipulator Spacecraft System Cost Estimate

	Development	First Flight Unit	Total
Hours:			
Engineers	203,108	52,176	255,284
Draftsmen	59,488	2,288	61,776
Hourly	81,536	28,496	110,032
Technicians	191,168	51,384	242,552
Total Hours	535,300	134,344	669,644
Dollars:			
Labor:			
Engineers	2,300,794	593,155	2,893,949
Draftsmen	403,329	15,513	418,842
Hourly	373,851	130,431	504,282
Technicians	1,274,166	346,411	1,620,577
Total Labor	4,352,140	1,085,510	5,437,650
Overhead 128%	5,570,739	1,389,452	6,960,192
Material	5,055,100	1,567,400	6,622,500
Subcontract (Manipulators	2,858,000	215,000	3,073,000
Sub Total	$ 17,835,979	$ 4,257,362	$ 22,093,342
CIRP 1.2%	214,032	51,088	265,120
Sub Total	$ 18,050,011	$ 4,308,450	$ 22,358,462
G&A 9.2%	1,660,601	396,377	2,056,979
Total Estimated Cost	$ 19,710,612	$ 4,704,827	$ 24,415,441
Fee	1,576,849	376,386	1,953.235
Total Estimated Cost and Fee	$ 21,287,461	$ 5,081,213	$ 26,368,676

Cost of each of next 10 flight units is estimated to be $ 4,064,970

APPENDIX A

MISSION SELECTION DETAILS

The list of spacecraft that were considered for maintenance missions during the study and the evaluation techniques used to reduce the group population to a manageable and meaningful number are described in this appendix. Table A-1 is a complete list of the candidate satellites examined for mission selection. The size of Table A-1 was reduced by imposing the following restrictions:

a. Spacecraft systems weighing less than 500 pounds were eliminated, since the weight, complexity and cost of the remote manipulator spacecraft would be at least as great as the spacecraft itself. Although it is not a part of this study, it should be noted that for a remote manipulator spacecraft which is kept in orbit and activated whenever a maintenance mission occurred, this restriction could be lifted, because the comparison then is no longer between the replacement satellite and a complete manipulator spacecraft, but simply the maintenance package containing parts, tools, and expendables.

b. All research satellites and probes such as the NASA Explorer Series and the USAF Orbiting Vehicle (OV) Series were eliminated because they are in general short life, low cost, special purpose systems which are not suitable candidates for on-orbit maintenance.

c. All non-earth-orbiters were eliminated because they required investigation of long communication time delay effects and special booster analyses which was outside of the scope of the study.

These restrictions resulted in the reduced listing shown in Table A-2.

Table A-3 shows the weight used to estimate the costs of the satellites of Table A-2 using the following equation and the estimated costs

$$C = 0.0625 \frac{W_T}{W_T - W_{P/L}} (W_{TTC,D} + W_P)$$

Table A-4 lists the calculations performed in the process of numerically assessing the candidate systems for the repair missions. The selection criteria and their weighted values appear across the top. Table A-5 provides similar information for the refurbishment missions.

Table A-1. Complete Candidate Satellite Listing

Designation	Agency	Weight (lb)
*Nimbus	NASA	1,260
*OAO	NASA	4,300
*OGO	NASA	1,130
SERT II	NASA	1,080
**ERTS	NASA	1,100
**ASTRA	NASA	4,306
*OSO	NASA	622
PEGASUS	NASA	23,100 (includes Boilerplate Apollo CSM)
VELA	USAF	297
ENVIRONMENTAL SCIENCES RESEARCH SATELLITE	USAF	176
LES 1	USAF	69
SECOR	USA	40
GGSE	USN	103.5
OSCAR	USAF	33
LES 2	USAF	82
EARLY BIRD	CSC	85
LCS	USAF	75
LES 3	USAF	35
LES 4	USAF	115
ESSA	ESSA	305
GGTS	USAF	104
IDCSP	USAF	100
TRAAC	USN	240
INTELSAT 2	CSC	192
DATS 1	USAF	100
DODGE	USN	430
LES 5	USAF	225
TTS	NASA	44
GEOS	NASA	460

*In Programs involving a series of satellites such as Nimbus, OSO, OGO, the Satellite weights vary between each satellite and an approximate weight is used.

**Programs in planning stages and weights are estimates.

Table A-1. Complete Candidate Satellite Listing (Cont'd)

Designation	Agency	Weight (lb)
ATS F/G	NASA	1,843
**DBS	NASA	1,000-3,300
DRSS	NASA	Undefined
USAM	NASA	Undefined
ATS A-E	NASA	758
INTELSAT III	CSC	286
INTELSAT IV	CSC	1,200
**NAV TRAFFIC CONTROL SAT	NASA	660
**ATS H-J	NASA	1,540-2,200
**ATS K-M	NASA	Undefined
**ADV SYNCH MET SAT	NASA	485 - 990
*LUNAR ORBITER	NASA	860
TRANSIT	USN	265
TIROS	NASA	287
TELSTAR	AT&T	170
ERS	USAF	166.5
RELAY	NASA	172
INJUN	USN	114
SYNCOM	NASA	86
LOFTI	USAF/USN	95.2
SOLRAD	USAF/USN	95.1
RADOSE	USAF/USN	95.2
SURCAL	USAF/USN	94.9
HITCH HIKER	USAF	132.6
ANNA	USN	350
GEOPHYSICAL RESEARCH SAT	USAF	220
ORBITING VEHICLE SERIES (OV)	USAF	14-427 (OV 4-3 was modified TITAN II first stage weighing 21,300 lbs)
EXPLORER SERIES	NASA	15-495

Table A-2. Reduced Candidate Satellite Listing

Designation	Agency	Weight (lb)
NIMBUS	NASA	1,260
OAO	NASA	4,300
OGO	NASA	1,130
SERT II	NASA	1,080
ERTS	NASA	1,100
ASTRA	NASA	4,306
OSO	NASA	622
PEGASUS	NASA	23,100
ATS-F/G	NASA	1,843
DBS	NASA	1,000-3,300
DRSS	NASA	Undefined
USAM	NASA	Undefined
ATS-A-E	NASA	758
INTELSAT IV	CSC	1,200
NAV TRAFFIC CONTROL SAT	NASA	660
ATS-H-J	NASA	1,540-2,200
ATS-K-M	NASA	Undefined
ADVANCED SYNCH MET SAT	NASA	485-990

Table A-3. Satellite Cost Estimates

DESIGN DATA SUBSYSTEM WEIGHTS (LBS)	NIMBUS	OAO	OGO	SERT II	ERTS	ASTRA	OSO	PEGASUS	ATS-F/G	DBS (VOICE/ TV DIST)	DRSS	USAM	ATS-A-E	INTELSAT IV	ATS-H-J	ATS-K-M	ADV SYNCH METSAT	NAV TRAF CONTROL SAT
TOTAL (W_T)	1260	4300	1130	1080	-	4306	622	-	1843	1126	-	-	753	1200	1540/ 2200	-	485/ 990	660
PAYLOAD ($W_{P/L}$)	330	1000	200	--	-	1035	252	-	421	188	-	-	223	--	-	-	-	-
TTC&D ($W_{TTC\&D}$)	191	430	135	64	-	140	56.1	-	72	60	-	-	108	--	-	-	-	-
POWER (W_P)	236	739	200	146	-	583	47.1	-	3706	235	-	-	86	--	-	-	-	-
FORMULA COST ($ M)	36	95	25.5	--	-	59.4	10.8	-	36	22.1	-	-	17	--	-	-	-	-
BEST KNOWN COST ($ M)	--	--	--	5.0	-	--	--	9.0	--	--	-	-	--	--	-	-	-	-
DATA SOURCE	GE PROG. OFF.	NASA PROG. OFF.	NASA PROG. OFF.	NASA PROG. OFF.	-	GE STUDIES	NASA PROG. OFF.	---	GE PROG. OFF.	GE PROG. OFF.	-	-	GE PROG. OFF. (ATT. STAB FOR A, D, E	--	NASA/ OSSA PROS-PECTUS 1966	-	NASA/ OSSA PROS-PECTUS 1966	NASA/ OSSA PROS-PECTUS 1966

NOTE: 1. FOR COMSATS, PAYLOAD WEIGHT INCLUDES ANTENNA WEIGHT

2. ON MANY ADVANCED SATELLITE CONCEPTS, LITTLE CONFIGURATION DESIGN AND SIZING HAS BEEN DONE AND CONSEQUENTLY LITTLE DATA IS AVAILABLE.

3. SERT II AND PEGASUS COSTS ARE STUDY TEAM ESTIMATES.

Table A-4. Satellite Evaluation for Repair Mission

ORBIT	SATELLITE DYNAMICS	CRITERIA																									
		SATELLITE COST ($M)					ACCUMULATED ENGRG DATA HDWE DWGS LIT PERS WEIGHTING				ACCUMULATED REL DATA				SATELLITE DESIGN LIFE (YRS)				PLANNED IN-ORB MAINT.		RELATED MAINT STUDIES		PROG COMPL	HARDWARE STATUS UNDWY HDWE	HARDWARE STATUS UNDWY DEVT	CONC	TOTAL
		>40	>30	>20	>10	<10					FLT DATA	FMEA	BLK DIAG & PRED	GENERIC DATA	3	2	1	5	YES	NO	YES	NO					
		10	8	6	4	2	4	3	2	1	4	3	2	1	5	4	3	1	5	0	5	0	1	3	4	5	
LOW	ACTIVE CONTROL																										
	NIMBUS		X				X	X	X	X	X	X	X	X				X		X	X			X			37
	OAO	X					X	X	X	X	X	X	X	X			X			X	X			X			41
	OGO IV			X				X	X					X			X			X		X		X			18
	SERT II					X			X					X		X				X		X		X			12
	ERTS	WILL BE EXISTING SPACECRAFT																									
	ASTRA	X							X	X				X		X			X			X				X	28
	SPIN STABILIZED																										
	OGO I, II, III			X				X	X					X			X			X		X		X			18
	OSO				X			X	X					X				X		X	X			X			19
	OTHER																										
	PEGASUS (UNCONTROLLED)					X			X	X	X			X			X			X		X	X				14
SYNCHRONOUS	ACTIVE CONTROL																										
	ATS F, G		X					X	X	X		X	X	X		X				X		X			X		28
	DBS			X				X	X	X				X	X				X			X				X	28
	DRSS	UNDEFINED							X	X				X	X					X		X				X	14
	USAM	UNDEFINED							X	X				X	X				X			X				X	19
	ATS H-J	UNDEFINED										UNDEFINED			X				UNDEFINED			X				X	10
	ATS K-M	UNDEFINED										UNDEFINED			X				UNDEFINED			X				X	10
	ATS SYNC METSAT	UNDEFINED										UNDEFINED			X				UNDEFINED			X				X	10
	SPIN STABILIZED																										
	ATS B, C				X			X	X	X			X	X	X					X		X	X	X			21
	INTELSAT IV				X				X					X	X					X		X		X			15
	OTHER																										
	ATS A, D, E				X			X	X	X			X	X	X					X		X		X			21
	NAV TRAFFIC CONTROL SAT	UNDEFINED										UNDEFINED			X											X	10

Table A-5. Satellite Evaluation for Refurbishment Missions

ORBIT	SATELLITE DYNAMICS	CRITERIA: ACCUMULATED ENGRG DATA				DESIGN LIFE (YRS)				PLANNED IN-ORB MAINT		RELATED MAINT STUDIES		PAYLOAD DES WEIGHT (LBS)			PAYLOAD POWER AVAIL			HARDWARE STATUS				TOTAL
		HDWE	DWGS	LIT	PERS	3	2	1	.5	YES	NO	YES	NO	<100	100-300	>300	<100	100-1KW	>1 KW	PROG COMP	UNDWY HDWE	UNDWY DEVT	CONC	
						WEIGHTING																		
		4	3	2	1	10	7	4	1	5	0	5	0	1	3	5	1	3	5	1	3	4	5	
LOW	ACTIVE CONTROL																							
	NIMBUS	X	X	X	X				X		X	X				X		X			X			27
	OAO	X	X	X	X				X		X	X				X		X			X			27
	OGO IV		X	X				X			X		X		X			X			X			18
	SERT II		X	X			X				X		X			X			X		X			25
	ERTS WILL BE EXISTING SPACECRAFT																							
	ASTRA			X	X		X			X			X			X		X					X	28
	SPIN STABILIZED																							
	OGO I, II, III		X	X				X			X		X		X			X			X			18
	OSO		X	X					X		X	X			X		X				X			18
	OTHER																							
	PEGASUS III (UNCONTROLLED)		X	X	X			X		X			X			X	X			X				22
SYNCHRONOUS	ACTIVE CONTROL																							
	ATS F,G		X	X	X		X				X		X			X		X				X		25
	DBS			X	X	X				X			X		X				X				X	31
	DRSS			X	X	X					X		X			X			X				X	28
	LSAM			X	X	X				X			X			X			X				X	33
	ATS H-J					X				UNDEFINED			X	UNDEFINED			UNDEFINED						X	15
	ATS K-M					X				UNDEFINED			X	UNDEFINED			UNDEFINED						X	15
	ADV SYNCH MET SATS					X				UNDEFINED			X	UNDEFINED				X					X	15
	SPIN STABILIZED																							
	ATS B,C		X	X	X	X					X		X		X			X			X			25
	INTELSAT III			X		X					X		X	X			X			X				15
	INTELSAT IV			X		X					X		X		X			X				X		22
	OTHER																							
	ATS A,D,E		X	X	X	X					X		X		X			X			X			25
	NAV TRAFFIC CONTROL SAT			X		X					X		X	UNDEFINED			UNDEFINED						X	17
OTHER	ACTIVE																							
	LUNAR ORBITER		X	X				X			X		X		X			X		X				16

APPENDIX B

DESCRIPTION OF SATELLITES

Descriptions of the four satellites selected for mission analysis are provided. These are the Orbiting Astronomical Observatory (OAO), the Orbiting Solar Observatory (OSO), the Direct Broadcast Satellite - Voice Broadcast Mission - UHF (DBS), and Nimbus.

B.1 OAO-A1 SATELLITE DESCRIPTION

The satellite is illustrated in Figures B-1 and B-2. The dimensions and weight are basically a function of the optics required for useful experimentation and limited by available launch vehicle capabilities. The 36-inch-diameter optical system (in the central 48-inch-diameter tube), is 10 feet long and the total spacecraft weight is about 4300 pounds which is within the launch capabilities of the Atlas-Agena D. After orientation in orbit, the sun shade is opened as shown in Figure B-1. It is automatically closed to protect the optical tube from sunlight when an angle less than 45 degrees to the sun-line is reached.

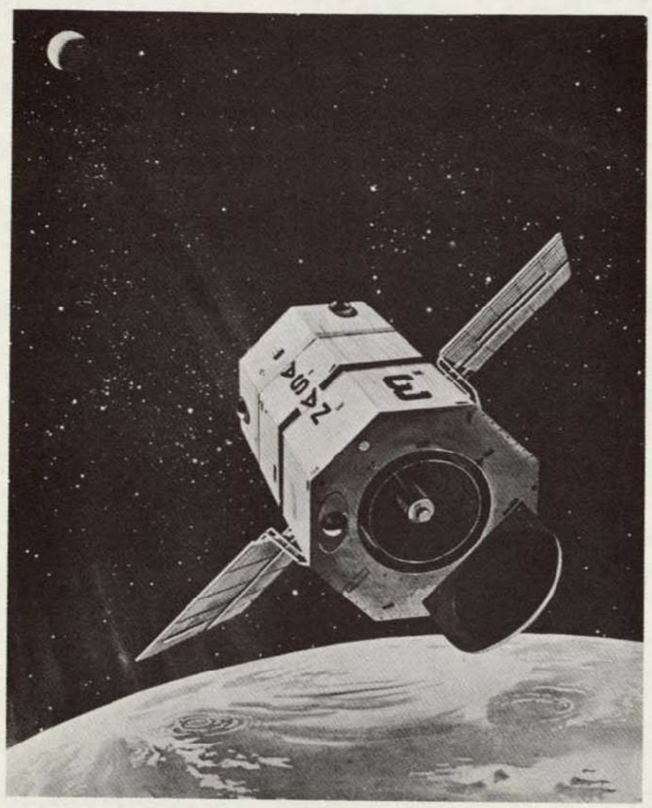


Figure B-1. Orbiting Astronomical Observatory (OAO)

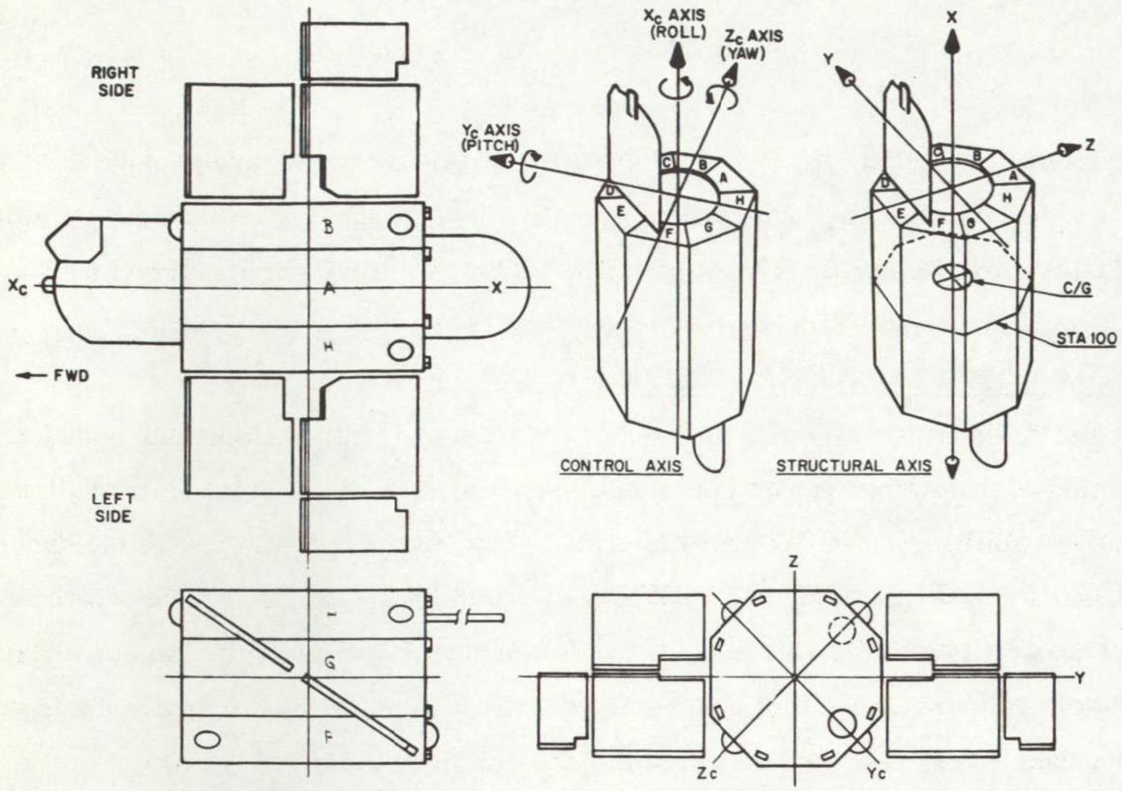


Figure B-2. OAO Coordinate Axis Reference

The internal structure consists of the central tube surrounded by eight vertical trusses and five horizontal shelves to form 48 equipment bays. Many sheet-metal members are chemically milled to save weight and aluminum honeycomb is used in the paddles, sun shades, and equipment shelves for high rigidity. The stabilization and alignment requirements demand extremely conservative design and manufacturing approaches; e.g., riveting, spot welding, and optical alignment. Two tungsten balance weights (40 pounds) are deployed on spring-loaded hinged booms from positions adjacent to bays A and E before separation of the OAO.

After ejection of a fiberglas fairing, the p-n type solar arrays are unfolded from around the octogonal body to a fixed position by torsion and compression springs. The 33-3/4 degree angle of the erected paddles and the saw-tooth angle of the 60,000 solar cells mounted on both sides of the paddles provide the maximum area and the minimum angle from the sun line normal. The OAO is rolled 180 degrees when the sun angle becomes excessive; the small obliquity was accepted to avoid mechanical rotation of the paddles with attendent problems of in-orbit bearings and mass shifts on spacecraft stabilization.

A trip arranged by Grumman personnel was made by study team members to the Goddard Space Flight Center to examine the OAO-hardware and obtain further data beyond the many documents studied regarding the feasibility of making the selected repairs on the OAO-A1 satellite using the remote manipulator spacecraft. The OAO "hangar-queen" configured as the OAO-TA2, the OAO-TA2 satellite itself, and the OAO vibration-test model were examined. The experimental OAO-TA2 model was particularly useful because many pertinent photographs were taken of it, although it contained a number of design changes to improve performance. Discussions with the competent Grumman OAO engineers and technicians also were of significant value.

B. 1. 1 THERMAL DESIGN

The thermal design is almost entirely passive: there is a minimum of electric heaters and there are no automatic louvers on the radiating surfaces. The spacecraft structure is isolated reigorously from sun, earth, space, and equipment heating by isolation techniques such as fiberglass, titanium, and teflon mounting hardware and by radiation shields such as multiple layers of aluminized mylar and highly reflective skin panels. The structure is not used as a heat sink for the electronic equipment. Instead, the electronic equipment radiate to the outer surface of their cases which function as a heat-sink and then as a radiator to the outer skin panels.

B. 1. 2 STABILIZATION AND CONTROL

The stabilization system first reduces the separation tumbling rates by stabilizing on the sun. Then a stellar reference using one of six gimballed startrackers is established. The OAO-A1 is then rotated to the desired pointing direction for the experiment which must be maintained with great precision (1 minute of arc) for long periods of time (maintain the pointing direction within 15 arc-seconds for 50 minutes of time). Using the star experiment as an error source, the fine momentum wheels are then capable of holding an accuracy of 0. 1 arc-second.

The primary sensors consist of rate gyros to measure initial tumbling rates, solar sensors to establish sun direction, and six gimbaled startrackers which acquire selected guide stars and track them continuously. The torquing system consists of a high-thrust nitrogen gas-jet for initial stabilization on the sun, a course momentum wheel system for slewing to other experimental stars and coarse pointing, and a fine momentum wheel system primarily for fine pointing.

B. 1. 3 DATA PROCESSING

The data processing system handles all data going to and from the observatory including commands, experimental data, and flight status on both the spacecraft and the experiment. It also includes two magnetic core storages: one for storing delayed commands, and the other for experimental data while operating out of line-of-sight of a ground station.

B. 1. 4 POWER SUPPLY

The excess of paddle power (total of 680 watts, 24 volts, at 37 degrees inclination from normal) is stored in Ni-Cd batteries for the orbit night. A Battery Charge and Sequence Controller (BCSC) controls the charging of the three 20 amp-hour batteries and selects the proper battery for use. A central power supply system consisting of a voltage regulator-converter and an inverter supplies all observatory and experiment requirements.

B. 2 OSO-D SATELLITE DESCRIPTION

B. 2. 1 CONFIGURATION

The OSO-D has a 44-inch wheel-like main body comprised of nine wedge-shaped compartments (see Figures B-3, B-4 and B-5). The fan-shaped solar array sail is attached to a central rotating shaft. The overall height is 38 inches; the total weight is 622 pounds. The wheel contains apparatus for seven experiments, electronic controls, batteries, telemetry equipment, and radio command equipment. The sail which is stationary during orbit daytime also carries the pointed experiments gimballed about pitch and yaw axes for orientation of primary experiments to within 1 arc-minute of the center of the sun.

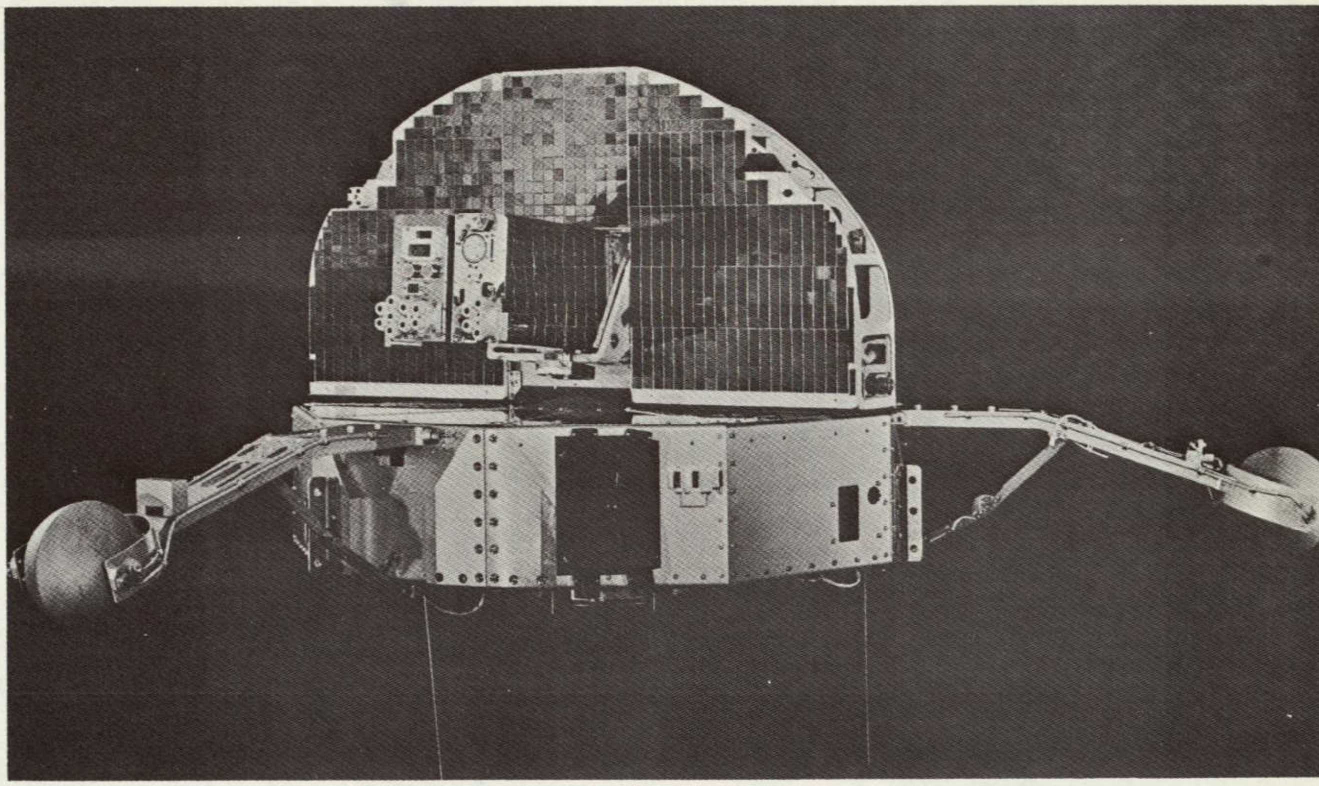


Figure B-3. Orbiting Solar Observatory

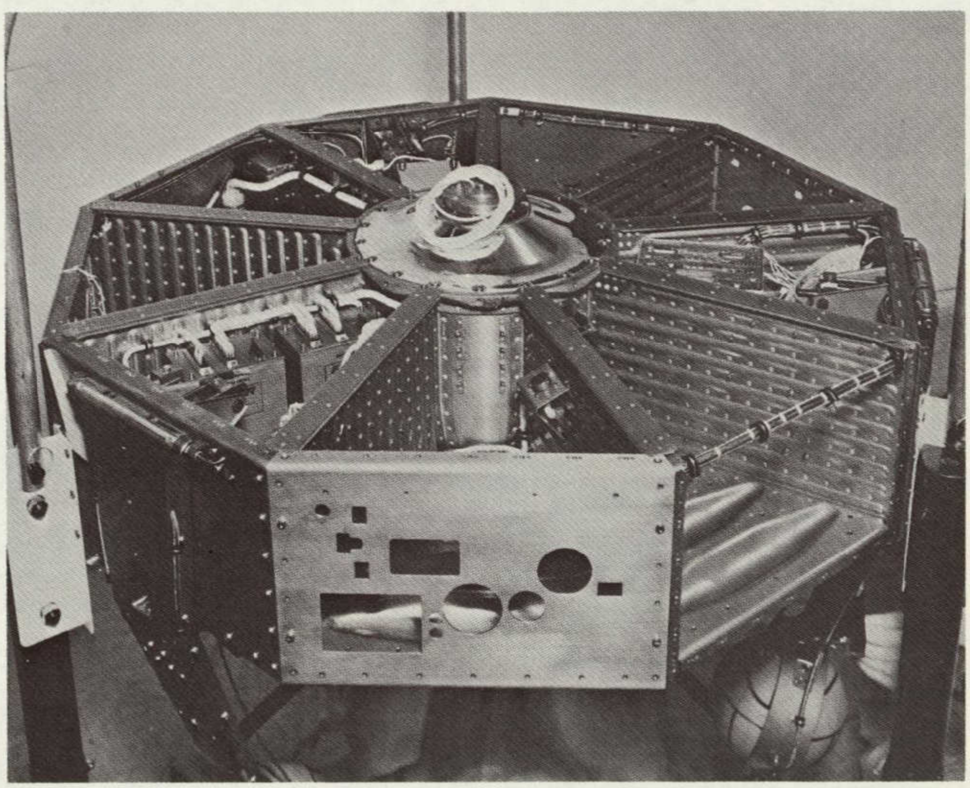


Figure B-4. Orbiting Solar Observatory Interior

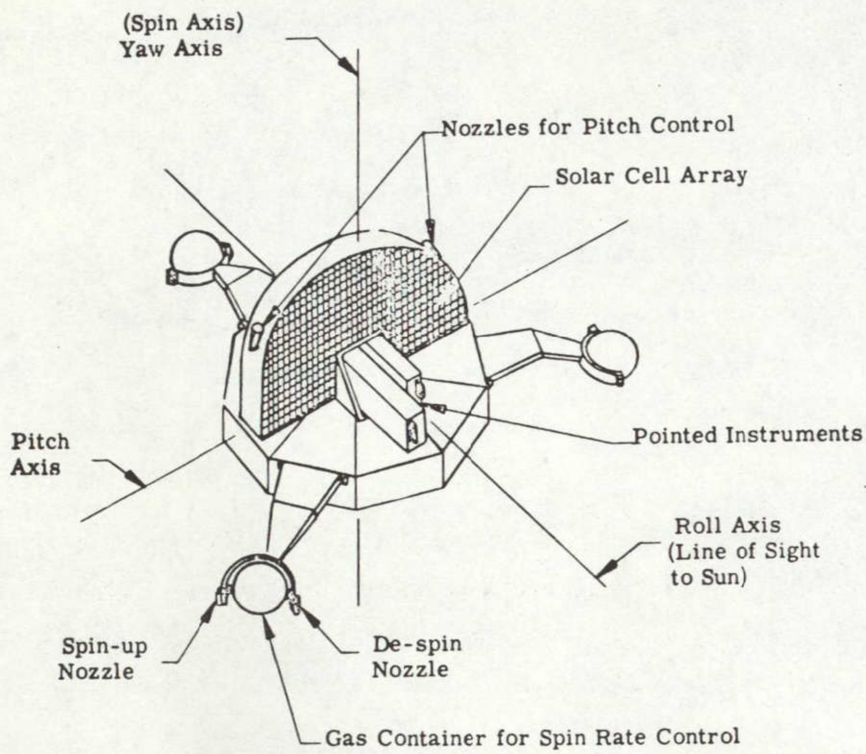


Figure B-5. Orbiting Solar Observatory Physical Characteristics

The study team visited the OSO Project Office at Goddard Space Flight Center to discuss the OSO satellite design, flight history, future OSO plans, and to examine and obtain photographs of the satellite. The photographs lent a great deal of insight into the actual assembly tasks. No hardware was available for inspection at GSFC. All available vehicles were at Ball Brothers Corporation in Denver, Colorado.

B.2.2 STABILIZATION AND CONTROL

The satellite is spin-stabilized by rotating the wheel at 30 rpm; spherical gas bottles are located on radial arms which are erected in orbit and provide torques to maintain this rate (see Figure B-5). At orbit night, the spin rate falls to 25 rpm in approximately 5 minutes when the sail point control is disabled and the sail is dragged up to speed. The wheel plane maintains the satellite sun-line to within $\pm$ 3 degrees. The wheel rotates about the roll axis approximately 1 degree per day. Progressive changes in the satellite attitude permit observation of almost all the celestial sphere and portions of the earth's surface during its 6 month life. A circular magnetic coil, used to conserve the attitude control gas supply, is mounted in the wheel structure normal to the spin axis. Gas jets triggered by solar sensors in the sail automatically control the attitude so that the pitch angle never exceeds the limits by $\pm$ 3.5 degrees.

B.2.3 DATA PROCESSING AND COMMUNICATIONS

The communication system of the satellite performs both command and data transfer functions. Data handling is by pulse-code modulated digital telemetry, which multiplexes the data from the many experiments on each mission and spacecraft housekeeping subsystems. Data are digitized and stored on a tape recorder during each orbit and read back at a high rate during 5-minute passes over the ground stations. Data acquisition rates are adjusted to each experiment, and range from approximately 3 to 100 bits per second. The command system operates on a pulse-width modulated audio tone and has the capability of 70 commands. Typical functions controlled by the command system are tape recorder playback and record, transmitter selection, experiment turn-off and turn-on, and individual experiment control.

B.3 DIRECT BROADCAST SATELLITE (DBS-VBM/UHF) DESCRIPTION

B.3.1 CONFIGURATION

The spacecraft is shown in the launch configuration in Figure B-6 and in the orbital configuration in Figure B-7.

The roll-out solar array panels would be deployed by extendable rod devices along the principles of Hunter spirator, the deHaviland STEM, or possibly the Ryan foldable beam concept, depending upon the natural frequency requirements of the system.

The antenna structure is deployed by means of pneumatic system to pressurize the wire grid tube members. The antenna feed is a fairly rigid telescoping-type structure to support the necessary additional equipment and maintain a desired natural frequency.

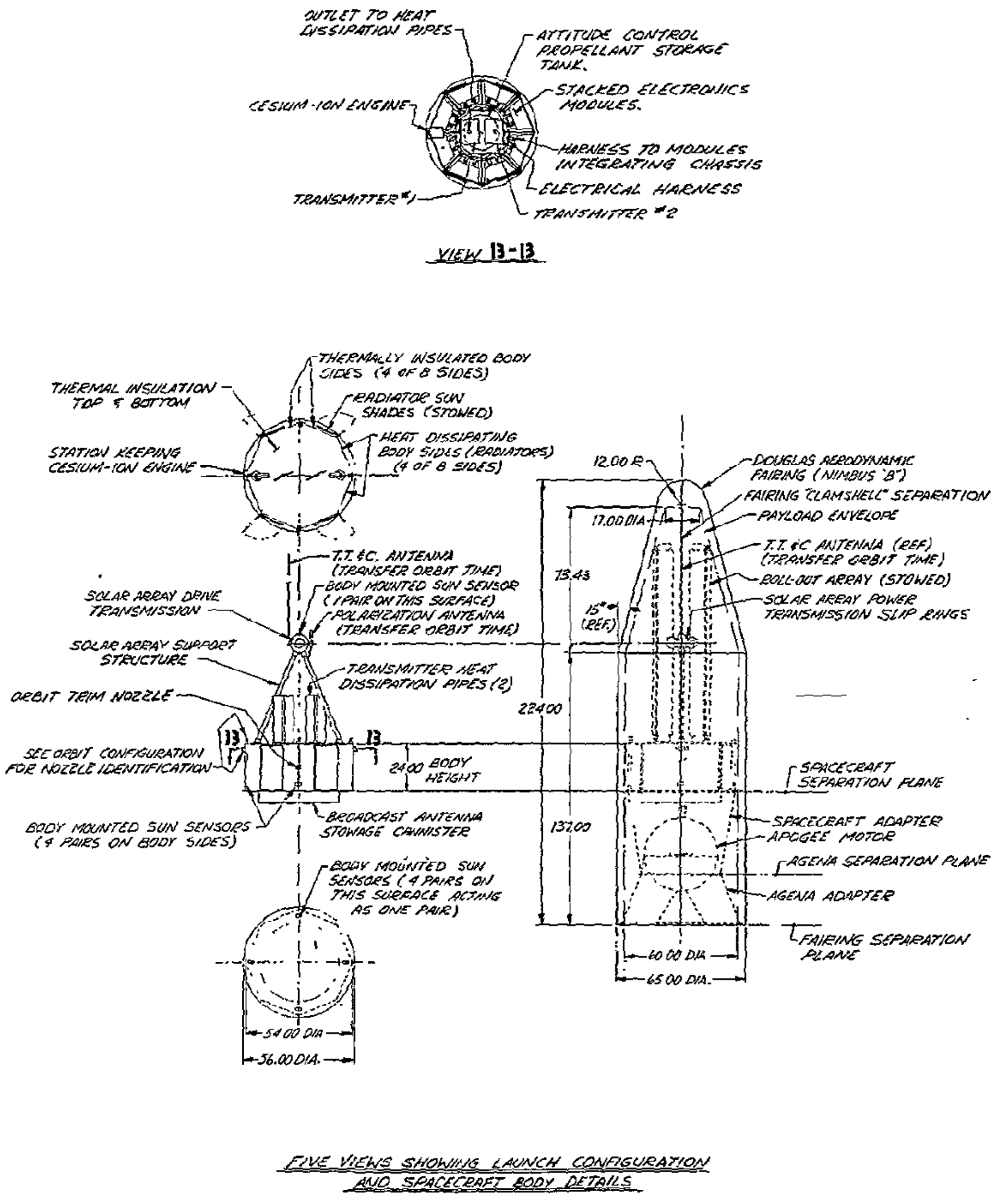


Figure B-6. Launch Configuration of the DBS Satellite

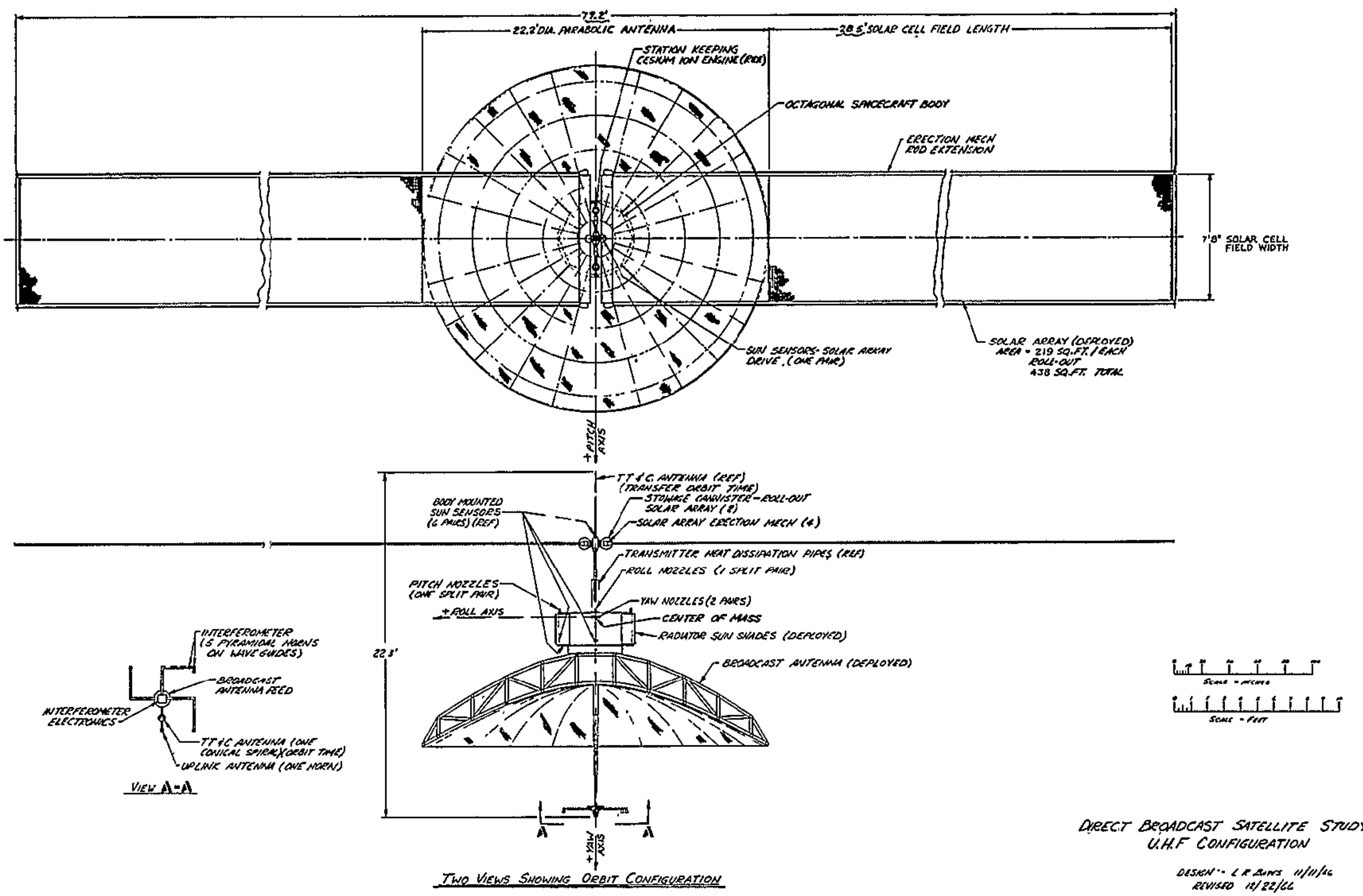


Figure B-7. Direct Broadcast Satellite - Voice Broadcast Mission UHF Configuration

The selection of an 8-sided structure rather than 4 or 16 sides was made based upon an 8-point back-up structure in the booster. The other considerations made were packaging volume related to package size, surface area, and orientation requirements for thermal control. Flat sides rather than circular were selected to facilitate standardized component packaging techniques (such as used on Mariner, Ranger, Voyager, and Nimbus) that are compatible with thermal control requirements.

The heat pipes for transmitter temperature control were mounted on the end of the spacecraft nearest the solar array. The vacuum tube operates at a high temperature enabling a compact design of this system. Two pipes are shown for redundancy.

The housekeeping system utilized four radiating surfaces and four automatically operated sun shades to prevent sun impingement on the surfaces and to act as insulation during the occultation periods.

A satellite weight breakdown is provided in Table B-1 and description of the major satellite subsystems is provided in Table B-2.

B.3.2 SATELLITE COMMUNICATIONS SUBSYSTEM

The UHF satellite transmits in two modes. Mode one is the TV-aural model. The satellite provides only voice material in this mode. However, two transmitters are required because industry has universally adopted the intercarrier method of sound reception. Very few, if any, of the separate channel receivers are still in use, so that probably 99.9 percent of the receivers used today are of the intercarrier sound type. These receivers have but one intermediate frequency channel; this channel passes both picture IF and sound IF. The second detector is an envelope detector producing output signals for the video pattern of the receiver as well as an FM aural output of 4.5 MHz. The picture portion, usually not extending beyond 4.2 MHz, is filtered out of the detector composite signal and is portrayed on the picture tube. The sound portion, a narrow band some 120 kHz wide, is filtered out, amplified, and applied to a suitable discriminator-detector to obtain audio frequency, which, after amplification, is applied to a loud-speaker. Thus, although it is theoretically possible to

Table B-1. Weight Payload Summary

Item	Weight (lb)
Satellite Subsystems	
Transmitter/Receiver System	(97)
Transmitters	86
Receivers	11
Broadcast Antenna System	(91)
Power System	(338)
Power Generation	197
Batteries	54
Conditioning	87
Array Pointing	--
Attitude Control, Stabilization, & Orbit Correction System	(277)
Autopilot	9
Mass Expulsion	71
Stabilization	186
Stationkeeping	11
Telemetry, Tracking and Command	(60)
Structure	(120)
Thermal Controls	(63)
Electrical Distribution	(80)
Spacecraft Weight (in orbit)	1126
Spacecraft Adapter	78
Apogee Motor	1330
Spacecraft at Booster Separation	2534
Booster Adapter	107
Launch Weight	2641
Booster	SLV-3A/Imp. Agena D & Apogee Kick
Payload Capability	2750

Table B-2. Subsystem Design Values

Subsystem	Design Value
<u>Antenna</u>	
Type	Parabola
Size	22.5 foot diameter
HPBW, Degrees	3.75
Gain, db	33.0
<u>Transmitter</u>	
Type	Vacuum Tube
RF Power Out, kw	1.15
Efficient, %	59
Power In, kw	1.95
<u>Power</u>	
Prime Power for Transmitter, kw	2.29
Prime Power for S/C, kw	0.25
Tot. Prime Power, kw	2.54
Type of System	Oriented Solar Array
Array Area, ft^2	438
<u>Thermal Control</u>	
Radiator Area, ft^2:	
Transmitter	2.6
Housekeeping	16.5
<u>Attitude Control</u>	
Type:	
Antenna/Body	Stabilized to Interferometer ground station/sun
Solar Array Module	Oriented to sun by 1 DOF gimbal + vehicle yaw
Accuracy, Degrees per axis	$\pm$ 0.1
<u>Antenna Pointing</u>	Electrical axis is determined by ground test and mechanically aligned to interferometer axis which is maintained by closed loop control

transmit only the FM aural carrier to provide sound, actually, in practice, receivers rely on both carriers being present: (1) an FM aural carrier, modulated according to TV standards and (2) an unmodulated picture carrier wave.

Mode two is a wideband FM mode, and in this mode the transmitter broadcasts a wideband FM signal.

The satellite transmitter section consists of three separate transmitters: one for the wideband transmission and one each for the picture carrier and sound signal. Each transmitter requires a separate input.

B. 3. 2. 1 Satellite Transmitter

The UHF transmitter will be required to operate at approximately 870 MHz and a power level of approximately 575 watts for each carrier. The wideband FM transmitter requires approximately 1150 watts.

The UHF audio transmitter must provide two carriers. One carrier will be CW at the visual carrier frequency in the television channel selected. The other carrier will be frequency-modulated by the audio signal, and its frequency will be that of the aural carrier in the selected channel (4. 5 MHz higher in frequency than the visual carrier). It has been determined that the two carriers should have equal power levels.

The UHF transmitter block diagram is illustrated in Figure B-8.

B. 3. 2. 2 Receiver

The receiver input will be an X-band signal. For the wideband mode, the uplink signal will have the required downlink modulation.

In the TV-aural mode, the picture carrier and sound signal will have the required downlink modulation and frequency difference. The receiver is required to translate these inputs at the appropriate UHF frequencies.

The block diagram of Figure B-9 illustrates the selected receiver configuration.

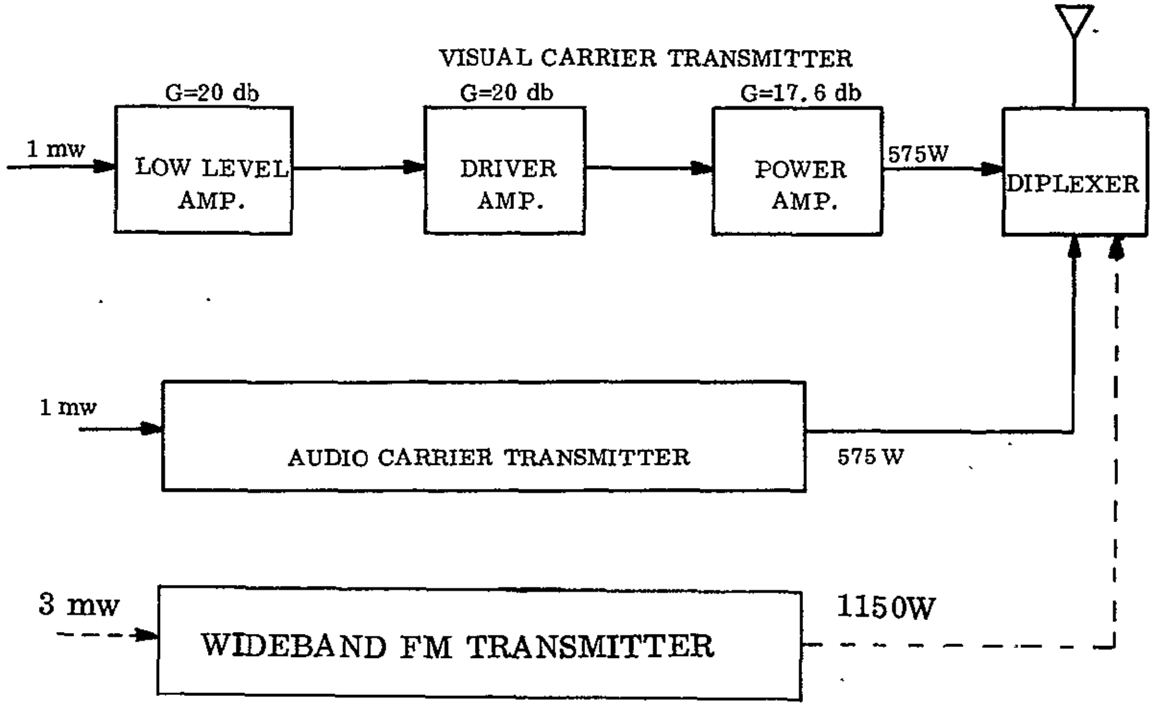


Figure B-8. UHF Transmitter Block Diagram

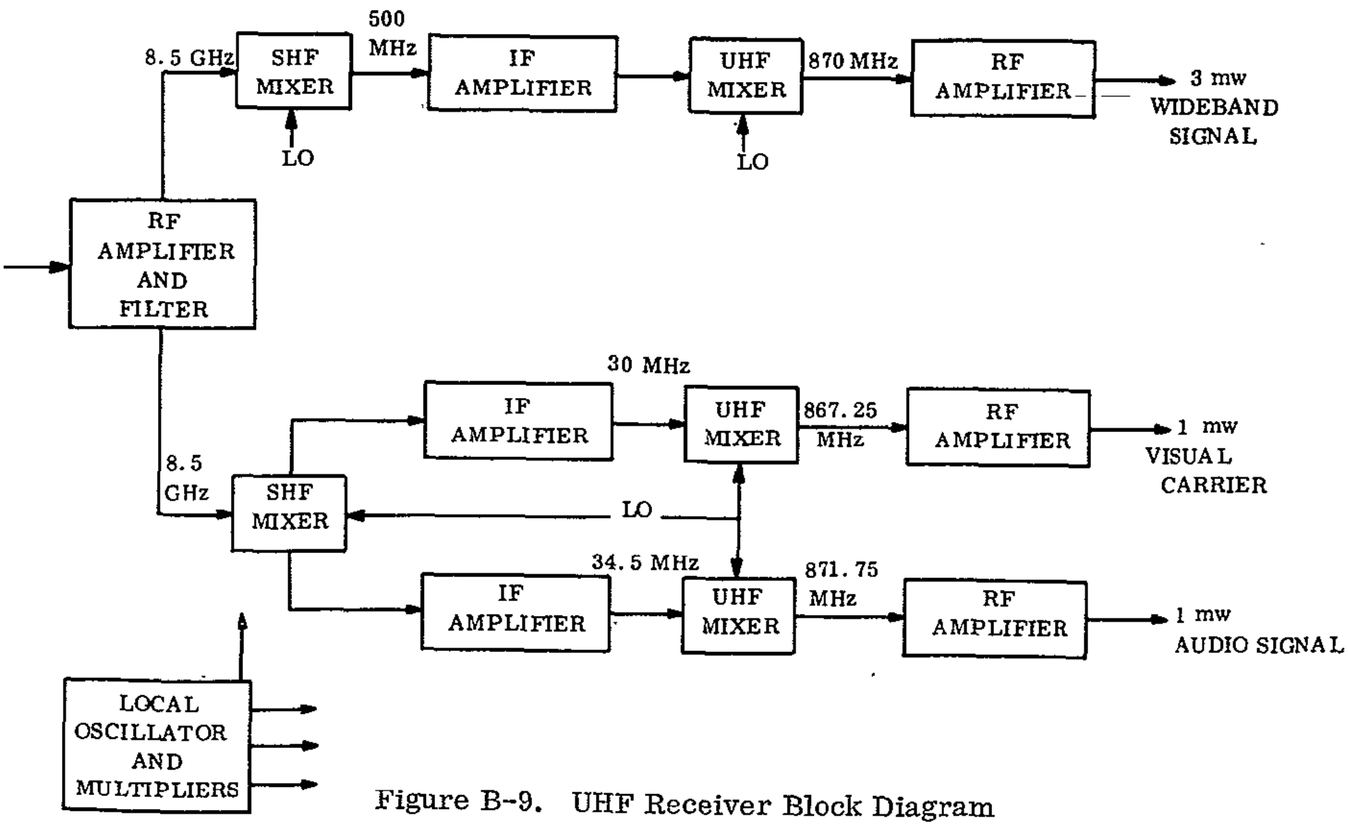


Figure B-9. UHF Receiver Block Diagram

B. 3. 2. 3 DBS Component Packaging

In order to perform the component replacement tasks, the satellite component packaging technique was examined. Electronic components were assigned to a toroidal section around the periphery of the body to accommodate heat dissipation requirements. Flat sides were chosen for the body to permit usage of modularized electronic packaging technqiues similar to those employed on Mariner and Ranger Programs, and Voyager studies. Harnessing was a ring harness around the inside of the equipment section. The modular packaging design facilitates repair and rework of assemblies. It is possible to replace any subassembly without mechanically or electrically disturbing any other subassembly. The electronic equipment is illustrated in Figures B-10 and B-11. The electronic assembly is attached to the vehicle longerons by means of bolts passed through the integral bathtub fittings on the individual subassemblies. After the assembly is in place, the thermal control/shear panel is bolted to the subassemblies and the spacecraft structure, completing the load and thermal paths to the spacecraft frame. Any subassembly may be removed and replaced by removing the outer panel, and disconnecting its fasteners and connectors.

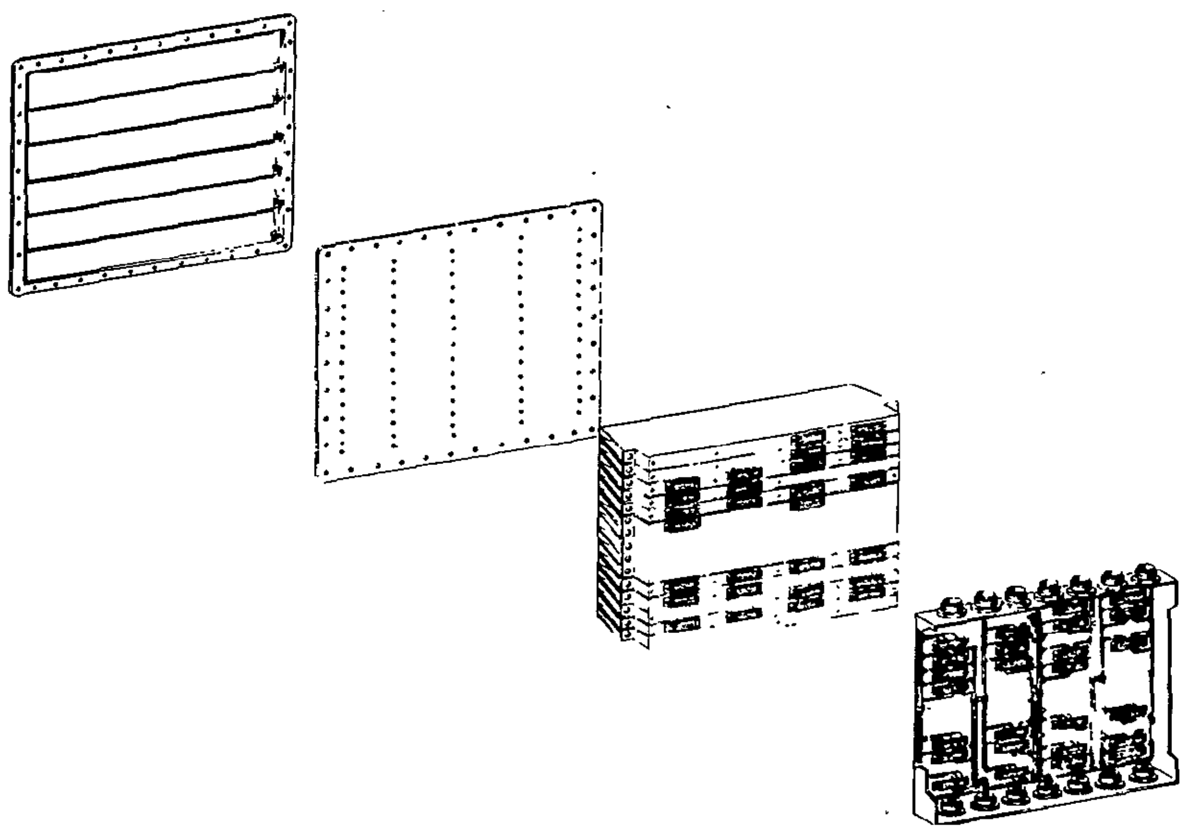


Figure B-10. Electronic Equipment Assembly (Exploded View)

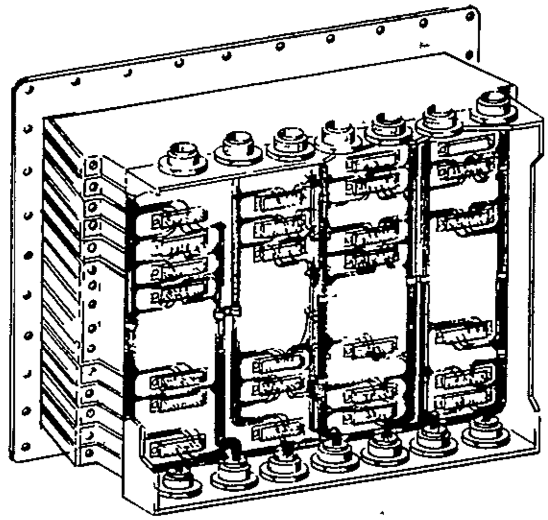


Figure B-11. Electronic Equipment Assembly

The electronic assemblies are joined to the system harness through a series of system interface connectors, located at the upper end of the harness tray which is accessible from the top of the equipment bay.

B.4 SATELLITE CONFIGURATION - COMMUNITY BROADCAST FOR INDIA

The preliminary configuration of a Community Broadcast Satellite for India derived in-house by the General Electric Company is provided for comparison with the DBS-VBM (UHF) satellite configuration. The satellite configuration is shown in Figure B-12 and a payload weight summary appears in Table B-3. Table B-4 describes the various satellite subsystems

B.5 NIMBUS SATELLITE DESCRIPTION

B.5.1 CONFIGURATION

The 1260-pound Nimbus spacecraft (Figures B-13 and B-14) is approximately 10 feet high from the base of the sensory ring to the top of the command antenna which results in a long torque arm for the controls because of the low cneter of gravity. The spacecraft consits of three major elements: the rotating solar paddles provide the basic electric power supply, the upper hexagonal package contains the complete attitude stabilization and control system, and the 54 inch diameter sensory ring contains eighteen standard-size bays in the heat-sink, toroidal structure which houses the major electronic systems (Figures B-15, B-16 and B-17).

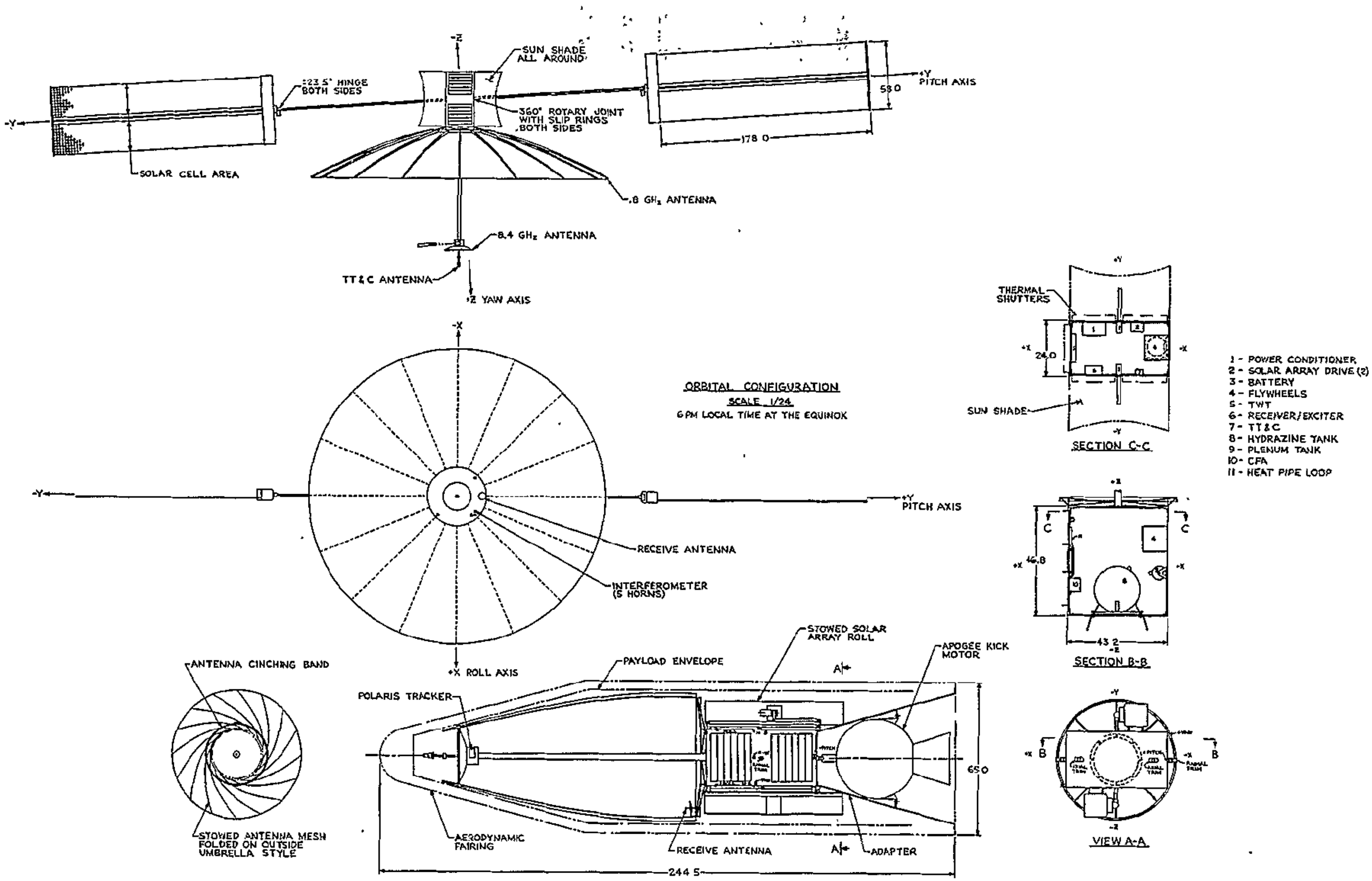


Figure B-12. Television Broadcast Satellite Community Service to India

Table B-3. Payload/Weight Summary

Satellite Subsystems	Weight (lb)
Power Supply	150
Receiver Exciter	28
Transmitter	33
Antenna	120
Attitude Control	86
Thermal Control	43
Structure	137
Propulsion	135
TT&G	20
Total	752

Table B-4. Subsystem Design Values

Subsystem Antenna	Design Value Community Broadcast	Design Value Distribution
Type	Parabola	Parabola
Size	21.0 feet diam.	2.0 feet diam.
HPBW	4.1°	4.1°
Gain, db	32	32
Transmitter		
Type	Cross Field	TWT
RF Power Out, w	416	20
Efficiency, %	58	35
Power In, w	718	57
Power		
Prime Power for XMTR, w	843	67
Prime Power for S/C, w	190	--
Total Prime Power, kw	1.1	--
Type of System	Oriented Solar Array	--
Array Area, ft^2	124 ft^2	--
Thermal Control		
Radiator Area, ft^2	9.8	
Attitude Control	3-Axis Active-Momentum Wheel	

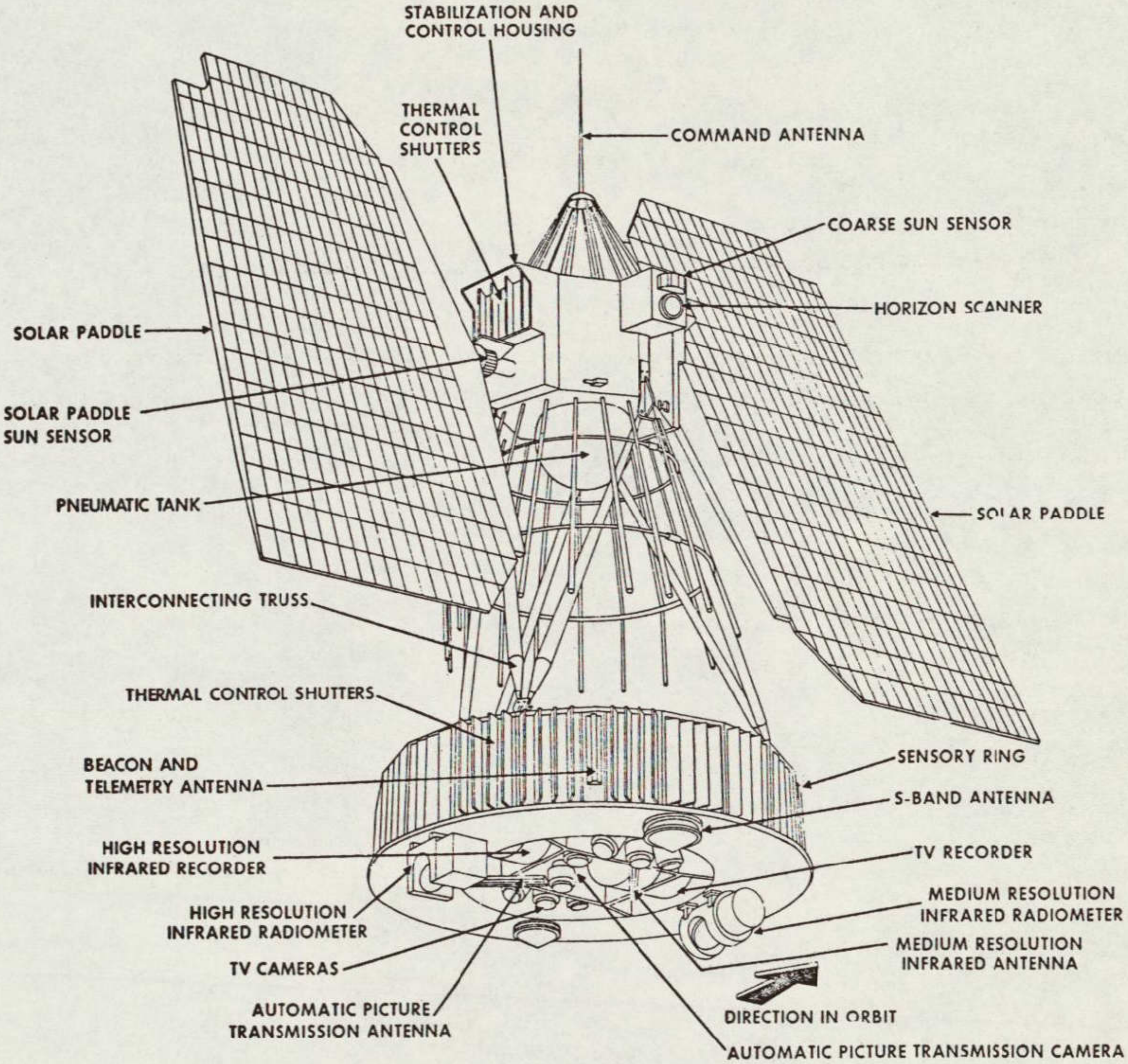


Figure B-13. Nimbus Spacecraft

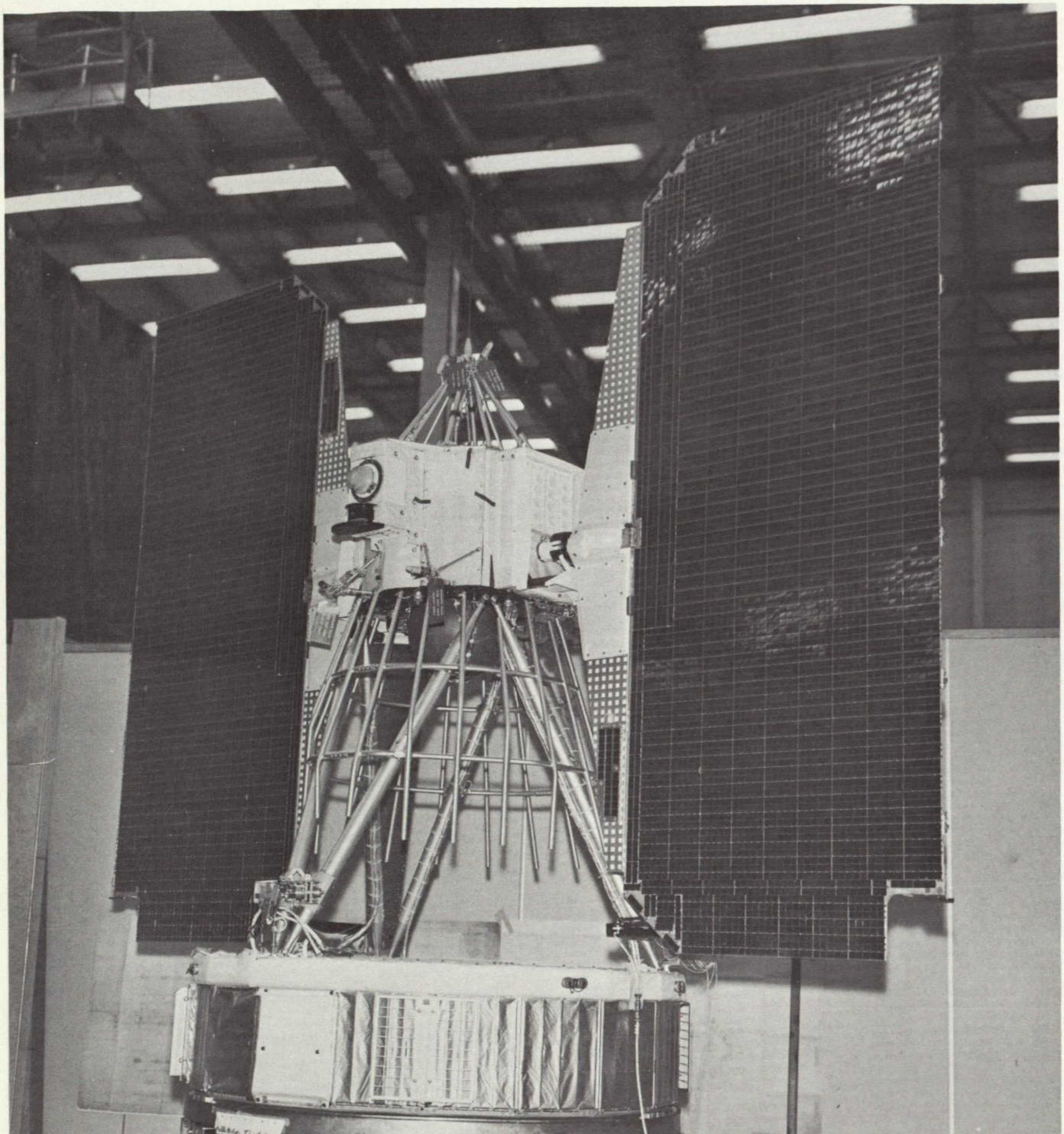


Figure B-14. Nimbus Spacecraft

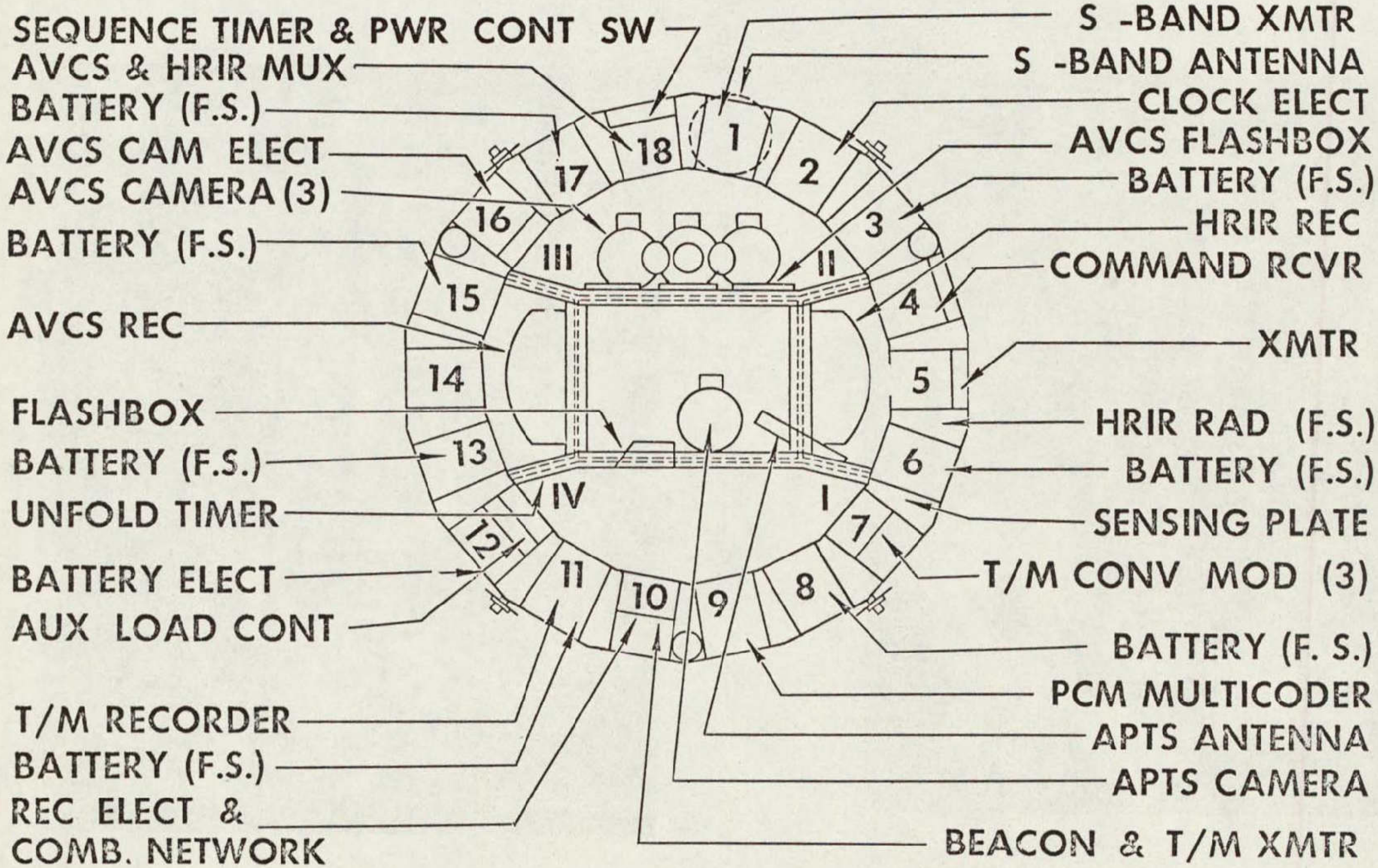


Figure B-15. Nimbus A Component Locations in Sensory Ring - Top View

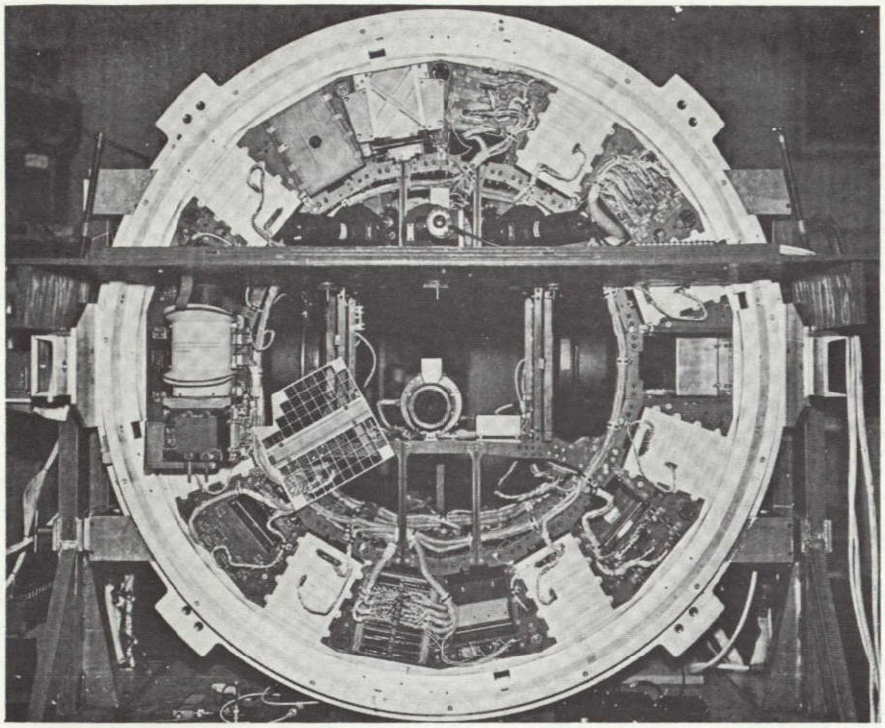


Figure B-16. Nimbus A Component Locations in Sensory Ring - Bottom View

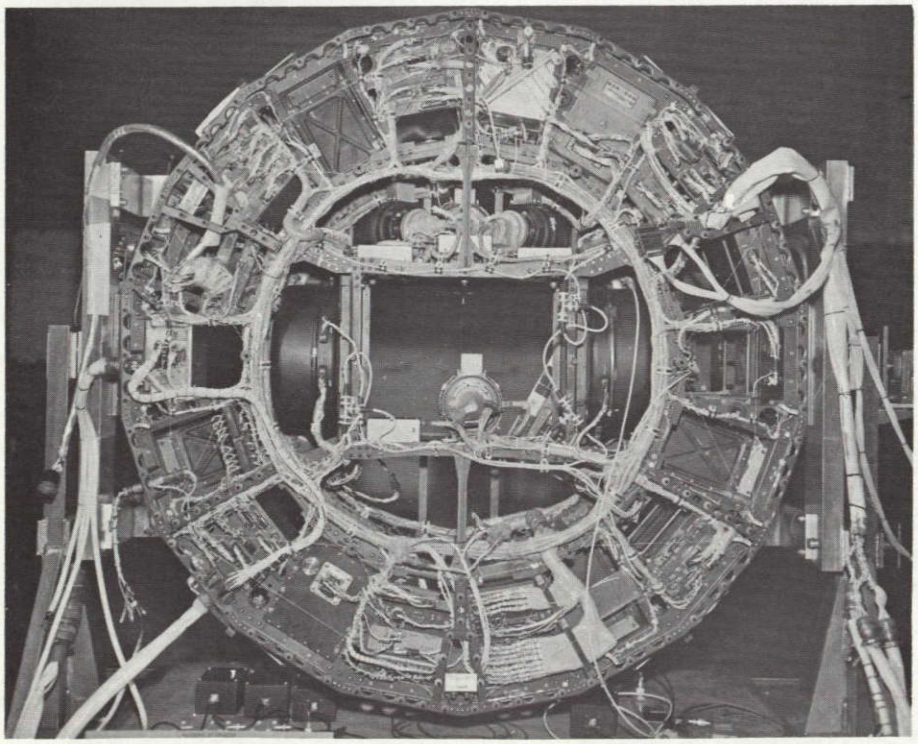


Figure B-17. Nimbus A Component Locations in Sensory Ring - Top View

This toroidal structure is quite rigid and functions as a heat-sink. Large and bulky elements such as cameras and tape recorders are mounted on the H-frame in the center of the toroidal structure. The Nimbus spacecraft is a completely modularized design which allows for separate development, evolution, and improvement of individual subsystems with a minimum of interface problems. There is a high ratio of module packaging density to structural weight allowing the standardization of subsystem size. Furthermore, the modules can be easily arranged to obtain optimum c.g. location and thermal balance. This flexibility feature greatly enhances in-orbit refurbishment of the Nimbus A and Nimbus D spacecraft into Nimbus C and Nimbus E spacecraft, respectively, in all aspects of design: mechanical, thermal, and electrical. For example, the modules have external Cannon connectors, and all electrical integration is accomplished by means of an external wiring harnesses using a connector-to-connector (Cannon) principle (i.e., no hard wiring). Furthermore, each of the 18 peripheral flat faces on the sensory ring can readily accommodate a shuttered thermal-control assembly which is primarily sensitive to the thermal state of the modules in that compartment and helps maintain the temperature of all components at 25 $\pm 3^{\circ}$C at all times. The structural system is thermally homogenous and well-damped; mechanical tolerances are closely held by spring-loaded pressure-plates and shimming.

B.5.2 STABILIZATION AND CONTROL

The control system is housed in an independent structure and is attached to the sensory section by truss members of adjustable length which results in a nearly equal moment of inertia distribution and a rigid structure between the controls and mission sensors. A bearing-supported shaft extends through and protrudes from the control housing. This shaft has slip rings for the solar paddles attached to it. It is driven by a motor gear-head drive assembly in the control housing; a 165 in.-lb slip clutch protects the gear-train mechanism. The attitude control system keeps the spacecraft pointed towards the earth with a pointing accuracy better than 1 degree in all three aces. In addition, the spacecraft rotational rates are extremely slow in pitch and roll, less than 0.05 degree/second and 3 degrees per second in yaw.

B.5.3 POWER SUPPLY

The solar power system consists of batteries located in the sensory section and charged by the two rotating solar cell paddles (approximately 3 by 8 feet each). Solar sensors on the shaft continuously point the paddles at the sun until orbit night whereupon they are rotated backwards to await the next solar day. Each paddle is folded on a lengthwise hingeline near the inboard edge of the paddle in order to fit within the shroud envelope of the booster. They are fastened together and to the sensory ring by a paddle lock mechanism. After separation, the paddles unlock and are erected by mechanical drive systems on the triangular transition sections of the paddles.

B.5.4 COMMAND AND COMMUNICATIONS

The spacecraft contains a command system and a housekeeping telemetry system (4000 data points per minute) for monitoring the in-orbit performance of the spacecraft. These housekeeping data include spacecraft attitude error measurements which are required for the interpretation and accurate position reference of the sensory data.

APPENDIX C

OAO FAILURE ANALYSIS DATA

The data used in the OAO failure analysis is summarized in this appendix. Also included is Solder-Alloy Composition Table from Federal Specification QQ-S-571 which describes the composition of the solders used in OAO.

A summary of the tests conducted in the post-flight failure analysis program by Grumman and cognizant sub-contractors and the results of each test is given in Tables C-1 through C-5.

Table C-1.
OAO A-1 Failure Verification Power Supply Subsystem Tests

Test	Objective	Test Conditions	Results
1. Normal operation	To demonstrate nor- operation	Simulated Orbital operations. Ground environment with batteries in a thermal enclosure	During 12 hr. test period, battery temperatures rose from 79°F to 104°F. Normal transfer of batteries occurred, normal regulator turn-on and turn-off occurred. The batteries charged and discharged normally and heat dissipation was normal.
2. Simulated BCSC failure	To determine performance with BCSC relay contact (K201) permanently open	Simulated Orbital operations. Ground environment with batteries in a thermal enclosure	During 7.5 hr. test period, battery temperature rose from 92°F to 140°F. Over voltage and undervoltage transfers from battery did not occur and attempts to transfer by command were unsuccessful. The regulators turned on when temperature reached 110°F.
3. Simulated battery cell short	To determine performance with one cell of battery No. 2 shorted and its thermostat lead open	Simulated Orbital operations. Ground environment with batteries in a thermal enclosure	During 11.5 hr. test period, battery temperature rose from 83°F to 112°F. Normal over voltage and undervoltage transfers occurred and normal regulator turn-on and turn-off occurred. Batteries transferred successfully by command in both light and dark periods until battery temp. reached 110°F. Above, this temp. transfer by command was successful only in the dark.
4. System run down and battery transfer.	To investigate operation of BCSC logic with an open K201 relay contact and to determine battery transfer mechanism.	Simulated Orbital operations. Ground environment with batteries in a thermal enclosure	Under steady state load conditions, battery 1 did not transfer either automatically or on command. When the fault was cleared, successful transfer to battery 2 was accomplished. When battery 1 was removed from the system, transfer did not occur when its voltage was above 10V, but did occur at voltages between 5-10V each time the battery was reapplied into the system. With unregulated load applied transfer to battery 2 occurred when battery voltage dropped to 6.5V and unregulated load was removed. Chattering between relays for batteries 2 and 3 was also observed. Transfer of regulated load from +18V to +28V and removal of +28V load did not result in battery transfer.
5. Susceptibility	To determine the effect of audio ripple and transient voltages on power subsystem lines	Simulated Orbital operations. Ground environment with batteries in a thermal enclosure	The power subsystem was not susceptible to the audio voltages or to ± 405V transients injected on the ± 28V and 10V regulated lines. Certain transients on the ± 28V unregulated line and the + 10V regulated line caused battery switching and BCSC regulator turn on.

Table C-2.
OAO A-1 Failure Verification BCSC and Battery Tests

Test	Objective	Test Conditions	Results
1. Simulated Faults	To verify BCSC logic and determine the effect of simulated faults.	Ambient laboratory conditions. Battery temperature was simulated.	BCSC logic was checked per conditions in the truth table and was verified by tests. Nine individual BCSC faults were simulated and although several failure modes were identified only one failure resulted in performance similar to that experienced on OAO A-1: an open in relay K-201, K-301, or associated wiring.
2. Simulated star tracker arcing	To determine if star tracker arcing could cause BCSC failure.	Ambient laboratory conditions. Battery temperature was simulated.	The High voltage transients impressed at various BCSC inputs produced some anomalous behavior but none that would explain the operation seen during the OAO A-1 flight.
3. Battery high temperature tests	To determine the effect of high temperature on charge-discharge characteristics	Elevated temperatures.	Cells operated up to 170°F however capacity continuously decreased as charge-discharge cycles progressed and dropped below undervoltage transfer point after the 22nd cycle.

Table C-3.
OAO A-1 Failure Verification Stabilization and Control Subsystem Tests

Test	Objective	Test Conditions	Results
1. Star tracker high voltage power supply breadboard test	To determine effects of star tracker arcing and to measure the resulting transients	Vacuum chamber and laboratory ambient conditions	Arcing occurred at 50 mm of Hg. In the vacuum chamber. Induced arcing on the bench produced transient spikes of large amplitudes and nanosecond rise times.
2. Star tracker altitude test at Kollsman Instrument Corp.	To determine effects of star tracker arcing and to measure the resulting transients	Altitude chamber.	Arcing occurred at 65 mm of Hg. and caused failure of 2 transistors in the inverter module.
3. Star tracker vacuum test at GE	To determine effects of star tracker arcing and to measure the resulting transients	Vacuum chamber. SDHE module connected.	Arcing occurred when chamber was back filled with nitrogen and also when back filled with argon. Firing N_2 gas jet also caused arcing. The SDHE module did not fail.

Table C-4.
OAO A-1 Failure Verification Data Processing Sybsystem Tests

1. PPDS/PSSC transient tests	To determine the effect of transients on power and signal lines	Laboratory ambient	Some circuit failures occurred indicating that level of transients was higher than experience on A-1. None of the A-1 anomalies were reproduced.
2. RRG transient test	To determine the effect of transients on PPDS/PSSC - RR6 supply voltages	Laboratory ambient	-6V transient on the +18V line caused system clock and command memory resets. This transient is higher than considered possible for existing design.
3. SDHE analog gate tests.	To obtain information on mode and mechanism of SDHE analog gate failure	Laboratory ambient	Failures were induced by discharging a capacitor across the gates. Damage to gates is cumulative until sufficient gates have failed to act as a shunt. Gates with low AC impedance fail first.
4. SDHE mode change tests.	To investigate ways of producing uncommanded mode changes		Mode changes were produced by injecting voltage transients on the case (signal ground) of the SDHE.

Table C-5.
OAO A-1 Failure Verification Spacecraft Tests

Test	Objective	Test Conditions	Results
1. Spacecraft Simulator Tests.	a. To determine anomalies due to star tracker arcing.	Spacecraft simulator at room ambient conditions except for a single star tracker optical package in vacuum chamber.	Eleven tests were run. The vacuum chamber was operated at critical pressure to produce arcing. The following observations were made during the tests: SDHE rows failed, startrackers turned on in advertently, system clock resets occurred, RRG's were initiated, batteries switched, inverter switched, regulator switched, startrackers gimbal command angles changed, SDHE detailed status was lost, SDHE changed mode, SCU turned on and off. Neither the startrackers nor the BCSC was damaged during the tests.
	b. To determine if the BCSC would be damaged when two startrackers are simultaneously	Same as above except for an additional startracker added to spacecraft simulator.	Similar to above. The BCSC was not damaged
	c. To determine if startracker arcing causes roll jet firing	Same as above except for an additional startracker added to spacecraft simulator.	Fifteen tests performed with various test configurations indicated that roll jet firing is caused by inverter switching which in turn is caused by startracker arcing
2. Spacecraft B tests	a. To determine if jet solenoid transients cause clock resets	Laboratory ambient	All jets were fired simultaneously. Transients observed were not considered to have caused A-1 anomalies.
	b. To determine if gyro turn on/off transients cause clock resets.	Laboratory ambient	No transients of sufficient magnitude were observed on lines which could have caused A-1 anomalies.
	c. To determine the effect of grounded vs. ungrounded IFCM and the effect of injecting noise on selected power and signal lines	Laboratory ambient	Tests simulated the first 16 minutes of operation on nine consecutive runs. None of the A-1 flight anomalies occurred either due to normal command or transient injection. However, during the test, a high thrust roll jet reversed direction when startrackers were turned on, and telemetry data indicated abnormal variations (+ 200 mv on SDHE 0-5V range) in power supply battery and cell voltages.
	d. To determine if discharge of static charge build-up on skin surfaces could cause an interference problem	Laboratory ambient	Electrical bonding measurements (point to point ohmeter readings) indicate that potential differences can exist and could result in potential discharges between isolated points.

C.2 Solder Compositions

The solders used in the manufacture of spacecraft are generally procurred to Federal Specification QQ-S-571 and are usually type SN60 (60 percent tin). Table C-6 shows the composition of solder-alloys according to QQ-S-571.

Table C-6. Solder-Alloy Compositions

Composition	Tin	Lead	Antimony	Bismuth max	Silver	Copper max	Iron max	Zinc max	Aluminum max	Arsenic max	Cadmium max	Total of all others, max
Sn96	Remainder	0.10, max	----	----	3.6 to 4.4	0.20	----	0.005	-----	0.005	0.005	----
Sn70	69.5 to 71.5	Remainder	0.20 to 0.50	0.25	---	0.08	0.02	0.005	0.005	0.03	-----	0.06
Sn63	62.5 to 63.5	Remainder	0.20 to 0.50	0.25	---	0.08	0.02	0.005	0.005	0.03	-----	0.06
SN62	61.5 to 62.5	Remainder	0.20 to 0.50	0.25	1.75 to 2.25	0.08	0.02	0.005	0.005	0.03	-----	0.08
SN60	59.5 to 61.5	Remainder	0.20 to 0.50	0.25	---	0.08	0.02	0.005	0.005	0.03	-----	0.08
SN50	49.5 to 51.5	Remainder	0.20 to 0.50	0.25	---	0.08	0.02	0.005	0.005	0.025	-----	0.08
SN40	39.5 to 41.5	Remainder	0.20 to 0.50	0.25	---	0.08	0.02	0.005	0.005	0.02	-----	0.08
SN35	34.5 to 36.5	Remainder	1.6 to 2.0	0.25	---	0.08	0.02	0.005	0.005	0.02	-----	0.08
SN30	29.5 to 31.5	Remainder	1.4 to 1.8	0.25	---	0.08	0.02	0.005	0.005	0.02	-----	0.08

APPENDIX D

OSO-D FAILURE EFFECTS ANALYSIS

TABLE D-1. OSO Failure Effects Analysis

1.0 SPIN CONTROL SYSTEMS

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
1.1 Automatic Spin Control System	Automatically maintain spin rate between 27.0 & 39.6 rpm (30 rpm ideal)	None - Use command backup system			
1.1.1 Sensors	Generate pulse when exposed to sun	None - Redundant	1	1	1
1.1.2 Electronics	Convert sensor pulses into signals to operate solenoid valves	None - Use backup system	1	2	2
1.1.3 Pneumatic System	Produce torque for spin/despin by flow of nitrogen (N_2) through nozzles	Mission failure	5	2	10
1.2 Command Backup Spin Control System	Control spin rate by ground command	None if automatic system is operating			
1.2.1 Sensors	Generate pulse when exposed to sun	None if automatic system is operating	1	1	1
1.2.2 Electronics	Convert sensor pulses to permit monitoring of spin rate and command activation of pneumatic system	None if automatic system is operating	1	2	2

TABLE D-1. OSO Failure Effects Analysis (Cont)

2.0 COARSE AZIMUTH CONTROL SYSTEM

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
2.1 Sensors					
2.1.1 Coarse Eyes	Detect sun for coarse orientation of sail each spacecraft day (within 3° of sun)	Inability to stabilize sail; severe mission degradation	4	1	4
2.1.2 Target Eyes	Provide increased pointing accuracy	Loss of pointing accuracy; some mission degradation	3	1	3
2.2 Electronics	Receive error signals from sensors and convert to driving power for azimuth torque motor	Severe mission degradation	4	2	8
2.3 Servomotor	Provide torque for orientation of sail	Severe mission degradation	4	1	4

TABLE D-1. OSO Failure Effects Analysis (Cont)

3.0 PITCH CONTROL SYSTEMS

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
3.1 Automatic Pitch Control System.	Keep spin axis of spacecraft normal to solar vector ($\pm 3^{o}$)	None - Use backup system (3.2)			
3.1.1 Sensors	Provide angular pitch error signal to pitch control system	None - Use backup system	1	1	1
3.1.2 Electronics	Converts error signals from sensors to signals for operation of pitch gas jet solenoids.	None - Use backup system	1	2	2
3.1.3 Pneumatics	Provide torque for changing pitch attitude by flow of nitrogen through nozzles on rim of the sail.	Slightly degraded performance. Pitch attitude controlled by magnetic bias coil.	2	2	4
3.2 Command Backup Pitch Control System	Provide pitch control through ground command	None, if automatic system is operating			
3.2.1 Sensors	Provide pitch error output signal to spacecraft data handling system for transmission to ground	None, if automatic system is operating	1	1	1

3.2.2 Electronics	Monitor auto system and switch to manual mode if pitch error correction continues for 107 ±15 sec.	None, if automatic	1	2	2

TABLE D-1. OSO Failure Effects Analysis (Cont)

4.0 POINTING CONTROL SYSTEM

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
4.1 Sensors	Generate error signals to control pointing of experiments.	Degraded mission - loss of pointed experiments			
4.1.1 Fine Control	Generate signals to provide final servo control for azimuth and elevation	Degraded mission - loss of pointed experiments	3	1	3
4.1.2 Target Eye	Generate signal to switch azimuth control from coarse to fine eye output	Degraded mission - loss of pointed experiments	3	1	3
4.2 Electronics	Convert sensor signals to voltage for driving torque motors	Degraded mission - loss of pointed experiments	3	2	6
4.3 Torque Motors	Provide torque for azimuth and elevation correction	Degraded mission - loss of pointed experiments	3	1	3

TABLE D-1. OSO Failure Effects Analysis (Cont)

5.0 SCANNING CONTROL SYSTEM

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
5.1 Sensors, Readout	Provide azimuth and elevation error signals	Degraded mission	3	1	3
5.2 Electronics	Through ground command, turn on and off the raster signal to scan the solar disc	Degraded mission	3	2	6

6.0 OTHER CONTROL EQUIPMENTS

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
6.1 Nutation Damper	Damp S/C nutation	Degraded mission	2	1	2
6.2 Magnetic Bias Coil	Assist in control of pitch attitude; conserve N_2	Degraded mission - shortened life due to more rapid use of N_2	2	1	2
6.3 Aspect Measuring System	Determine 3-axis aspect of S/C with respect to celestial sphere				
6.3.1 Spin Orientation and Rate Electronics		Degraded mission	4	2	8
6.3.2 Magnetometer		Degraded mission	4	2	8

TABLE D-1. OSO Failure Effects Analysis (Cont)

7.0 TELEMETRY SYSTEM

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
7.1 Digital Multiplexer & Encoders (2)	Convert analog and digital signals to proper format for storage and transmission	None - If one unit fails. (Block redundancy.) Mission failure if both units fail	5	1	5
7.2 Analog Subcommutators (2)	Sample data from multiple sources and multiplex for input to digital multiplexer and encoder	Degraded mission if 1 unit fails. Mission failure if both units fail.	5	1	5
7.3 Tape Recorders (2)	Store information during orbit and high speed playback for transmission to ground on command.	None if one unit fails. (Block redundancy.) Mission failure if both units fail.	5	2	10
7.4 Frame Counter and Submultiplexer	Provides timing for data multiplexing	Mission failure	5	1	5
7.5 P.C.M. Junction Box	Provides electrical interface between data units and communication subsystem.	Mission failure	5	1	5
7.6 Transmitters (2)	Provide beacon signal for tracking or transmit telemetry data.	None if one unit fails (Block redundancy). Mission failure if both units fail.	5	1	5

TABLE D-1. OSO Failure Effects Analysis (Cont)

8.0 COMMAND SYSTEM

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
8.1 Command Receivers (2)	Receive and demodulate ground command signals	None if one unit fails (Block redundant). Mission failure if both units fail	5	1	5
8.2 Command Decoders (3)	Decode signals from command receiver and activate commanded functions.	Degraded mission if 1 or 2 units fail. Mission failure if all encoders fail.	5	1	5

TABLE D-1. OSO Failure Effects Analysis (Cont)

9.0 POWER SUPPLY & DISTRIBUTION

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
9.1 Solar Array	Convert solar energy to electrical power	Mission failure	5	1	5
9.2 Battery Pack	Provide electrical power during off sun condition	Mission failure	5	1	5
9.3 Continuous Power Bus	Connects power from battery to spacecraft loads	Mission failure	5	1	5
9.4 Launch Power Bus	Applies battery to selected circuits during launch	Mission failure	5	1	5
9.5 Orbit Power Bus	Control power distribution to either day power bus or night power bus	Mission failure	5	1	5
9.6 Day Power Bus	Provide power to appropriate equipments during S/C day	Mission failure	5	1	5
9.7 Night Power Bus	Remove power to selected equipments during S/C night	Degraded (shortened) mission	4	1	4
9.8 15V Regulator	Provide regulated voltage to selected S/C loads	Mission failure	5	1	5
9.9 Azimuth Shaft Assembly	Connects wheel and sail structures	Mission failure	5	3	15

TABLE D-1. OSO Failure Effects Analysis (Cont)

10.0 ANTENNA SYSTEM

Item	Function	Effect of Complete Item Failure	Criticality Rank	Prob. of Failure Rank	Total Rank
10.1 Antenna Array (2)	Receive & transmit signals	None if one unit fails. Mission failure if complete array fails.	5	1	5
10.2 Match Boxes (2)	Match impedance between antenna and transmission line	None if one unit fails. Mission failure if both fail	5	1	5
10.3 Power Divider	Provides equal power distribution to two active antennas	Mission failure	5	1	5
10.4 Diplexer	Provides connection and isolation to allow trans-mitting and receiving through a common antenna array	Mission failure	5	1	5
10.5 Coaxial Relay	Selects one of the two transmitter outputs to be applied to the antenna system	Mission failure	5	1	5
10.6 Hybrid Circulator	Permits simultaneous operation of two receivers from same antenna	Mission failure	5	1	5

APPENDIX E

RENDEZVOUS REQUIREMENTS

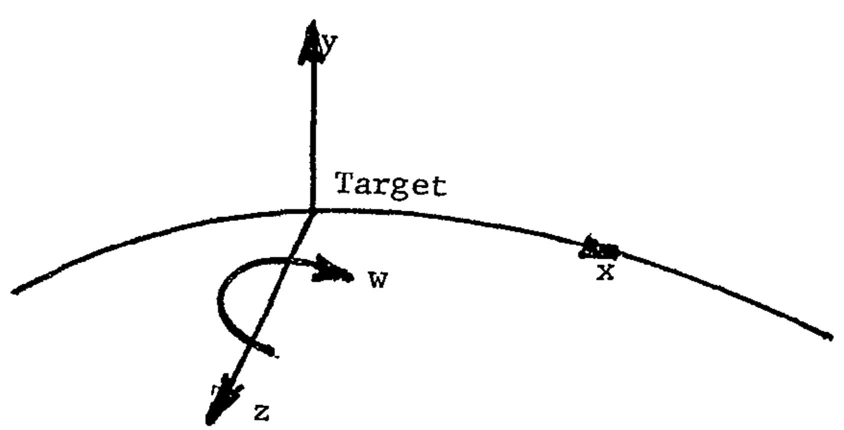
General Electric Company self-funded studies performed in the past were used to examine rendezvous requirements. Although this data was not requested, some of the data derived from these studies were used to establish the propulsion and tracking requirements for this study and is therefore presented.

E.1 RENDEZVOUS AT LOW ORBITS

For this analysis, it was assumed that the orbit of interest is 435 nm circular and that the Remote Manipulator Spacecraft was injected into a 467 nm by 435 nm orbit. After orbit stabilization and tracking data acquisition, a gross rendezvous maneuver was initiated to place the Spacecraft in a 419 nm circular orbit. This maneuver required approximately 100 fps ΔV. At the completion of this maneuver, the remote manipulator would be in a position behind and below the target vehicle. Experience in other programs has shown that this is a preferred orientation because visual acquisition of the target vehicle is facilitated when it is viewed against a star background.

E.1.1 DISPERSION ANALYSIS

The coordinate system employed is a target centered rotating system oriented as follows:



The time history of remote manipulator spacecraft position relative to the target is:

$$x = x_o - \frac{2\dot{y}_o}{w} + (-3\dot{x}_o - 6wy_o)t - 2\,(-3y_o - \frac{2\dot{x}_o}{w}\ \text{sinwt} + 2\ \frac{\dot{y}}{w}\ \text{coswt} \tag{E-1}$$

$$y = 4y_o + \frac{2\dot{x}_o}{w} + (-3y_o - \frac{2\dot{x}_o}{w})\ \text{coswt} + \frac{\dot{y}_o}{w}\ \text{sinwt} \tag{E-2}$$

$$z = \frac{\dot{z}_o}{w}\ \text{sinwt} + z_o\ \text{coswt} \tag{E-3}$$

where:

w = angular rate of the target

$x_o, \dot{x}_o$ = the position and velocity of the remote manipulator spacecraft in the x direction at $t = 0$ (initiation of rendezvous maneuver).

$y_o, \dot{y}_o$ = the position and velocity in the y direction at $t = 0$

$z_o, \dot{z}_o$ = the position and velocity in the z direction at $t = 0$

If the desired time to rendezvous is t, the velocity required to rendezvous is given by:

$$\frac{\dot{x}_o}{w} = \frac{x_o\ \text{sinwt} - y_o\left[6wt\ \text{sinwt} - 14\ (1-\text{coswt})\right]}{3\ wt\ \text{sinwt} - 8\ (1-\text{coswt})} \tag{E-4}$$

$$\frac{\dot{y}_o}{w} = \frac{-2\ x_o\ (1-\text{coswt}) + y_o\ (4\ \text{sinwt} - 3wt\ \text{coswt})}{3\ wt\ \text{sinwt} - 8\ (1-\text{coswt})} \tag{E-5}$$

$$\frac{\dot{z}_o}{w} = -z_o\ \cot\ (wt) \tag{E-6}$$

Using a Monte Carlo technique, errors in the initial position, and velocity of the vehicle as well as errors in the pointing of the ΔV were modeled and the resulting dispersions in the position at rendezvous were calculated. For the case investigated, the terminal rendezvous maneuver was initiated at $x_o = -40$ nm and $y_o = -16$ nm. The standard deviation of the tracking errors employed were:

$$\sigma x_o = 40 \text{ ft} \qquad \sigma \dot{x}_o = 0.013 \text{ fps}$$

$$\sigma y_o = 18 \text{ ft} \qquad \sigma \dot{y}_o = 0.02 \text{ fps}$$

$$\sigma z_o = 22 \text{ ft} \qquad \sigma \dot{z}_o = 0.04 \text{ fps}$$

One sigma pointing errors of 0.33 degree and 1 degree were considered, and the angular travel of the target was chosen as 216 degrees. This transfer angle nearly minimizes the required ΔV for the terminal rendezvous maneuver considered; the total nominal ΔV required is approximately 70 fps.

E.1.2 TWO IMPULSE TERMINAL RENDEZVOUS

In the two-impulse rendezvous the initial impulse is employed to place the spacecraft on the transfer trajectory with the second maneuver being used to match velocity with the target. The standard deviations of position and rendezvous for the two velocity pointing errors are:

σ pointing $= 0.33^{\circ}$

$\sigma x = 1290$ ft	$\sigma \dot{x} = 0.27$ fps	$\sigma x = 3150$ ft	$\sigma \dot{x} = 0.47$ fps
$\sigma y = 150$ ft	$\sigma \dot{y} = 0.21$ fps	$\sigma y = 237$ ft	$\sigma \dot{y} = 0.58$ fps
$\sigma z = 83$ ft	$\sigma \dot{z} = 0.18$ fps	$\sigma z = 284$ ft	$\sigma \dot{z} = 0.56$ fps

E.1.3 MULTI-IMPULSE MANEUVERS

Because of the fairly large standard deviations in x, y, z at rendezvous resulting from a two impulse maneuver, it was decided to determine if these dispersions could be reduced by employing a multiple impulse rendezvous scheme. After the first impulse places the vehicle on the transfer trajectory, tracking measurements would be obtained during transit and would be used to determine if a second and, perhaps a third impulse, would be required to correct any execution errors. Because of the addition of velocity impulses to the remote manipulator spacecraft, there would be, for a period of time, a degradation in subsequent tracking accuracy. For this study, it was assumed that after a velocity impulse, there was no degradation in position measurements and that velocity accuracies were reduced to:

σ pointing $= 0.33^\circ$	$\sigma = 1^\circ$
$\sigma \dot{x}_o = 0.1$ fps	$\sigma \dot{x}_o = 0.3$ fps
$\sigma \dot{y}_o = 0.1$ fps	$\sigma \dot{y}_o = 0.3$ fps
$\sigma \dot{z}_o = 0.1$ fps	$\sigma \dot{z}_o = 0.3$ fps

Because of time limitations, the modeling of the multiple impulse maneuver was simplified in the following manner: before each impulse, the remote manipulator spacecraft was assumed to be located at the mean position and possess the mean velocity at the time of impulse. This assumption should influence only the total velocity impulse required and not the dispersions of the position at rendezvous.

E.1.4 THREE IMPULSE MANEUVERS

The second impulse is applied after an angular travel of 108 degrees. At intercept the dispersions are:

σ pointing $= 0.33^\circ$	σ pointing $= 1^\circ$
$\sigma x = 340$ ft	$\sigma x = 914$ ft
$\sigma y = 290$ ft	$\sigma y = 775$ ft
$\sigma z = 90$ ft	$\sigma z = 270$ ft

E.1.5 FOUR IMPULSE MANEUVERS

The second impulse is applied after an angular travel of 108 degrees and the third at 162 degrees. At intercept the dispersions are:

σ pointing = 0.33°	σ pointing = 1°
σx = 100 ft	σx = 280 ft
σy = 115 ft	σy = 340 ft
σz = 80 ft	σz = 235 ft

To reduce the probability of the remote manipulator spacecraft colliding with the target at the completion of the terminal rendezvous maneuver, the maneuver may have to be biased. Although no attempt was made to determine the required biasing, the velocity requirements for a bias of 1000 feet was determined. It was assumed that before the attempt to rendezvous from the biased position, tracking measurements had again reduced the uncertainties in position and velocity to:

σx_o = 40 ft	$\sigma \dot{x}_o$ = 0.013 fps
σy_o = 18 ft	$\sigma \dot{y}_o$ = 0.02 fps
σz_o = 22 ft	$\sigma \dot{z}_o$ = 0.04 fps

For x_o = -1000 feet, a one sigma pointing error of 1 degree, and a transfer of 9 and 18 degrees the velocity requirements and position uncertainties at rendezvous are:

<u>9° Transfer</u>

σx = 40.4 ft.	$\sigma \dot{x}$ = 0.025 fps	
σy = 23.3 ft.	$\sigma \dot{y}$ = 0.12 fps	ΔV = 13.3 fps
σz = 25.2 ft.	$\sigma \dot{z}$ = 0.12 fps	

18^{o} Transfer

$\sigma x = 42$ ft. $\quad \sigma \dot{x} = 0.026$ fps

$\sigma y = 25$ ft. $\quad \sigma \dot{y} = 0.067$ fps $\quad \Delta V = 6.7$ fps

$\sigma z = 26.5$ ft. $\quad \sigma \dot{z} = 0.066$ fps

E.1.6 SUMMARY

After injection into orbit, and stabilization and tracking acquisition is completed, a gross rendezvous maneuver is employed to place the remote manipulator spacecraft in a position below and behind the target vehicle before initiation of the terminal rendezvous maneuver. Of the several terminal maneuvers analyzed, the Four Impulse Maneuver places the remote manipulator spacecraft in the best position to visually acquire the target vehicle. This maneuver reduces the dispersion to $\sigma x = 280$ ft, $\sigma y = 340$ ft, and $\sigma z = 235$ ft (σ pointing $= 1^{o}$).

E.2 RENDEZVOUS AT SYNCHRONOUS ORBIT

The ΔV requirements to achieve rendezvous at synchronous altitude are described. The total ΔV required to remove injection error is approximately 95 feet per second. Additional ΔV to achieve rendezvous after injection is given in Table E-1.

Table E-1. Rendezvous ΔV Requirements at Synchronous Altitude

Time to Rendezvous	Additional ΔV
1 Day	170 fps
2 Days	86 fps
3 Days	56 fps
4 Days	40 fps

E.3 GROUND TRACKING OF THE REMOTE MANIPULATOR SPACECRAFT VEHICLE

The time available to track the vehicle was determined for both the STADAN and Manned tracking networks. The vehicle was assumed to be in a 35 degree inclined 500 nm circular orbit; the vehicle was tracked for a total of four orbits (400 minutes) and the total tracking time is 172 minutes for the STADAN network and 102 minutes for the Manned network.

STADAN NETWORK: A total of 16 tracking stations were considered; the deviation angle limit was taken as 7.5 degrees.

The locations of the sites are given in Table E-2.

Table E-2. Location of STADAN Stations

Station Number	Location	Latitude	Longitude
1	Alaska	64.977°N	212.485°E
2	NESC/CDA	64.979°N	212.505°E
3	Barstow, Cal.	35.330°N	243.101°E
4	Carnarvon, Aust.	24.904°S	113.716°E
5	Mobile Station Darwin	12.289°S	130.816°E
6	Fort Meyers, Fla.	26.548°N	278.134°E
7	Johannesburg, South Afr.	25.883°S	27.708°E
8	Kauai, Hawaii	22.124°N	202.331°E
9	Lima, Peru	11.777°S	282.850°E
10	Orroral, Aus.	35.631°S	148.956°E
11	Quito, Ecuador	0.623°S	281.421°E
12	Rosman, N.C.	35.200°N	277.128°E
13	Saint Johns, Nfld.	47.741°N	307.279°E
14	Santiago, Chile	33.150°S	289.331°E
15	Transnarive, Madagascar	19.008°S	47.300°E
16	Winkfield, England	51.446°N	359.304°E

MANNED TRACKING NETWORK: A total of 13 sites were included in the Manned Tracking network. The locations of the sites are given in Table E-3.

Table E-3. Location of Manned Sites

Station Number	Location	Latitude	Longitude
17	Cape Kennedy	28.482°N	279.423°E
18	Bermuda	32.348°N	295.347°E
19	Canary Islands	27.735°N	344.400°E
20	Zanzibar	7.33 °S	39.330°E
21	Mucheci, Aus.	31.599°S	115.928°E
22	Kano, Nigeria	11.580°N	8.28 °E
23	Woomera, Aus.	30.819°S	136.837°E
24	Canton, Is.	2.790°S	188.330°E
25	Hawaii	22.125°N	200.329°E
26	Southern Cal.	34.583°N	239.439°E
27	Guaymas, Mexico	27.958°N	249.279°E
28	White Sands	32.358°N	253.631°E
29	South, Texas	27.655°N	262.620°E

In the first case investigated, the vehicle was tracked for a total of two orbits with the starting latitude and longitude of the vehicle being 0 and 330 degrees E respectively. In the orbit, the vehicle is tracked a total of 50 minutes by the STADAN network and 22.8 minutes by the Manned network. During the second orbit, the vehicle is tracked 55.4 minutes by STANDAN AND 35.1 minutes by the Manned network.

In the second case considered, the vehicle was again tracked for two orbits (200 minutes) with a starting latitude of 0 degree and longitude of 60 degrees. With these initial conditions, the vehicle is viewed a total of 66 minutes by the STANDAN network and 45 minutes by the Manned network.

In both the cases investigated, there were periods during which the vehicle was viewed concurrently by more than one station. For instance, the Hawaii stations in the two networks viewed the vehicle at essentially the same time. Thus, of the total 274 minutes of tracking over the four orbits, there is concurrent tracking by more than one station for at least 75 minutes; therefore, the period in which the vehicle is out of view of all stations will exceed 126 minutes. The ground trace of the vehicle for the cases considered is shown in Figures E-1 and E-2 with the radar cones of the STADAN network superimposed on the mercator projection.

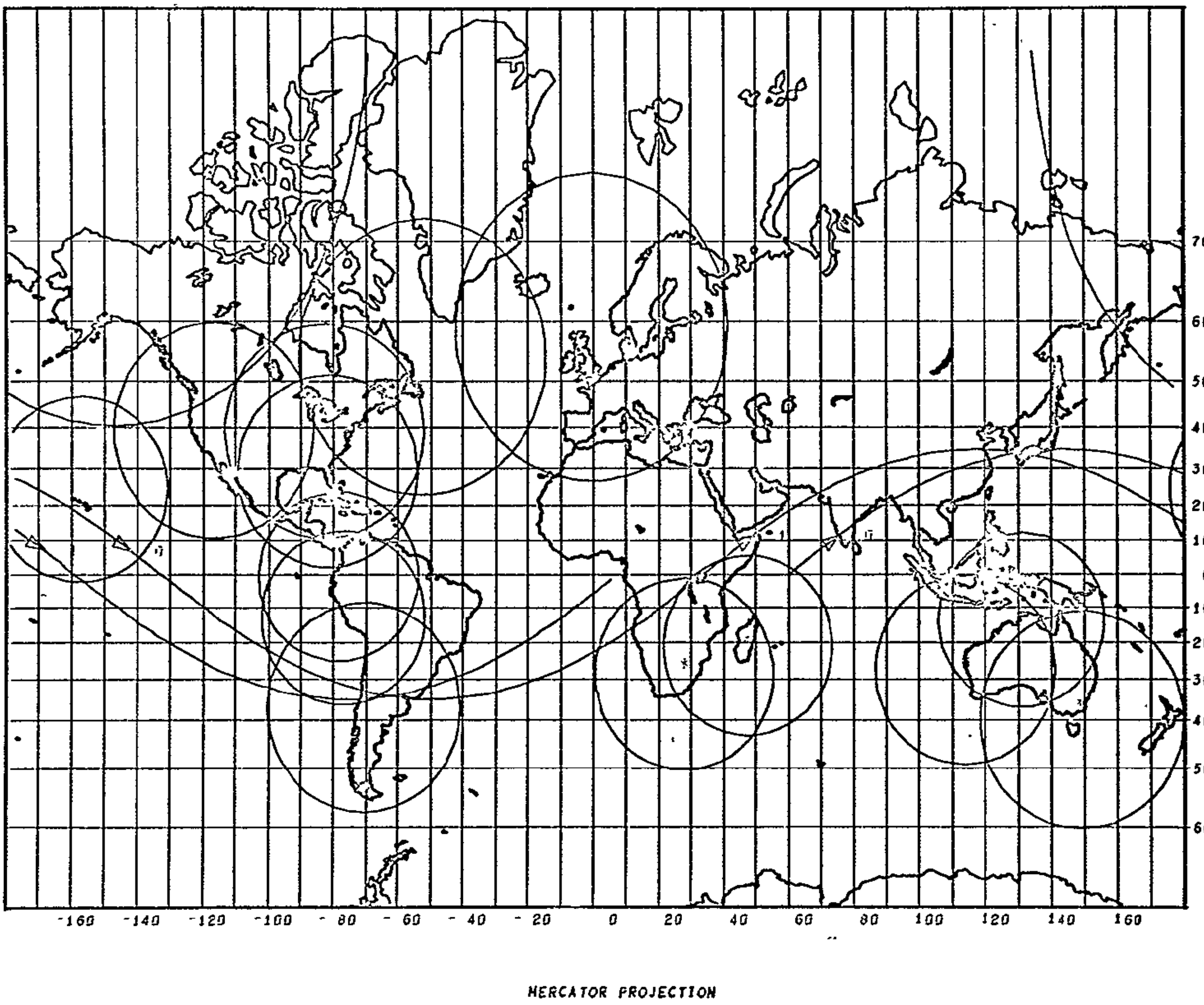


Figure E-1. STADAN Tracking of Remote Manipulator Spacecraft (Initial Longitude = 60 Degrees)

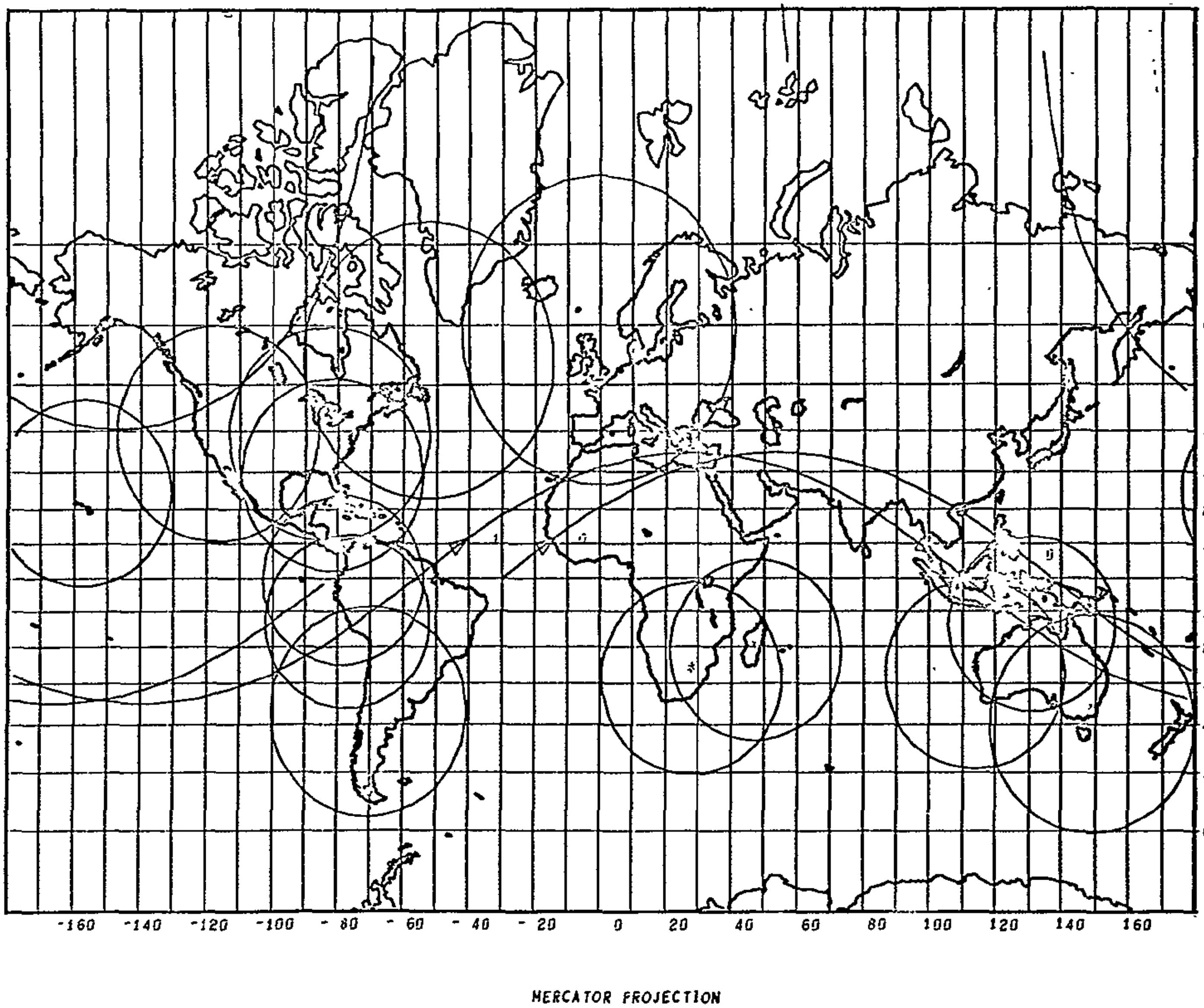


Figure E-2. STADAN Tracking of Remote Manipulator Spacecraft
(Initial Longitude = 330 Degrees)

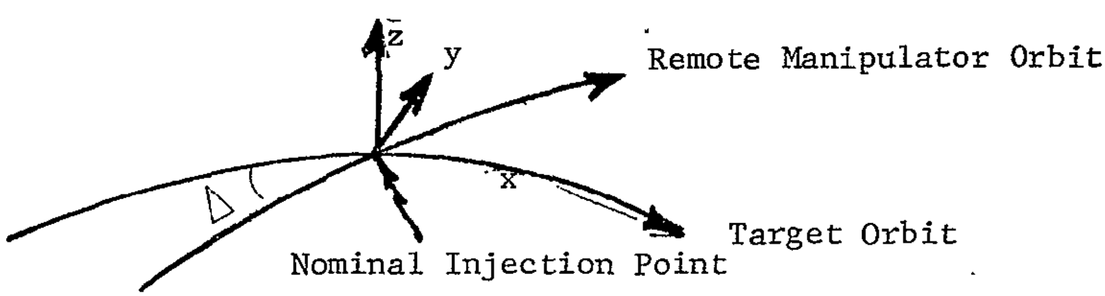
E.4 PROBABILITY OF REMOTE MANIPULATOR SPACECRAFT VEHICLE COLLIDING WITH TARGET AT INJECTION

At injection of the vehicle into the orbit of the target, there is a possibility of the two vehicles colliding. If the probability of this event occurring is sufficiently small, the injection point of the vehicle need not be biased; otherwise, the injection must be biased to reasonably assure that a collison does not occur. With the expected 3σ injection errors at 435 nm of 20 seconds in period, 0.3 degree in inclination and 34 nm in altitude biasing of the injection point is not warranted because the probability of collision is of the order of 10^{-11}.

Analysis: The 36 errors employed in the analysis were:

Altitude error $(h_a - h_p)$	=	34 nm
Inclination error	=	0.3 degree
Period error	=	20 seconds

Consider a coordinate system located at the nominal injection point with the position x-axis in the direction of motion of the target, the positive z-axis in the outward-radial direction, and the y-axis completing a righthanded system.



The velocity of the vehicle at injection is approximately 24440 ft/sec; this translates into the following dispersions:

$3\sigma x \quad = \quad 24,400 \text{ fps} \times 20 \text{ sec} \times \cos(0.3^{o}) = 488,000 \text{ ft}$

$3\sigma y \quad = \quad 24,400 \text{ fps} \times 20 \text{ sec} \times \sin(0.3^{o}) = 2,560 \text{ ft}$

$3\sigma z \quad = \quad 32 \text{ nm} = 194,000 \text{ ft}$

Assuming that the volume of impact is a 15 foot cube located at the center of the coordinate system, the probability of the vehicle being located in the impact volume (Pi) is given by:

$$Pi = P\left\{-7.5 < x < 7.5,\ -7.5 < y < 7.5,\ -7.5 < z < 7.5\right\}$$

Assuming that the position errors in each of the directions are independent and normally distributed, this reduces to:

$$Pi = P\left\{-7.5 < x < 7.5\right\} \times P\left\{-7.5 < y < 7.5\right\} \times \left\{P \quad -7.5 < z < 7.5\right\}$$

$$P\left\{-7.5 < x < 7.5\right\} = \int_{7.5}^{7.5} \frac{e^{-1/2\left(\frac{x}{\sigma x}\right)^2}}{x\sqrt{2\pi}}\, dx \doteq \frac{15 \text{ ft}}{\sigma x\sqrt{2\pi}} \qquad \text{(E-7)}$$

$$P\left\{-7.5 < x < 7.5\right\} = \int_{7.5}^{7.5} \frac{e^{-1/2\left(\frac{y}{\sigma y}\right)^2}}{y\sqrt{2\pi}}\, xy = \frac{15 \text{ ft}}{\sigma y\sqrt{2\pi}} \qquad \text{(E-8)}$$

$$P\left\{-7.5 < x < 7.5\right\} = \int_{7.5}^{7.5} \frac{e^{-1/2\left(\frac{z}{\sigma z}\right)^2}}{z\sqrt{2\pi}}\, dz = \frac{15 \text{ ft}}{\sigma z\sqrt{2\pi}} \qquad \text{(E-9)}$$

The probability of the vehicle being located in a 15-foot cube located at the center of the coordinate system becomes:

$$Pi = \frac{(15 \text{ ft.})^3}{\sigma_x \sigma_y \sigma_z\, 2\pi\sqrt{2\pi}} = 2.4 \times 10^{-11} \qquad \text{(E-10)}$$

This analysis has not accounted for the uncertainties in the targets position, if these are included the probability of the two vehicles colliding would be reduced further.

APPENDIX F

TIME DELAY

With a ground control station, it is necessary to cope with a communications time delay in the video and positional control information. The total delay (round trip) will be as much as much as 3/4 second for the low orbiters and 1/4 second for the synchronous orbiters. The longer length of the low orbiters results from the use of relay satellite needed to maintain contact. The time delay will have a twofold degrading effect. First, task times will be necessarily lengthened and second, when the manipulators are in a bilateral mode, the servo performance will be degraded to maintain stability. Investigations of both areas have been performed at MIT.

Consider first the increased task time factor. Using unilateral position controls; i.e., no force feedback, it was found that self paced tasks could be accomplish with time delays with the only degraded performance factor being completion time. (See References 1 and 2.) This was accomplished through use of an open-loop move-and-wait strategy; i.e., the operator would move the master to what was thought to be the proper distance and then wait the time delay to observe the result before making the next move. If this strategy was used, a method was found for predicting task times for any time delay. The procedure for any task requires the time to do the task with no delays, t_o, the time to do the task if the operator is only allowed to move when his eyes are closed, t_N, the number of pauses one would use in the move-and-wait strategy, N, and the operator reaction time, t_r. These last three numbers, t_N, N and t_r, are obtained by observing the operator performing the task in the open loop manner prescribed. The actual completion time could be estimated by:

$$t_c = \frac{t_{c_1} + t_{c_2}}{2} = \frac{t_o + t_N}{2} + (N+1)\left(t_d + \frac{t_r}{2}\right)$$

$$t_{c_1} = t_o + (N+1)\,(t_d + t_r)$$

$$t_{c_2} = t_N + (N+1)\,t_d$$

where t_c = completion time and t_d is the full-time delay.

The times given in the step by step analysis of Sections 3 and 4 were basically estimates of t_o, i.e., the time to accomplish that step with a remote manipulator with TV visual feedback, but no time delay. To estimate t_N and N would require extensive testing which time did not allow. However, one of the longer simulations was run to get an indication these values. The task of recharging the N_2 supply of the OAO was run in the open-loop, move only with eyes closed strategy. The result is the first entry in the table below:

	t_o	t_N	N	t_N/t_o	t_o/N
OAO Task	123 sec.	315 sec.	75	2.56	1.6
MIT Task	13	22	7.7	1.69	1.7

The second entry is the most complicated task considered in the MIT study (Reference 1). We use this data to estimate N and t_N as a function of t_o. The factor t_o/N was rather steady but t_N/t_o was not as consistent.

However, using $t_N/t_o = 2$ and 0.25 second as the reaction time we arrive at the on time delay factor as:

$$t_c = t_o \left[1.5 + \frac{t_d + 0.125}{1.65} \right] \tag{F-1}$$

or for $t_d = 0.75$ sec $\qquad t_c \doteq 2.03\, t_o$

$t_d = 0.25$ sec $\qquad t_c \doteq 1.73\, t_o$

where again t_o is the estimate without time delay and t_d is the total time delay. Thus, we predict task times will about double if the move-and-wait strategy is employed to cope with the time delay.

The effect of the time delay on force feedback manipulators is discussed in Reference 3. The problem here is that the force feedback signal comes back delayed to the operator's hand. Instability can occur by the operator reacting to a force feedback signal out of phase because of the time delay. The experimental work described in the reference needs further clarification; i.e., extension of the work to an actual manipulator system instead of the x-y recorder apparatus used, before we can quantitatively design the manipulator servo system. Preliminary work is under way to verify this work in a separate self-funded effort. Various solutions have been suggested, such as adaptive networks that adjust the master force feedback level. Of course, the force can be fed back in ways other than the conventional bilateral manipulator to avoid the stability problem. Whether these ways retain the effectiveness of the bilateral manipulator has not yet been evaluated.

References

1. Ferrell, W.R., "Remote Manipulation with Transmission Delay," NASA Tech. Note TN D-2665, February 1965.

2. "Experiments on Delayed Remote Manipulation," prepared by MIT in 1967 for General Electric. Extends previous reference to six dimensional work.

3. Ferrell, W.R., "Delayed Force Feedback," a paper at Symposium on Human Communication at Metropolity Chapter of the Human Factors Society, New York University, 12 June, 1965.

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